SA226 SERIES

M7 AEROSPACE, LLC

Maintenance Manual

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MAINTENANCE MANUAL

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27-10	27-10-01 PG 209	MAR 06/15	MAR 06/15	M7	OCT 07/17	M7	R123
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TEMPORARY REVISIONS

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MAINTENANCE MANUAL

LETTER OF TRANSMITTAL

TO: ALL HOLDERS OF SA226 SERIES MAINTENANCE MANUAL

REVISION 123, DATED OCTOBER 07, 2017

1. REVISION HIGHLIGHTS

A. Revised the following sections, many of which required conversion to new software used by M7 Aerospace Technical Publications. This conversion affects headers, format and general layout, so sections converted for the first time will show the new revision date on all pages. If technical content was not changed on a page of a newly converted section, a change bar will be located next to the page number and date at lower right corner of page.

(1) Introduction Section: Revised General Information section; added Supplemental Manuals/ Documents section and ATA 100 Assignments section.

(2) Incorporated all active Temporary Revisions: 27-09, 27-10, 29-02, 05-10, 71-02 and 27-11.

(3) Section 05-10-00: Changed CVR intervals to Ref. Mfg. CMM; Added A.D. 2016-25-12, A.D. 2016-15-02, A.D. 2016-10-01 and A.D. 2017-07-01.

(4) Section 05-50-10: Added statement suggesting use of FDR (if installed) to determine overweight/hard landing & severe turbulent air events.

(5) Section 12-20-10: Relocated and revised lubrication procedures; Revised Tables throughout for clarity; added Fig/Table for MLG Uplock Mechanism Lube.

(6) Section 20-40-00: Updated entire section to latest specifications and procedures.

(7) Section 21-00-00: Renumbered Figures and revised Figure references throughout; Corrected effectivity on pages 5, 8, 16, 18 and 19.

(8) Section 21-10-10: Replaced outdated Inspection/Check instructions to SB226-21-022 reference.

(9) Section 21-30-60: Corrected trip point altitude in Description and Operation; Added Cabin Altitude Warning Operational Test.

(10) Section 23-70-00: Added G Switch Removal/Installation procedures.

(11) Section 26-10-10: Expanded test procedure for overheat detectors in wheel well and engine compartment same as SB226-26-005.

(12) Section 26-10-20: Added application of RTV734 sealer to 250 degree thermal switch installation procedure.

(13) Section 27-00-00: Added 1/8" cable to B. for tension tolerance on Figure 209.

(14) Section 27-10-01: Incorporated TR27-10 revising Warning in Push-Pull Tube Installation, and TR27-11 deleting obsolete aileron drum repair and referring to M7 Service Engineering for repairs; Revised Figure 204 for clarity.

(15) Section 27-10-03: Corrected Figure 1 to show aileron trim on left wing only; Added freeplay and travel check requirement to aileron trim tab and actuator installation procedures.

(16) Section 27-30-01: Revised cable numbering throughout illustrations and text, added Figure 2 and revised procedure, for clarity.

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(17) Section 27-30-02: Revised Servicing instruction or clarity; Revised Walking Beam Removal/ Installation procedures and Figure 202 for clarity; Replaced Approved Repair section with Adjustment/Test or column clearance and side play correction.

(18) Section 27-30-03: Added two shim options to align quadrant with stop assembly.; Revised remaining procedure including Figures 201 and 202 for clarity.

(19) Section 27-40-01: Incorporated TR27-09 - Corrected Steps C.(5) and (7) conflict with Figure 204 for degrees of travel between Zero Degree rivet and Top/Bottom of Takeoff Band.

(20) Section 27-40-02: Corrected Table 201 to show 27-19008-006 and -007 are acceptable replacements for all actuators, removed autopilot restriction from DL5040M6 and M8 actuators as this is determined by autopilot system installation, and added note requiring 27K44063-001 Kit prior to installation of DL5040M8.

(21) Section 27-42-03: Corrected Pg 205 step (10) to read, Adjust Nose Up end of band.

(22) Section 29-00-00: Added an external hydraulic power connect/disconnect procedure to this section, deleted servicing procedure from this section as this is duplicated and belongs in 12-10-20 only.

(23) Section 29-10-04: Incorporated TR29-02 and revised step 14 for clarity.

(24) Section 32-10-10: Revised Fig 203 to add axle locking bolt and protective washer; deleted strut inspection procedure as this is called out in SB226-32-033, which is referred to by ST-UN-M002.

(25) Section 32-20-00: Added R&I of NG Door Actuator roller and brackets to this section; Revised remaining procedures as required for clarity.

(26) Section 32-20-10, pg 201: Deleted "and lock washers" from 1.B.(1).

(27) Section 32-20-30: Revised Figure 1 and added Figure 201 (moved previous 201 to 202) for clarity; Added NOTE: Upper and Lower drag braces must be assembled as matched sets; Revised remaining instructions for clarity.

(28) Section 34-10-00: Added alternate means for Pitot Leak Check other than manometer; Corrected step 2.E.(16).

(29) Section 55-00-00, pg 2: Removed NOTE at bottom of page 2 referring to Chapter 27 for removal, installation and inspection of vertical stabilizer as there are no such instructions in this manual.

(30) Section 56-10-00: Changed Title of Figure 201A; Added NOTEs referring to Figure 201A to Side Window Installation instructions; Added alternate for window trim paint; Revised Side Window Installation instructions for clarity.

(31) Section 71-00-30: Incorporated TR71-02 and corrected step L. reference to Figure 208, SHEET 1 of 2.

MAINTENANCE MANUAL

INTRODUCTION TO THE MAINTENANCE MANUAL

1. General Information

A. Except where noted, maintenance instructions in this manual apply to all SA226 Series airplanes manufactured by Fairchild Aircraft. As a general rule, part numbers will not be specified in this Maintenance Manual unless necessary for clarity. Refer to the SA226 Series Illustrated Parts Catalog (IPC) for a specific part number (P/N).

B. Aircraft 406, 415, and 416 were originally Metro IIA (SA226-TC) aircraft converted to Metro III (SA227-AC) configuration. For these aircraft, refer to SA227 Series technical publications.

2. How To Use This Manual

A. Chapter-Section-Subject Designator

(1) M7 Aerospace SA226 Manuals comply with ATA 100 standards in the use of six digit chapter-section-subject numbers.

(2) Under this numbering system and using the example, 27-10-30:

(a) The first pair of numbers (27-) designates the Chapter/System (in this example, Flight Controls).

(b) The second pair of numbers (10-) designates the Section/Sub-system for that chapter (in this example, Aileron).

(c) The third pair of numbers (30) designates the Unit/Subject for the sub-system (in this example, Aileron Push-Pull Tubes).

(3) Page Numbering

(a) Maintenance Manual pages are numbered consecutively:

Description and Operation	Page blocks begin with page 0 (if the first page is an illustration) or page 1 (if the first page is text), and continue through page 99.
Fault Isolation	Page blocks begin with page 100 (if the first page is an illustration) or page 101 (if the first page is text), and continue through page 199.
Maintenance Practices	Page blocks begin on page 200 (if the first page is an illustration page), or page 201 (if the first page is text), and continue through page 299.

(b) Illustrations and tables are numbered consecutively.

(c) Illustrations used in this manual may be assigned a serial number. The serial number will appear in a small box (Example: 6M233030201) appearing on the face of the illustration. This serial number is for M7 Aerospace Technical Publications internal use only.

(d) In some cases, **INTENTIONALLY LEFT BLANK** pages may be used within the manual. This is done to allow room for chapter expansion and to provide a back page, where necessary.

(4) Service Information

(a) The Maintenance Manual contains a Service Information section at the end of some chapters.

(b) This section, designated as SERVC (e.g., 27-SERVC, 32-SERVC, 61-SERVC), contains special tools, useful maintenance tips, and other information not contained in the body of a chapter.

(5) Abbreviations

(a) Abbreviations are used throughout the SA226 Series Maintenance Manual. The following list contains some of the most common recurring abbreviations:

AFM – Airplane Flight Manual
AC – Alternating Current
Amp – Amperage
ANT – Antenna
APR – Automatic Power Recovery
B.L. – Butt Line
C – Centigrade (Celsius)
CB – Circuit Breaker
CW – Clockwise
CCW – Counterclockwise
CAWI – Continuous Alcohol Water Injection
CM – Centimeter
Dia – Diameter
EA – Each
EPU – External Power Unit
F – Fahrenheit
FAR – Federal Aviation Regulation
F.S. – Fuselage Station
GSE – Ground Support Equipment
GTEC – Garrett Turbine Engine Company
IAW – In Accordance With
IPC – Illustrated Parts Catalog
INBD – Inboard
ID – Inside Diameter
KIAS – Knots Indicated Airspeed
KGM - Kilogram
LDG – Landing Gear
LH – Left Hand
MLG – Main Landing Gear
MM – Millimeter
NLG – Nose Landing Gear
N.S. – Nacelle Station
OUTBD - Outboard
PPH – Pounds Per Hour
POT – Potentiometer
RH – Right Hand
TB – Terminal Board
VAC – Volts Alternating Current
VDC – Volts Direct Current

(b) Other abbreviations may be used in this manual. However, when an abbreviation is to be used that does not appear in the list above, the proper name of the item/component/system will be provided, followed by its assigned abbreviation in parentheses; for example, Stall Avoidance System (SAS). After the initial abbreviation has been defined, it may again be used within the section / subsection without further explanation (i.e., SAS Indicator).

3. Airplane Description

A. The SA226 series airplanes, built by Fairchild Aircraft, are low wing monoplanes, consisting of the Merlin IV SA226-T model, Merlin IV SA226-AT model, and Metro model SA226-TC. Merlin models are corporate executive aircraft; the Metro is an airline commuter model.

B. Merlin and Metro models in these series are powered by two turboprop engines equipped with constant speed, full-feathering, reversible propellers. The engines are built by Garrett Turbine Engine Company (GTEC) / Honeywell International Inc.

(1) Metro SA226-TC airplanes incorporate an automatic operating alcohol-water injection (AWI) system.

(2) Use of the AWI system provides augmented emergency power of 940 shaft horsepower to the operable engine in event of engine power loss during takeoff.

C. The forward retracting landing gear is hydraulically actuated. Emergency extension is free fall, assisted by the airstream and an emergency hydraulic hand pump.

D. The nose wheel steering system is electrically actuated by a steering knob on the pilot's left console.

NOTE: On later models, the nose wheel steering is hydraulically actuated and controlled by rudder pedals.

E. The cabin is pressurized, heated and cooled by bleed air ducted through valves, heat exchangers, and refrigeration units.

F. Ice protection includes pneumatic deicing of the wing and horizontal stabilizer leading edge surfaces, electrical deicing of the propellers, windshield, and pitot tubes. Engine bleed air is used to deice engine air intakes.

G. The Metro Model SA226-TC airplane is designed to carry passengers, cargo, or a combination of passengers and cargo. A movable bulkhead provides for isolation of the cargo from the passengers when a combination load is desired.

4. Supplemental Manuals/Documents

A. The following manuals/documents are recommended by M7 Aerospace to supplement this Airplane Maintenance Manual:

(1) SA226 Series Illustrated Parts Catalog.
(2) SA226/227 Series Structural Repair Manual.
(3) SA226 Series Inspection Procedures Manual.
(4) SA226 Series Airframe Airworthiness Limitations Manual, ST-UN-M002.
(5) SA226 Series Aircraft Wiring Manual.
(6) SA226 Series Service Information Manual.
(7) Honeywell Service Bulletin TPE331-72-0180, Engine Service Life Limits of Critical Life Limited Components and Periodic Maintenance Schedules.
(8) Honeywell TPE331-3 and -10 Maintenance Manual.
(9) Hartzell Propeller Instructions for Continued Airworthiness.

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CHAPTER 05 – TIME LIMITS/MAINTENANCE CHECKS

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CHAPTER 05 – TIME LIMITS/MAINTENANCE CHECKS

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LIGHTNING STRIKE – DESCRIPTION AND OPERATION

1. General

A. If the aircraft is struck by lightning or a lightning strike is suspected, the following procedure should be applied:

(1) Carefully inspect the exterior of the aircraft. Evidence of a lightning strike will normally appear as a burned hole or a series of burned holes in surfaces. Plastic parts may be delaminated and/or deformed due to high internal pressures. Normally two or more points will be found, the entry and exit points. Antennas are frequently struck by lightning and should be carefully inspected for evidence of arcing, sooting, or pitting.

(2) From the point of entry, the strike usually spreads aft in a series of small holes or burn marks. After the point of entry and exit are found, the structure between these points should be carefully inspected. Attention should be given to hinges and hinge pins for possible damage. Antennas, electrical, and electronic equipment should be visually checked for damage and functionally checked for operation. If the strike was near the fuel vent, all plumbing should be carefully inspected for damage.

(3) Normal structural repair methods should be used for metal surfaces damaged by lightning. Non-metallic parts may be repaired, but will generally require replacement. Minor pitting can be carefully dressed out. Normal corrosion protective measures should be applied to areas reworked. Damaged equipment or electrical leads should be carefully inspected or replaced as required.

(4) If an engine or propeller has experienced an actual or suspected lightning strike, comply with the special inspections contained in Chapter 72-00-00 of the applicable Manufacturers' Manuals.

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TIME LIMITS - MAINTENANCE PRACTICES

1. General

A. Time Limited Components

(1) M7 Aerospace has established inspection, replacement and/or overhaul time limits for specific aircraft components. These time limits are based on recommendations or requirements from the manufacturer of the specific component as well as M7 Aerospace experience.

(2) Most components listed in Table 201 require inspection, overhaul or replacement at the intervals indicated. We have also included items in this chapter that are only our recommendation, which can and should be adjusted based on the operators experience and environment. The items that are recommended and not mandatory are identified with an asterisk (*) next to the stated interval.

(3) Items in the ACTION column preceded with (ALI) identify Airworthiness Limitation Items which are also listed in the Airworthiness Limitations Manual (ST-UN-M002). The time interval stated for these items cannot be exceeded.

(4) Should a revision to this manual cause a change in retirement life or inspection interval, aircraft that are over the revised time limit(s) should be brought into compliance with this manual at the next scheduled access, in conjunction with the Phase or Letter Check inspection program the airplane is operated under.

Time Limited Aircraft Components
Table 201

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 11 - PLACARDS AND MARKINGS			
ALL.	Emergency Phosphorescent Signs.	Inspect Minimum Brightness.	3 Years, Then Annually.
ATA 21 - AIR CONDITIONING			
203745-3-1.	Cooling Turbine.	Service.	300 FH.
204755-4-1; 204755-4-3; 204755-4-4; 204755-4-5; 204755-4-6.	Cooling Turbine.	Service.	200 FH.
CU79030; 79030.	Pressurization Controller External Filter.	Replace.	1,000 FH*.
12 Ply Gauze.	Pressurization Controller Internal Filter.	Replace.	1,000 FH*.
CU79030; 79030.	Outflow Valve Filter.	Replace.	1,000 FH*.
	Cabin Altitude Warning System.	Operational Test IAW 226-SL-053.	3 Years.

TIME LIMITS - MAINTENANCE PRACTICES

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 23 – COMMUNICATIONS			
A100.	Cockpit Voice Recorder.	Overhaul.	Ref. CVR Mfg. CMM.
89090.	Cockpit Voice Recorder (B&D).	Audio System Check.	Ref. CVR Mfg. CMM.
		Replace Tape.	
		Overhaul.	
ATA 24 – ELECTRICAL POWER			
ALL.	Essential & Non-Essential Buses in LH/RH Consoles & at Battery Bus in J-Box.	Inspect 4 Gauge Bus Wire at Terminations and Inspect 4 Gauge Bus Wire Terminations.	500 FH*.
ALL.	Generator Brushes.	Inspect.	500 FH*.
23070-001.	Starter-Generator Alternator Assy.	Overhaul.	1,000 FH.
23069 Series.	Starter-Generator.	Overhaul.	1,000 FH.
23079 Series.	Starter-Generator.	Overhaul.	1,000 FH.
300SG Series.	Starter-Generator.	Overhaul.	1,000 FH.
27-62208.	Starter Generator.	Overhaul.	1,000 FH.
	Main Battery (Nicad).	Deep Cycle.	Ref. Battery Mfg. CMM.
		Capacitance Check.	
	Main Battery (Lead Acid).	Replace.	Ref. Battery Mfg. CMM.
		Capacity Test.	
8781-K11; AN3230-2.	Bus Tie Switches (Left Essential, Right Essential and Non-Essential).	(ALI) Replace.	16,000 FH.
ATA 25 – EQUIPMENT/FURNISHINGS			
ALL.	Life Preserver.	Inspect.	60 Months after placed in service or no later than 60 Months from date of manufacture. Then every 60 Months.
DK100.	Underwater Locator Beacon (B&D CVR).	Inspect.	Ref. CVR Mfg. CMM.
		Clean and Test.	
		Replace Battery.	
ALL.	Emergency Locator Transmitter.	Inspect per FAR 91.207(d).	1 Year.

PART NUMBER	PART NAME	ACTION	INTERVAL
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452-0130; 452-0130-02.	Emergency Locator Transmitter Battery.	Replace.	2 Years (expiration date label on beacon).
452-0133.	Emergency Locator Transmitter Battery.	Replace.	5 Years (expiration date label on beacon).
ATA 26 – FIRE PROTECTION			
30301100; 30300020.	Fire Extinguisher Bottle.	Hydrostatic Test.	5 Years.
13083-5.	Cartridge (Squib).	(ALI) Replace.	6 Years, 10 Years Max. Life (Storage/Service).
30600-22.	Cartridge (Squib).	(ALI) Replace.	8 Years, 10 Years Max. Life (Storage/Service).
1211.	Fire Extinguisher, Hand Held (Halon).	Weight Check, Recharge, Hydrostatic Test.	In Accordance with Nameplate Instructions.
RT A1200.	Fire Extinguisher, Hand Held (Halon).	Weight Check, Recharge, Hydrostatic Test.	In Accordance with Nameplate Instructions.
ATA 27 – FLIGHT CONTROLS			
ALL.	Pitch Trim Actuator.	(ALI) Travel Time Check. (Ref. AMM 27-40-02, Operational Test-Actuator Time Check).	400 FH.
DL5040M2-3.	Pitch Trim Actuator.	(ALI) Overhaul.	1,200 FH.
DL5040M2-4.	Pitch Trim Actuator.	(ALI) Overhaul.	2,000 FH.
DL5040M2-6.	Pitch Trim Actuator.	(ALI) Overhaul.	2,000 FH.
DL5040M5 (Original actuator installed).	Pitch Trim Actuator.	(ALI) Measure actuator freeplay & inspect for rod slippage. (Ref. SL 226-SL-005).	Ref. A.D. 2007-16-03.
		(ALI) Replace. **NOTE 1**.	Ref. A.D. 2007-16-03.

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 27 – FLIGHT CONTROLS (continued)			
DL5040M5 (Replacement actuator installed).	Pitch Trim Actuator.	(ALI) Measure actuator freeplay & inspect for rod slippage. (Ref. SL 226-SL-005).	Ref. A.D. 2007-16-03.
		(ALI) Replace. **NOTE 1**.	Ref. A.D. 2007-16-03.
DL5040M5 (Overhauled actuator with P/N AA56142 nut assemblies NOT replaced).	Pitch Trim Actuator.	(ALI) Measure actuator freeplay & inspect for rod slippage. (Ref. SL 226-SL-005).	Ref. A.D. 2007-16-03.
		(ALI) Replace. **NOTE 1**.	Ref. A.D. 2007-16-03.
DL5040M5 (Overhauled actuator with P/N AA56142 nut assemblies replaced).	Pitch Trim Actuator.	(ALI) Measure actuator freeplay & inspect for rod slippage. (Ref. SL 226-SL-005).	Ref. A.D. 2007-16-03.
		(ALI) Replace. **NOTE 1**.	Ref. A.D. 2007-16-03.
DL5040M6.	Pitch Trim Actuator.	(ALI) Measure actuator freeplay & inspect for rod slippage. (Ref. SL 226-SL-005).	Ref. A.D. 2007-16-03.
		(ALI) Replace. **NOTE 1**.	Ref. A.D. 2007-16-03.
DL5040M8.	Pitch Trim Actuator.	(ALI) Measure actuator freeplay & inspect for rod slippage. (Ref. SL 226-SL-005).	Ref. A.D. 2007-16-03.
		(ALI) Replace. **NOTE 1**.	Ref. A.D. 2007-16-03.
27-19008-001; 27-19008-002; 27-19008-004; 27-19008-005.	Pitch Trim Actuator.	(ALI) Mech., Elec. Stops Functional Check and No-Back Functional Check . (Ref. SL 226-SL-014).	Ref. A.D. 2007-16-03.
		(ALI) Replace. **NOTE 1**.	Ref. A.D. 2007-16-03.
27-19008-006; 27-19008-007.	Pitch Trim Actuator.	(ALI) Overhaul.	Ref. A.D. 2007-16-03.

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 27 – FLIGHT CONTROLS (continued)			
27-43062-011; 27-43062-013; 27-43062-015; 27-43062-019.	Upper Attach Fitting, Pitch Trim Actuator.	(ALI) Inspect. (Ref. SB 226-27-081).	Ref. A.D. 2016-25-12.
AN23-12; NAS1103-6D.	Bolt, FWD Rudder Cable Attachment at Rudder Pedal Arm.	(ALI) Replace. (Ref. SB 226-27-027).	Ref. A.D. 92-19-08.
NAS75-3-004.	Bushing, FWD Rudder Cable Attachment at Rudder Pedal Arm.	(ALI) Replace. (Ref. SB 226-27-027).	Ref. A.D. 92-19-08.
CF-5/8-S; CRS-10-1; MS21440-102.	Bearing, Control Column Pivot.	(ALI) Replace. **NOTE 2**. (Ref. SB 226-27-060).	Ref. A.D. 2015-19-10.
MS21044N4; MS21042L4.	Nut, Control Column Pivot Bearing.	(ALI) Torque Check. **NOTE 2**. (Ref. SB 226-27-060). (Ref. SB 226-27-061).	Ref. A.D. 2015-19-10.
	Control Column Pivot Bearing Configuration.	(ALI) Modify. (Ref. SB 226-27-061).	Ref. A.D. 2015-19-10.
AN4-22; AN4-22A; NAS6604D31; NAS6604D34.	Bolt, Elevator Rod to Walking Beam Attachment (and Associated Hardware).	(ALI) Replace. (Refer to SB 226-27-060 for replacement P/N's).	Ref. A.D. 2015-19-10.
26-71004-003.	Control Column Horizontal Tube.	(ALI) Inspect. (Ref. SB 226-27-078).	Ref. A.D. 2016-15-02.
27-71026-001. **NOTE 3**.	Chain, Aileron Control Columns.	(ALI) Replace. (Ref. SB 226-27-074).	Ref. A.D. 2014-09-02.
26-71010-001; 26-71010-003.	Sprocket, Aileron Control Columns.	(ALI) Replace. (Ref. SB 226-27-074).	Ref. A.D. 2014-09-02.
KP16B; MS27642-16.	Bearing, Aileron Control Wheels.	(ALI) Replace. (Ref. SB 226-27-074).	Ref. A.D. 2014-09-02.
HF-4M (Alt. MW-4M-17), HM-4M (Alt. MM-4M-18).	Elevator Control Links.	(ALI) Inspect. (Ref. SB 226-27-080).	Ref. A.D. 2016-10-01.
		(ALI) Lubricate. (Ref. SB 226-27-080).	
ALL.	Flight Control Cables.	(ALI) Replace or Inspect.	Ref. A.D. 87-02-02.
		Tension Check.	Initial 200 and 500 FH, then every 1,000 FH*.

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 27 – FLIGHT CONTROLS (continued)			
	Flight Control Cables (Aircraft with belly mounted camera installation, S/N T265, T267, T(B)348, TC277, AT071, AT072 and AT073 Only).	(ALI) Replace. (Ref. SB 226-27-072 for specific cables and instructions).	Ref. A.D. 2015-13-10.
	Stall Avoidance, Conrac System.	(ALI) Visual Inspection. (Ref. SB 226-27-037, paragraph 2.A.).	Ref. A.D. 85-22-06.
		(ALI) Functional Check Pusher Servo. (Ref. SB 226-27-037, paragraph 2.B.).	
		(ALI) Recalibrate. (Ref. SB 226-27-037, paragraph 2.C.).	
	Stall Avoidance, Rosemount System.	(ALI) Visual Inspection. (Ref. SB 226-27-038, paragraph 2.A.).	Ref. A.D. 85-22-06.
		(ALI) Functional Check Pusher Servo. (Ref. SB 226-27-038, paragraphs 2.B. and C.).	
		(ALI) System Recalibration. (Ref. SB 226-27-038, paragraph 2.D.).	
27-44045-005.	Elevator Down Spring.	(ALI) Visual Inspection of Spring and Attach Hardware.	Ref. A.D. 93-09-05.
		(ALI) Relocate spring IAW SB 226-27-032.	Ref. A.D. 93-09-05.
		(ALI) Check for wear and broken cable wires.	5,000 FH.
		Tension Check (Ref MM 27-30-00, Adjustment - Elevator Down Spring).	2,250 FH.*
27-70066-001.	Rudder Gust Lock Assembly.	(ALI) Replace. (Ref. SB 226-27-066).	5,000 FH.

TIME LIMITS - MAINTENANCE PRACTICES

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 29 - HYDRAULIC POWER			
27-81009-015.	Hydraulic Power Pack.	Overhaul.	15,000 FH.
AN6235-3A.	Hydraulic System Filter.	Replace.	450 FH*.
24600-6.	Gear Selector Valve.	Overhaul.	7,500 FH.
ATA 32 - LANDING GEAR			
235607-1; TRE-8-T.	Rod End, Landing Gear Actuator.	(ALI) Replace. (Ref. SB 226-A32-014).	Ref. A.D. 99-21-05.
01-571-081D.	Rod End, Landing Gear Actuator.	(ALI) Replace. (Ref. SB 226-A32-014).	Ref. A.D. 99-21-05.
OAS5453 Up To and Including -019.	Main Gear Yoke.	(ALI) Inspect (Ultrasonic). (Ref. SB 226-32-065).	Ref. A.D. 96-19-05.
OAS5451 Up To and Including -017.	Nose Gear Yoke.	(ALI) Inspect (Ultrasonic). (Ref. SB 226-32-065).	Ref. A.D. 96-19-05.
AC3255F-8Y14.	Filter Element, Nose Wheel Steering.	Replace.	1,000 FH*.
350506.	Filter Elements, Nose Wheel Steering Servo.	Clean or Replace.	900 FH*.
5453001-1; 5453001-3.	Main Gear Strut.	(ALI) Inspect (Dye Penetrant). (Ref. SB 226-32-033).	800 FH if not cracked; 50 FH if cracked and reworked.
ALL.	Landing Gear Struts.	Leak Check.	450 FH*.
V1-15-1000.	Brake Master Cylinders.	(ALI) Overhaul. (Ref. SB 226-32-069).	Ref.A.D. 2002-08-01.
ATA 34 - NAVIGATION			
ALL.	Transponders.	Test IAW FAR Part 43, Appendix F.	2 Years.
ALL.	Altimeters.	Test IAW FAR Part 43, Appendix E.	2 Years.
	Pitot Static Systems.	Leak Check IAW FAR Part 43, Appendix E.	2 Years.
27-82584-003; MS17983-2; C2300 L41.	Magnetic Compass.	Compass Swing. (Ref. AC 43.13-1B and AMM 34-20-10).	2 Years.

TIME LIMITS - MAINTENANCE PRACTICES

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 35 - OXYGEN			
176000-22; 176124-22; 176127-22; 176177-22; 6350-A4.	Oxygen Bottle, 22 Cu. Ft. (DOT 3AA).	(ALI) Hydrostatic Test.	**NOTE 4**.
		(ALI) Replace.	
ZC373; 176000-49; 176124-49; 176127-49; 176177-49.	Oxygen Bottle, 49 Cu. Ft. (DOT 3HT).	(ALI) Hydrostatic Test.	**NOTE 4**.
		(ALI) Replace.	
176000-64; 176127-64; 176177-64.	Oxygen Bottle, 64 Cu. Ft. (DOT 3HT).	(ALI) Hydrostatic Test.	**NOTE 4**.
		(ALI) Replace.	
ZC415; 176000-76; 176127-76; 176177-76.	Oxygen Bottle, 76 Cu. Ft. (DOT 3HT).	(ALI) Hydrostatic Test.	**NOTE 4**.
		(ALI) Replace.	
ZC428; 176000-115; 176124-115; 176127-115; 176177-115.	Oxygen Bottle, 115 Cu. Ft. (DOT 3HT).	(ALI) Hydrostatic Test.	**NOTE 4**.
		(ALI) Replace.	
21507-02.	Oxygen Bottle, 50.1 Cu. Ft. (Composite, DOT SP 8162-1850).	(ALI) Hydrostatic Test.	**NOTE 4**.
		(ALI) Replace.	
ATA 52 - DOORS			
H2949; H931; 27-24135; 27-24151.	Cargo Door Latch Actuator Assembly (Lower Two Only). **NOTE 5.**	(ALI) Replace. (Ref. SB226-52-008).	3,000 FH (T, AT).
			10,000 FH (TC).
H2949; H931; 27-24135; 27-24151.	Passenger Door Latch Actuator Assembly (Top Latch Only).	(ALI) Replace.	3,000 FH (T, AT).
			10,000 FH (TC).

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 56 - WINDOWS			
ALL.	Acrylic Single Pane Cockpit Side Windows.	(ALI) Inspect for cracks. (Ref. SB 226-56-003).	Ref. A.D. 96-20-08.
		(ALI) Replace.	
ALL.	Acrylic Cabin and Dual Pane Cockpit Side Windows.	(ALI) Inspect for cracks. (Ref. SB 226-56-001, SB 226-56-002 and SB 226-56-003).	Ref. A.D. 96-20-08.
26-21126; 27-19442.	Glass Heated Windshields, LH/RH.	(ALI) Inspect. (Ref. SB 226-56-011).	Ref. A.D. 2011-02-04.
ATA 57 - WINGS			
	Upper Wing Skin at Battery Box Aft Corner.	(ALI) Inspect. (Ref. SB 226-57-018).	Ref. A.D. 94-07-10.
		(ALI) Modify. (Ref. SB 226-57-018).	
ATA 61 - PROPELLERS			
ALL.	Propeller.	Overhaul.	Ref. latest Service Information from Propeller manufacturer.
		Retire.	
		Balance.	
ATA 71 - POWER PLANT			
1337-1.	JATO Rocket.	Replace.	1,000 FH or 12 Months, if installed.
			36 months, if not installed.
LM-827-SA2, -SA7, -SA10 and -SA11.	LORD Engine Isolator.	Inspect. (Ref. SA226-SL-034 and LORD SIL LM827-71-001).	Engine Removal.
ATA 72 - ENGINE			
894040; 895880; 3102550.	-3 Engine.	Gearbox, Hot Section and Overhaul Inspections.	Ref. latest Service Information from Engine manufacturer.
3102940.	-10 Engine.		
ATA 76 - ENGINE CONTROLS			
ALL. (See A.D.)	Propeller Pitch Control Lever.	(ALI) Inspect attachment.	Ref. A.D. 2017-07-01.
	Propeller Pitch Control.	(ALI) Replace or rework. (Ref. SB 226-76-012).	

TIME LIMITS - MAINTENANCE PRACTICES

PART NUMBER	PART NAME	ACTION	INTERVAL
ATA 78 - EXHAUST			
27-62080-047.	Exhaust Duct (Inconel).	Inspect flange radius for cracks.	3,600 FH*.
ATA 82 - WATER INJECTION			
71126-9.	Emergency AWI Pump (4.5 U.S Gallon usable).	Replace.	6,000 FH (T, TC).
	CAWI Distribution System.	Inspect.	9,000 FH or 9 Years*.

TABLE 201 NOTES:

1. Refer to AD 2007-16-03 or Airplane Maintenance Manual Section 27-40-02 for acceptable replacement actuator part numbers.

2. Control Column Pivot Bearing Configuration modification IAW SB 226-27-061 terminates the repetitive replacement of the control column pivot bearing and the repetitive nut torque check.

3. The P/N 27-71026-001 Aileron Control Chain is a section of the aileron control cable assembly (P/N 27-70001-011 for AT and TC models; P/N 27-70002-011 for T model). Replacement of these aileron control cable assemblies (2 per aircraft) satisfies this chain replacement requirement as long as they are replaced at the interval stated for the chain.

4. Oxygen cylinders marked with DOT 3HT require a hydrostatic test every 3 years and are limited to a maximum service life of 24 years after original manufacture or 4,380 pressurization cycles whichever comes first. If a cylinder is recharged on average more than once every other day, an accurate record of the number of rechargings must be maintained.

   Oxygen cylinders marked with DOT 3AA and DOT 3AL require a hydrostatic test every 5 years and are not life limited.

   Composite oxygen cylinders are limited to a maximum service life of 15 years after original manufacture. If hydrostatically tested after June 30, 2006, then they are required to be hydrostatically tested every 5 years. If hydrostatically tested before July 1, 2006, then they are required to be hydrostatically tested every 3 years.

5. Cargo door latch (click-clack) assemblies consist of an actuator and a receptacle. Retirement life applies only to the latch actuator and does not apply to the steel latch receptacle after compliance with SB 226-52-008. Note that AD 95-18-05 requires replacement of the cargo door lower two aluminum latch receptacles with steel receptacles IAW with SB 226-52-008.

UNSCHEDULED MAINTENANCE CHECKS – DESCRIPTION AND OPERATION

1. General

A. The SA226 series aircraft are designed to withstand more than normal takeoff, flight, and landing loads.

(1) A number of safety factors have been incorporated to prevent structural failure under such abnormal conditions as hard landings and flight through severe turbulence.

(2) Visible exterior damage, unusual noises, changes in controllability, etc., may indicate a need for a special inspection following a particularly hard landing or flight through extremely turbulent air.

B. Any overweight landing requires the same structural inspection as a hard landing.

C. Those aircraft equipped with a Flight Data Recorder measuring N_2 (vertical inertia) may use these readings to help determine if the aircraft experienced an overweight/hard landing or flight into turbulent air, if visual inspections are less obvious. All readings above 3.0 g for landings and turbulent air require further inspection as outlined in section 05-50-10.

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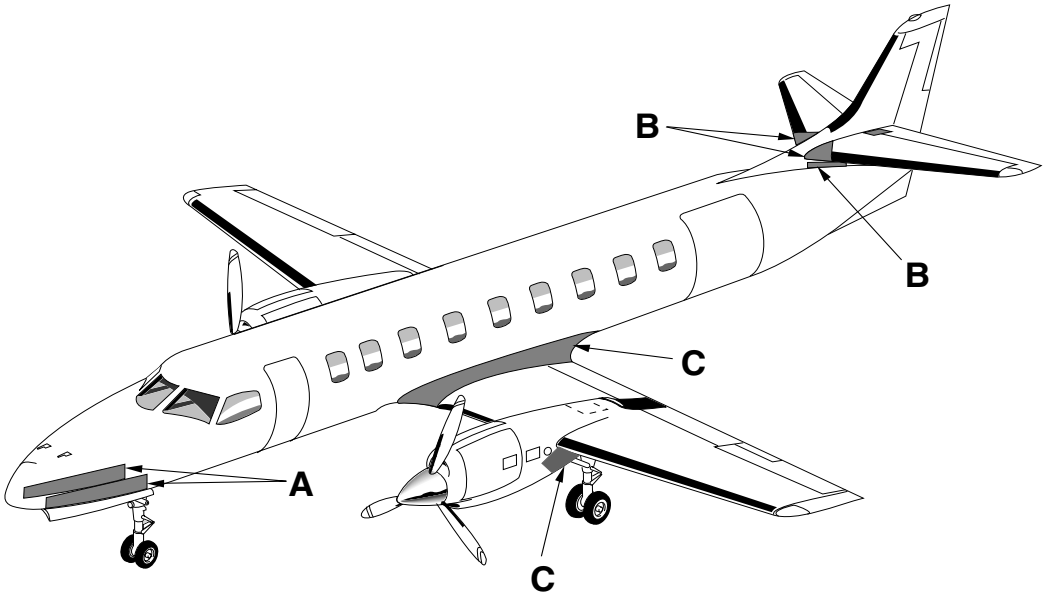
1. Structural Inspections

A. Under various conditions and circumstances, different structural inspection requirements may apply. The following table lists the areas of the airplane and a brief description of the inspection to be conducted after a hard landing or flight through turbulent air.



A INSPECT AFTER HARD, OVERLOAD, OR OVERWEIGHT LANDING

B INSPECT AFTER TURBULENT AIR

C INSPECT AFTER HARD, OVERLOAD, OVERWEIGHT LANDING, OR TURBULENT AIR

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Inspection Areas
Figure 201

STRUCTURAL INSPECTION GUIDE – MAINTENANCE PRACTICE

AREA	REQUIREMENT	HARD LANDING	TURBULENT AIR
NACELLE	Inspect wing-to-nacelle attachment for cracks and loose or sheared attachment.	X	X
	Inspect engine mount attachment fittings for condition.	X	
	Inspect engine truss assembly for bent or cracked tubes, cracked gussets, and welds.	X	
	Inspect nacelle keelson webs for evidence of permanent buckling.	X	
	Inspect nacelle upper and lower longerons for evidence of buckling.	X	
	Inspect aft nacelle bulkhead.	X	
LANDING GEAR	Inspect MLG drag strut and support fittings for cracks and sheared or loose attachment.	X	
	Inspect actuators, bellcranks and rollers for cracks and damage.	X	
	Inspect MLG axle for permanent set.	X	
	Perform dye penetrant inspection of wheels for cracks in hub and rim flanges.	X	
	Perform landing gear retraction with aircraft on jacks, and check for freedom of motion and clearance.	X	
WING	Inspect top and bottom of wing for cracks, permanent wrinkles and/or loose rivets.	X	X
	Inspect wing spar webs for evidence of buckling if external wing skin buckling is present after encountering a hard landing or flight through turbulent air.	X	X
	Inspect leading edge of wing for evidence of skin buckling and/or loose and damaged structure.	X	X
	Inspect wing-to-nacelle aft attachment for cracked or sheared attachments.	X	X
	Remove fairing and inspect area of wing enclosed by fairing.	X	X
	Check for clearance throughout flap travel.	X	X

STRUCTURAL INSPECTION GUIDE – MAINTENANCE PRACTICES

AREA	REQUIREMENT	HARD LANDING	TURBULENT AIR
FUSELAGE	Inspect top side and underside of fuselage skin for evidence of buckling.	X	X
	Inspect skin splices for loose or sheared attachments.	X	X
	Remove wing fairing and inspect fuselage skin and frames, in area of wing spars, for permanent set or cracks. With wing fairing removed, perform dye penetrant inspection of visible upper, outboard fillet radii on both halves of wing-to-fuselage fitting at main and aft spars, checking for cracks.	X	X
	Inspect all doors and door alignment.	X	X
	Pressure check cabin to 7.0 psi.	X	X
TAIL	Inspect vertical stabilizer-fuselage attachment by removing fillets. Inspect fillets and tubes for cracks, buckling, and loose attachment.		X
	Inspect vertical stabilizer spar fuselage frames.		X
	Inspect horizontal stabilizer trim actuator.		X
	Inspect rudder surface and hinges for distortion, cracks, and sheared or loose attachment.		X
	Check for clearance over full rudder travel.		X
CONTROLS	Check flight control system from cockpit; operate all controls and trim tabs through full travel, checking for freedom of movement.	X	X

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MAINTENANCE MANUAL

CHAPTER 06 – DIMENSIONS AND AREAS

LIST OF EFFECTIVE PAGES

SUBJECT	SECTION	PAGE	DATE	SUBJECT	SECTION	PAGE	DATE
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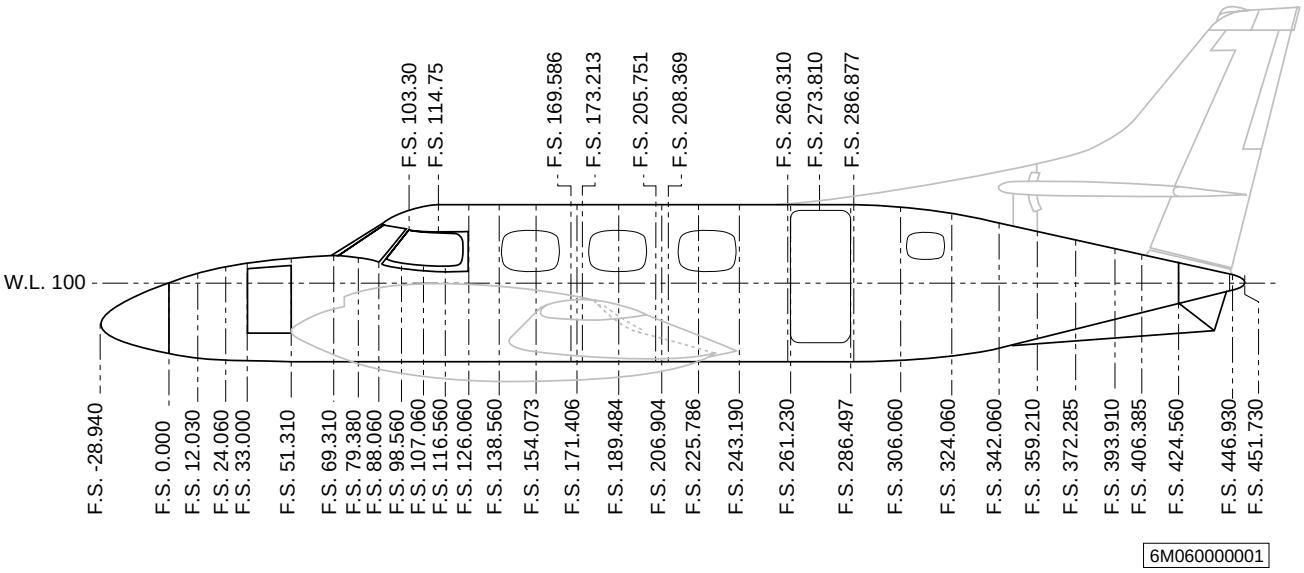
06-CONTENTS	-	1	AUG 01/98	06-00-00	DO	5	AUG 01/98
06-00-00	DO	1	AUG 01/98	06-00-00	DO	6	AUG 01/98
06-00-00	DO	2	AUG 01/98	06-00-00	DO	7	AUG 01/98
06-00-00	DO	3	AUG 01/98	06-00-00	DO	8	OCT 01/06

MAINTENANCE MANUAL

CHAPTER 06 – DIMENSIONS AND AREAS

TABLE OF CONTENTS

SUBJECT	CHAPTER SECTION SUBJECT	PAGE	MODEL	EFFECTIVITY
DIMENSIONS AND AREAS DESCRIPTION AND OPERATION	06-00-00	1	T AT TC	201-999 001-999 201-999



Station Locations – SA226 – T and T(B)
Figure 1

1. General

A. This chapter defines and identifies station locations and lines used in locating reference points of major components of the SA226 series aircraft.

B. Definitions

(1) Stations are planes crossing at points along the longitudinal or lateral axis established in reference to datums or stations 0.00. Lines are planes defined as either horizontal-parallel to the lateral stations, or vertical-parallel to the longitudinal stations.

C. Fuselage Stations (see Figures 1 and 2)

(1) Fuselage stations (F.S.) are measured along the longitudinal axis of the fuselage. Those forward of the datum (F.S. 0.00) are negative; those aft of the datum are positive. The radome bulkhead is located at F.S. 0.00. F.S. minus 28.940 is the forward most point of the fuselage. When the optional JATO is installed, the most aft position would be extended approximately 5.0 inches (12.7 cm).

D. Nacelle Stations (see Figure 3)

(1) Nacelle stations (N.S.) are measured along the longitudinal axis of the nacelle. All nacelle stations are located aft of the datum (N.S. 0.00), located 24.75 inches forward of the spinner dome tip.

E. Wing Stations (see Figure 3)

(1) Wing Stations (W.S.) are measured along the lateral axis of the wing. They are measured out to the left or right wing tip from the centerline of the fuselage, W.S. 0.00.

F. Water Lines (see Figure 4)

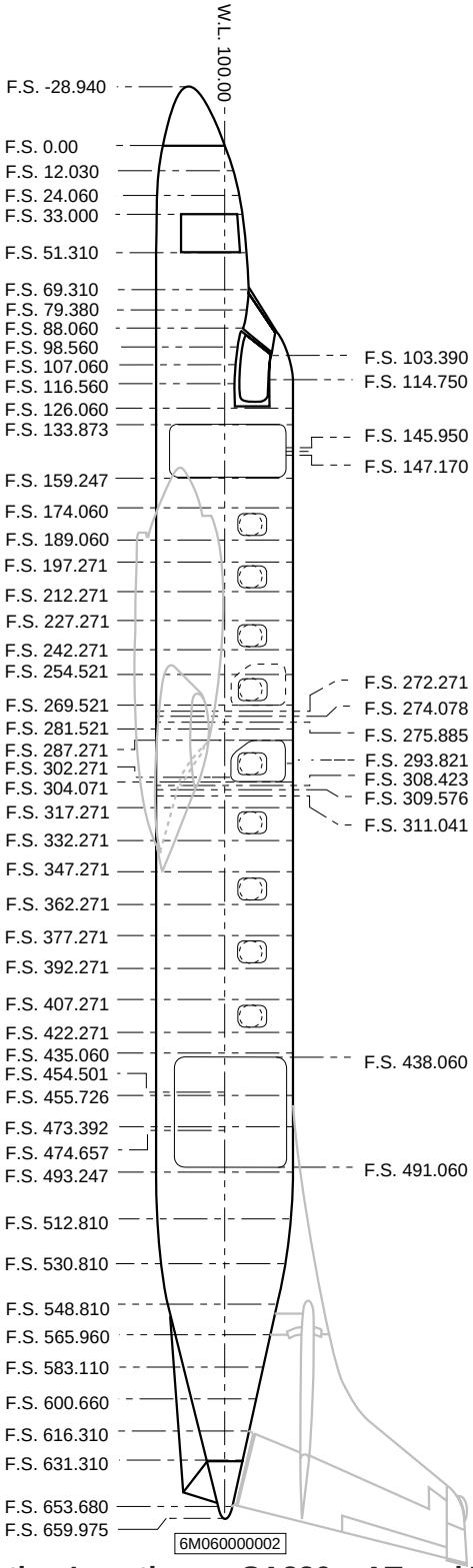
(1) Water lines (W.L.) measure along the lateral axis of the fuselage and the vertical stabilizer and rudder. The fuselage centerline is at W.L. 100.00. The fuselage is 66 inches in diameter (frame to frame). Starting from W.L. 100.00, each inch upward is measured until top of fuselage is reached at W.L. 133.00 (100 plus 33 inches). Measuring downward from W.L. 100.00, the lowest point of the fuselage measures W.L. 67.00 (100 minus 33 inches).

G. Butt Lines (see Figure 4)

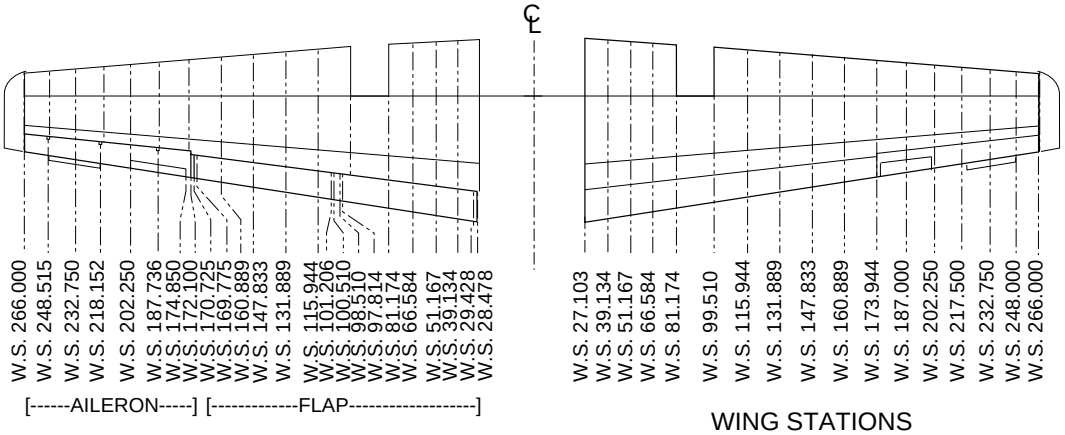
(1) Butt lines (B.L.) are defined as longitudinal planes parallel to the wing stations. Butt lines and wing stations use the same reference datum (B.L. 0.00) and therefore the values of both coincide; however, butt lines are used for fuselage locations, and wing stations are used for wing locations. The centerline of the fuselage is B.L. 0.00. Butt lines are measured left and right from the centerline with each butt line being one inch. The center of each nacelle is B.L. 90.00 (90 inches from the fuselage centerline).

H. General Dimensions

(1) A three-angle view and principle dimensions of each aircraft are shown in Figures 5 and 6.

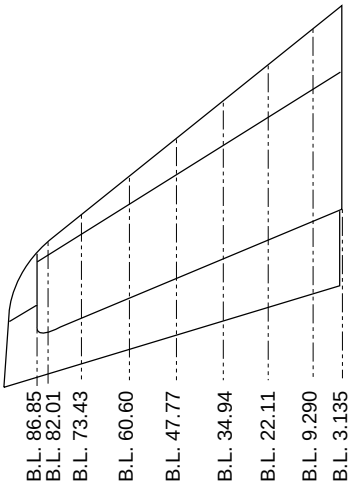


Station Locations – SA226 – AT and TC
Figure 2

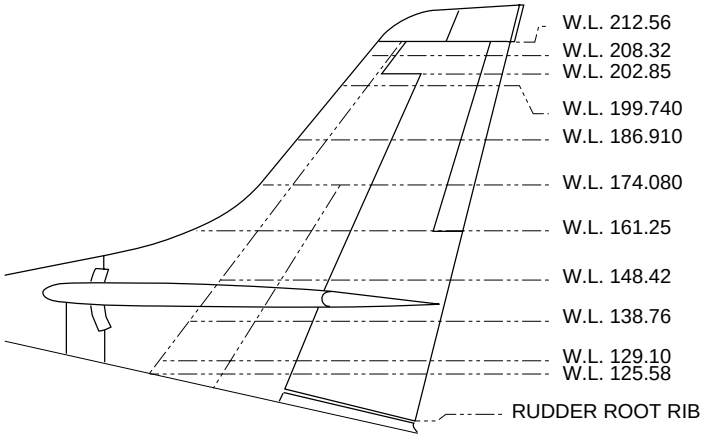


AILERON AND FLAP STATIONS

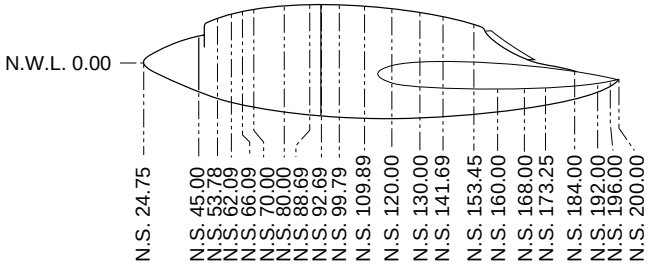
WING STATIONS



HORIZONTAL STABILIZER
AND ELEVATOR



VERTICAL STABILIZER
AND RUDDER



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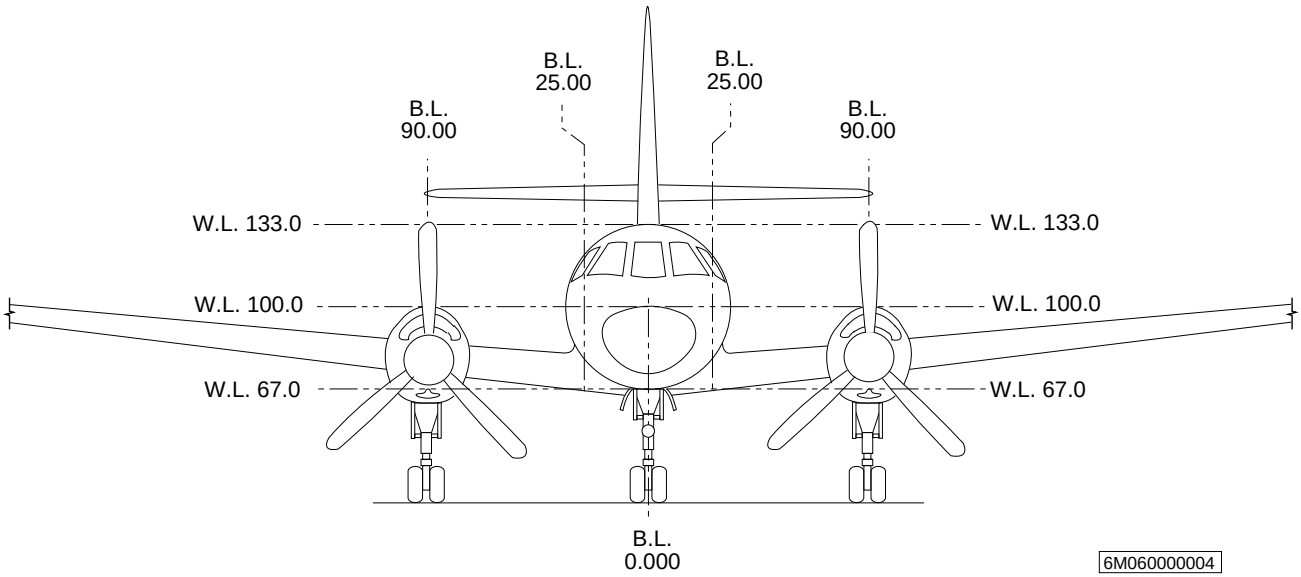
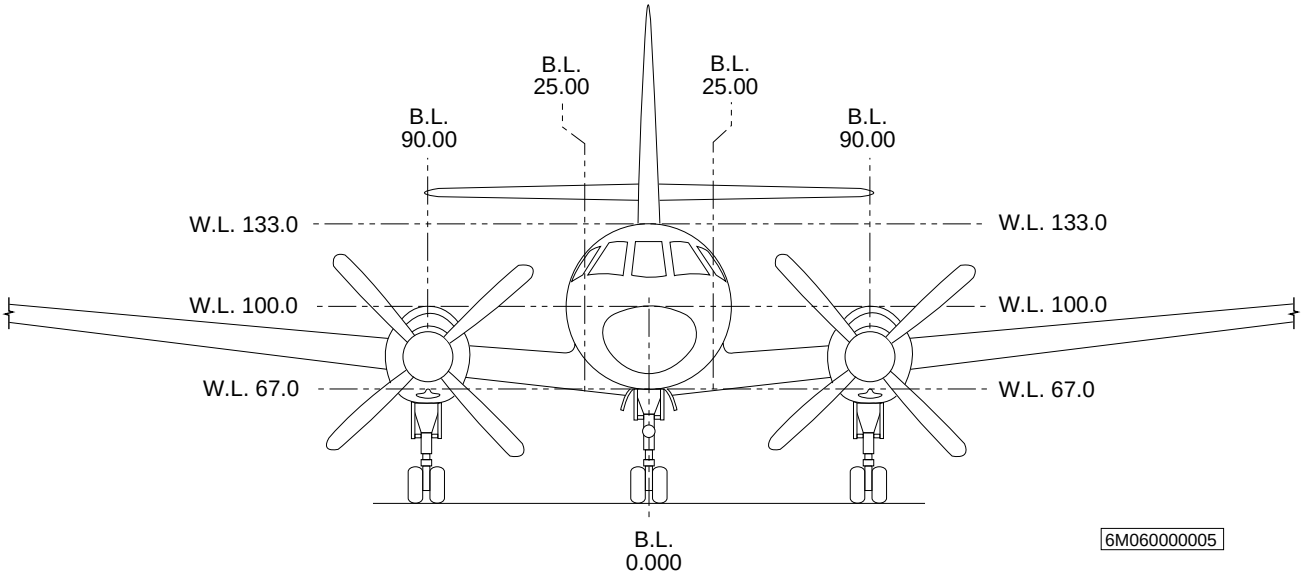
NACELLE STATIONS

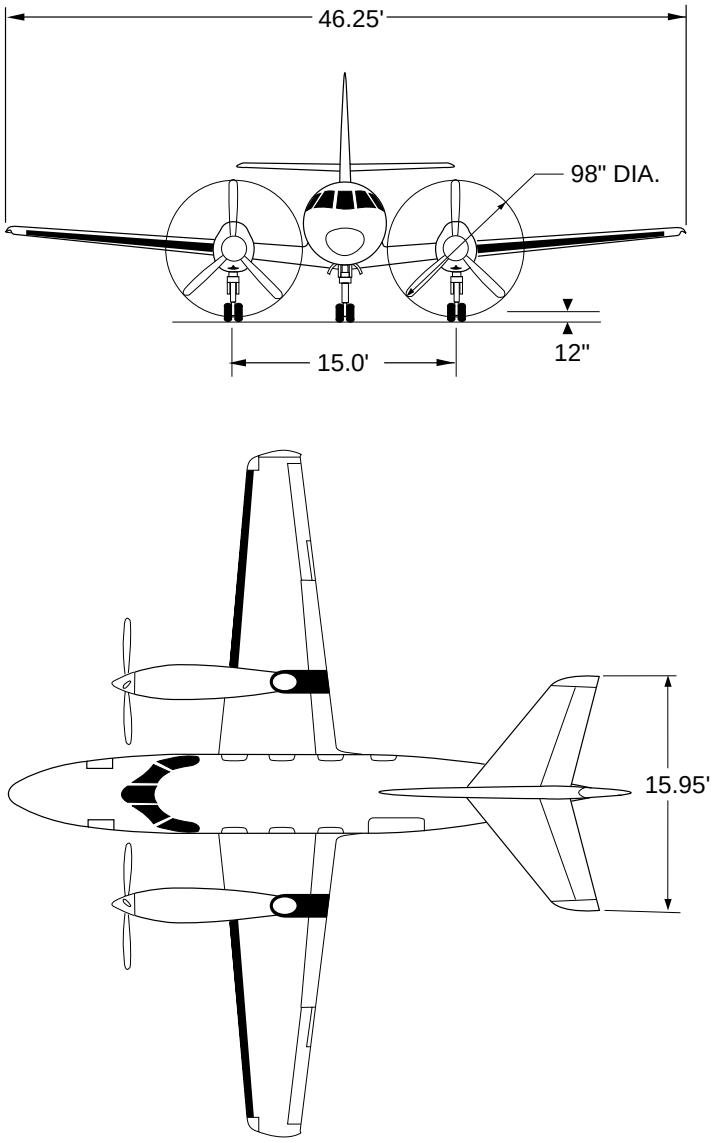
Other Station Locations – SA226 – T, AT, and TC
Figure 3



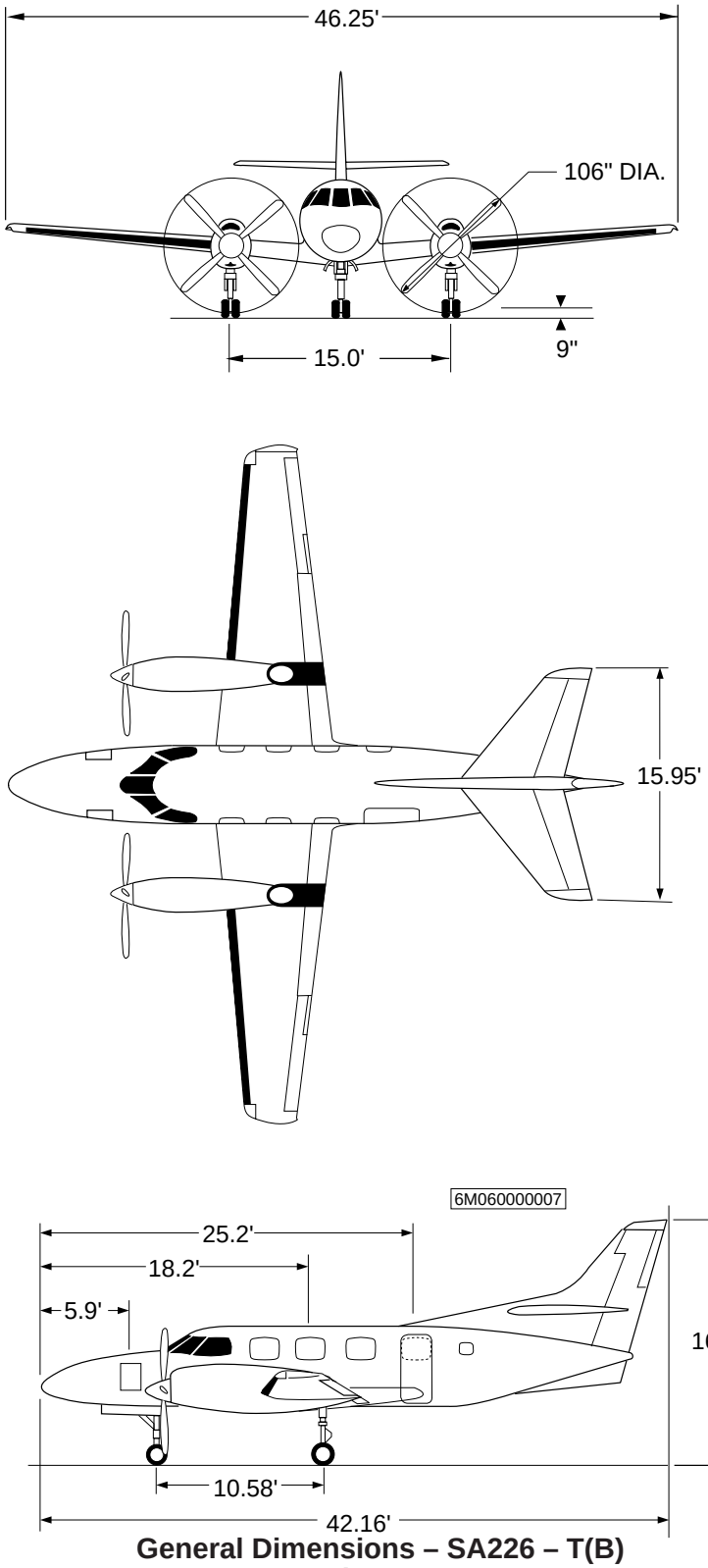
Water Lines and Butt Lines – SA226 – T, AT, and TC
Figure 4



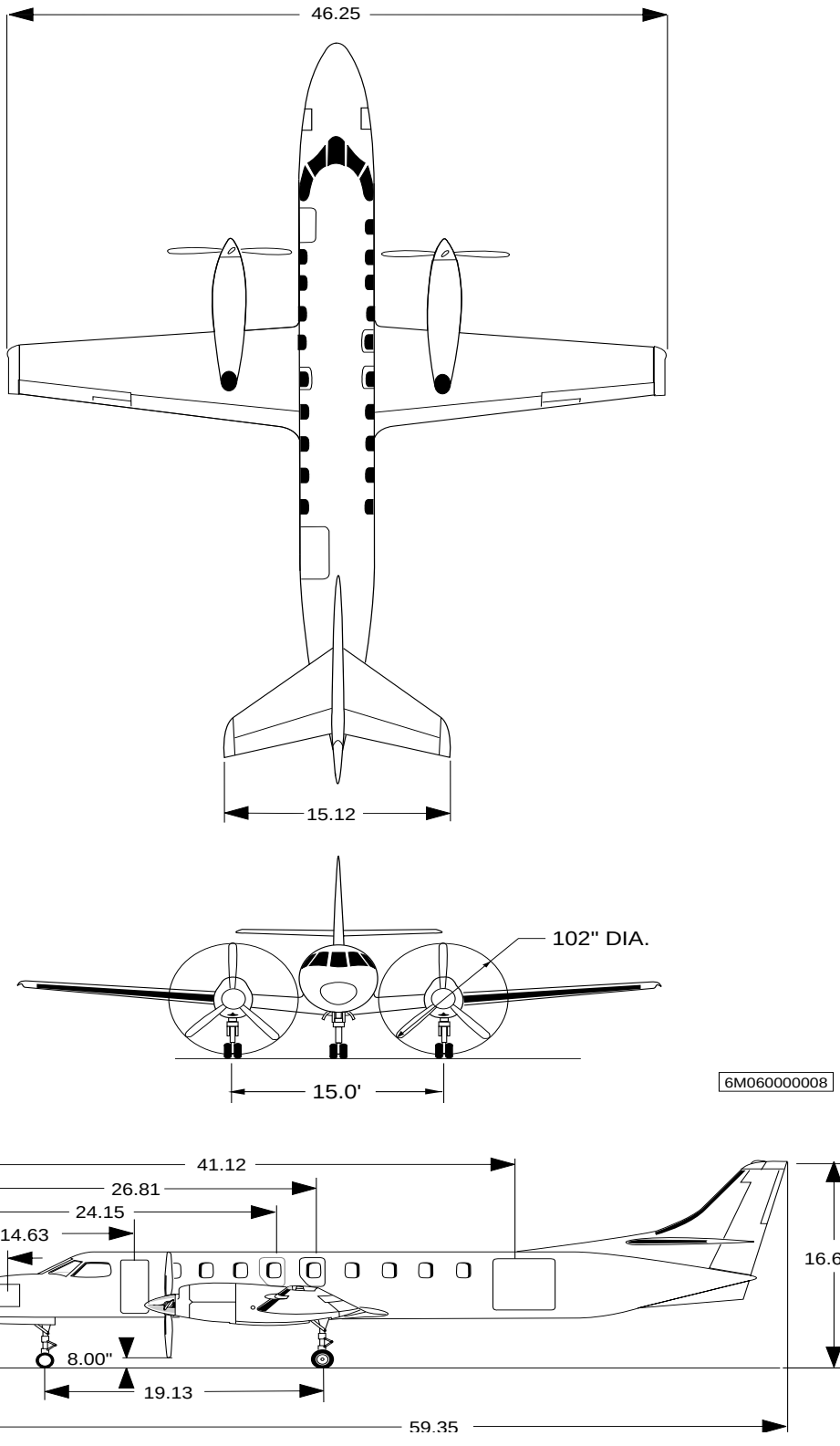
Water Lines and Butt Lines – SA226 – T(B)
Figure 5



General Dimensions – SA226 – T
Figure 6



General Dimensions – SA226 – T(B)
Figure 7



General Dimensions – SA226 – AT and TC
Figure 8

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CHAPTER 07 – LIFTING AND SHORING

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CHAPTER 07 – JACKING AND SHORING

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AIRCRAFT SHORING DESCRIPTION AND OPERATION	07-20-00	1	T AT TC	201-999 001-999 201-999
AIRCRAFT SHORING MAINTENANCE PRACTICES	07-20-00	201	T AT TC	201-999 001-999 201-999

MAINTENANCE MANUAL

CHAPTER 07 – JACKING AND SHORING

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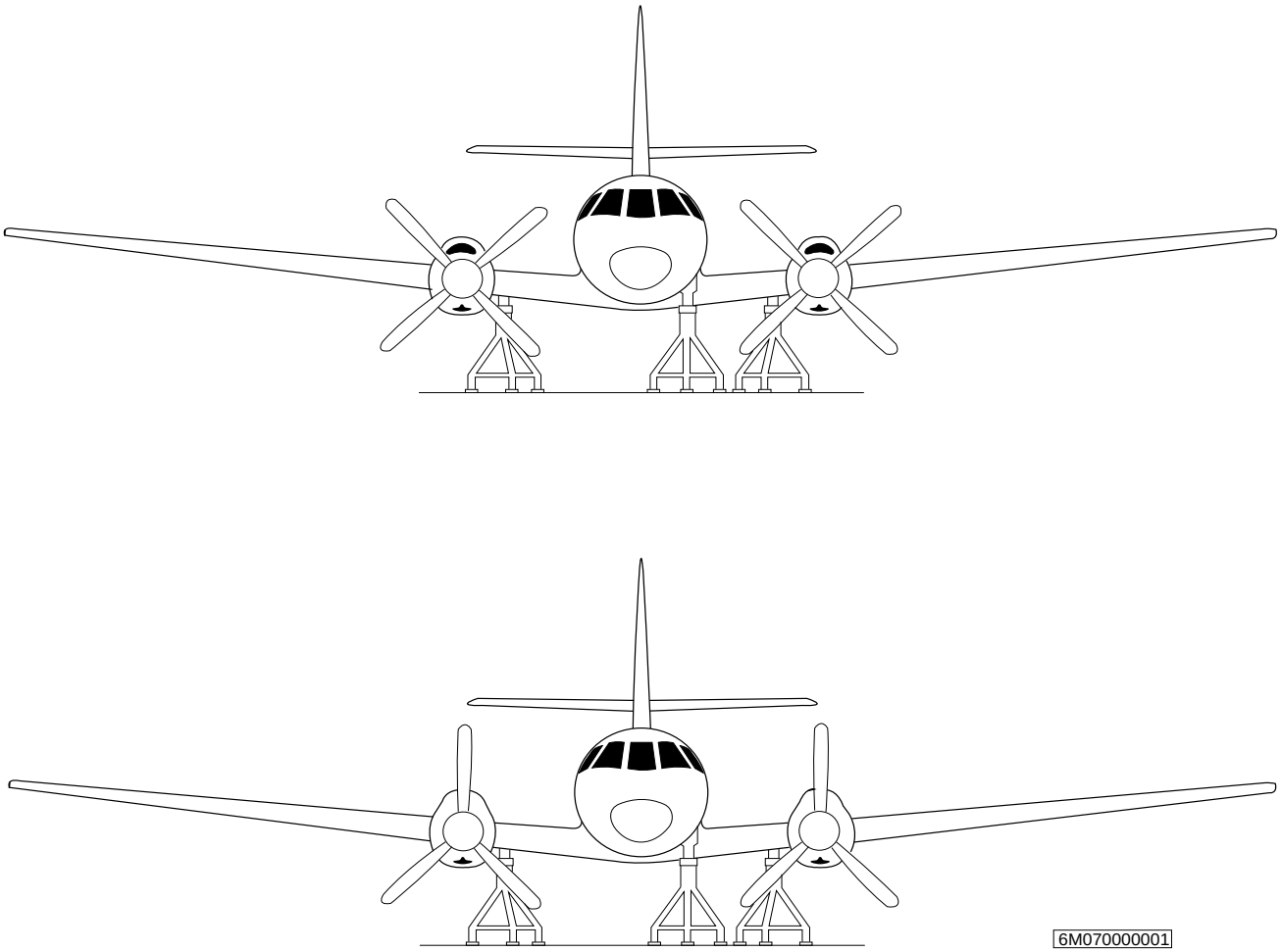
AIRCRAFT JACKING – DESCRIPTION AND OPERATION

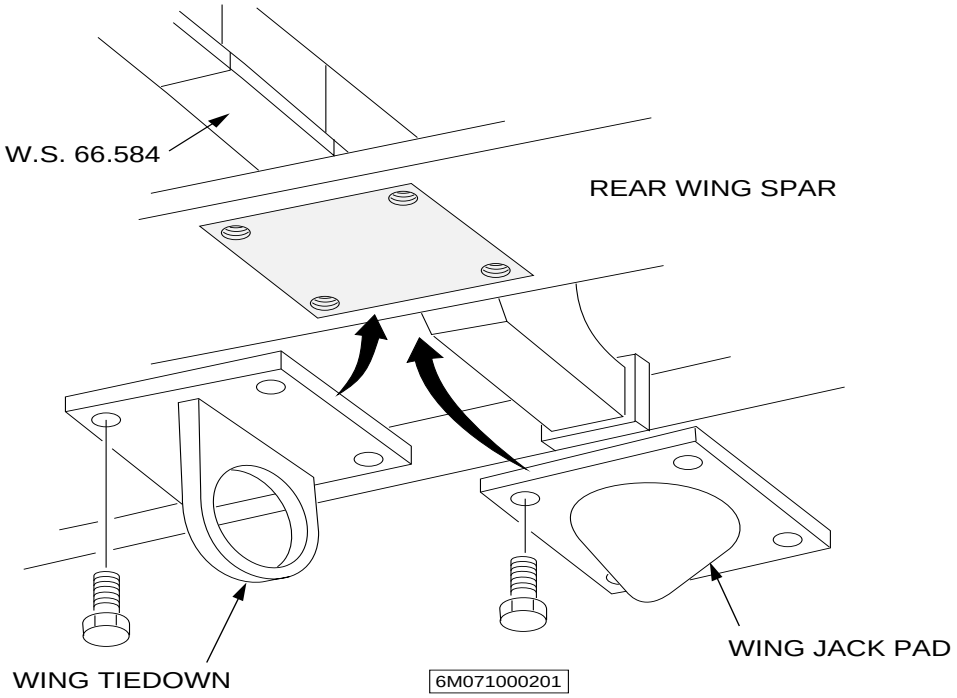
1. General

A. Standard jacking methods are employed for the aircraft and/or major components. Jack pads are located in each wing center section at the rear spar inboard of each nacelle, and in the nose fuselage aft of and below the left forward baggage compartment door. Each wing jack point consists of a jack pad and four bolts. The nose fuselage jack point consists of a permanent conical projection protruding through the fuselage skin.

B. The wing jack pads are constructed of heavy gauge aluminum: 2 1/2 x 2 1/2 x 1/4 inches. The jack pads (Figure 201) are attached to the rear spar by four 3/16 bolts. Wing tie-down rings, furnished as loose equipment (Figure 201), may be installed in place of wing jack pads.



Jacked Aircraft
Figure 1



Jack Pad and Wing Tiedown
Figure 201

GEAR	CONDITION	MAXIMUM HEIGHT
Nose	Tire and Strut Normal Inflation	57 Inches
	Tires Flat	54 inches
	Shock Strut Deflated	53 Inches
	Shock Strut Deflated And Tires Flat	50 Inches
Main	Tire and Strut Normal Inflation	50 Inches
	Tires Flat	45 Inches
	Shock Strut Deflated	47 Inches
	Tires and Shock Strut Deflated	42 Inches
	One Tire Flat	48 Inches

Maximum Retracted Jack Heights
Table 201

AIRCRAFT JACKING – MAINTENANCE PRACTICES

1. Jacking

A. Jacking Precautions

NOTE: Refer to Table 201 for maximum retracted jack heights.

(1) Jacking the aircraft outside of hangar is not recommended.

(2) The aircraft should be on hard, level surface when jacking.

(3) Loose cargo, tools, and equipment should be removed from the aircraft before jacking.

(4) All three jack points must be used and jacks raised smoothly and evenly during jacking operations.

(5) When one or both engines are removed, provide adequate ballast and tail support to maintain aircraft balance.

B. Nose Jacking

(1) Jacking the nose requires a jack with a minimum capacity of 3,000 pounds.

(2) In order to jack at three points and maintain a level attitude, the minimum recommended extended jack height is 66 inches.

C. Wing Jacking

(1) Jacking requires jacks with a minimum capacity of 6,000 pounds each. The minimum recommended extended jack height in jacking a wing level attitude is 64 inches.

D. Jacking For Gear And Wheel Maintenance

(1) The aircraft is jacked at three points for all wheel, tire, brake, and landing gear maintenance that requires transferring the aircraft weight from any of the three landing gears, (see Figure 1).

(2) Raise jacks simultaneously to maintain aircraft in a level attitude.

CAUTION: SUPPORT THE AFT FUSELAGE ANYTIME PERSONNEL OR CARGO ARE IN OR ON THE AIRPLANE AFT OF F.S. 350 (TC OR AT) OR WHEN THE NOSE LANDING GEAR IS REMOVED TO PREVENT TAIL STRIKE.

(3) M7 Aerospace recommends the use of a support under the aft fuselage.

(4) As an alternative, a tire may be changed by running the adjacent wheel up a ramp of suitable material, 6 inches high, 10 inches wide, and 24 inches long. This provides adequate clearance from the ground for a wheel or tire to be changed.

NOTE: Do not attempt to use a ramp if both tires on the same gear are flat.

(5) If performing gear functional checks (gear swings), the aircraft shall be jacked to ensure there is adequate clearance between the gear and ground.

E. Down Jacking

(1) Lower the aircraft jacks simultaneously, keeping the aircraft level, until the landing gear are firmly on the ground and have taken the weight of the aircraft. Continue lowering the jacks until they are free of the aircraft and remove.

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MAINTENANCE MANUAL

AIRCRAFT SHORING – DESCRIPTION AND OPERATION

1. General

A. Standard shoring methods are employed for the aircraft and/or major components. Establish the shoring requirements before starting. Ensure that the ground is firm and level. Avoid shoring out of doors. See Figure 1 for shoring example.

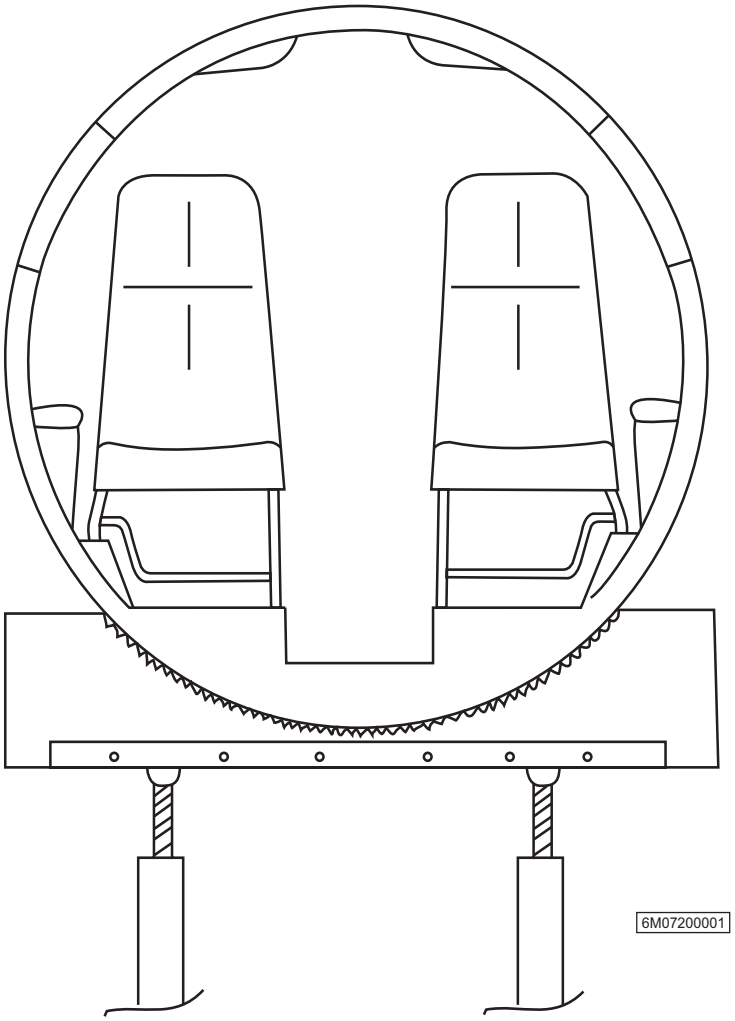
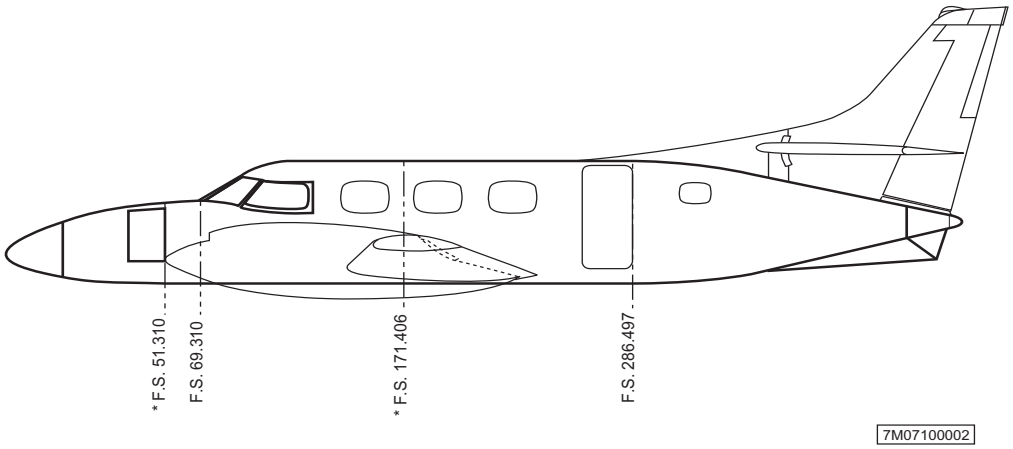
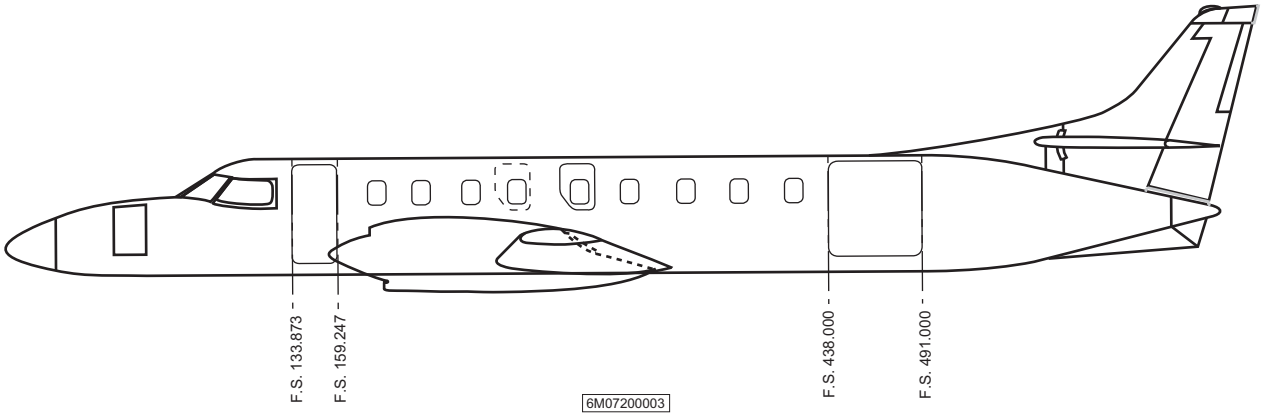


Figure 1
Aircraft Shoring
(View Looking Aft)



* Alternate Locations if F.S. 69.310 is not available

Aircraft Shoring Locations
SA226-T/-T(B)
Figure 2



Aircraft Shoring Locations
SA226-AT/-TC
Figure 3

AIRCRAFT SHORING – MAINTENANCE PRACTICES

1. Shoring

A. Shoring Precautions

(1) Jack the aircraft to the required height. See chapter 07-10-00.

(1) Shoring the aircraft outside of hangar is not recommended.

(2) The aircraft should be on hard, level surface when shoring.

(3) Loose cargo, tools, and equipment should be removed from the aircraft before shoring.

B. Recommended Stabilization and Shoring Locations

(1) **SA226-T/-T(B) (See Figure 2)**

(a) Jack aircraft in accordance with chapter 07-10-00.

(b) Locate cradle at F.S. 69 fuselage lower belt frame.

NOTE: If strucutural maintenance is being performed at F.S. 69, shoring locations are F.S. 51 and F.S 171 in addition to B.(1)(d) below.

(c) Locate cradle at F.S. 286 fuselage lower belt frame.

(d) Locate wing cradles at LH and RH wing lower surface under W.S. 266.

(e) Down jack aircraft into cradles slowly ensuring aircraft is level in accordance with chapter 08-10-00.

(2) **SA226-AT/-TC (See Figure 3)**

(a) Jack aircraft in accordance with chapter 07-10-00.

(b) Locate cradles at F.S. 133 and/or F.S. 159 fuselage lower belt frame.

(c) Locate cradles at F.S. 438 and/or F.S. 491 fuselage lower belt frame.

(d) Locate wing cradles at LH and RH wing lower surface under W.S. 266.

(e) Down jack aircraft into cradles slowly ensuring aircraft is level in accordance with chapter 08-10-00.

C. Removing Aircraft from Shoring

(1) Jack the aircraft to the required height. See chapter 07-10-00.

(a) Remove wing cradles at LH and RH wing lower surface under W.S. 266.

(b) Remove cradle at F.S. 286 fuselage lower belt frame.

(c) Remove cradle at F.S. 69 fuselage lower belt frame.

(2) Remove jacks from the aircraft per chapter 07-10-00.

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CHAPTER 08 – LEVELING AND WEIGHING

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LEVELING AND WEIGHING – DESCRIPTION AND OPERATION

1. General

A. Leveling and weighing of aircraft is required to perform certain maintenance functions or to re-establish the basic weight of the aircraft after a modification or alteration. The formula for calculating weight and balance due to weight changes is included in this chapter.

B. Definitions

(1) Arm – The horizontal distance from the reference datum to the center of gravity (C.G.) of an item.

(2) Basic Empty Weight – Standard empty weight plus optional equipment.

(3) Center of Gravity Arm – The distance along the longitudinal axis from the aircraft center of gravity location to the reference datum.

(4) Center of Gravity (C.G.) – The point at which an aircraft would balance if suspended. Its distance from the reference datum is found by dividing the total moment by the total weight of the aircraft.

(5) Maximum Ramp Weight – Maximum weight approved for ground maneuver; includes weight of start, taxi, and run up fuel.

(6) Maximum Takeoff Weight – Maximum weight approved for the start of the takeoff run.

(7) Maximum Zero Fuel Weight – Maximum weight exclusive of usable fuel.

(8) Moment – The product of the weight of an item multiplied by its arm; moment divided by a constant is used to simplify balance calculations by reducing the number of digits.

(9) Payload – Weight of occupants, cargo, and baggage.

(10) Standard Empty Weight – Weight of a standard aircraft including unusable fuel, full operating fluids, and full oil.

(11) Tare – The weight of the chocks, blocks, stands, etc., used when weighing an aircraft; included in the scale reading. Tare is subtracted from the scale reading to obtain the actual (net) aircraft weight.

WEIGHING AND BALANCING – MAINTENANCE PRACTICES

1. Leveling Aircraft – Weighing and Balancing

A. Leveling – Laterally

(1) Jack aircraft.

(2) Locate leveling holes in nose baggage compartment. Holes are drilled in door frame, Figure 201, item A.

(3) Insert cleco or #40 drill bit into hole, one each door frame.

(4) Place straight edge onto clecos, joining leveling points.

(5) Place level onto straight edge.

(6) Level aircraft by adjusting height of the wing jacks.

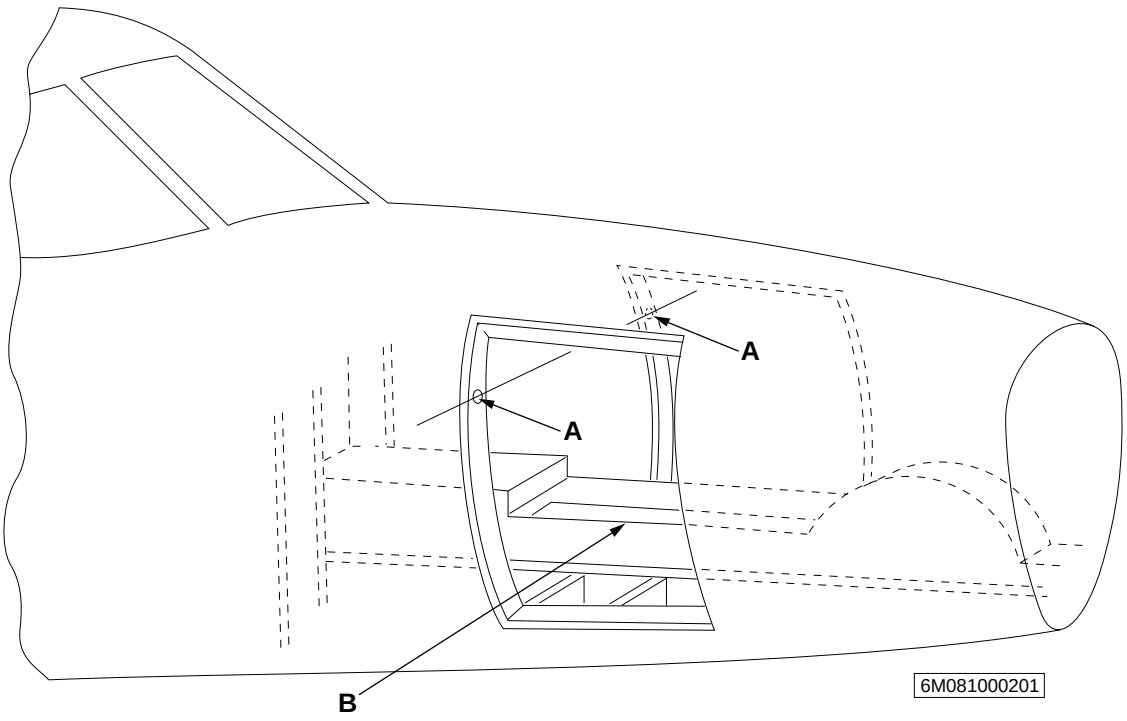
B. Leveling – Longitudinally

(1) Method 1:

(a) Jack aircraft.

(b) Standing on left wing facing fuselage, place level on centerline, top of fuselage. Level should be at least 24 inches long.

(c) Level aircraft by adjusting height of nose jack.



Leveling Points
Figure 201

(2) Method 2 (Figure 201, item B):

(a) Jack aircraft.

(b) Inside nose baggage compartment, gain access to top right side of keelson by removing equipment and carpeting as necessary.

(c) Make a platform to accommodate a level by stacking two 0.25 inch washers over rivets at each end of level. Level should be at least 18 inches long.

(d) Level aircraft by adjusting height of the nose jack.

2. Weighing Aircraft – Weighing and Balancing

A. Weighing Methods

(1) Two methods of weighing the aircraft are recommended; the platform method and jackstand method. After weighing is completed, enter corrected weight, arm, and moment data on Weighing Forms provided in the Aircraft Flight Manual.

B. Platform Method

(1) Place aircraft wheels onto three (3) platform scales.

(2) Align nose gear in normal, forward rolling, centered position.

(3) Release parking brake to prevent binding of scales.

NOTE: The aircraft must be longitudinally level.

(4) Combine scale readings for nose, left and right gear to determine total aircraft weight.

(5) Complete forms provided in Tables 201 or 202 to compute weight, moment, and center of gravity.

C. Jackstand Method

NOTE: This method requires the use of electronic load cells and support jacks.

(1) With the load cells in place on the jacks and zeroed, raise the aircraft until gear is clear of the floor.

(2) Verify gear is down and locked during weighing.

(3) Level aircraft laterally and longitudinally.

(4) Complete forms provided in Tables 201 or 202 to compute weight, moment, and center of gravity.

NOTE: Due to the off-center location of the nose jack pad and locations of wing jack pads, there will be considerable differences in individual scale readings when comparing weights obtained using the jackstand method with weights obtained using the platform method.

JACKSTAND METHOD – SA226 – T AND T(B)

Jack	Scale Reading (Pounds)	– Tare (Pounds)	= Net Weight (Pounds)	X Arm (Inches)	= Moment (In Lbs)
Nose				68.35	
Left				203.81	
Right				203.81	
TOTALS					

PLATFORM METHOD – SA226 – T AND T(B)

	Scale Reading (Pounds)	– Tare (Pounds)	= Net Weight (Pounds)	X Arm (Inches)	= Moment (In Lbs)
Nose				64.10	
Left				191.02	
Right				191.02	
TOTALS					

NOTE: Center of Gravity Arm = $\frac{\text{TOTAL MOMENT}}{\text{TOTAL WEIGHT}}$

Weight, Arm, and Moment Table
SA226 – T and T(B)
Table 201

JACKSTAND METHOD – SA226 – AT, TC

Jack	Scale Reading (Pounds)	– Tare (Pounds)	= Net Weight (Pounds)	X Arm (Inches)	= Moment (In Lbs)
Nose				68.35	
Left				306.50	
Right				306.50	
TOTALS					

PLATFORM METHOD – SA226 – AT, TC

	Scale Reading (Pounds)	– Tare (Pounds)	= Net Weight (Pounds)	X Arm (Inches)	= Moment (In Lbs)
Nose				64.10	
Left				293.69	
Right				293.69	
TOTALS					

NOTE: Center of Gravity Arm = $\frac{\text{TOTAL MOMENT}}{\text{TOTAL WEIGHT}}$

Weight, Arm, and Moment Table
SA226 – AT, TC
Table 202

D. Weight and Balance Calculation

(1) Table 3 is a formula for calculating weight and balance to determine aircraft center of gravity location.

AIRCRAFT WEIGHT (W) Determined by crew since it will vary dependent upon aircraft payload, fuel load, etc. Different for each aircraft.	X ARM (A) = Aircraft center of gravity (C.G.). Determined by crew since it is also dependent upon aircraft weight, payload, etc.	MOMENT (M1) (Inch-Pounds) Obtained by multiplying aircraft weight (W) by arm (A)
WEIGHT CHANGE (WC)(Pounds) Plus or minus rounded to nearest pound. Less than one pound is negligible.	X ARM (A) = Obtained by multiplying weight(WC) by arm (A).	MOMENT (M) (Inch-Pounds) Location of weight change (fuselage station number obtained from station diagram).
NEW WEIGHT (NW) (Pounds) Crew adds (W) and (WC) to obtain (NW) for weight increase or subtracts (WC) from (W) for weight reduction.		NEW MOMENT (NM) (Inch-Pounds) Crew adds (M) and (M1) to obtain (NM) for weight increase or subtracts (M1) from (M) for weight reduction.
	NEW C. G. (NCG) LOCATION Crew divided (NM) by (NW) to obtain (NCG).	

Weight and Balance Calculation Formula Table 203

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CHAPTER 09 – TOWING AND TAXIING

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TAXIING DESCRIPTION AND OPERATION	09-20-00	1	T AT TC	201-999 001-999 201-999
TAXIING MAINTENANCE PRACTICES	09-20-00	201	T AT TC	201-999 001-999 201-999

TOWING AND TAXIING – DESCRIPTION AND OPERATION

1. General

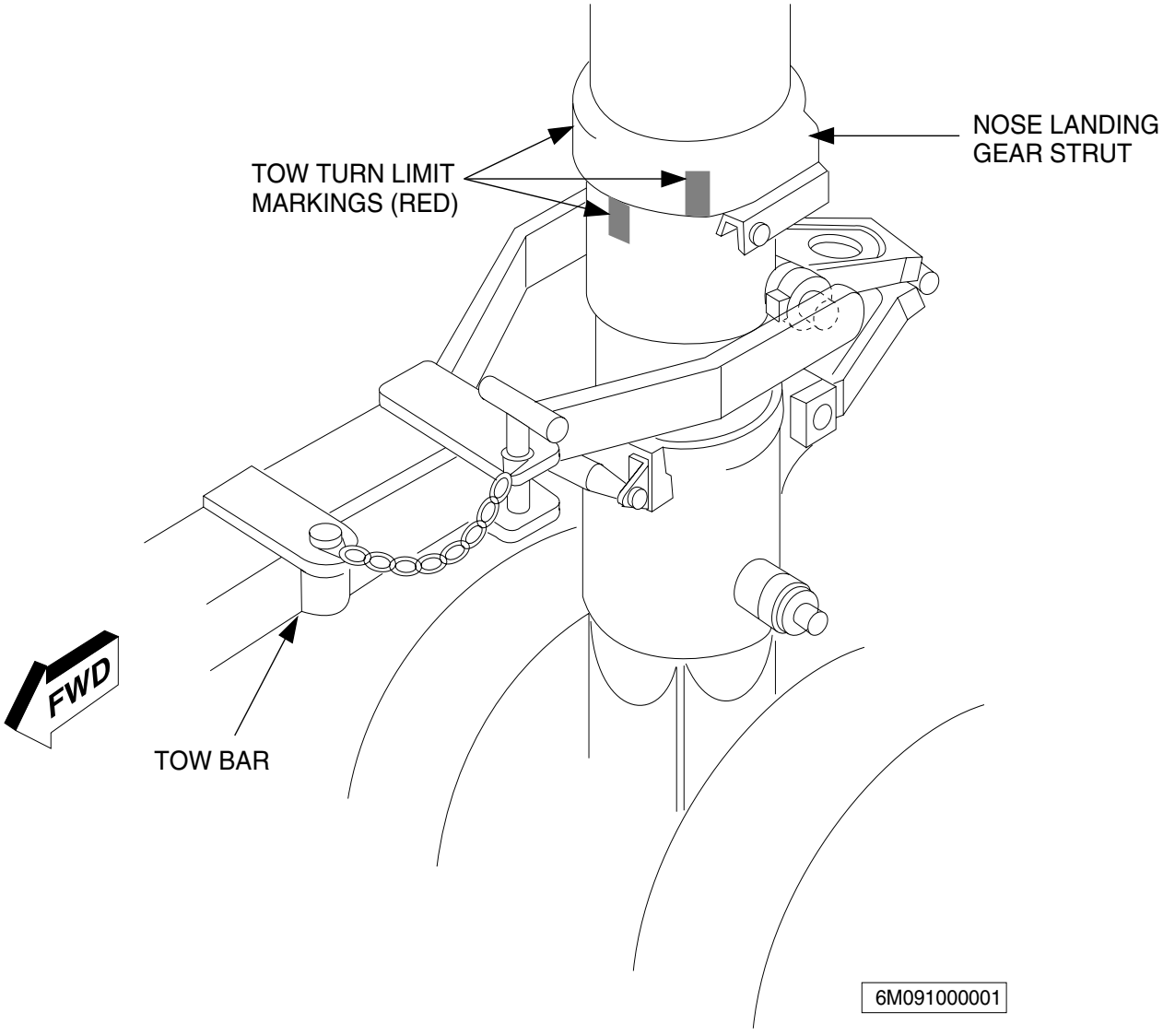
A. Towing and taxiing should only be done by qualified personnel to avoid damage to equipment or personnel. Consult the Airplane Flight Manual for specific instructions on preflight, starting engines, and general ground handling. Personnel should familiarize themselves with the specific instructions in Chapter 27 before locking and unlocking flight controls.

TOWING – DESCRIPTION AND OPERATION

1. General

A. Moving aircraft on the ground may be accomplished by attaching a tow bar to the nose landing gear (see Figure 1). Towing turn limits are 50° to each side of center. These turn limits are marked on the nose strut and must be strictly observed to prevent nose gear damage.

NOTE: For taxiing purposes, the nose wheel steering is allowed to deflect nose wheels an additional 13° past the towing turn limits, to a maximum of 63°.



Tow Bar Attachment
Figure 1

1. Procedures – Towing

A. Safety Precautions

(1) Station a qualified crew member in the pilot seat to man the brakes in case of emergency.

(2) Station observers at each wing tip and the tail to verify obstacle clearance. Check the clearance requirements in Figure 201.

CAUTION: DAMAGE WILL OCCUR TO THE NOSE WHEEL CENTERING CAM AND CAM FOLLOWER IF THE TURN LIMITS ARE EXCEEDED.

(3) Verify all three landing gear drag braces are in overcenter locked position.

(4) Close all aircraft doors.

(5) Tow at low speeds and avoid jerky motions.

(6) Move aircraft in a straight line before turning.

(7) Observe turning limits marked on the nose strut.

CAUTION: DO NOT TOW AIRCRAFT WITH NOSE WHEEL STEERING ARMED AND BATTERIES ON. DAMAGE TO THE NOSE WHEEL STEERING ACTUATOR CYLINDERS IS PROBABLE.

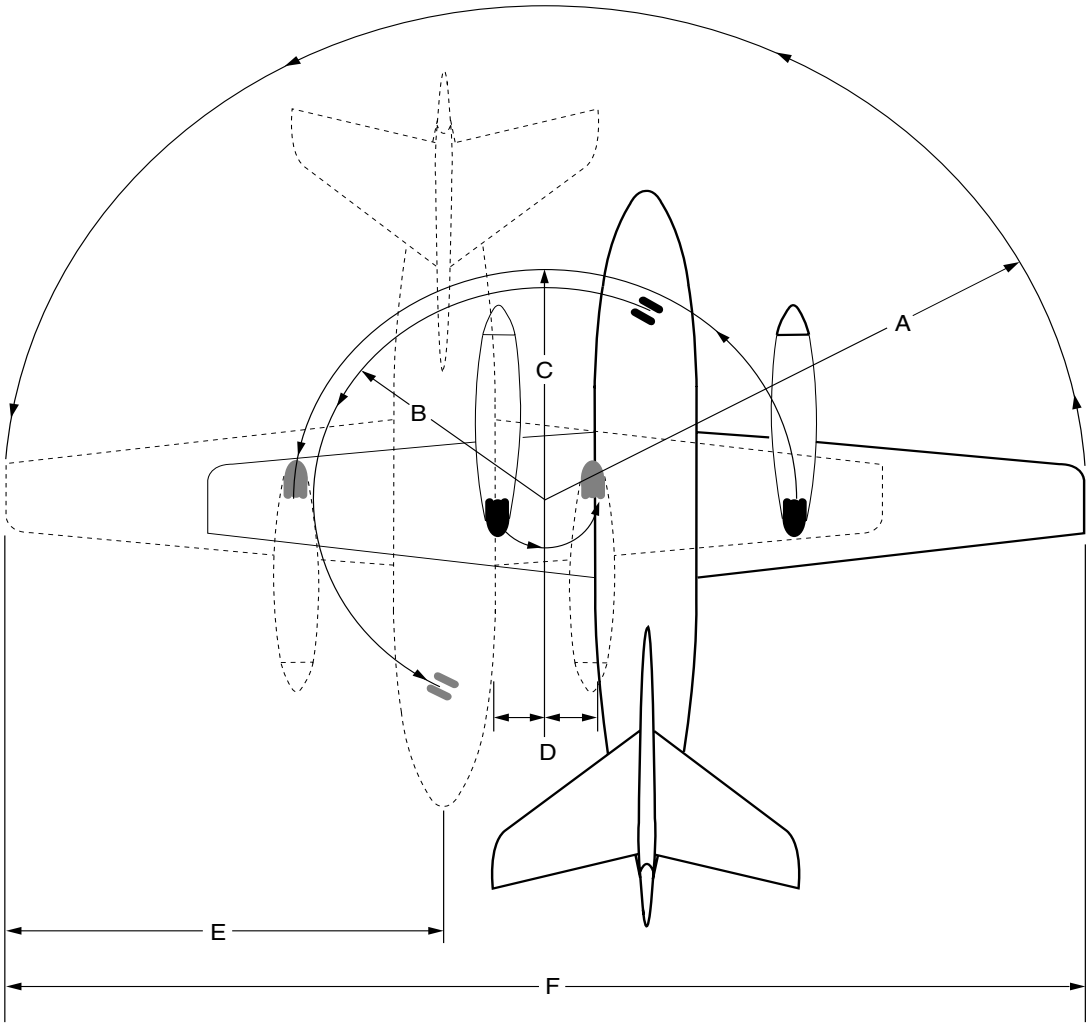
(8) Aircraft may be towed with gust locks engaged.

TAXIING – DESCRIPTION AND OPERATION

1. General

A. Aircraft taxiing should be done by qualified personnel acquainted with engine starting and shutdown procedures, aircraft/engine limitations and taxiing techniques. Special attention should be given to taxiing conditions insofar as danger areas and space required for maneuvers are concerned.

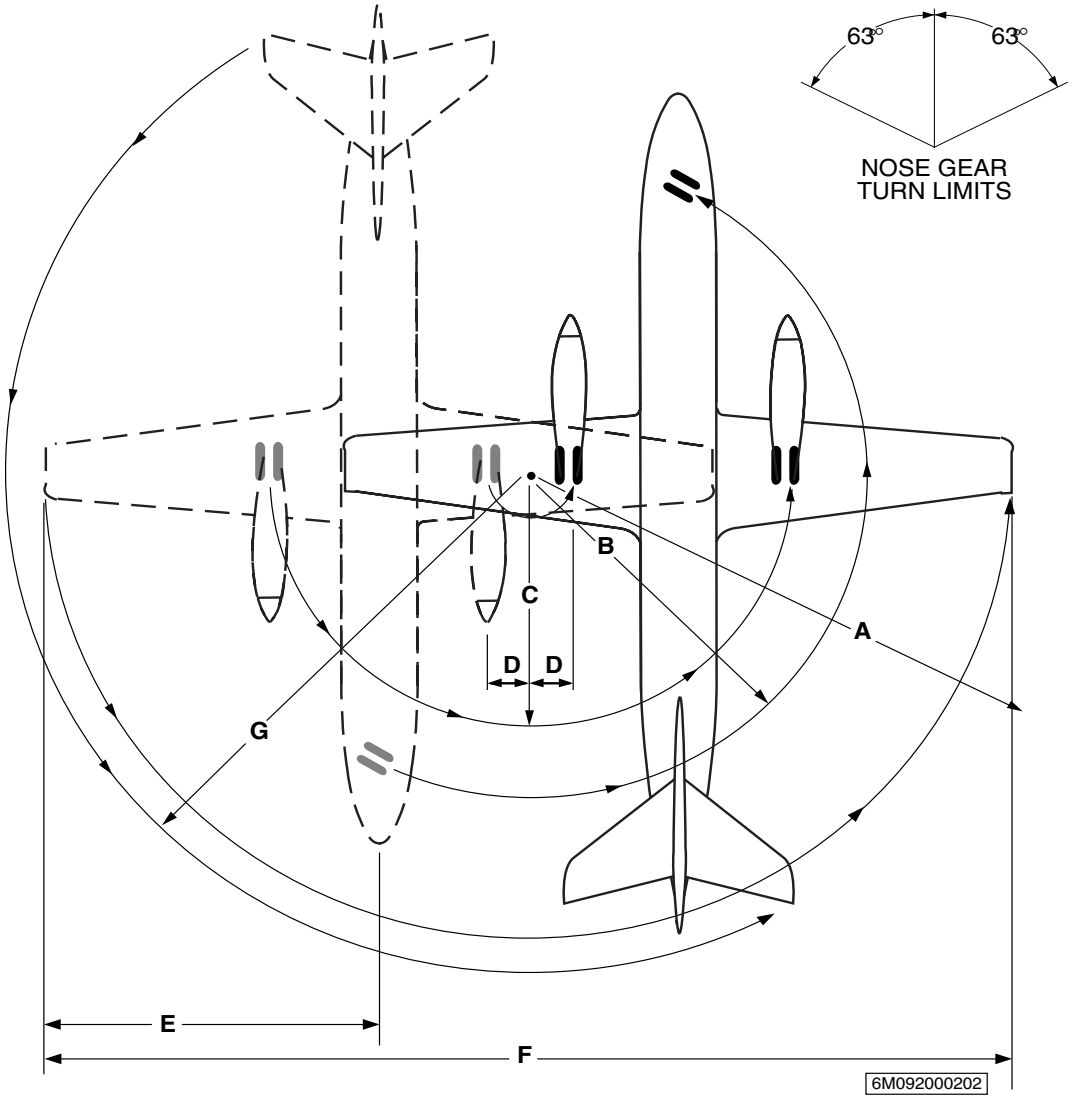
B. Consult Airplane Flight Manual for specific instructions on preflight, starting engines, and general ground handling. Personnel should familiarize themselves with the specific instructions in Chapter 27 before locking and/or unlocking flight controls.



A.	WING TIP ARC	28.4 Ft.	D.	INBOARD MAIN GEAR ARC (CENTER TO CENTER)	2.1 Ft.
B.	NOSE GEAR ARC	11.9 Ft.			
C.	OUTBOARD MAIN GEAR ARC (CENTER TO CENTER)	12.9 Ft.	E.	CENTERLINE OF AIRCRAFT TO WING TIP	23.0 Ft.
			F.	MINIMUM DIAMETER FOR 180° TURN	56.8 Ft.

NOTE: Main Gear On Inboard Side Of Turn Moves Aft.

Minimum Taxi Turn Radii
SA226 – T and T(B)
Figure 201



A.	WING TIP ARC AIRCRAFT TO WING TIP	38.5 Ft.	E.	CENTERLINE OF AIRCRAFT TO WING TIP	28.5 Ft.
B.	NOSE GEAR ARC	21.5 Ft.	F.	MINIMUM DIAMETER FOR 180 TURN	77.0 Ft.
C.	OUTBOARD MAIN GEAR ARC (CENTER TO CENTER)	19.8 Ft.	G.	MINIMUM RADIUS TO CLEAR HORIZONTAL STABILIZER TIP	35.4 Ft.
D.	INBOARD MAIN GEAR ARC (CENTER TO CENTER)	2.4 Ft.			

Minimum Taxi Turn Radii
SA226 – AT, TC
Figure 202

1. Taxiing Procedures

A. Pre-Taxi Checks

(1) Remove wheel chocks.

(2) Disengage flight control gust locks.

(3) Start engines IAW AFM.

(4) Take propellers off start locks.

(5) Release parking brake.

B. Taxiing

(1) Check surrounding area for aircraft, vehicle, or pedestrian traffic that may have entered the area.

(2) Advance power levers to sufficient power to taxi aircraft at low speed.

C. Steering

NOTE: Use brakes as necessary.

(1) Aircraft with electric control wheel steering:

(a) Hold button down on control wheel and turn wheel in desired direction.

(2) Aircraft with hydraulic rudder pedal nose steering:

(a) Place nose gear steering arm switch to ARMED.

(b) Hold power lever button in.

NOTE: For those aircraft with RPM lever switch, it is not necessary to hold power lever button in.

(c) Use rudder pedals to steer aircraft.

(d) Use park button as desired.

(3) Aircraft with hydraulic control wheel steering:

(a) Place nose gear steering arm switch to ARMED.

(b) Hold button down on control wheel and turn wheel in desired direction.

C. Post Taxi Parking

(1) Stop aircraft with brakes.

(2) Apply parking brake.

(3) Shut down engines IAW AFM.

(4) Engage flight control gust lock.

(5) Install wheel chocks.

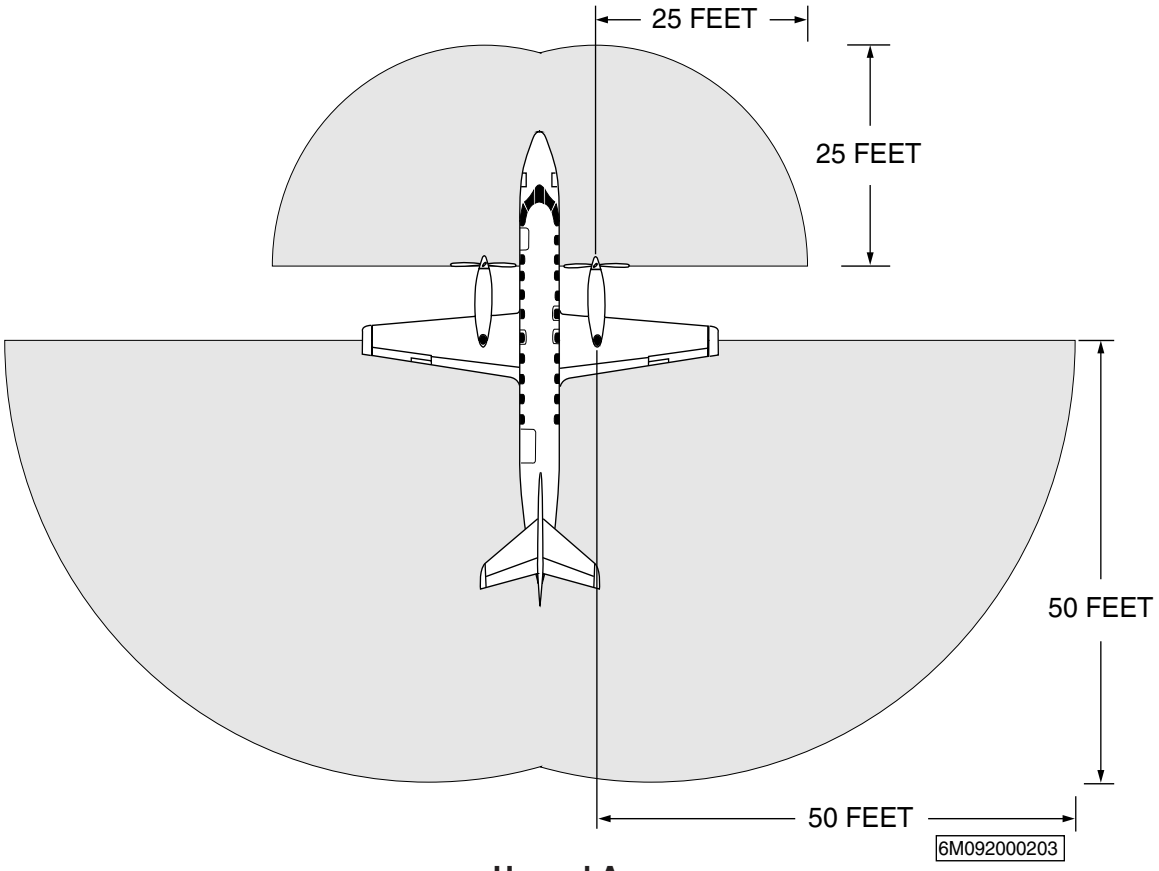
(6) Release parking brake.

(7) Install ground wire.

(8) install protective plugs as required.

2. Safety Precautions – Taxiing

A. Caution should be exercised during taxiing to ensure safe proximity of personnel and objects that could be damaged by propeller wash. To avoid damage to equipment/objects and injury to personnel, become familiar with the hazard areas as shown in Figure 203.



Hazard Area
Figure 203

CHAPTER 10 – PARKING AND MOORING

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PARKING MAINTENANCE PRACTICES	10-10-00	201	T AT TC	201-999 001-999 201-999
PRESERVATION AND STORAGE MAINTENANCE PRACTICES	10-10-10	201	T AT TC	201-999 001-999 201-999
MOORING MAINTENANCE PRACTICES	10-20-00	201	T AT TC	201-999 001-999 201-999
RETURN TO SERVICE MAINTENANCE PRACTICES	10-30-00	201	T AT TC	201-999 001-999 201-999

PARKING AND MOORING – DESCRIPTION AND OPERATION

1. General

A. This chapter contains parking, preservation, storage, and mooring procedures for SA226 – Series airplanes. Wind conditions and the period of time the aircraft will remain in position generally dictate the applicable procedure.

B. For information pertaining to preservation and storage of engines, see Garrett Turbine Engine Company (GTEC) Maintenance Manual.

2. Parking (10-10-00)

A. This section describes procedures for short term and long term parking. Short term parking is defined as a time period less than 12 hours. Long term parking may be defined as a time period in excess of 12 hours, but less than three days.

3. Preservation and storage (10-10-10)

A. This section contains procedures to properly preserve and store the airplane during periods of extended inactivity.

4. Mooring (10-20-00)

A. Mooring instructions are provided in this section. Mooring is defined as securing the airplane to help prevent high wind from upsetting the airplane. The most common form of mooring is to secure the airplane, with tie down ropes or chains, to a ground anchor point.

5. Return to Service (10-30-00)

A. This section contains maintenance instructions to return the airplane to servicable status following a period of storage.

1. Parking

A. Procedure

(1) Park aircraft heading into the wind, if possible.

CAUTION: DO NOT SET PARKING BRAKE WHEN THE BRAKES ARE OVER-HEATED OR IN COLD WEATHER WHEN ACCUMULATED MOISTURE MAY FREEZE THE BRAKES.

(2) Depress both pedals at pilot position and apply parking brake.

(3) Engage gust lock:

(a) Power levers ..... AFT OF FLIGHT IDLE

(b) Control Lock Lever ..... LOCKED

NOTE: The control lock lever is spring-loaded down when in the OFF detent. It is not spring-loaded when in the locked position, therefore, a positive aft and downward force on the lever is required to ensure positive engagement of the lever in the locked detent.

(c) Rudder, Ailerons, and Elevator Controls NEUTRAL

NOTE: Ensure positive engagement of all flight control surfaces and the power levers by gently moving the controls against their locks.

(4) Install Pitot and engine intake covers.

CAUTION: ALLOW ENGINES TO COOL AT LEAST TEN MINUTES BEFORE INSTALLING ENGINE INTAKE COVERS.

(5) Install SAS vane protector.

(6) Chock all wheels, fore and aft.

(7) Ensure all doors are closed.

(8) To unlock flight controls:

CAUTION: IF THE ELEVATOR CONTROL IS NOT RESTRAINED WHEN UNLOCKING THE FLIGHT CONTROLS, THE CONTROL COLUMN WILL SNAP FORWARD. THE TOP OF THE COLUMN MAY TRAVEL FAR ENOUGH FORWARD TO STRIKE INSTRUMENTS ON THE INSTRUMENT PANEL.

(a) Control Column ..... RESTRAIN

(b) Control Lock Lever ..... OFF

NOTE: Determine that the elevator, rudder, and ailerons unlock before the control lock lever reaches the forward, OFF detent. Failure of any control to unlock indicates a requirement to re-rig the respective control lock.

(c) Flight Controls ..... FREE

PRESERVATION AND STORAGE – MAINTENANCE PRACTICES

1. Preparation – Preservation and Storage

A. Fuel System

(1) Prior to storing aircraft, mix Biobor anti-algae additive to the fuel storage tanks. Mix two (2) ounces of Biobor per 1000 pounds of jet fuel.

NOTE: Biobor is an anti-micro biological agent, effective in preventing algae growth in fuel systems.

(2) Run engines for fifteen (15) minutes to ensure all fuel system components are well coated with Biobor/fuel mixture.

B. Landing Gear

(1) Rotate all tires 90° every 30 days.

NOTE: To aid in keeping track of last tire rotation, mark a line on the tire with chalk. Align chalk mark with the point of contact between tire and ground. Date chalk mark for future reference.

C. Electrical System

(1) Batteries need not be removed during storage. If batteries remain installed, remove access cover and disconnect battery power cable. Install access cover.

D. Airframe

(1) Protect windshield and windows from scratches by applying self adhering paper to external window/windshield surfaces.

E. Engines

(1) Preserve engines IAW GTEC Maintenance Manual.

2. Checklist – Preservation and Storage

A. This checklist defines the requirements and procedures necessary to preserve and store aircraft IAW the preceding instructions.

B. Blank spaces are provided to record date of accomplishment and/or quantities of fluids added.

AIRCRAFT STORAGE CHECKLIST

Fuel System

Fuel Preparation (Pre-ferry Flight) Date _____

1. Biobor added to fuel system. _____fL.oz.
2. Total fuel on board. L _____R _____

Fuel Preparation (Post ferry Flight) _____

1. Fuel tanks topped off at storage site. Date _____
2. Biobor added to fuel system. _____fl. oz.

3. Total fuel added. L _____ R _____
4. Run engines 15 minutes. L _____ R _____
5. Top off fuel tanks after engine run. Date _____
6. Biobor added to fuel system after run. _____ fl. oz.

Landing Gear

1. Main and nose struts cleaned. L_____R_____N_____
2. Tire inflation. L_____R_____N_____
3. Tires rotated 90 degrees. L_____R_____N_____
4. Tires marked with chalk. L_____R_____N_____
5. Gear down locks installed. L_____R_____N_____
6. Install wheel chocks. L_____R_____N_____

Electrical System

1. Batteries disconnected. L_____ R_____
2. Battery access doors/radome installed. L_____ R_____
3. All electrical switches OFF. Date_____

Airframe

1. Set flight control gust locks. Date_____
2. Release parking brake. Date_____
3. Connect static ground cable. Date_____
4. Service hydraulic system. Date_____
5. Service brake system. Date_____
6. Install Pitot covers. Date_____
7. Cover static ports. Date_____
8. Cover fuel vents at wing tips. Date_____
9. Cover all nacelle drains. Date_____
10. Cover dorsal fin openings. Date_____
11. Cover air conditioning inlet and exhaust openings. Date_____
12. Cover air conditioning drain. Date_____
13. Cover flammable fluids drain. Date_____

14. Cover all other drains. Date_____

15. Close and secure cargo door. Date_____

16. Close and secure passenger door. Date_____

17. Close and secure baggage doors. Date_____

18. Tie down airplane. Date_____

Engines

1. Engines preserved IAW GETC Maintenance Manual.

   A. 1 to 60 day period. Date_____

   B. 60 to 180 day period. Date_____

   C. 180 to 360 day period. Date _____

2. Install engine inlet and exhaust covers.

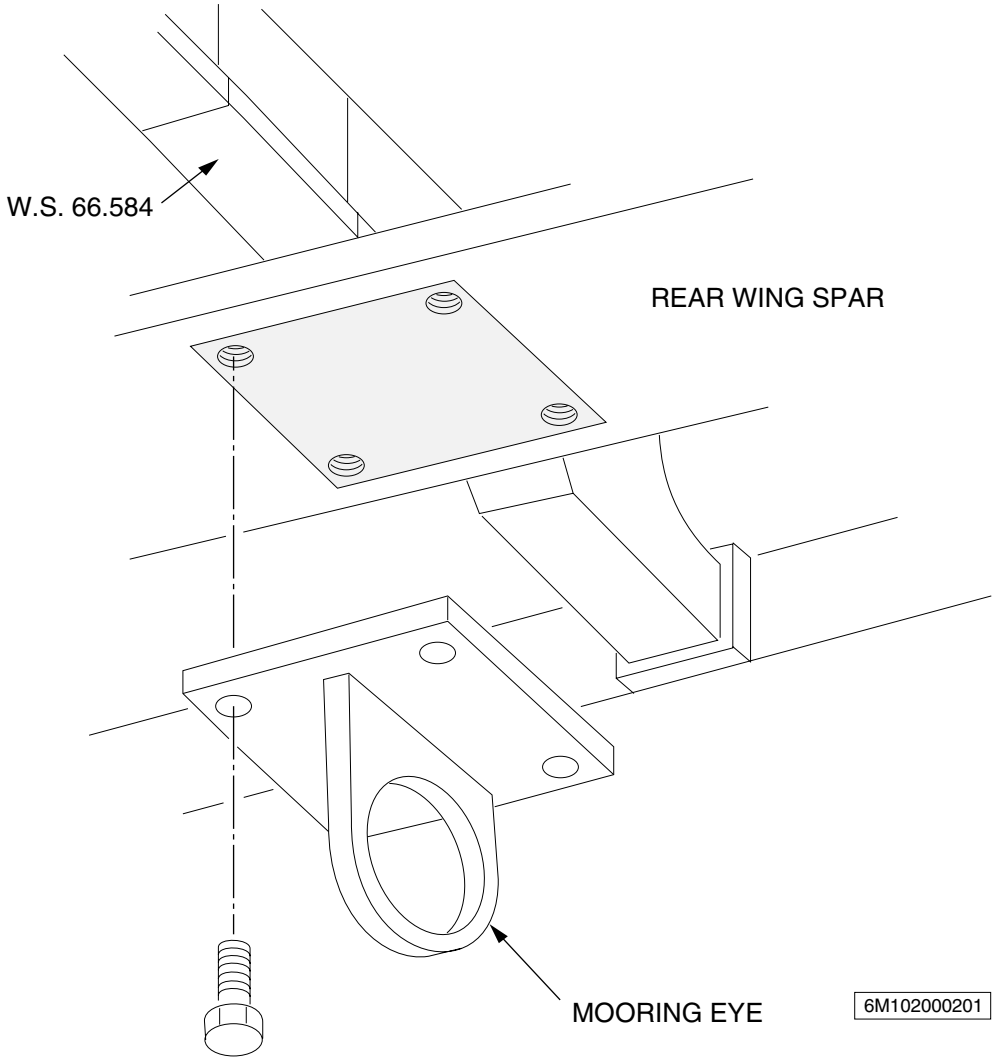
1. Mooring

A. Procedure

(1) Park aircraft IAW instructions contained in Parking section of this chapter.

(2) Mooring attachment points are the same for all model aircraft.

(3) Install the mooring eye in each wing. The mooring eye is part of the aircraft loose equipment. Mooring eyes are installed in the same locations as the jack pads, using the same attaching hardware. Locations are on the rear wing spar, inboard of the nacelles, (see Figure 201).



Mooring Eye
Figure 201

(4) Wing tiedown.

(a) Secure line at mooring eye underside of wing and anchor each at an outboard angle, toward wing tip.

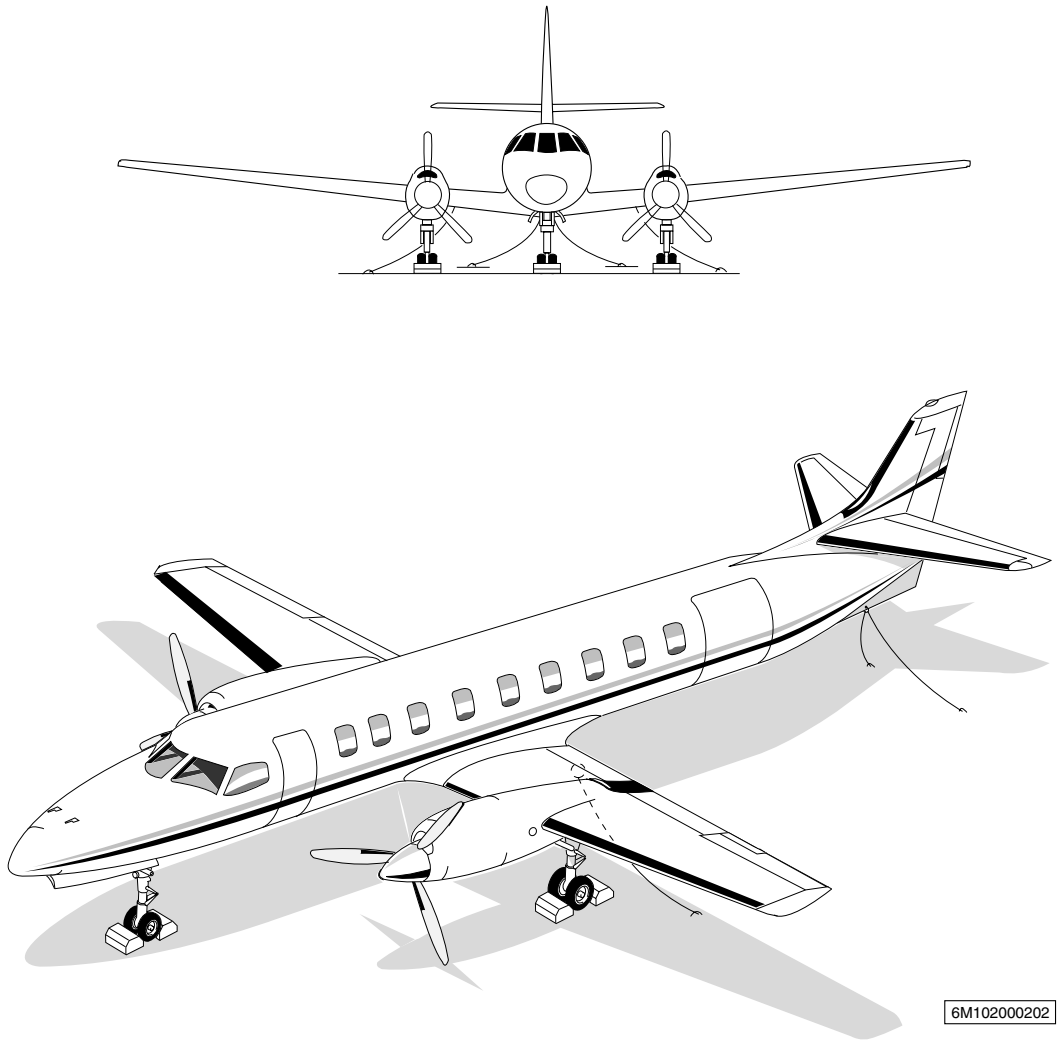
(5) Tail tiedown.

(a) Loop mooring line through mooring eye in ventral fin and anchor at two points. Anchor aft and outboard of eye.

(6) Pull all lines snug.

CAUTION: DO NOT USE CHAINS TO SECURE THE LANDING GEAR. USE OF CHAINS WILL SCRATCH OR DAMAGE POLISHED STRUT SURFACES.

(7) As an alternate method for mooring, nylon or Manila hemp ropes can be used around the three landing gear.



Aircraft Moored and Chocked
Figure 202

RETURN TO SERVICE – MAINTENANCE PRACTICES

1. Preparation – Return to Service

A. When returning aircraft to service after storage, perform the following checks and services.

(1) If an aircraft has been stored or is operated less than 200 flight hours within a 12 calender month period, perform inspection checks IAW your inspection procedures.

(2) Remove all barrier paper, self adhering paper, covers, caps, and/or other protective devices from windows, drains, static ports, pitots, and vents.

(3) Service tires, struts, hydraulic system, and brake system. Check all fluid levels and service as necessary.

(4) Look for evidence of bird nesting and any unreported damage that may have occurred.

(5) Since stored airplanes become susceptible to parts cannibalization, be alert for missing parts and/or components.

(6) Connect power cables to batteries.

(7) Perform engine run IAW GTEC Maintenance Manual(s).

NOTE: During engine run, visually check for leaks.

AEROSPACE
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CHAPTER 11 – PLACARDS AND MARKINGS

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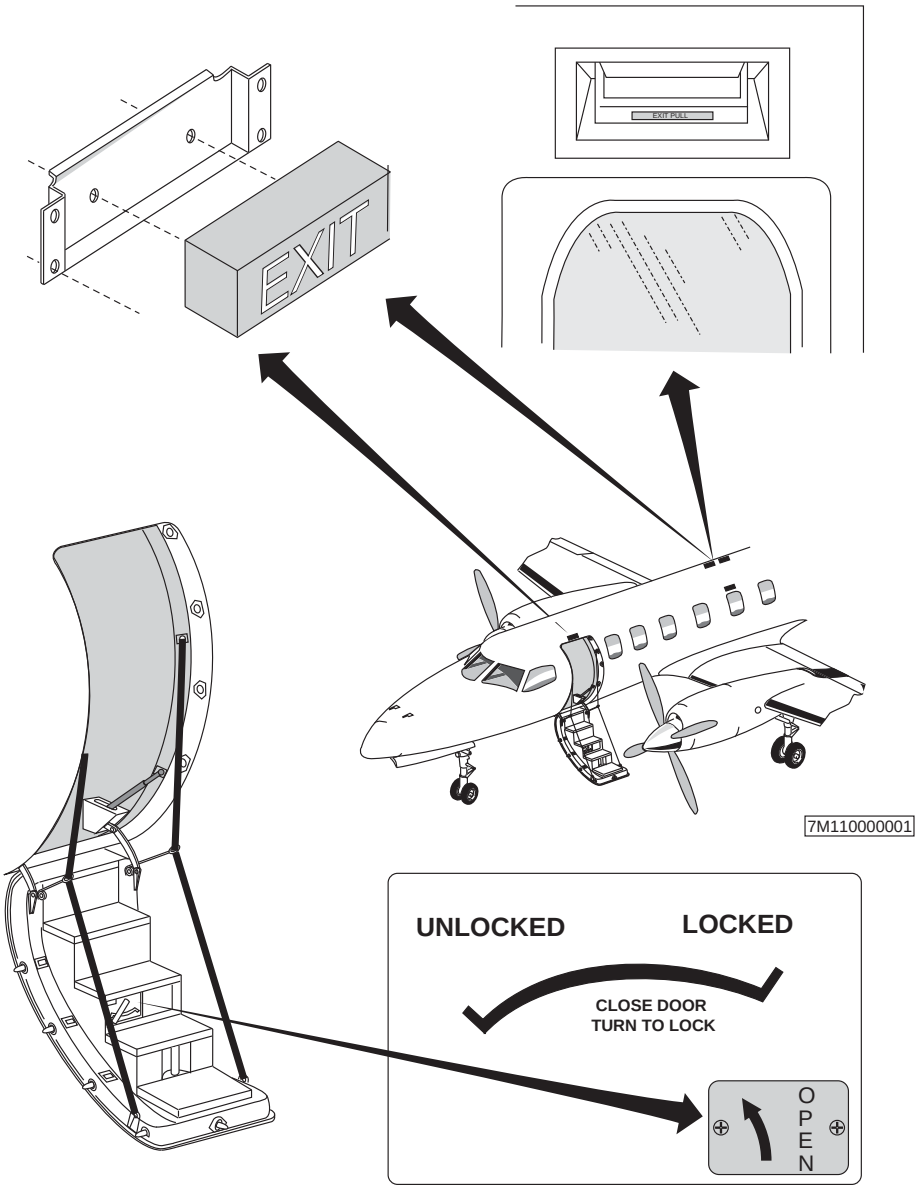
EMERGENCY PHOSPHORESCENT SIGNS – DESCRIPTION AND OPERATION

1. General

A. SA226 Series aircraft are equipped with phosphorescent signs to indicate emergency exit locations at the cabin door and escape hatches.



(ACTUAL LOCATION OF EMERGENCY EXIT SIGNS WILL VARY DEPENDING ON AIRCRAFT MODEL AND INTERIOR CONFIGURATION)

Emergency Phosphorescent Signs
Figure 1

EMERGENCY PHOSPHORESCENT SIGNS – MAINTENANCE PRACTICES

1. Removal/Installation – Emergency Phosphorescent Signs

A. Removal – Emergency Exit Sign

NOTE: Emergency exit signs are attached to interior paneling sections of the aircraft. To remove a sign assembly, the entire panel section the sign is attached to must be removed.

(1) Remove applicable panel.

(2) Remove washers and nuts from exit sign studs.

(3) Remove exit sign.

B. Installation – Emergency Exit Sign

(1) Position exit sign to bracket.

(2) Secure with washers and nuts.

(3) Install panel.

2. Inspection/Check – Emergency Phosphorescent Signs

A. Inspection – Emergency Phosphorescent Signs

(1) Verify calibration date of tool being used (comparator light or light meter) is current.

NOTE: Minimum intensity of light emitted by phosphorescent sign is determined by your local operating regulations and must be verified at time of inspection.

(2) Inspect sign for brightness using either of the following:

(a) A comparator light available from the following.

SRB Technologies
320 - 140 Boundary Road
Pembroke, Ontario
Canada K8A 6W5
Telephone: (613) 732-0055

(b) A portable light meter capable of measuring below minimum level called out in local regulations.

1 When using a portable light meter, all ambient light must be prevented from entering aircraft interior.

2 Setup light meter per manufacturers instructions.

(3) Replace any sign that does not meet the minimum requirement of local airworthiness regulations.

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CHAPTER 12 – SERVICING

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12-10-00	DO	4	APR 01/05
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12-10-10	MP	204	APR 01/05
12-10-15	MP	201	APR 01/05
12-10-15	MP	202	APR 01/05
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SERVICING – DESCRIPTION AND OPERATION

1. General

A. This chapter includes instructions for replenishing fuel, oil, hydraulic, and brake fluids, and for servicing tires and the oxygen system.

2. Replenishing (12-10-00)

A. This section includes instructions for replenishing fuel, oil, hydraulic and brake fluids, tires and tire pressure, landing gear struts, oxygen system, lavatory sink fluids, alcohol/water injection (AWI) fluid, and external power receptacle.

3. Scheduled Servicing (12-20-00)

A. This section includes instructions for scheduled servicing such as the periodic lubrication of components, and aircraft external and internal cleaning.

4. Unscheduled Servicing (12-30-00)

A. This section includes instructions for unscheduled servicing, such as the removal of ice and snow, and procedures for defueling an aircraft.

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REPLENISHING – DESCRIPTION AND OPERATION

1. **General (Figures 1 and 2)**

   A. This chapter includes instructions for replenishing fuel, oil, hydraulic and brake fluids, alcohol/ water (AWI) mixture, tires and tire pressure, and oxygen.

2. **Fuel Replenishing (12-10-10)**

   A. This section includes those procedures used in fueling an aircraft, fuel tank capacity, recommended fuels and safety precautions.

3. **Oil Replenishing (12-10-15)**

   A. This section includes instructions to check/replenish oil, tank and reservoir capacities, recommended products, and alternates.

4. **Hydraulic Fluid Replenishing (12-10-20)**

   A. This section includes procedures to replenish hydraulic fluids, reservoir capacity, and recommended fluids and alternates.

5. **Tire Pressure Replenishing (12-10-25)**

   A. This section includes a chart showing the tire sizes, ply ratings and pressure, and tire changing precautions.

6. **Landing Gear Strut Replenishing (12-10-30)**

   A. This section includes instructions for inflating the landing gear using dry nitrogen.

7. **Brake Fluid Replenishing (12-10-35)**

   A. This section includes procedures for checking and filling the brake reservoir, and brake reservoir capacity.

8. **Passenger Conveniences Replenishing (12-10-40)**

   A. This section includes procedures for replenishing the fluids for the lavatory compartment sink, and refreshment bar/galley.

9. **Oxygen System Replenishing (12-10-45)**

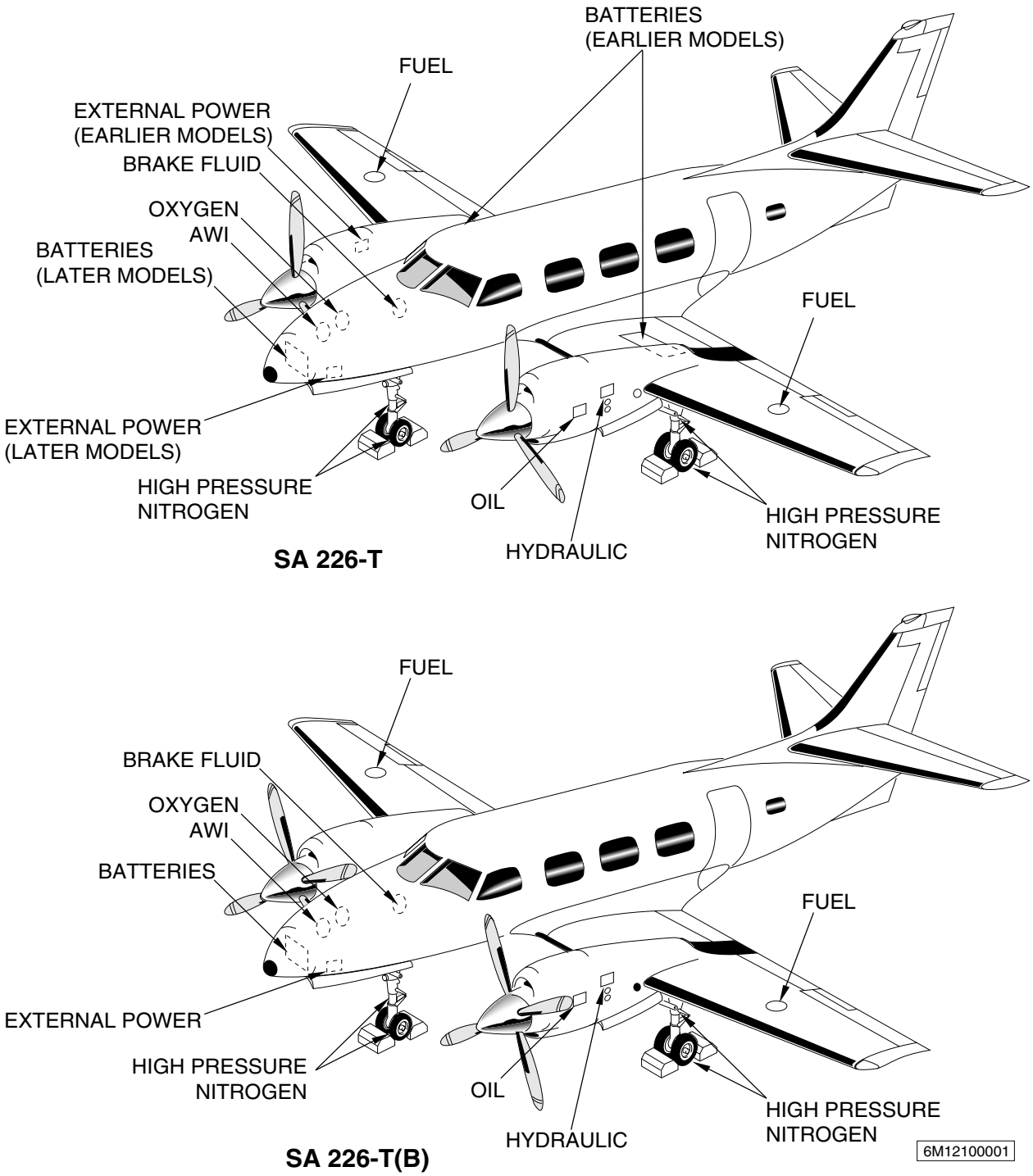
   A. The oxygen filler valve is located in the nose compartment. Refer to Chapter 35, this Manual, for servicing procedures.

10. **Alcohol/ Water Fluid Replenishing (12-10-50)**

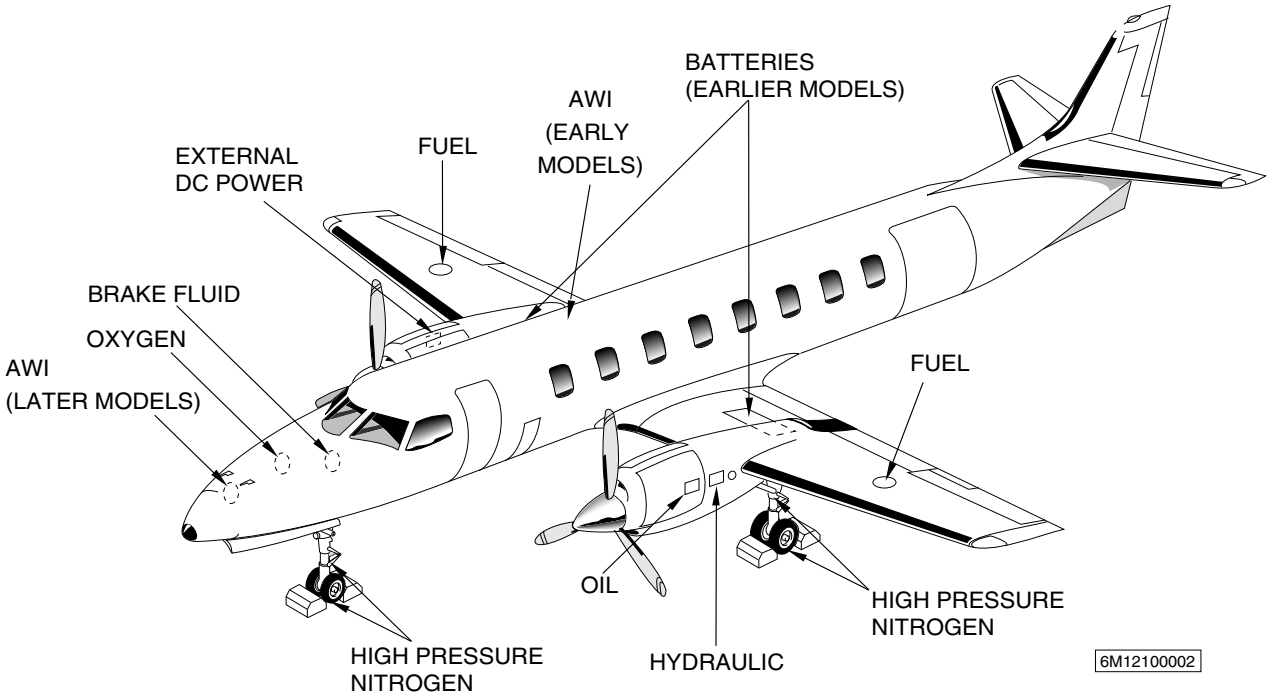
    A. This section includes procedures for replenishing the alcohol/water fluid containment tank.

11. **Batteries**

    A. Refer to Chapter 24, this Manual, for service and maintenance of batteries.



Replenishing – SA226-T and T(B)
Figure 1



Replenishing – SA226-AT and TC
Figure 2

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FUEL REPLENISHING – MAINTENANCE PRACTICES

1. Servicing – Fuel Replenishing

A. Safety Precautions – Fuel Servicing

NOTE: Observe the following safety precautions prior to, during, and after servicing.

(1) Permit no smoking or open flames within 50 feet of aircraft, hangar, fuel, or oil storage areas.

(2) Keep the refueling area clear of all unnecessary equipment and materials.

(3) Avoid operation of gasoline engines or electric motors in the proximity of the refueling area when possible. If operation of such equipment is essential to the refueling procedure, ensure proper safety devices are used (e.g., lightning, spark and exhaust flame arrestors, and explosion proof motors).

(4) Remove any spilled fuels and liquids from work stands and servicing areas immediately. Flush contaminated area(s) with water.

(5) Prevent fuel spillage(s) from coming in contact with the skin. If spillage occurs, wash affected area immediately with soap and water.

(6) Do not use fuel as a cleaning agent.

(7) Allow sufficient time, after refueling, for clothing to air prior to smoking, working near power equipment, or working near open flames.

(8) Allow only qualified personnel to service the aircraft.

B. Fueling Procedures

(1) Observe safety precautions.

(2) Ensure aircraft electrical power is OFF.

WARNING: GROUNDING PROCEDURES ARE FAA RECOMMENDED BY ADVISORY CIRCULAR 00.43A, BUT ARE NOT MANDATORY. NON-COMPLIANCE OF THESE INSTRUCTIONS MAY RESULT IN PERSONAL INJURY OR PROPERTY DAMAGE.

(3) Ground aircraft as follows:

(a) Aircraft to ground.

(b) Truck to ground.

(c) Truck to aircraft.

(d) Fuel line to aircraft.

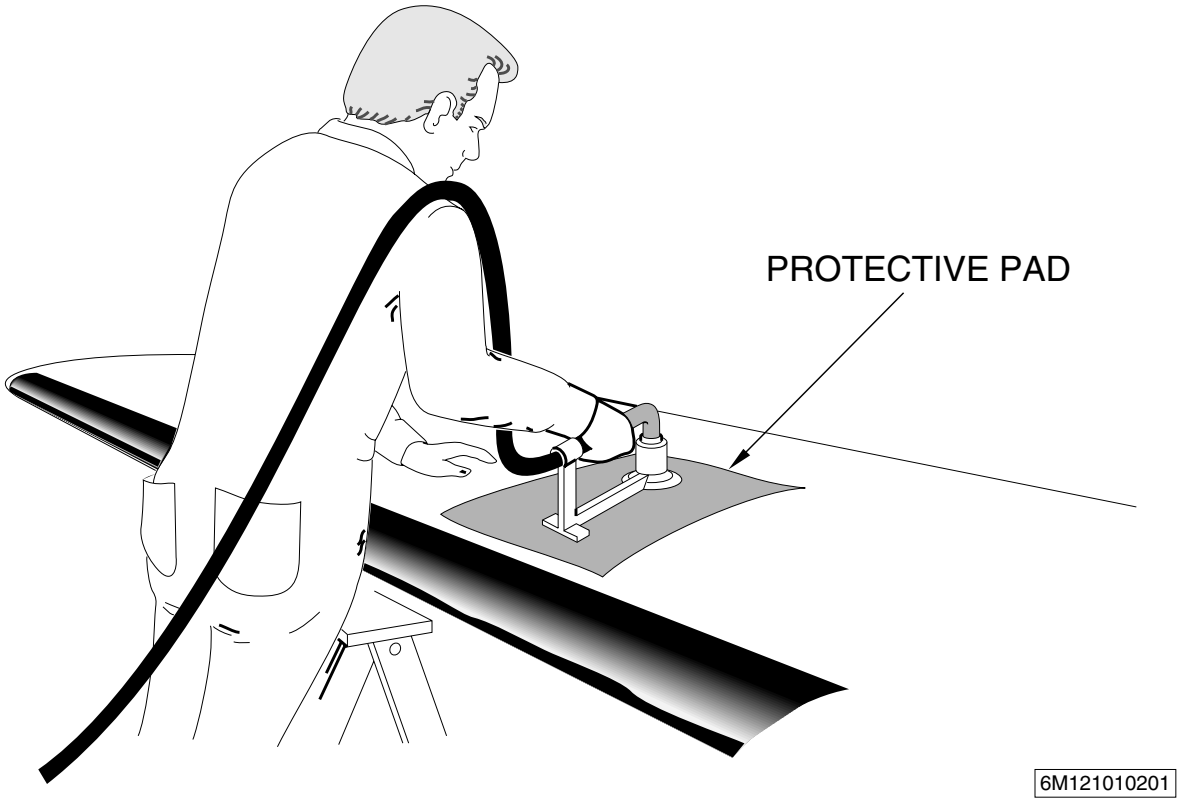
CAUTION: DO NOT ALLOW AFT PORTION OF FUEL NOZZLE OR HOSE TO REST ON THE DEICE BOOT. THE DEICE BOOTS ARE MADE OF SOFT, FLEXIBLE, SYNTHETIC RUBBER WHICH MAY BE DAMAGED IF HOSES ARE DRAGGED OVER THE SURFACE OR IF LADDERS AND PLATFORMS ARE RESTED AGAINST THEM.

(4) Using a small stepladder, bring refueling hose over wing and service tank to desired quantity.

NOTE: On a sloping ramp, the uphill tank may not accept a full fuel load if the downhill wing tank is filled first. When refueling on a sloping ramp, the uphill tank can be filled to maximum capacity by refilling both tanks simultaneously, by refilling the uphill wing tank first, or by adding fuel to the wings alternately in approximately 125 gallon increments. When less than maximum capacity is required, this special fueling procedure is not necessary.

(5) When tank is serviced, ensure tank cap is securely seated and locked (latch facing aft).

(6) Remove fuel servicing nozzle before disconnecting ground.



Fueling Aircraft
Figure 201

C. Fuel Grades and Types

(1) Use aviation fuel conforming to AiResearch Installation Manual IMS117 (Jet A, Jet A1, Jet B, JP-4, JP-5 and JP-8).

D. Emergency/Alternate Fuels

CAUTION: THE USE OF AUTOMOTIVE GASOLINE, DIESEL FUELS, HEATING FUELS OR ANY COMBINATION OF THESE PRODUCTS WITH JET FUELS IS PROHIBITED.

(1) Grade 80/87 aviation gasoline (AVGAS) is permitted for emergency use, but restricted to no more than 1,000 US gallons (3,785 liters) per engine for each 100 hours of engine operation.

NOTE: The amount of aviation gasoline used must be recorded in the engine log book.

(2) Grade 100LL aviation gasoline is permitted for emergency use at a rate not to exceed 250 US gallons (964 liters) per 100 hours of operation with the total use limited to 7,000 US gallons (26,495 liters) per engine overhaul period.

(3) Jet fuel and aviation gasoline may be mixed in any proportion. If 25% or more aviation gasoline is used, add one US quart (0.95 liter) of MIL-L-6082 specification grade 1065 to 1100 piston engine oil per 100 US gallons (379 liters) of aviation gasoline to provide fuel pump lubrication.

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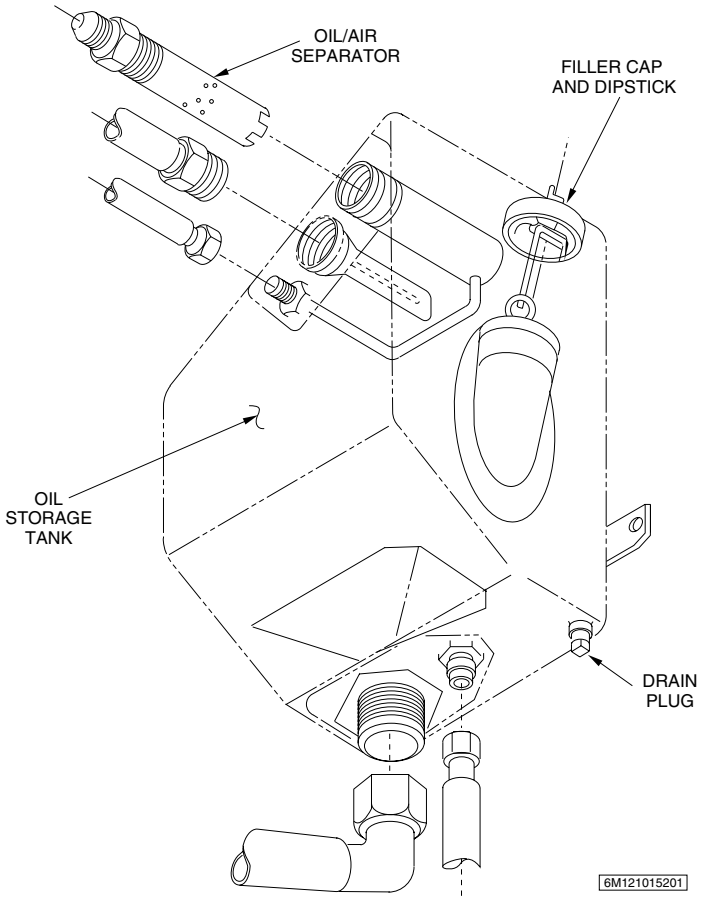


Engine Oil – Storage Tank
Figure 201

1. Servicing – Oil Replenishing

WARNING: PROLONGED EXPOSURE TO MIL-L-23699B ENGINE LUBRICATING OIL MAY BE HAZARDOUS TO YOUR HEALTH. FOLLOW MANUFACTURERS' SAFETY PRECAUTIONS.

A. Oil Level Check and Filter

(1) Oil Level Check (Figure 201).

CAUTION: NEVER OPERATE AIRCRAFT WITHOUT OIL VISIBLE IN SCREEN AT BOTTOM OF OIL TANK FILLER NECK.

(a) To obtain the most accurate reading, engine oil level should be checked within first ten minutes after engine shutdown with propellers rotated against start locks.

NOTE: If the propellers are not on start locks, unfeather the propellers to engage the start locks. In order to scavenge oil back to the storage tank, motor the engine twice to 15% RPM. Pulling the propeller through by hand may not provide sufficient scavenge pressure for correct oil level indication.

(b) Remove LH side access panel from engine.

(c) A dip stick is attached to the oil tank filler cap and is marked to indicate various oil levels. Verify correct oil level.

(d) After servicing the oil tank, replace filler cap and install side engine cowling.

NOTE: Overservicing is not hazardous, but it may leave oil deposits in the flap area.

(2) Oil Filter.

(a) A 10 Micron oil filter is installed on an adapter on the aft left side of the reduction gear case above the fuel control. It filters all oil entering the oil system from the oil pump. Inspection and replacement of the filter element should be accomplished at the interval specified in the inspection program.

(b) For removal and installation of oil filter see Chapter 79 of this manual.

B. Oil Tank Capacities

(1) External oil tank capacity is 3.6 U.S. quarts (3.4 liters). Internal oil passages and oil lines contain an additional 4.5 U.S. quarts (4.3 liters) for a total system capacity of 8.1 U.S. quarts (7.7 liters).

C. Recommended Oils

(1) Refer to Table 201 for approved and recommended oils.

D. Oil Changes

CAUTION: SA226 SERIES AIRPLANES USE TYPE II ENGINE OIL ONLY.

(1) Change engine oil at 900 hours. Refer to Chapter 79, this manual, for oil change instructions.

MIL SPECIFICATION	BRAND NAME
MIL-L-23699	Mobil Jet II
	Mobil 254
	Exxon (Enco/Esso) 2380 Turbo Oil
	Castrol 5000
	Aeroshell/Royco Turbine Oil 500
	Aeroshell/Royco Turbine Oil 560
	Chevron Jet Engine Oil No. 5
	Stauffer Jet II
	Cal Tech RPM Jet Engine Oil No. 5
	Castrol 205

Approved Oils
Table 201

HYDRAULICS REPLENISHING – MAINTENANCE PRACTICES

1. Servicing – Hydraulics Replenishing

A. Hydraulic Fluid – Replenishment

(1) The hydraulic system may be replenished by adding fluid to the hydraulic reservoir, or through the use of a hydraulic servicing cart.

B. Hydraulic Fluid – Capacity

CAUTION: DO NOT USE MIL-H-5606 HYDRAULIC FLUID IN AIRCRAFT HYDRAULIC SYSTEM. MIL-H-5606 HYDRAULIC FLUID IS USED FOR GEAR STRUT SERVICING ONLY.

(1) The system fluid capacity, including lines, actuators, valves and reservoir, is two U.S. gallons (7.57 liters). Aircraft must be serviced with approved hydraulic fluid MIL-H-83282 only.

NOTE: If hydraulic fluid contamination is suspected, system must be purged and replenished with MIL-H-83282 hydraulic fluid.

(2) A sight gauge, is located on the outboard side of the left nacelle. Add more fluid whenever fluid level drops to the “ADD” mark on the sight gauge.

C. Servicing – With Hydraulic Cart (Figure 201)

NOTE: Use this procedure when servicing with a hydraulic cart, and when aircraft is not supported on jacks.

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING OFF ELECTRICAL POWER.

(1) Rotate emergency gear release handle fully aft and secure.

(2) Turn aircraft electrical power ON. Bleed hydraulic pressure in accumulator by moving flap handle up and down several times, until hydraulic pressure indicator indicates zero.

NOTE: Ensure flap position handle is returned to FULL UP position.

WARNING: HYDRAULIC SYSTEM IS UNDER PRESSURE. USE CAUTION WHEN LOOSENING OR REMOVING FILLER CAP.

(3) Bleed air pressure from hydraulic reservoir as follows:

(a) Locate and open hydraulic reservoir access door on outboard side of left nacelle.

(b) Remove spring lock and loosen filler cap. SLOWLY remove cap.

(4) Replace and lock filler cap and close access door.

(5) Remove lower hydraulic access panel immediately aft of lower left engine cowling.

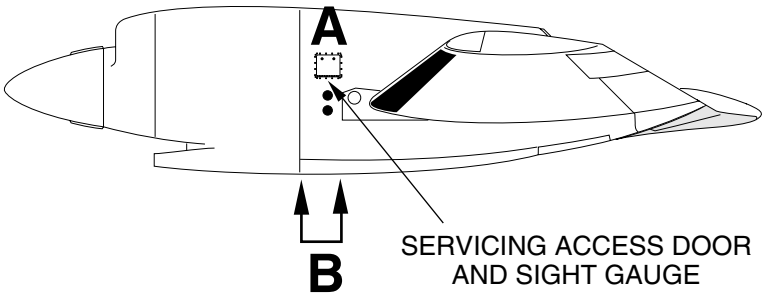
(6) Ensure hydraulic cart, aircraft lines, and connectors are clean. Connect hydraulic cart pressure and return lines.

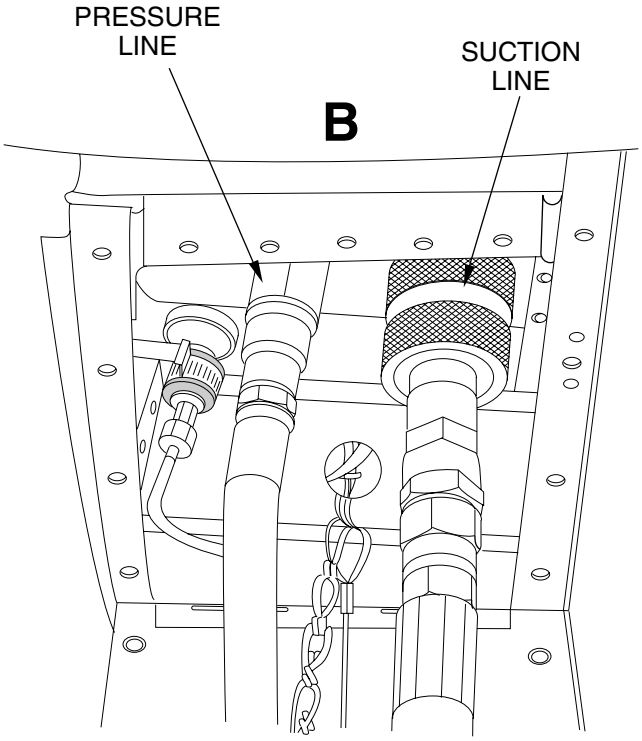
(7) Move hydraulic cart controls to FILL position. Observe hydraulic fluid sight gauge on the outboard side of the left nacelle. Reservoir is full when upper window is half full.

NOTE: Overfilling of the hydraulic reservoir may result in hydraulic fluid spilling overboard during landing gear or flap operation.

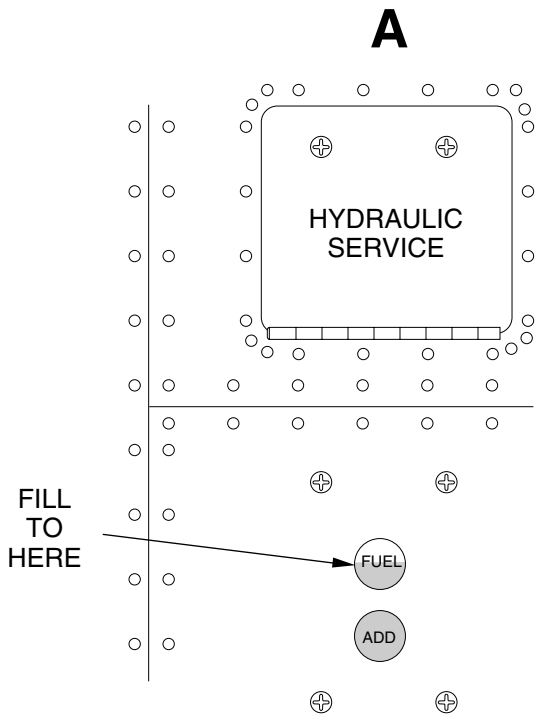
(8) Neutralize hydraulic cart controls. Disconnect hoses from pressure and return ports and reinstall access panel.

(9) When service is complete, remove hydraulic cart and return emergency gear release handle to normal position.





EXTERNAL HYDRAULIC POWER ACCESS PANEL



SERVICING ACCESS DOOR AND SIGHT GAUGE

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Servicing Hydraulic System With Hydraulic Cart Figure 201

D. Servicing – Without Hydraulic Cart

(1) Move flap position handle FULL UP.

(2) Open access door on outboard side of left nacelle. Unlock and remove filler cap. Remove SLOWLY to allow air pressure to escape. Ensure that filler screen is in place before servicing.

WARNING: HYDRAULIC SYSTEM IS UNDER PRESSURE. USE CAUTION WHEN LOOSENING OR REMOVING FILLER CAP.

(3) Observing sight gauge, add hydraulic fluid until upper window is half full.

NOTE: Overfilling hydraulic reservoir may result in hydraulic fluid spilling overboard during landing gear or flap operation.

(4) Replace and lock filler cap.

NOTE: If fluid level is below bottom window, check system for leaks or empty hydraulic lines. It may be necessary to service system with a hydraulic cart.

(5) Close and secure hydraulic reservoir access door.

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TIRE PRESSURE REPLENISHING – MAINTENANCE PRACTICES

1. Servicing – Tire Pressure Replenishing

A. Tire Sizes and Air Pressures

NOTE: Main tires, other than 18 X 5.5, and nose tires, other than 16 X 4.4, are not compatible with standard installation. Aircraft structure must be modified IAW SB32-007 before other tires may be used.

(1) Other than the Metro IIA, aircraft originally equipped with low pressure tires may use high pressure tires.

CAUTION: DO NOT MIX PLY RATINGS BETWEEN TIRES ON NOSE GEAR OR BETWEEN TIRES ON THE MAIN GEAR.

ALUMINUM AND MAGNESIUM WHEELS MAY BE INTERMIXED ON AN AIRCRAFT; HOWEVER, TUBE AND TUBELESS TIRES MAY NOT BE INTERMIXED.

(2) Table 201 depicts recommended tire sizes and inflation air pressures for high pressure tires at an assumed base temperature of 70°F (21.1°C).

SA226 Series Aircraft Factory Standard Tires – High Pressure

TIRE SIZE	PLY RATING	GEAR	PRESS PSI	T	AT	TC
18 X 5.5	8	Main	105	201-397	001-397	201-397
18 x 5.5	10	Main	140	(Customer Option)		
16 x 4.4	6	Nose	80	201-397	– – – – –	– – – – –
16 x 4.4	6	Nose	70	– – – – –	001-397	201-397

SA226 Series Aircraft Factory
Standard Tires – High Pressure
Table 201

(3) Table 202 depicts recommended tire sizes and inflation air pressures for low pressure tires at an assumed base temperature of 70°F (21.1°C).

SA226 Series Aircraft Factory Standard Tires – Low Pressure

TIRE SIZE	PLY RATING	GEAR	PRESS PSI	T	AT	TC
19.5 x 6.75-8	10	Main	62	201-397	– – – – –	– – – – –
		Main	70	– – – – –	001-397	201-397
		Main	75	– – – – –	– – – – –	398-419
18 x 4.4	6	Nose	70	201-397	– – – – –	– – – – –
		Nose	46	– – – – –	001-397	201-397
		Nose	50	– – – – –	– – – – –	398-419
18 x 4.4	10	Nose	50	– – – – –	001-397	201-397
(Chine)		Nose	54	– – – – –	– – – – –	398-419

SA226 Series Aircraft Factory Standard Tires – Low Pressure Table 202

LANDING GEAR STRUT REPLENISHING – MAINTENANCE PRACTICES

1. Servicing – Landing Gear Strut Replenishing

A. Strut Pressure Servicing

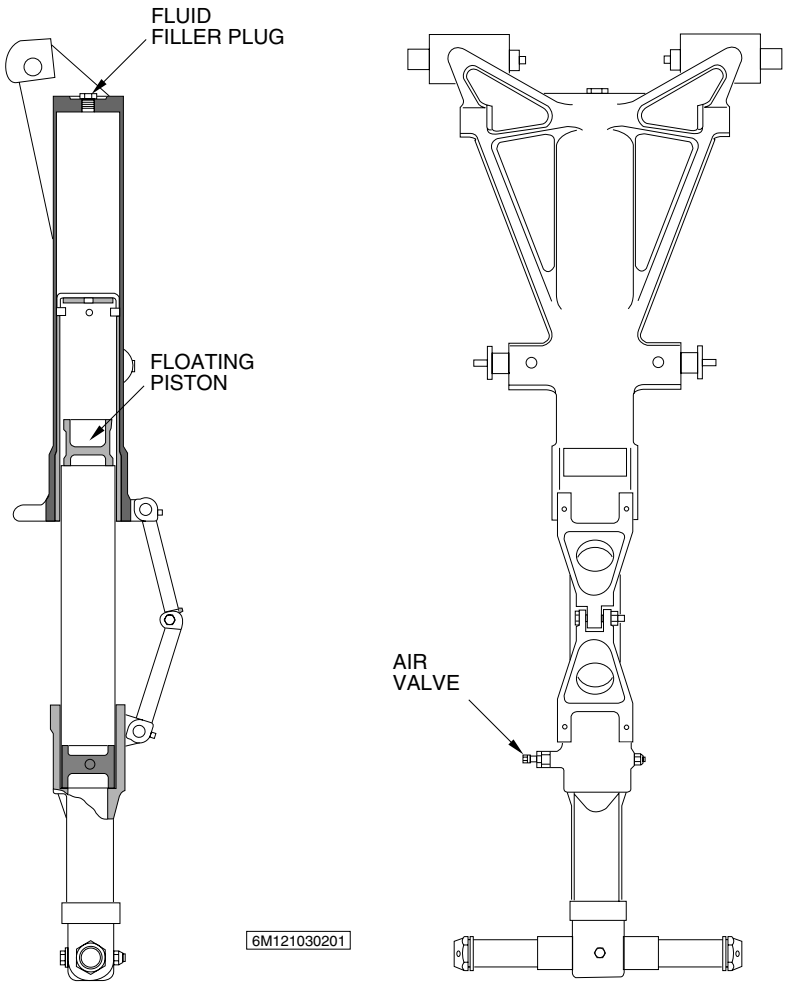
NOTE: Varying aircraft weights and air temperatures will change the strut height, which will affect the relative stiffness of the strut action.

CAUTION: IMPROPER SERVICING OF LANDING GEAR STRUTS MAY RESULT IN DAMAGE TO STRUTS.

(1) A properly serviced strut operates best in a range of approximately two (2) inches (5.08 cm) extended from the fully compressed position with the weight of the aircraft on the strut.

(2) With aircraft weight on gear, apply nitrogen or dry air to air valve and Inflate strut to achieve 2.0 inches (5.08 cm) of strut extension.

NOTE: Approximately 1.5 inches (3.81 cm) of piston will be visible on a normally fully compressed Nose Landing Gear strut. Inflate nose gear so that 3.5 inches (8.89 cm) of piston is visible.

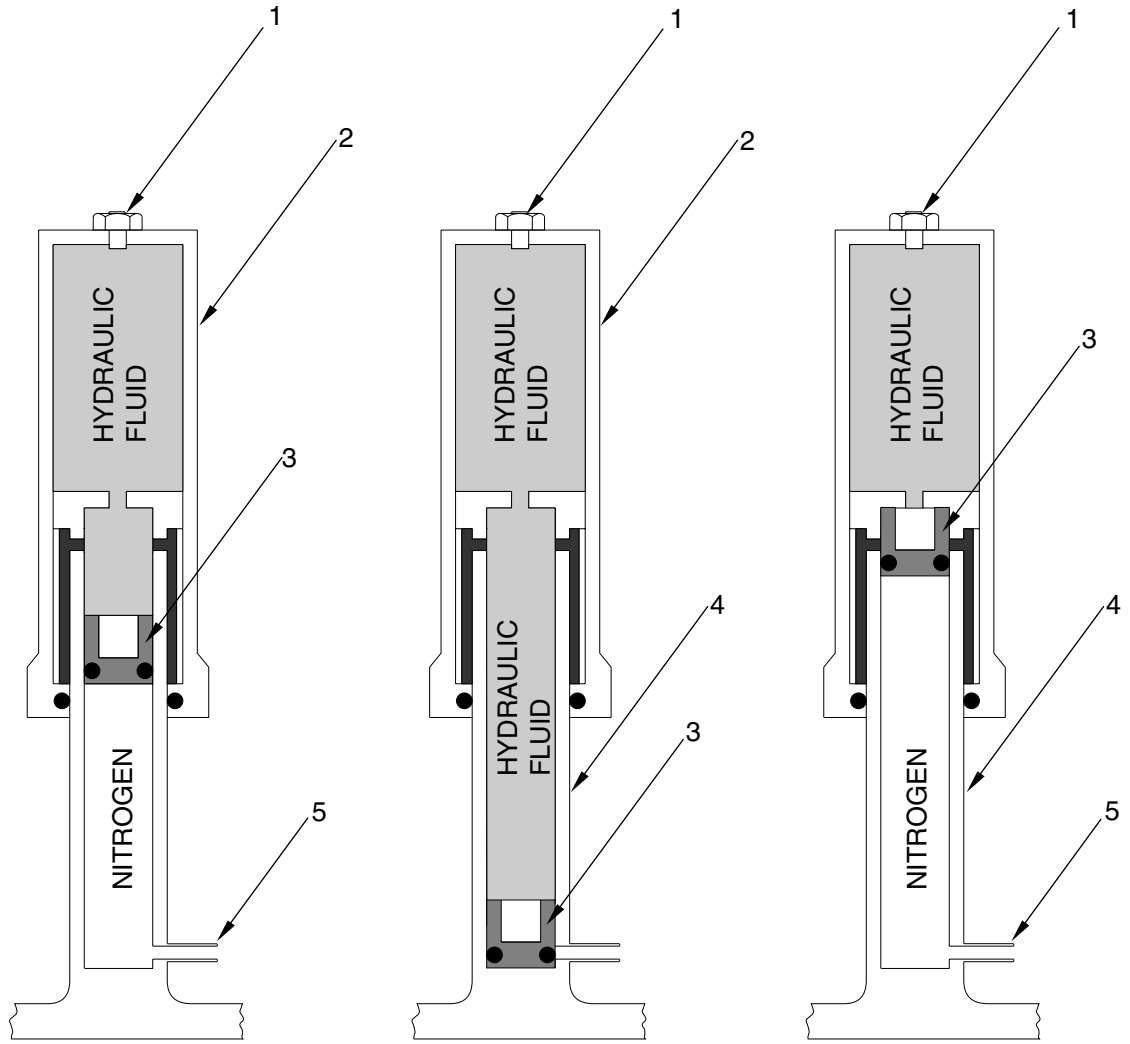


Landing Gear Strut
Figure 201

B. Fluid Servicing – Landing Gear (Figure 202)

NOTE: This procedure may be performed in one of two methods. Method 1 (Fluid Serving with Hydraulic Hand Pump) is preferred. If no pump is available use Method 2 (Servicing without Hydraulic Hand Pump).

Service landing gear strut with MIL-H-5606 hydraulic fluid ONLY.



Landing Gear Strut Fluid Servicing Figure 202

(1) Method 1 (Fluid Servicing with Hydraulic Hand Pump).

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING OFF ELECTRICAL POWER.

(a) Jack aircraft.

(b) Remove pots IAW Chapter 32.

WARNING: NITROGEN CHARGE IN STRUT IS UNDER HIGH PRESSURE. USE CAUTION WHEN BLEEDING OFF NITROGEN.

(c) Slowly open nitrogen filler valve (item 5) to release nitrogen in strut.

(d) Remove both hydraulic filler plugs, (use as fill and vent), (item 1), at top of cylinder (item 2).

(e) Insert hose into hydraulic filler hole (item 1). Connect opposite end of hose to hand hydraulic pump.

(f) Pump in hydraulic fluid (MIL-H-5606 only) until piston (item 3) moves to bottom of strut (item 4). Do not exceed 200 psi (14.06 kg/cm^2).

(g) Connect nitrogen to the filler valve (item 5) and apply 3 to 10 psi (0.211 to 0.703 kg/cm^2) nitrogen (air) until piston (item 3) travels to the top of the strut (item 4). Verify piston stops when fluid flow stops.

(h) Watch excess hydraulic fluid for air entrapment (foam in hydraulic fluid).

(i) Repeat steps **1.**, B., (1). (f) and (g) until fluid is free of foam.

(j) Bleed nitrogen pressure from the filler valve (item 5) and repeat step **1.**, B., (1). (f).

(k) Compress gear to discharge excess fluid. Install hydraulic filler plugs (item 1). Torque 130 to 180 inch-pounds (14.69 to 21.34 N·m) and lockwire.

(l) Connect nitrogen to the filler valve (item 5) and inflate strut.

(m) With aircraft sitting on gear, inflate strut to achieve 2.0 inch (5.08 cm) strut extension.

(n) Close nitrogen filler valve (item 5). Torque 50 to 80 inch-pounds (5.65 to 9.04 N·m).

(o) Install nitrogen filler valve dust cap.

(p) Install pots IAW Chapter 32.

(2) Method 2 (Fluid Servicing without Hydraulic Hand Pump).

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING OFF ELECTRICAL POWER.

(a) Jack aircraft.

(b) Remove pots IAW Chapter 32.

WARNING: NITROGEN CHARGE IN STRUT IS UNDER HIGH PRESSURE. USE CAUTION WHEN BLEEDING OFF NITROGEN.

(c) Slowly open nitrogen filler valve (item 5) to release nitrogen in strut.

(d) With hydraulic filler plug (item 1) installed, compress gear, forcing piston against hydraulic fluid.

(e) Remove both hydraulic filler plugs, (use as fill and vent), extend gear and fill with clean hydraulic fluid (MIL-H-5606 only).

(f) Repeat steps **1.**, A., (2), (c) and (d) until fluid is free of air (no foam).

(g) Compress gear to discharge excess hydraulic fluid.

(h) Install hydraulic filler plug (item 1). Torque 130 to 180 inch-pounds (14.69 to 20.34 N·m) and lockwire.

(i) With aircraft sitting on gear, inflate strut to achieve 2.0 inch (5.08 cm) of strut extension.

(j) Close nitrogen filler valve (item 5) and torque 50 to 80 inch-pounds (5.65 to 9.04 N·m).

(k) Install nitrogen filler valve dust cap.

(l) Install pots IAW Chapter 32.

BRAKE FLUID REPLENISHING – MAINTENANCE PRACTICES

1. Servicing – Brake Fluid Replenishing

NOTE: SA226 series aircraft have two brake systems: the standard brake system and the anti-skid brake system. The anti-skid system is a customer option. Separate maintenance procedures apply to each system.

A. Brake Fluid – Anti-Skid Brake System

(1) To service the anti-skid brake system, refer to Chapter 32.

B. Brake Fluid – Standard Brake System

(1) The standard brake system reservoir capacity is one (1) U.S. pint (0.473 liter). The type of fluid used is MIL-H-83282 hydraulic fluid.

(a) Before each flight, remove filler cap located on left side of aircraft, aft of nose baggage door.

(b) Verify fluid is level with bottom of filler cap port. Add fluid as necessary to maintain this level.

(c) For further servicing of the standard brake system, refer to chapter 32.

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PASSENGER CONVENIENCES REPLENISHING – MAINTENANCE PRACTICES

1. Servicing – Passenger Conveniences Replenishing

A. Lavatory Compartment Sink – Replenishment

(1) Depress release button located outboard of mirror. Swing door open.

(2) Disconnect hose at bottom of stainless steel container, using quick-disconnect attachment at end of hose.

(3) Remove container from shelf.

(4) Replenish lavatory sink.

(5) Replace container.

(6) Attach hose.

B. Refreshment Bar/Galley – Replenishment

(1) The standard refreshment bar/galley is located on the left side of the passenger compartment, immediately behind the flight deck, and forward of the first seat.

(2) The water dispenser reservoir is replenished in a manner similar to the dispenser reservoir found in the lavatory.

(3) Replenish the hot/cold liquid refreshment/reservoir as follows:

(a) Release latch.

(b) Lower door.

(c) Turn fluid dispenser knob clockwise, as far as it will go. This will lock dispenser button.

(d) Remove reservoir.

(e) Unlatch lid.

(f) Refill reservoir.

(g) Close and secure lid.

(h) Replace reservoir.

(i) Turn dispenser knob all the way counterclockwise and close door.

NOTE: Door will not close unless knob is turned.

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OXYGEN SYSTEM REPLENISHING – MAINTENANCE PRACTICES

1. Servicing – Oxygen System Replenishing

A. Replenishment – Oxygen Cylinder

WARNING: OIL OR GREASE, WHEN PUT IN CONTACT WITH PURE OXYGEN, WILL IGNITE. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN WHEN PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

CAUTION: IF OXYGEN IS ALLOWED TO FLOW INTO THE CYLINDER TOO FAST, EXCESSIVE HEATING OF THE CYLINDER WILL OCCUR, CAUSING EXPANSION OF THE OXYGEN IN THE CYLINDER. THIS WILL RESULT IN A HIGHER PRESSURE INDICATION THAN ACTUALLY EXISTS.

(1) The filler valve is located inside the nose baggage compartment next to the forward upper corner of the R/H door frame.

(2) Remove the dust cover from the oxygen filler fitting. Connect servicing cylinder to filler fitting (pure breathing oxygen only). Tighten fitting enough to prevent leakage.

(3) Open the servicing cylinder valve slowly until oxygen is heard going into the aircraft cylinder.

NOTE: When an oxygen cylinder has been completely depressurized, condensation may form in the cylinder unless recharged within two (2) hours. If the cylinder **IS NOT** charged within two (2) hours, it must be purged of any condensation prior to servicing, in accordance with FAA Advisory Circular AC 43.13 -1B Chg. 1 or latest FAA approved revision.

The new composite cylinders are DOT approved by exemption DOT-E8162. The service pressure is 1850 psi. At present the approved service life is twenty four (24) years, or after 4,380 pressurization cycles. Hydrostatic test for new composite cylinders is every three (3) years.

(4) When the aircraft oxygen cylinder is fully charged to 1850 psi as indicated on the cockpit pressure gauge, close the servicing cylinder valve.

(5) Remove servicing cylinder from the oxygen filler fitting. Replace the fitting dust cover and secure.

NOTE: If time and circumstances permit, allow the cylinder to cool down for thirty (30) minutes to one (1) hour and recheck the cylinder pressure. If the pressure hasdropped appreciably, repeat steps **1.**, A., (2) through (5).

B. Bleeding Oxygen Cylinder – Cylinder Installed In Aircraft

WARNING: OIL OR GREASE, WHEN PUT IN CONTACT WITH PURE OXYGEN, WILL IGNITE. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN WHEN PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(1) Turn regulator control valve OFF.

(2) Bleed trapped oxygen pressure from supply line by inserting an oxygen mask fitting into an outlet and breathing through the mask.

NOTE: Complete discharge of the oxygen cylinder is not necessary except to remove/install the oxygen cylinder regulator.

(3) Disconnect supply line from oxygen regulator.

(4) Connect a hose to the supply line port on the regulator. Hose should be clean and long enough to reach four (4) to five (5) feet outside the aircraft nose or aft compartment.

(5) Securely grasp the free end of the hose and have an assistant slowly open the regulator control valve.

(6) When the cylinder is empty, remove the bleed hose from regulator control valve.

(7) Securely close regulator control valve and reconnect supply line to the regulator.

ALCOHOL/WATER REPLENISHING – MAINTENANCE PRACTICES

1. Servicing – Alcohol/Water Replenishing

A. Replenishment – AWI Storage Tank

(1) Locate AWI tank filler.

(a) On early models, the cap is located under the right aft wing/fuselage fairing.

(b) On some models, the cap is located just inside the right nose baggage door.

(c) On later models with 16 U.S. gallon tanks, the cap is on the right side of nose section, just forward of the right nose baggage door.

(2) On early models, open fairing access door, and remove cap slowly. Ensure that filler screen inside tank is in place. Fill tank to overflowing.

(3) On those aircraft equipped with 14 or 16 gallon tanks, fill tank to desired level using the sight gauge to determine quantity. If desired, the tank may be filled to overflowing.

(4) Replace filler cap and close fairing door.

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SCHEDULED SERVICING – DESCRIPTION AND OPERATION

1. General

A. This section contains lubrications charts, illustrations, and instructions for periodic lubrication and cleaning of the aircraft.

2. Lubrication (12-20-10)

A. This section contains lubrication charts, illustrations, and instructions for periodic lubrication, lubricant type and method of application.

3. Cleaning (12-20-15)

A. This section covers the external and internal cleaning, including the engines, fuselage, wings, empennage, chemical and electric toilets, deice boots, and interior windows.

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LUBRICATION – MAINTENANCE PRACTICES

1. Servicing – Lubrication

A. Lubricants:

(1) The following is a list of recommended lubricants:

MIL-PRF-81322 - Grease, General Purpose, Wide Temperature Range.

MIL-PRF-23827 - Grease, General Purpose For Operations in Extreme Cold Conditions.

MIL-PRF-83261 - Grease, General Purpose For Operations in Extreme Cold Conditions.

MIL-PRF-83282 - Hydraulic Fluid, Fire Resistant, Synthetic Hydrocarbon Base.

MIL-PRF-21164 - Grease, Aircraft and Instrument, Gear and Actuator Screw.

MIL-PRF-5606 - Hydraulic Fluid, General Purpose, Low Temperature.

MIL-PRF-7870 - Lubricating Oil, General Purpose, Low Temperature.

Tri-Flow Aerosol - Click-Clack Lubricant.

903 DRI-SLIDE - Liquid Hydrocarbon Carrier, Click-Clack Lubricant.

MIL-L-23398 - Type II (Aerosol Propelled), Lubricant, Solid Film, Air - Cured, Corrosion Inhibiting.

NOTE: A letter at the end of a MIL specification denotes the revision level of that specification. Lubricants which meet any revision of the above specifications are permitted.

B. Lubrication Procedures

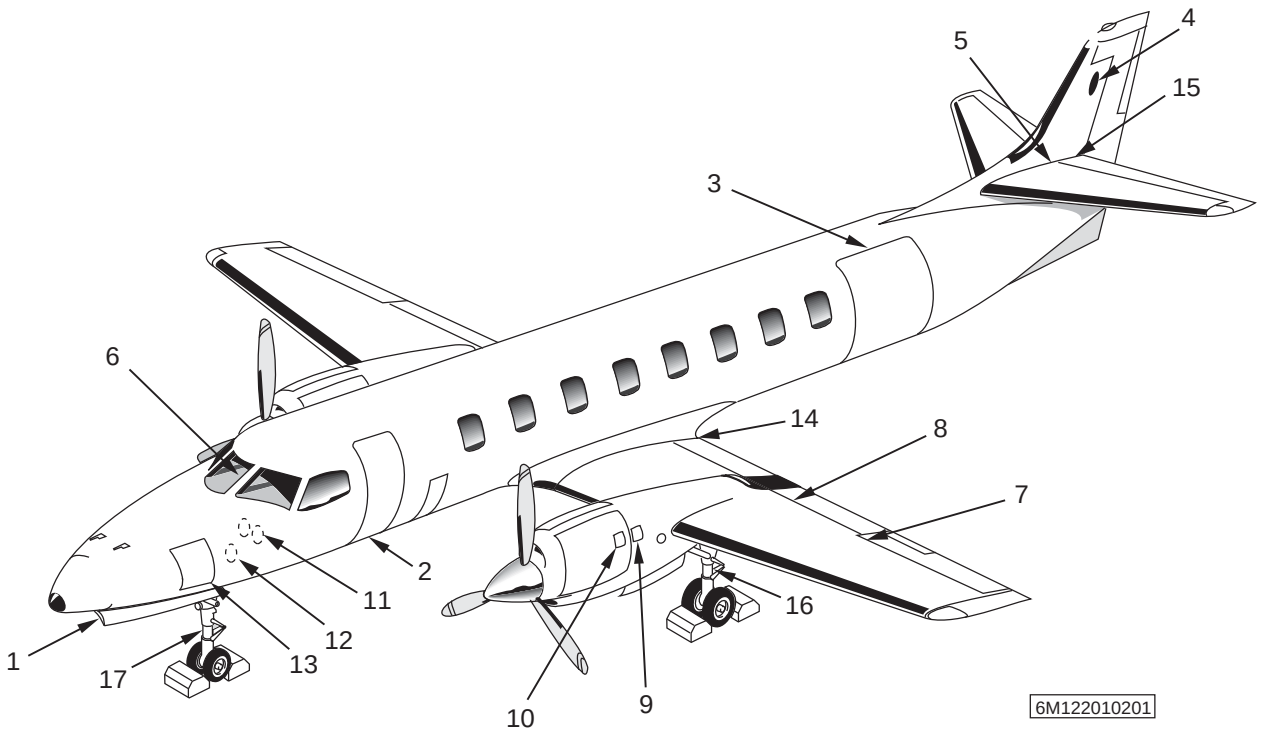
(1) Fill gun or can with lubricant specified in Tables 201 through 205.

(2) Lubricate points specified in Figures 201 through 205.

(3) Clean off excess lubricant with clean cloth.

(4) Wheel Bearings - See Section 32-42 for main wheel and nose wheel bearing lubrication procedures.

C. Lubrication Charts: Refer to the following pages for lubricant to be used at each point shown and the recommended application method.



Airframe Lubrication Locations
Figure 201 (See Table 201)

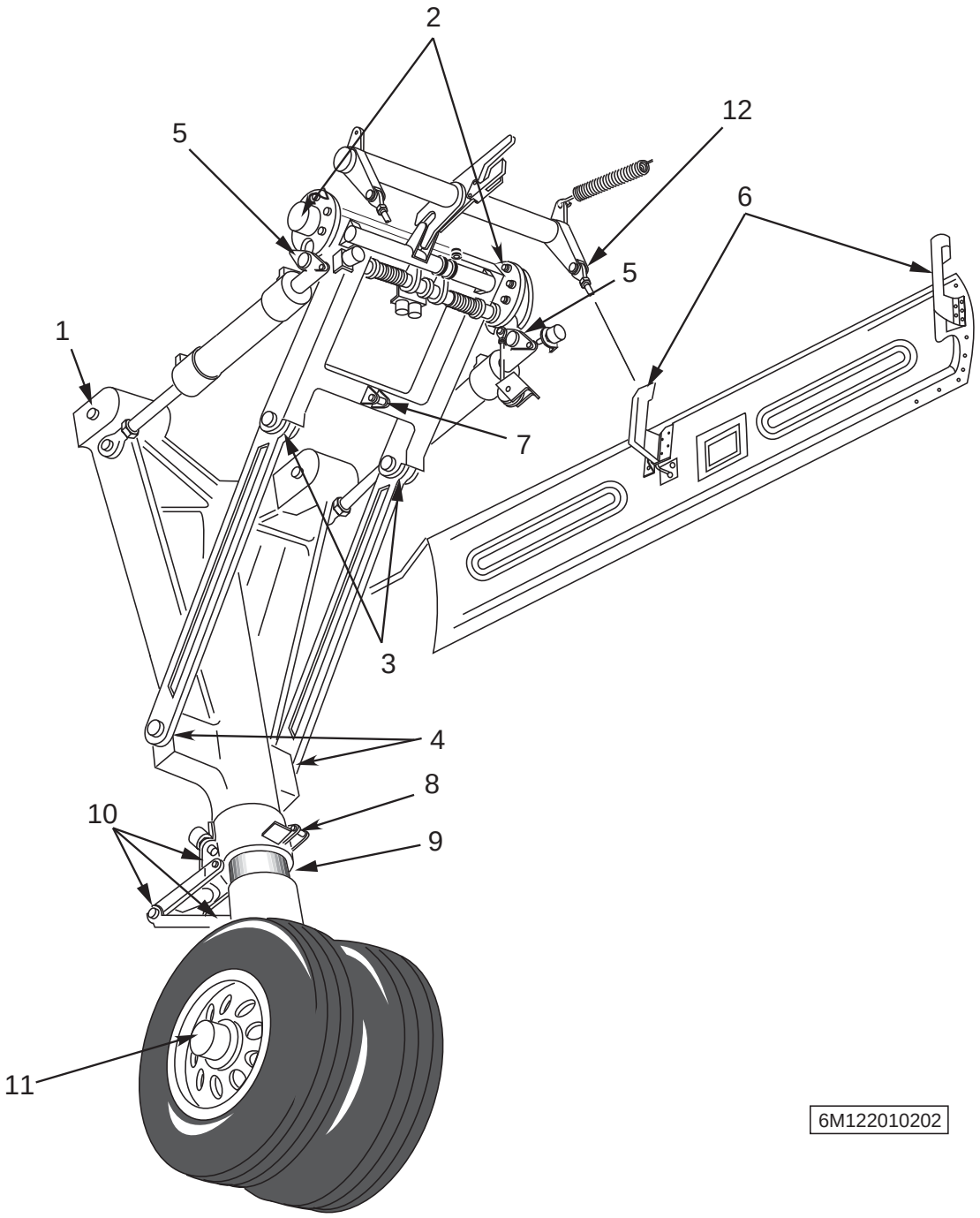
ITEM NO.	LOCATION	LUBRICATION*	NO. OF PLACES	APPLICATION METHOD
1	Nose Gear Door Hinges	MIL-PRF-7870	4	Oil Can
2	Cabin Door Hinge	Tri-Flow	1	Hand
	Cabin Door Click-Clacks, Bayonets and Guide Pins	Tri-Flow, Dri-Slide or MIL-L-23398	9	Hand
	Cabin Door Latch Bellcranks	MIL-PRF-7870	6	Oil Can
3	Cargo Door Hinge	Tri-Flow	1	Hand
	Cargo Door Click-Clacks and Guide Pins	Tri-Flow, Dri-Slide or MIL-L-23398	10	Hand
	Cargo Door Latch Bellcranks	MIL-PRF-7870	6	Oil Can
4	Rudder Trim Actuator Chain	MIL-PRF-81322**	1	Hand
5	Horizontal Stabilizer Trunnions	MIL-PRF-81322**	2	Gun
6	Aileron Control Column Chains (Refer to SB 226-27-074)	MIL-PRF-81322**	2	Hand
7	Aileron Trim Actuator and Chain	MIL-PRF-81322**	2	Gun/Hand
8	Flap Actuator	MIL-PRF-81322**	4	Gun
9	Hydraulic Fluid	MIL-PRF-83282	1	Filler Can
10	Engine Oil	MIL-L-23699 (See 12-10-15, Table 201)	2	Filler Can
11	Rudder Pedals	MIL-PRF-81322**	4	Gun
12	Brake Fluid	MIL-PRF-83282	1	Filler Can
13	Nose Baggage Door Hinges	MIL-PRF-7870	4	Oil Can
	Nose Baggage Door Latches	Tri-Flow, Dri-Slide or MIL-L-23398	4	Hand
14	Flap Walking Beam OTBD Bearings	MIL-PRF-81322**	2	Hand
15	Elevator Control Link Rod Bearings (Refer to SB 226-27-080)	MIL-PRF-81322**	4	Hand
16	Main Landing Gear (See Figure/Table 202 and 203 this section)			
17	Nose Landing Gear (See Figure/Table 204 and 205 this section)			

* A letter at the end of a MIL specification denotes the revision level of that specification. Lubricants which meet any revision of the above specifications are permitted.

** For operations in extreme cold conditions, MIL-PRF-83261 and/or MIL-PRF-23827*** may be substituted for MIL-PRF-81322 provided they are not mixed and lubrication point is purged of one type before using the other.

*** MIL-PRF-23827 Type I or Type II is acceptable to use provided they are not mixed and lubrication point is purged of one type before using the other.

Airframe Lubrication Locations
Table 201 (See Figure 201)



Main Landing Gear Lubrication Locations
Figure 202 (See Table 202)

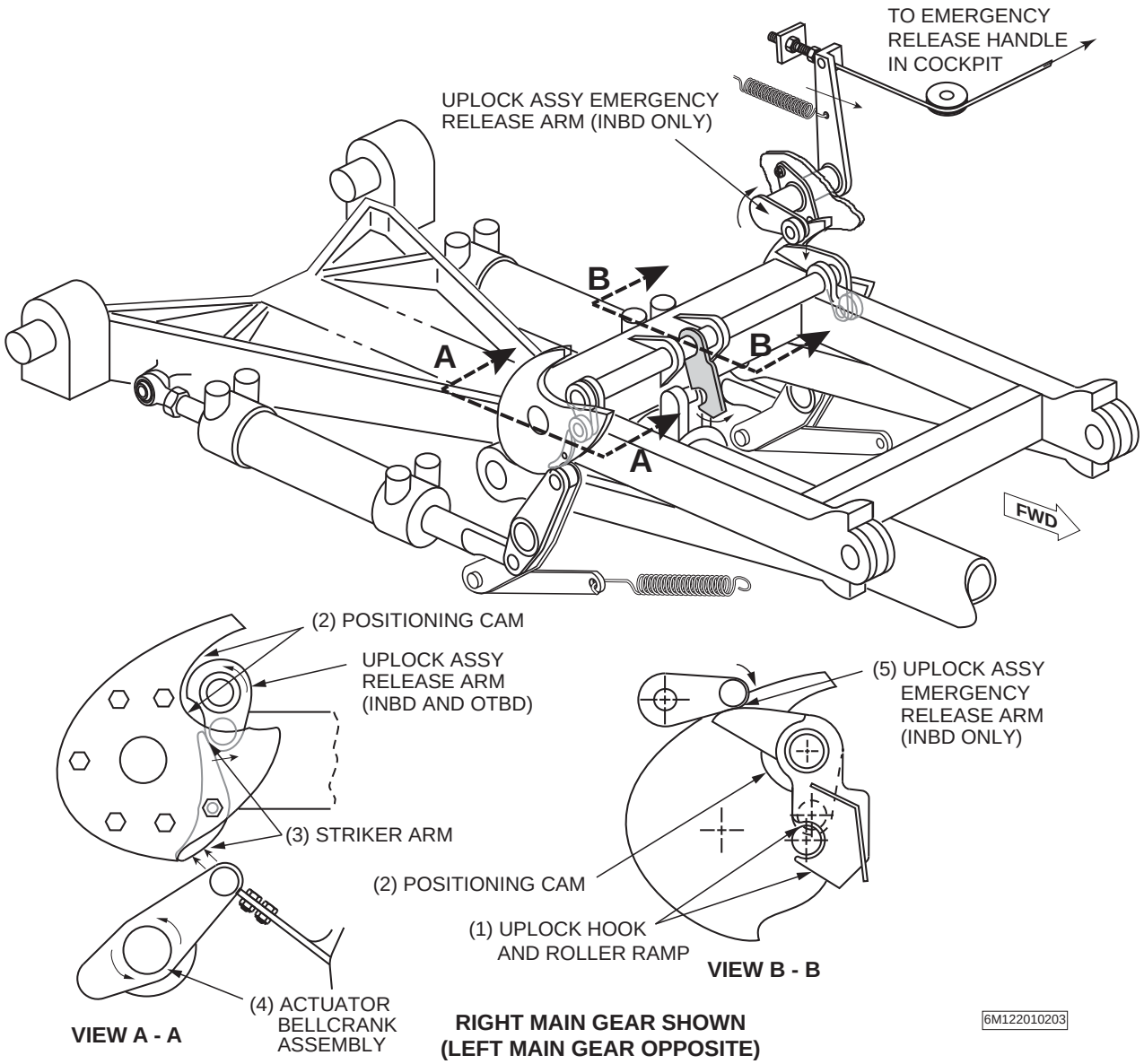
ITEM NO.	LOCATION	LUBRICATION*	NO. OF PLACES (per gear)	APPLICATION METHOD
1	Main Gear Strut Trunnions	MIL-PRF-81322**	2	Gun
2	Drag Brace Upper Trunnions	MIL-PRF-81322**	2	Gun
3	Drag Brace Center Joints	MIL-PRF-81322**	2	Gun
4	Drag Brace Lower Trunnions	MIL-PRF-81322**	2	Gun
5	Actuator Bellcrank Assemblies	MIL-PRF-81322**	2	Gun
6	Door Hinges	MIL-PRF-7870	4	Oil Can
7	Door Roller	MIL-PRF-81322**	1	Hand
8	Gear Uplock Roller	MIL-PRF-81322**	1	Hand
9	Shock Strut Oleo	MIL-PRF-5606	1	Hand
10	Torque Links	MIL-PRF-81322**	5	Gun
11	Wheel Bearings	MIL-PRF-81322**	4	Hand
12	Door Actuator Linkage	MIL-PRF-7870	4	Oil Can

* A letter at the end of a MIL specification denotes the revision level of that specification. Lubricants which meet any revision of the above specifications are permitted.

** For operations in extreme cold conditions, MIL-PRF-83261 and/or MIL-PRF-23827*** may be substituted for MIL-PRF-81322 provided they are not mixed and lubrication point is purged of one type before using the other.

*** MIL-PRF-23827 Type I or Type II is acceptable to use provided they are not mixed and lubrication point is purged of one type before using the other.

Main Landing Gear Lubrication Locations (LH and RH Same)
Table 202 (See Figure 202)



Main Landing Gear Uplock Mechanism Lubrication Locations
Figure 203 (See Table 203)

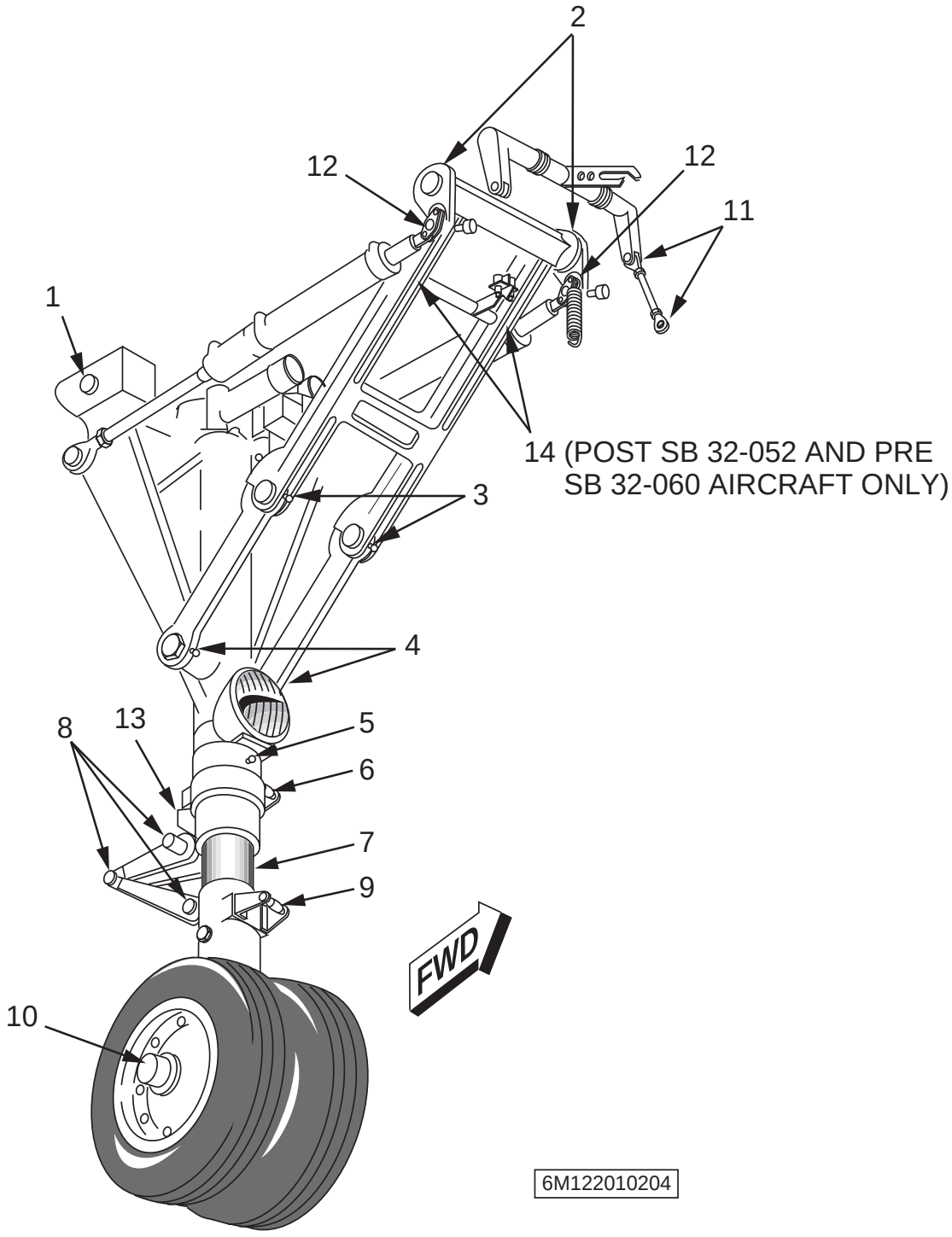
ITEM NO.	LOCATION	LUBRICATION*	NO. OF PLACES (per gear)	APPLICATION METHOD
1	Uplock Hook and Roller Ramp	MIL-G-21164	1	Hand
2	Positioning Cam	MIL-G-21164	2	Hand
3	Striker Arm	MIL-G-21164	2	Hand
4	Actuator Bellcrank Assembly	MIL-PRF-81322**	2	Gun
5	Uplock Assembly Emergency Release Arm	MIL-G-21164	1	Hand

* A letter at the end of a MIL specification denotes the revision level of that specification. Lubricants which meet any revision of the above specifications are permitted.

** For operations in extreme cold conditions, MIL-PRF-83261 and/or MIL-PRF-23827*** may be substituted for MIL-PRF-81322 provided they are not mixed and lubrication point is purged of one type before using the other.

*** MIL-PRF-23827 Type I or Type II is acceptable to use provided they are not mixed and lubrication point is purged of one type before using the other.

Main Landing Gear Uplock Mechanism Lubrication Locations
Table 203 (See Figure 203)



Nose Landing Gear Lubrication Locations
Figure 204 (See Table 204)

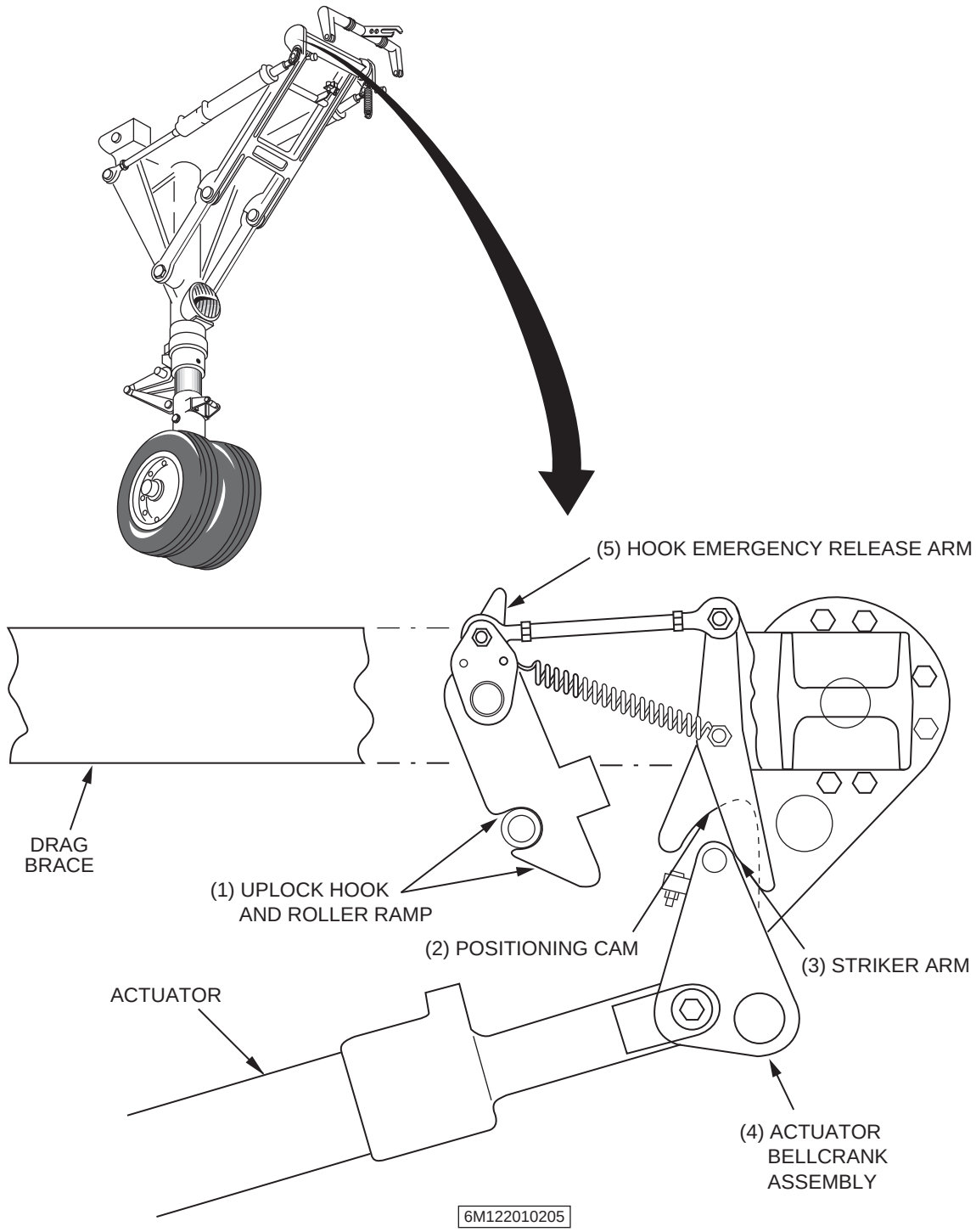
ITEM NO.	LOCATION	LUBRICATION*	NO. OF PLACES (per gear)	APPLICATION METHOD
1	Nose Gear Strut Trunnions	MIL-PRF-81322**	2	Gun
2	Drag Brace Upper Trunnions	MIL-PRF-81322**	2	Gun
3	Drag Brace Center Joints	MIL-PRF-81322**	2	Gun
4	Drag Brace Lower Trunnions	MIL-PRF-81322**	2	Gun
5	Strut Housing Lower Bearing	MIL-PRF-81322**	1	Gun
6	Uplock Roller	MIL-PRF-81322**	1	Hand
7	Shock Strut Oleo	MIL-PRF-5606	1	Hand
8	Torque Links	MIL-PRF-81322**	5	Gun
9	Door Roller	MIL-PRF-81322**	1	Hand
10	Wheel Bearings	MIL-PRF-81322**	4	Hand
11	Door Actuator Linkage	MIL-PRF-7870	4	Oil Can
12	Actuator Bellcrank Assemblies	MIL-PRF-81322**	2	Gun
13	Nose Gear Centering Arm Roller	MIL-PRF-81322**	1	Gun
14	Uplock Hook Assy. Torque Tube (Post SB 32-052 and Pre SB 32-060 aircraft only)	MIL-PRF-81322**	2	Gun

* A letter at the end of a MIL specification denotes the revision level of that specification. Lubricants which meet any revision of the above specifications are permitted.

** For operations in extreme cold conditions, MIL-PRF-83261 and/or MIL-PRF-23827*** may be substituted for MIL-PRF-81322 provided they are not mixed and lubrication point is purged of one type before using the other.

*** MIL-PRF-23827 Type I or Type II is acceptable to use provided they are not mixed and lubrication point is purged of one type before using the other.

Nose Landing Gear Lubrication Locations
Table 204 (See Figure 204)



Nose Landing Gear Uplock Mechanism Lubrication Locations
Figure 205 (See Table 205)

ITEM NO.	LOCATION	LUBRICATION*	NO. OF PLACES (per gear)	APPLICATION METHOD
1	Uplock Hook and Roller Ramp	MIL-G-21164	1	Hand
2	Positioning Cam	MIL-G-21164	2	Hand
3	Striker Arm	MIL-G-21164	2	Hand
4	Bellcrank Pivot Assembly	MIL-PRF-81322**	2	Gun
5	Hook Emergency Release Arm	MIL-G-21164	1	Hand

* A letter at the end of a MIL specification denotes the revision level of that specification. Lubricants which meet any revision of the above specifications are permitted.

** For operations in extreme cold conditions, MIL-PRF-83261 and/or MIL-PRF-23827*** may be substituted for MIL-PRF-81322 provided they are not mixed and lubrication point is purged of one type before using the other.

*** MIL-PRF-23827 Type I or Type II is acceptable to use provided they are not mixed and lubrication point is purged of one type before using the other.

Nose Landing Gear Uplock Mechanism Lubrication Locations
Table 205 (See Figure 205)

1. Servicing – Cleaning

A. Engine – Cleaning

WARNING: ALWAYS WEAR SAFETY GLASSES WHEN USING COMPRESSED AIR.

CAUTION: COVER ENGINE MOUNT ASSEMBLIES TO PROTECT THEM FROM CONTAMINATION BY CLEANING FLUIDS OR SOLVENT. SHOULD ANY SOLVENT GET ON RUBBER PORTION OF ENGINE MOUNTS, WIPE OFF WITH A CLEAN CLOTH.

(1) Apply a dry cleaning solvent to the engine in an non-atomizing spray until clean. It may be necessary to vigorously brush areas where heavy grease and dirt deposits have collected in order to thoroughly clean them.

CAUTION: KEEP CLEANING TO A MINIMUM IN THE VICINITY OF ELECTRICAL EQUIPMENT. ELECTRICAL COMPONENTS ARE EASILY DAMAGED.

(2) Allow solvent to remain on engine from five to ten minutes, then rinse engine clean with additional solvent and wipe engine dry with a clean cloth.

B. Fuselage, Wing, and Empennage Cleaning

(1) Use a standard non-abrasive soap and water solution. Avoid entry of solutions and/or flooding of soap or water into scoops or static vents. As a precaution, scoops should be plugged and vents taped.

NOTE: Do not flood the horizontal stabilizer trim opening.

C. Cleaning Deice Boots

CAUTION: DO NOT SCRUB BOOTS. SCRUBBING WILL REMOVE CONDUCTIVE COATING FROM SURFACE AREAS OF DEICE BOOTS.

(1) Deice boot surfaces should be checked for any oil spots. Oil can be removed by using a non-abrasive soap and water solution.

D. Cleaning Aircraft Interior

(1) Rugs, upholstery panels, and head linings should be frequently vacuumed to remove surface dust. Occasional shampooing, using a commercial foam-type cleaner, is acceptable. Clean and polish metal trim with any commercial metal cleaner/polish. Clean and polish cabinet, table, interior bulkhead, etc., and all laminated plastic surfaces (Formica) with any commercial furniture cleaner/polish.

E. Cleaning Windows

(1) Use any non-abrasive material on plastic windows. To prevent scratches, wash windows carefully with plenty of soap and water, using palm of hand to dislodge dirt and mud.

(2) Grease, stains, and insect can be removed using a good grade of plastic cleaner and polisher. Using a soft clean cloth and light pressure, apply a thin even coat. Let dry. Wipe off with an up and down motion only. Buff with a soft clean cloth.

F. Cleaning Chemical Toilet

(1) The chemical toilet is a standard dry chemical type. After each period of use, it should be emptied and cleaned by using a stiff bristle brush and a soap and water solution. Any approved commercial chemical may be used in the toilet.

G. Cleaning Electric Toilet

NOTE: Because of the chemicals involved and for sanitary reasons, it is a good idea to wear elbow length gloves when servicing the toilet.

(1) Remove back rest pad.

(2) With toilet seat cover raised and electrical power ON, activate toilet pump by inserting a coin, key or similar metal object into slot located on top right hand corner of raised cover. Maintain contact for several seconds until flushing liquid had been pumped into washer container.

NOTE: Depending upon toilet usage, the flushing liquid needs changing only occasionally, (usually once every five to ten times that the waste container is serviced). If flushing liquid appears to have correct chemical balance and has been changed recently, it does not need changing.

(3) Lift strap at rear of toilet seat cover and release friction lock.

(4) Pull seat cover forward to full open and rest seat cover on retaining strap.

(5) Remove round cover from bracket and snap into place on waste container.

(6) Using strap handle, lift waste container out and dispose of content. Rinse waste container.

(7) Clean toilet inside and out with a disinfectant (Lysol or similar product) each time it is serviced.

(8) Mix disinfectant and water according to instructions on label. Be sure to measure liquids carefully as too strong or too weak a mixture can result in an offensive odor.

(9) Replace container by firmly pressing down at corners to ensure proper seal.

(10) Remove cover and stow in bracket.

(11) Replenish flushing reservoir by adding proper mixture of water and chemical until it reaches FILL TO HERE line inside tank.

(12) Close seat cover and press fastener into place.

(13) With aircraft electrical power ON and cover CLOSE, check for proper flushing cycle (eight to fourteen seconds).

MAINTENANCE MANUAL

UNSCHEDULED SERVICING – DESCRIPTION AND OPERATION

1. General

A. This section includes instructions necessary to carry out unscheduled servicing, such as removal of ice and snow and procedures for defueling an aircraft.

B. For engine preservation and depreservation, see GTEC Maintenance Manual, Chapter 72.

2. Ice Removal (12-30-05)

A. This section covers precautions and procedures for removing ice from aircraft.

3. Snow Removal (12-30-10)

A. This section includes precautions and procedures for removing snow from aircraft.

4. Defueling (12-30-15)

A. This section includes procedures for defueling an aircraft using the quick release drain.

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ICE REMOVAL – MAINTENANCE PRACTICES

1. Servicing – Ice Removal

A. Observe the following precautions during ice removal:

(1) Do not use tools to scrape or chip ice from surfaces or attempt to loosen ice by beating; damage is certain to result.

(2) Avoid temperatures in excess of 200°F (93.3°C) when heating aircraft surfaces. Apply externally ducted heat carefully to window areas. High temperatures on cold windows will crack or craze windows.

(3) Do not use isopropyl alcohol on windows or above window line on fuselage as alcohol causes crazing and cracking of acrylic windows.

(4) In all applications of heat for deicing purposes, care must be exercised to avoid overheat damage to painted surfaces, rubber hoses, acrylic or fiberglass, landing gear and wheel well areas, hydraulic lines, fabrics, and lubricated surfaces. Remove heat source immediately after surfaces are dry, or mechanisms are functioning normally.

(5) If snow, slush, or ice are suspected in seal or control areas forward of control surfaces, aircraft should not be cleared for flight until noted areas and all control components therein are completely clear and dry.

B. Apply heat for snow/ice removal in accordance with the following:

CAUTION: HAND ACTUATION OF ALL FLIGHT CONTROL SURFACES THROUGH FULL RANGE OF MOTION SHOULD BE ACCOMPLISHED PRIOR TO ANY ATTEMPT TO OPERATE COCKPIT CONTROLS. AFTER REMOVAL OF SNOW AND/OR ICE, PERFORM A COMPLETE OPERATIONAL CHECK.

(1) When icing conditions occur at ground level, aircraft must be checked thoroughly. Check for iced surfaces, frozen controls and mechanisms, plugged orifices, frozen landing gear components, and ice or snow clogged air inlet and outlet ports.

(2) Engine cowl inlet and exhaust areas must be checked for ice before attempting to start engines.

(3) Remove ice from fuselage areas by application of heat ducted to affected area from a large capacity heater unit.

NOTE: A heat source providing a large volume/area of warm dry air is more effective than a small volume of hot air, and can be used with less danger of overheating.

(4) Ice and frost formations can be removed from wing and empennage surfaces by spraying an application of heated deicing solution. Loosened ice should be swept from surfaces immediately. Residual water should be dried to prevent refreezing.

(5) Ice formations on wing, empennage, landing gear, or wheel well areas can be removed by heat from large capacity heater units.

CAUTION: AFTER DEICING WING AND EMPENNAGE, CHECK BALANCE BAYS AND REMOVE ANY SLUSH AND/OR ICE ACCUMULATIONS. ENSURE BAY DRAIN HOLES ARE NOT PLUGGED.

(6) Ice formation(s) in control areas can also be removed by using warm air until ice is melted and area is dry. Ensure drain holes in panels below control areas are open.

(7) To prevent ice formation in cargo door side or lower jamb areas, apply periodic applications of Fluorocarbon Teflon MS-122 spray or Frekote No. 33 spray.

(8) After completion of ice removal, a complete operational check of all flight controls is mandatory.

C. Anti-Icing – Procedures

NOTE: Anti-icing is defined as the preventative treatment of aircraft surfaces to keep them clear of ice and/or snow accumulations.

CAUTION: NEVER SPRAY FLUID OR WATER ON OR INTO PITOT TUBES, STATIC PORTS, ENGINE AIR INLETS, CABIN INTAKES, OR AGAINST TRAILING EDGE OF WINGS OR CONTROL SURFACES.

(1) Be sure that all windows and doors are closed.

(2) Keep fluid clear of the following items by avoiding direct spraying on or into:

(a) Engine air inlets.

(b) ECS air intakes.

(c) Fuel tank vents.

(d) Instrument heads and ports.

(e) Louvered openings in the fuselage tail cone.

(3) Be sure to spray the aircraft symmetrically, both wings and tailplanes on both sides.

(4) Wipe down nose of aircraft after spraying to ensure that any excess fluid will not flow back onto cockpit windows.

(5) Remove fluid from cockpit windows prior to departure.

(6) Do not use deicing fluid for anti-icing, as melting snow and ice will further dilute the solution and refreezing may take place on all surfaces.

(7) Maintain a ten (10) foot separation between de-icing fluid truck and aircraft.

(8) When operating fluid dispensing equipment for anti-icing extreme care must be exercised while operating the truck around the aircraft.

(9) Aircraft that have been hangared should be deiced, when removed from hanger and exposed to any precipitation, if skin temperature has not cooled to 32°F.

(10) Avoid spraying deicing fluids in the louvered area of the fuselage tail cone. If overspray impinges on pressurization outflow valve, valve may fail.

(11) Be careful not to spray over Static ports. These openings are not heated and in the event of refreezing, it may become clogged.

(12) Avoid spraying directly onto the SAS vane. Check SAS vane for freedom of operation prior to taxi.

D. Anti-Icing personnel protection precautions:

WARNING: ANTI-ICING FLUIDS ARE TOXIC. WASH THOROUGHLY BEFORE EATING OR DRINKING AFTER WORKING WITH THESE FLUIDS.

(1) Anti-icing and deicing fluids generally contain ethylene and/or propylene glycol. Ethylene and propylene glycol are considered toxic when taken orally.

(2) Eyes are moderately irritated by deicing fluids. In event fluid enters eyes, eyes should be washed immediately with clean, running water.

(3) Deicing fluids are not absorbed through the skin in harmful amounts, nor are they regular skin irritants.

(4) In cold weather, particular care should be taken to avoid splashing of fluids on the skin, as this increases the risk of frost bite.

(5) Position operator upwind of spray so that mist will not drift towards him/her whenever possible.

(6) Wear suitable protective clothing and gloves.

(7) If glycol spraying is performed indoors, area must be well ventilated.

E. For best results, apply anti-icing fluids in accordance with the following procedures:

WARNING: ANTI-ICING FLUIDS ARE TOXIC. WASH THOROUGHLY BEFORE EATING OR DRINKING AFTER WORKING WITH THESE FLUIDS.

(1) Anti-icing and deicing fluids generally contain ethylene and/or propylene glycol. Ethylene and propylene glycol are considered toxic when taken orally.

(2) Eyes are moderately irritated by deicing fluids. In event fluid enters eyes, eyes should be washed immediately with clean, running water.

(3) Deicing fluids are not absorbed through the skin in harmful amounts, nor are they regular skin irritants.

(4) In cold weather, particular care should be taken to avoid splashing of fluids on the skin, as this increases the risk of frost bite.

(5) Position operator upwind of spray so that mist will not drift towards him/her whenever possible.

(6) Wear suitable protective clothing and gloves.

(7) If glycol spraying is performed indoors, area must be well ventilated.

F. Approved Deicing Fluids

NOTE: Typically, Type I fluids are unthickened and contain a minimum of 80% glycol. Type I fluids are approved for use on CC, DC aircraft.

(1) TYPE I FLUIDS

(a) SAE and ISO Type I

(b) AMS 1425

(c) AMS 1427

(d) AEA Type I

(e) MIL-A-8243D Type I and Type II (MIL-A-8243D Type II is a Type I fluid specification).

(2) TYPE II FLUIDS

NOTE: Type II fluids are thickened and are **NOT APPROVED** for use on SA226 aircraft.

Type II fluids contain a minimum of only 50% glycol.

Type II fluids have a greater affect on lift and drag than Type I fluids.

G. Application guidelines for SAE Type I fluid mixtures

NOTE: Type I fluid is the only approved deice fluid.

Concentrations in % V/V			
OAT	One-step Procedure	Two-step Procedure	
	Deicing/Anti-icing	First step: Deicing	Second step: Anti-icing (1)
26.6°F (-3°C) and above	FP of heated fluid (2) mixture shall be at least 50°F (10°C) below OAT	Water heated to 140°F (60°C) minimum at the nozzle or a heated mix of fluid and water	FP of fluid mixture shall be at least 50°F (10°C) below actual OAT
below 26.6°F (-3°C)		FP of heated fluid mixture shall not be more than 26.6°F (-3°C) above OAT	

NOTE: For heated fluids, a fluid temperature not less than 140°F (60°C) at the nozzle is desirable. Upper temperature limit shall not exceed fluid and aircraft manufacturers recommendations.

CAUTION: WING SKIN TEMPERATURES MAY DIFFER AND IN SOME CASES MAY BE LOWER THAN OAT. A STRONGER MIX CAN BE USED UNDER THE LATTER CONDITIONS.

(1) To be applied before first step fluid freezes, typically within 3 minutes.

(2) Clean aircraft may be anti-iced with cold fluid.

Type I Deice Fluid Application
Table 1

H. Guidelines for Holdover Times Anticipated by SAE Type I and ISO Type I Fluid Mixtures as a Function of Weather Conditions and OAT.

NOTE: Freezing point of Type I fluid mixture used must be at least 10°C (18°F) below OAT.

CAUTION: THE FOLLOWING TABLE IS FOR USE IN DEPARTURE PLANNING ONLY AND IT SHOULD BE USED IN CONJUNCTION WITH PRETAKEOFF CHECK PROCEDURES.

OAT		Approximate Holdover Times Anticipated Under Various Weather Conditions (HRS:MIN)				
Degrees C	Degrees F	Frost	Freezing Fog	Snow	Freezing Rain	Rain On Cold Soaked Wing
0 & above	32 & above	0:18 to 0:45	0:12 to 0:30	0:06 to 0:15	0:02 to 0:05	0:06 to 0:15
below 0 to	below 32 to	0:18 to 0:45	0:06 to 0:15	0:06 to 0:15	0:01 to 0:03	CAUTION! ice may require touch confirmation
below -7	below 19	0:12 to 0:30	0:16 to 0:15	0:16 to 0:15		

NOTE: This Table does not apply to other than SAE or ISO Type I FPD Fluids.

The responsibility for the application of these data remains with the user.

Holdover Times Guidelines
Table 2

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SNOW REMOVAL – MAINTENANCE PRACTICES

1. Servicing – Snow Removal

A. Observe the following precautions during snow removal operations:

(1) Work from support stands placed near aircraft when sweeping snow from wing or stabilizer.

(2) Wear rubber or fabric footwear to aid in preventing personnel from slipping and/or sliding off aircraft surfaces. Use safety harness when available.

(3) Use only approved solutions for snow/ice removal and cleaning on acrylic windows.

(4) Remove all snow accumulations on fuselage nose forward of windshield; snow might blow back and stick to heated windshield restricting pilot's visibility during takeoff. Be careful not to damage pitot tubes when removing snow.

CAUTION: DO NOT USE SNOW REMOVAL TOOLS OR IMPLEMENTS TO REMOVE SNOW FROM TUBES, ANTENNAS, PROBES, OR ANY OTHER EXTERNAL COMPONENTS, WING OR EMPENNAGE CONTROL SURFACES, AND ANY FIXED OR MOVABLE SURFACES AND/OR COMPONENTS.

DO NOT ATTEMPT TO LOOSEN SNOW BY BEATING. DAMAGE IS CERTAIN TO RESULT.

(5) Avoid using deicing solutions for snow removal. Dilution of deicing solutions could result in a weak mixture refreezing and creating an icing condition more difficult to remove.

(6) When using deicing solutions for removal of iced snow or iced-over surfaces, or to prevent further accumulation on wings and control surfaces, avoid excessive flow. Use special care to prevent flushing of slush into control areas forward of control surfaces.

B. For best results, follow these snow removal procedures:

NOTE: Before heating aircraft interior, ensure that as much snow as possible is removed from upper fuselage area.

CAUTION: WHEN CLEANING SNOW FROM WINGS AND HORIZONTAL STABILIZER FIXED SURFACES, SWEEP SNOW OVER LEADING EDGE AND TIP. AVOID SWEEPING SNOW OVER CONTROL SURFACES ON TRAILING EDGES.

(1) With soft bristle brushes and brooms, remove as much residual or light snow as possible.

(2) By working a length of cotton rope back and forth over wing surfaces, snow accumulations can be removed from fuselage and wing upper surfaces.

(3) Pull snow accumulations over wing surface leading edges.

(4) Remove snow from engine cowls.

CAUTION: WHEN HEATING AIRCRAFT SURFACES AND/OR COMPONENTS, AVOID TEMPERATURES IN EXCESS OF 200°F (93.3°C).

(5) When using heated air to remove snow, continue heat application until surface/area is completely dry. Exercise care not to overheat structure or system components.

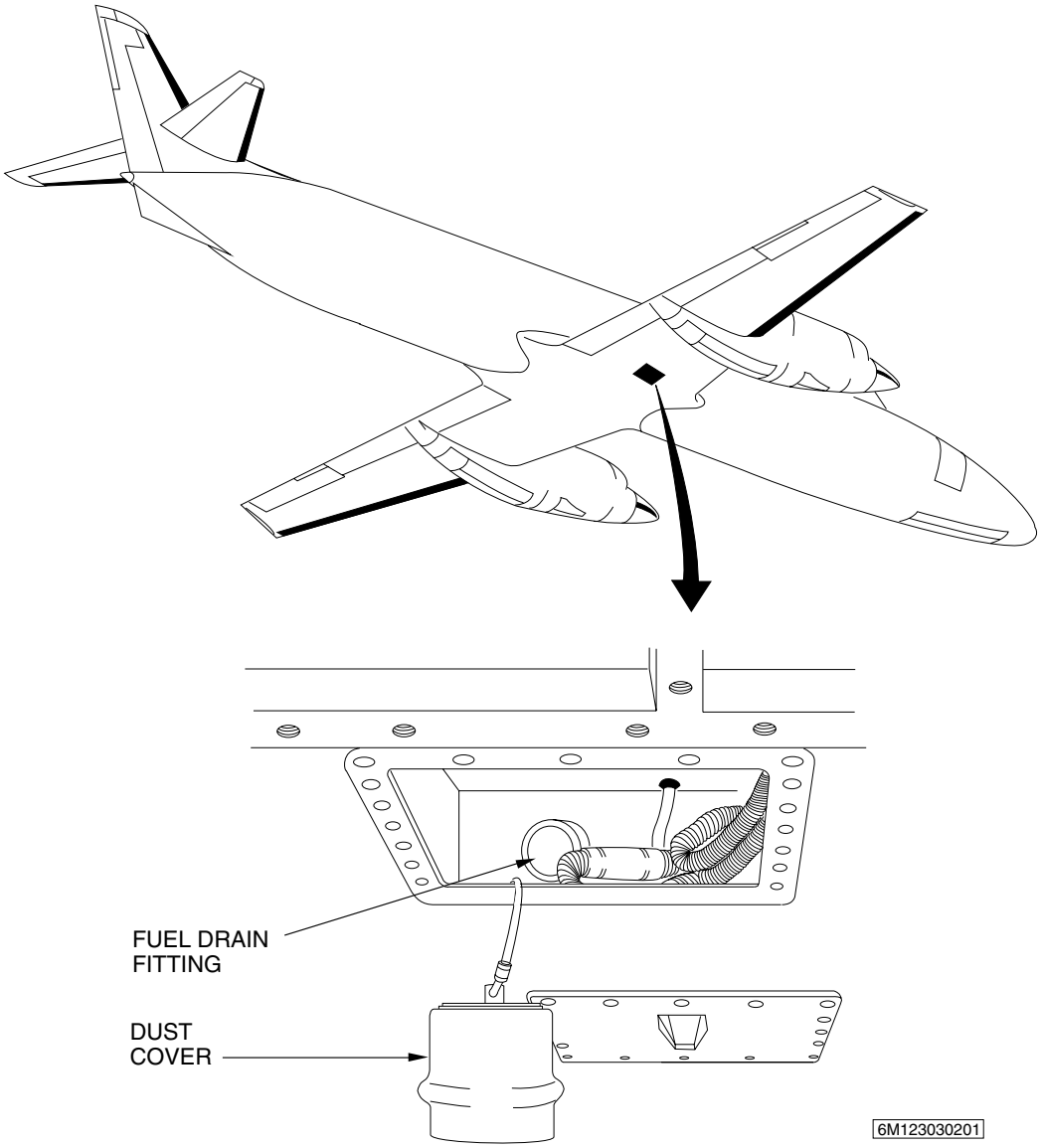
(6) Use a heat source providing a large volume of warm, dry air.

(7) Ensure that drain holes are open and drain freely.

(8) After completion of snow removal, a complete operational check of primary and secondary flight controls is mandatory.

(9) Ensure cargo floor at threshold is clean and dry. Remove any moisture that may freeze threshold guide/restraint assemblies.

(10) Ensure all passenger and service door jamb areas are clean and dry.



Fuel Drain
Figure 201

1. Servicing – Defueling Aircraft

A. Defueling Aircraft

(1) Position aircraft at least 50 feet from any potential source of ignition.

(2) Static ground aircraft; static ground defueling equipment to ground and to aircraft.

(3) Open crossflow valve.

(4) At center section, lower wing skin, remove access panel located just forward of the aft spar. Locate fuel drain fitting with dust cover (red nipple) attached.

(5) Remove dust cover by depressing the end cap and pulling straight out.

(6) Remove fuel filler caps.

(7) Install fuel drain tube assembly or comparable drain hose assembly.

(8) Defuel as required.

(9) After defueling operation, install fuel filler caps.

(10) Disconnect drain hose assembly and replace dust cover by pressing straight on.

(11) Install access panel.

NOTE: Residual fuel may be drained at sump drains (2) located in each wing tank lower skin between nacelle and center section, and one on each nacelle, located outboard side, under wing at main skin.

MAINTENANCE MANUAL

CHAPTER 20 – AIRFRAME STANDARD PRACTICES

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20-10-10	MP	202	JUN 01/93
20-10-20	MP	201	OCT 01/05
20-10-20	MP	202	OCT 01/05
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20-30-00	MP	201	JUN 01/93
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20-40-00	MP	213	OCT 07/17
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CHAPTER 20 – AIRFRAME STANDARD PRACTICES

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STANDARD PRACTICES – DESCRIPTION AND OPERATION

1. General

A. The procedures specified in this chapter apply to general airframe maintenance practices pertaining to the following:

(1) Torquing of threaded fasteners, and connections.

(2) Safetying procedures.

(3) Control cables and pulleys.

(4) Solvents and sealants.

(5) Conversion factors.

(6) General safety precautions.

B. Exceptions or modifications to these procedures are noted in the applicable chapter(s) of this Maintenance Manual. The information provided in this chapter applies to all fasteners and plumbing.

C. The final section of this chapter provides conversion tables to convert units of measurement used on SA22 Series airplanes into other standard measurements.

TORQUE PROCEDURES AND VALUES – MAINTENANCE PRACTICES

1. Procedures – Torque Procedures and Values

A. General

(1) To ensure components are securely installed without over-stressing nuts, bolts, screws, or pipe connections, use torque values provided in tables within this chapter.

CAUTION: BEFORE USING A TORQUE WRENCH, ENSURE THAT IT IS CORRECTLY CALIBRATED AND THAT THE PERMISSIBLE TIME BETWEEN BENCH CHECKS HAS NOT BEEN EXCEEDED.

(2) There is no satisfactory method of determining the torque previously applied to a threaded fastener. When re-torquing, always back off approximately one quarter (1/4) turn or more before reapplying torque.

(3) Torque values in Tables 201 thru 204 are for clean non-lubricated parts. Do not use lubricants unless specified. When threads are lubricated, decrease torque by 10%.

(4) If threads are cadmium plated or lubricated with graphite, decrease torque by 15%.

CAUTION: AVOID CONTACT BETWEEN CHLORINE BASED SOLVENTS AND TITANIUM SINCE IT MAY CAUSE BRITTLENESS.

(5) Standard titanium bolts are coated with molybdenum disulfide to prevent corrosion or seizure and no extra treatment is required when fitting. Should the coating be damaged, change the bolt. If threads are coated with molybdenum disulfide, decrease torque by 15%.

(6) When using a pre-set torque wrench, apply the load gradually until the wrench gives its indicating snap. Load must be removed immediately as wrench snaps. The snapping of the wrench does not prevent additional torque from being applied if more pressure is applied to the handle.

(7) Torque accessible nuts on threaded screws to the values in Table 201. Slotted and Phillips head screws that do not have accessible nuts or are attached to nut plates, and screws with threads other than those shown in Table 201, need not be torqued.

(8) Whenever possible, apply torque to nut. If necessary to torque a fastener from the head end, increase the maximum torque value by an amount equal to the shank friction. Measure shank friction with a torque wrench.

(9) Initially tighten nuts requiring a cotter pin to the minimum torque value indicated in the tables. The torque may be increased to insert the cotter pin but DO NOT exceed the maximum specified value.

(10) If parts are to be heated or cooled during assembly, torque nuts and bolts after the parts have returned to ambient temperature.

(11) Hand tighten nuts or bolts that must be sequentially tightened, then torque in the sequence shown in Figure 202.

CAUTION: DO NOT USE TORQUE WRENCHES TO LOOSEN NUTS OR BOLTS.

(12) Nuts and bolts fitted with flat seals or gaskets are to be initially torqued to slightly less than the required value. A brief pause will allow the seal or gasket to compress before final torquing.

(13) Install straight threaded connections, such as flared tubes and hose end fittings, and torque to values listed in Table 202.

(14) Straight threaded connections using O-rings or gaskets for seal such as AN924 nut, AN6289 nut, and fittings conforming to MS33656 Style E, need not be torqued. Install them in accordance with AND10064.

(15) When fitting self-locking nuts comply with the following instructions:

(a) Self-locking nuts are only to be fitted to screws or bolts that have a domed or chamfered start to the thread.

(b) Self-locking nuts are not to be used on bolts that have drillings in the threads to accept cotter pins.

(c) The re-tapping of a self-locking nut is not permitted.

(d) All metal self-locking nuts with thread cutters may only be used once.

(e) When fitting a self-locking nut check that the frictional resistance of the locking section is sufficient. If the frictional resistance can be overcome by hand, discard the nut.

(f) Do not lubricate the threads of the nut or bolt.

(g) Self-locking nuts with nylon-metallic inserts (NYLSTOP nuts) must not be used where temperatures exceed 248°F (120°C).

(h) When torquing self-locking nuts, add standard torque to the torque necessary to run the nut down to the material (tare value).

(16) Assemble threaded fasteners (bolts, screws, and nuts) at wrench torque values listed in Table 201 and Table 203.

(17) When an extension or adapter is used with a torque wrench the torque is altered. Use the chart in Table 207 or the formula in Figure 201 to calculate the correct setting.

(a) The adapter length is the distance between the torque centers of the wrench and the adapter.

(b) The torque arm is the approximate distance from the center of the hand grip of the wrench to its torque center.

NOTE: The use of an adapter extender at right angles to the torque arm DOES NOT affect the torque reading.

(18) If a universal (swivel) socket is required to torque inaccessible nuts or bolts, use the chart in Table 206 to determine the correct torque value.

(19) When an adapter and a universal socket extends the wrench length at an angle, the correct torque is obtained by consulting Figure 201 or Table 206 for the torque reading, then multiplying by a factor for the adapter obtained from Table 207 to yield the appropriate torque.

(20) Sheet metal screws need not be torqued but should have the heads drawn down tight.

(21) Specific torque is not applicable to bolts, nuts, and screws used in control systems or other installations where the torques required would cause binding or interference with proper operation of parts. On such applications the assembly should be firm, but not tight enough to cause binding.

(22) Tighten swagelok fittings by installing fitting until finger tight, and while holding the fitting body steady with a back up wrench, tighten the nut one and one quarter (1 1/4) turns, for 1/16, 1/8, and 3/16 size fitting, tighten only three quarter (3/4) turn from finger tight.

(23) Tighten nylo-seal fitting by installing fitting until finger tight. Tighten the nut two to two and one half (2 to 2 1/2) full turns. The nut may be marked at the six o'clock position to assure the nut is tightened the required amount.

(24) Both the nylo-seal and swagelok may be disconnected and retightened. Retighten as outlined in steps (22) and (23) above. After retightening swagelok fittings IAW step (22) above, tighten an additional one quarter (1/4) turn.

(25) Tighten screw type hose clamps finger tight plus one half (1/2) additional turn.

B. Torque Tables

Bolt/Nut Thread Size	Self Locking and Castellated Nuts			Self Locking External Wrenching Nuts					
	Part Number			Steel Shear			Steel Shear		
	NAS679 AN310 NAS1021 MS2104 MS21044			AN320			LH14367 160,00 PSI (11,248 kg/cm^2)		
	daNm	in. lbs	kpm	daNm	in. lbs	kpm	daNm	in. lbs	kpm
8 - 32	0.14 - 0.17	12 - 15	0.14 - 0.17	0.08 - 0.10	7 - 9	0.08 - 0.10	-- -- -- -- --	-- -- -- -- --	-- -- -- -- --
10 -32	0.23 - 0.28	20 - 25	0.23 - 0.29	0.14 - 0.17	12 - 15	0.14 - 0.17	-- -- -- -- --	-- -- -- -- --	-- -- -- -- --
1/4 - 28	0.57 - 0.79	50 - 70	0.58 - 0.81	0.34 - 0.45	30 - 40	0.35 - 0.46	0.73 - 1.02	65 - 90	0.55 - 1.04
5/16 - 24	1.13 - 1.16	100 - 140	1.15 - 1.61	0.90 - 1.00	60 - 85	0.69 - 1.00	1.47 - 2.03	130 - 180	1.50 - 2.07
3/8 - 24	1.76 - 2.15	160 - 190	1.76 - 2.19	1.07 - 1.24	95 - 110	1.09 - 1.26	2.50 - 4.07	220 - 360	2.53 - 4.15
7/16 - 20	5.08 - 5.65	450 - 500	5.08 - 5.70	3.05 - 3.38	270 - 300	3.11 - 3.45	4.18 - 6.89	370 - 610	4.25 - 7 03
1/2 - 20	5.42 - 7.80	480 - 690	5.42 - 7.95	3.28 - 4.63	290 - 410	3.34 - 4.72	7.12 - 10 73	630 - 950	7.26 - 10.95
9/16 - 18	9.04 - 11.30	800 - 1000	9.22 - 11.52	5.42 - 6.78	480 - 600	5.53 - 6.91	11.30 - 14.69	1000 - 1300	11.52 - 14.98
5/8 - 18	12.42 - 14.69	1100 - 1300	12.67 - 14.98	7.46 - 8.81	660 - 780	7.60 - 8.98	15.82 - 19.21	1400 - 1700	16.82 - 19.59
3/4 - 16	25.99 - 28.25	2300 - 2500	26.50 - 28.80	14.69 - 16.95	1300 - 1500	14.96 - 17.28	27.12 - 35 03	2400 - 3100	27.65 - 35 71
7/8 - 14	29.38 - 33.90	2600 - 3000	29.96 - 34.56	16.95 - 20.33	1500 - 1800	17.28 - 20.73	41.80 - 55.36	3700 - 4900	42.62 - 56.45
1 - 14	41.80 - 62.14	3700 - 5500	42.62 - 63.33	24.86 - 37.28	2200 - 3300	25.34 - 38.01	57.62 - 89.26	5100 - 7900	58.79 - 91.02
1 1/8 - 12	56.49 - 79.90	5000 - 7000	57.61 - 80.65	33.90 - 47.45	3000 - 4200	34.56 - 48.39	77.95 - 107.34	6900 - 9500	79.50 - 109.45
1 1/4 - 12	101.69 - 124.29	9000 - 11000	103.69 - 112.98	61.07 - 74.57	5400 - 6600	62.22 - 76.04	129.93 - 158 18	11500 - 14000	132.49 - 161.30
1 3/8 - 12				79.90 - 101,69	7000 - 9000	80.65 - 103 69			
1 1/2 - 12				112.99 - 135.58	10000 - 12000	115.21 - 138.25			

6M201000201

Nut And Bolt Torques
Table 201

Tubing O.D. (Inches)	Aluminum Alloy Tubing - Flare MS33583 or MS33584			Steel Tubing Flare MS33584			Hose End Fittings and Hose Assemblies (Inch-Pounds) MS24587 or Similar Fitting End		
	daNm	in. lbs	kpm	daNm	in. lbs	kpm	daNm	in. lbs	kpm
1/4	0.45 - 0.68	40 - 60	0.46 - 0.69	0.57 - 1.02	50 - 90	0.58 - 1.04	0.57 - 1.02	50 - 90	0.58 - 1.04
4/16	0.68 - 0.90	60 - 80	0.69 - 0.92	0.79 - 1.37	70 - 120	0.81 - 1.38	0.79 - 1.37	80 - 120	0.81 - 1.38
3/8	0.85 - 1.41	75 - 125	0.86 - 1.44	1.02 - 1.70	90 - 150	1.04 - 1.73	1.02 - 1.70	90 - 150	1.04 - 1.73
1/2	1.70 - 2.82	150 - 250	1.73 - 2.88	1.75 - 2.82	155 - 250	1.79 - 2.88	1.75 - 2.82	155 - 250	1.79 - 2.88
5/8	2.26 - 3.96	200 - 350	2.30 - 4.03	3.39 - 4.52	300 - 400	3.47 - 4.61	3.39 - 4.52	300 - 400	3.47 - 4.61
3/4	3.39 - 5.65	300 - 500	3.47 - 5.76	4.86 - 6.50	430 - 575	4.96 - 6.58	4.86 - 6.50	430 - 575	4.96 - 6 58
1	5.65 - 7.91	500 - 700	5.76 - 8.07	6.21 - 8.47	550 - 750	6.34 - 8.64	6.21 - 8 47	550 - 750	6.34 - 8.64
1 1/4	6.78 - 10.17	600 - 900	6.91 - 10.37						

6M201000202

Coupling Nut Torques
Table 202

Torque values for screws and nuts with inch thread and a strength ot 125,000 psi (861.8 N/mm^2) and above. (NAS 464, NAS 1303 to NAS 1306, NAS 1308)

Screw Size No.	Standard Nuts AN 310, NAS 1021			Flat Nuts AN 316, AN 320		
	daNm	in. lbs	kpm	daNm	in. lbs	kpm
10 -32	0.45	40	0.46	0.28	25	0.28
1/4 - 28	0.79 - 1.13	70 - 100	0.81 - 1.15	0.34 - 0.45	30 - 40	0.35 - 0.46
5/16 - 24	1.58 - 2.54	140 - 225	1.61 - 2.59	0.68 - 0.96	60 - 85	0.69 - 0.97
3/8 - 24	2.15 - 4.40	190 - 390	2.19 - 4.49	1.07 - 1.24	95 - 110	1.09 - 1.26
7/16 - 20	5.08 - 6.21	450 - 550	5.18 - 6.33	3.05 - 3.38	270 - 300	3.11 - 3.45
1/2 - 20	5.42 - 7.90	480 - 700	5.53 - 8.06	3.28 - 4.63	290 - 410	3.34 - 4.72
9/16 - 18	9.04 - 14.12	800 - 1250	9.22 - 14.40	5.42 - 6.78	480 - 600	5.53 - 6.91
5/8 - 18	12.42 - 18.08	1100 - 1600	12.67 - 18.43	7.46 - 8.81	660 - 780	7.60 - 8.98
3/4 - 16	25.99 - 33.90	2300 - 3000	26.50 - 34.56	14.69 - 16.95	1300 - 1500	14.98 - 17.28
7/8 - 14	28.25 - 47.46	2500 - 4200	28.80 - 48.38	16.95 - 20.33	1500 - 1800	17.28 - 20.73
1 - 12	41.80 - 67.80	3700 - 6000	42.62 - 69.12	24.86 - 37.28	22.00 - 3300	25.34 - 38.01

6M201000203

Screw And Nut Torques
Table 203

Screw Size No.	Torque values for screws and nuts with inch threads and with a minimum strength of 125,000 psi (861.8 N/mm^2)			Torque values for light alloy rod ends with inch thread and with a minimum strength of 62,580 psi (431.5 N/mm^2)		
	daNm	in. lbs	kpm	daNm	in. lbs	kpm
5/16 - 24	0.54 - 0.73	50 - 65	0.55 - 0.75	0.29 - 0.36	26 - 32	0.30 - 0.37
3/8 - 24	0.79 - 0.98	70 - 85	0.80 - 0.98	0.49 - 0.57	45 - 50	0.52 - 0.58
9/16 - 20	1.36 - 1.69	120 - 150	1.38 - 1.73	0.68 - 0.88	60 - 80	0.69 - 0.90
1/2 - 20	1.76 - 2.59	160 - 230	1.84 - 2.65	0.98 - 1.18	85 - 105	0.98 - 1.21
9/16 - 18	2.65 - 3.27	240 - 290	2.77 - 3.34	1.37 - 1.57	120 - 140	1.48 - 1.61
5/8 - 18	3.14 - 3.92	280 - 350	3.23 - 4.03	1.76 - 1.96	160 - 170	1.83 - 1.96

6M201000204

Steel And Light Alloy
Rod End Torques
Table 204

Torque values for hollow screws with Anti-Seize compound on threads				
Size	daNm	in. lbs	ft. lbs	kpm
1/8 - 27	0.68 - 0.92	60 - 80	5.0 - 6.6	0.69 - 0.92
1/4 - 18	1.47 - 1.70	130 - 150	10.9 - 12.5	1.50 - 1.73
3/8 - 18	2.09 - 2.43	185 - 215	15.4 - 18.0	2.13 - 2.47
1/2 - 14	2.88 - 3.22	255 - 285	21.1 - 23.8	2.94 - 3.29
3/4 - 14	3.50 - 3.96	310 - 350	25.8 - 29.2	3.57 - 4.03

6M201000205

Hollow Screw Torques
Table 205

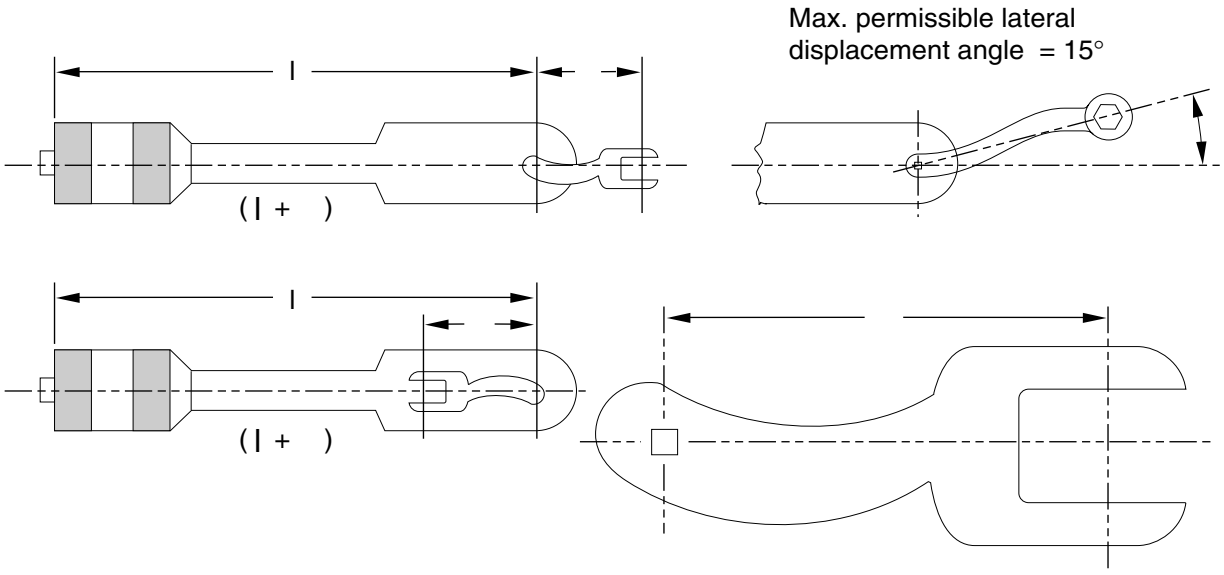
Formula: $Tx = \frac{Ta \times I}{I + I_1}$

Where: Ta = actual (required) torque value at end of extension (in. lbs or daNm)

Tx = Indicated or required value

I = length of arm (in. or m)

I_1 = effective length of extension (in. or m)



Effective Length of the adapter

6M201000206

Torque Wrench Extensions
Figure 201

Conversion Torque Table for Use with Universal Sockets							
To Obtain a Torque of	Dial Must Read – With Universal Held at These Angles						
Inch or Ft. Lbs.	10°	20°	30°	40°	45°	50°	60°
10	10.2	10.7	11.5	13.1	14.1	15.6	20.0
20	20.4	21.4	23.0	26.2	28.2	31.2	40.0
30	30.6	32.1	34.5	39.3	42.3	46.9	60.0
40	40.8	42.8	46.0	52.4	56.4	62.4	80.0
60	61.2	64.2	69.0	78.6	84.6	93.6	120.0
80	81.6	85.6	92.0	104.8	112.8	124.8	160.0
100	102.0	107.0	115.0	131.0	141.0	156.0	200.0
120	122.4	128.4	138.0	157.2	169.2	187.2	240.0
140	142.8	149.8	161.0	183.4	197.4	218.4	280.0
180	183.6	192.6	207.0	235.8	253.8	280.8	360.0
200	204.0	214.0	230.0	262.0	282.0	312.0	400.0
220	224.4	235.5	243.0	288.2	310.2	343.2	440.0
240	244.8	256.8	276.0	314.4	338.4	374.4	480.0
260	265.2	278.2	299.0	340.6	366.6	405.6	520.0
280	385.6	299.6	312.0	366.8	394.8	436.8	560.0
300	306.0	321.0	345.0	393.0	423.0	468.0	600.0
320	326.4	342.4	368.0	419.2	451.2	499.2	640.0
340	346.8	363.8	391.0	445.4	479.4	530.4	680.0
360	367.2	385.2	414.0	471.6	507.6	561.6	720.0
380	387.6	406.6	437.0	497.8	535.8	592.8	760.0
400	408.0	428.0	460.0	524.0	564.0	624.0	800.0
420	428.4	449.4	483.0	550.2	602.2	655.0	840.0
440	448.8	470.8	506.0	576.4	620.4	686.4	880.0
460	469.2	492.2	529.0	593.6	648.6	717.6	920.0
480	489.6	512.6	552.0	628.8	676.8	748.8	960.0
500	510.0	535.0	575.0	655.0	705.0	780.0	1000.0

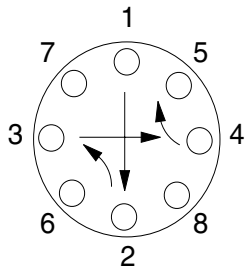
6M201000207

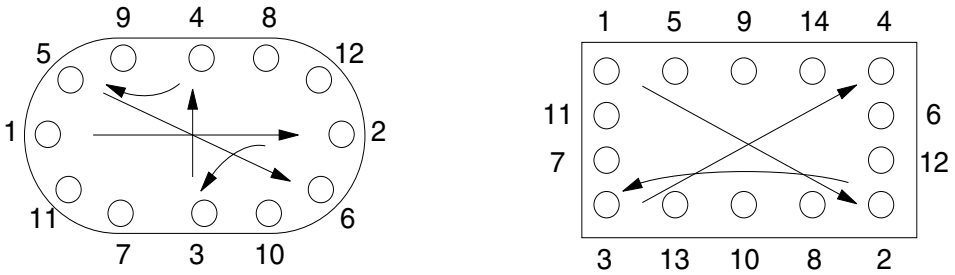
Universal Socket
Torque Conversion
Table 206

CORRECTION FACTORS FOR TORQUE WRENCHES USED WITH ADAPTORS WHICH EXTEND THE WRENCH							
Approx. Length of Torque Arm	Adaptor Lengths						
	1" thru 1 5/8"	Over 1 5/8" thru 2 3/8"	Over 2 3/8" thru 3 1/8"	Over 3 1/8" thru 3 7/8"	Over 3 7/8" thru 4 5/8"	Over 4 5/8" thru 5 3/8"	Over 5 3/8" thru 6"
7"	0.85	0.77	0.71	0.66	0.61	0.57	0.54
9 1/2"	0.89	0.83	0.78	0.74	0.69	0.67	0.64
13 1/2"	0.91	0.87	0.83	0.79	0.76	0.73	0.70
24"	0.95	0.92	0.90	0.87	0.85	0.83	0.81

6M201000208

Extension/Adapter Correction Factors Table 207





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Torquing Sequence Figure 202

kpm	To: in. lbs 89.796236	To: daNm 0.9806665	in. lbs	To: kpm 0.0115212	To: daNm 0.0112985	daNm	To: kpm 1.019716	To: in. lbs 88.507535
1	86.795	0.981	1	0.012	0.011	1	1.020	88.508
2	173.592	1.961	2	0.023	0.023	2	2.039	177.015
3	260.389	2.942	3	0.035	0.034	3	3.059	265.523
4	347.185	3.923	4	0.046	0.045	4	4.079	354.030
5	433.981	4.903	5	0.058	0.056	5	5.099	442.538
6	520.777	5.884	6	0.069	0.068	6	6.118	531.045
7	607.574	6.865	7	0.081	0.079	7	7.138	619.553
8	694.370	7.845	8	0.092	0.090	8	8.158	708.060
9	781.166	8.826	9	0.104	0.102	9	9.177	796.568
10	867.962	9.807	10	0.115	0.113	10	10.197	885.075
11	954.759	10.787	11	0.127	0.124	11	11.217	973.583
12	1041.555	11.768	12	0.138	0.136	12	12.237	1062.090
13	1128.351	12.749	13	0.150	0.147	13	13.256	1150.598
14	1215.147	13.729	14	0.161	0.158	14	14.276	1239.105
15	1301.944	14.710	15	0.173	0.169	15	15.296	1327.613
16	1388.740	15.691	16	0.184	0.181	16	16.315	1416.121
17	1475.536	16.671	17	0.196	0.192	17	17.335	1504.628
18	1562.332	17.625	18	0.207	0.203	18	18.355	1593.136
19	1649.128	18.633	19	0.219	0.215	19	19.375	1681.643
20	1735.925	19.613	20	0.230	0.226	20	20.394	1770.151
21	1822.721	20.594	21	0.242	0.237	21	21.414	1858.658
22	1909.517	31.575	22	0.253	0.249	22	22.434	1947.166
23	1996.313	22.555	23	0.265	0.260	23	23.453	2035.673
24	2083.110	23.536	24	0.277	0.271	24	24.473	2124.181
25	2169.906	24.517	25	0.288	0.282	25	25.493	2212.688
30	2603.887	29.420	30	0.346	0.339	30	30.591	2655.226
35	3037.868	34.323	35	0.403	0.395	35	35.690	3097.764
40	3471.849	39.227	40	0.461	0.452	40	40.789	3540.301
45	3905.831	44.130	45	0.518	0.508	45	45.887	3982.830
50	4339.812	49.033	50	0.576	0.565	50	50.986	4425.377
60	5207.774	58.840	60	0.691	0.678	60	61.183	5310.452
70	6075.736	68.647	70	0.806	0.791	70	71.380	6195.527
80	6943.699	78.453	80	0.922	0.904	80	81.577	7080.603
90	7811.661	88.260	90	1.037	1.017	90	91.774	7965.678
100	8679.624	98.067	100	1.152	1.130	100	101.974	8850.753
			200	2.304	2.260			
			300	3.456	3.390			
			400	4.609	4.519			
			500	5.761	5.649			
			600	6.913	6.779			
			700	8.065	7.909			
			800	9.217	9.039			
			900	10.369	10.169			
			1000	11.521	11.299			
			2000	23.042	22.597			
			3000	34.564	33.896			
			4000	46.085	45.194			
			5000	57.606	54.493			
			6000	69.127	67.791			
			7000	80.648	79.090			
			8000	92.170	90.388			
			9000	103.691	101.687			
			10000	115.212	112.985			

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Torque Conversion
Table 208
(Sheet 1 of 2)

ft. lbs.	To: kpm	To: daNm	To: in. lbs.	kpm	To: ft. lbs.	daNm	To: ft. lbs.	in. lbs	To: ft. lbs.
	0.138255	0.135	12.00		7.233011		7.407		0.0833
1	0.138	0.135	12.00	1	7.233	1	7.407	1	0.083
2	0.276	0.270	24.00	2	14.466	2	14.814	2	0.166
3	0.414	0.405	36.00	3	21.699	3	22.221	3	0.249
4	0.553	0.540	48.00	4	28.932	4	29.628	4	0.333
5	0.691	0.675	60.00	5	36.165	5	37.035	5	0.416
6	0.829	0.810	72.00	6	43.398	6	44.442	6	0.499
7	0.967	0.945	84.00	7	50.631	7	51.849	7	0.583
8	1.106	1.080	96.00	8	57.864	8	59.256	8	0.666
9	1.244	1.215	108.00	9	65.097	9	66.663	9	0.749
10	1.382	1.350	120.00	10	72.330	10	74.070	10	0.833
15	2.073	2.025	180.00	15	108.495	15	111.105	15	1.249
20	2.765	2.700	240.00	20	144.660	20	148.140	20	1.666
25	3.456	3.375	300.00	25	180.885	25	185.175	25	2.082
30	4.147	4.050	360.00	30	216.990	30	222.210	30	2.499
35	4.838	4.725	420.00	35	253.155	35	259.245	35	2.915
40	5.530	5.400	480.00	40	289.320	40	296.280	40	3.332
45	6.221	6.075	540.00	45	325.485	45	333.315	45	3.748
50	6.912	6.750	600.00	50	361.650	50	370.350	50	4.165

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Torque Conversion
Table 208
(Sheet 2 of 2)

MAINTENANCE MANUAL

SAFETY DEVICES – MAINTENANCE PRACTICES

1. General

A. Safetying devices are used to prevent loosening of attaching hardware subjected to vibration, tension, torque, or stress.

2. Safetying Devices

A. Turnbuckle Safety Springs

CAUTION: TURNBUCKLE SAFETY SPRINGS MAY BE USED ONE TIME ONLY. REPLACE PREVIOUSLY USED SAFETY SPRINGS

(I) After cable tension adjustment, not more than three (3) threads of the cable rod must be visible at tumbuckle barrel ends.

(2) Align grooves in cable rod and turnbuckle barrel and insert leg of safety spring until looped end of spring is locked over hole in center of turnbuckle barrel.

(3) Press looped end of safety spring into hole of turnbuckle barrel; the looped end will latch in barrel hole.

(4) The looped ends of the two safety springs may be inserted in the same side or in opposite sides of the turnbuckle barrel.

(5) To check correct seating of safety springs, attempt to remove looped ends with fingers only.

B. Safety Wiring Turnbuckles

NOTE: This method is to be used only if safety springs cannot be used.

(I) Insert two safety wires through hole in center of turnbuckle barrel.

(2) Bend each wire towards turnbuckle cable rods; insert wires through safety holes in rods.

(3) Bend one end of each wire back towards barrel and wind the wire ends around shaft of tumbuckle cable rod.

(4) Wrap each wire a minimum of four (4) times around shaft of turnbuckle cable rod. Do not twist wire ends together.

C. Safety Wiring Nuts and Screws

(1) Screw heads that require locking are generally locked with safety wire using the double twist method except in the case of small screws in a closely spaced, closed geometrical pattern.

(2) When securing multiple groups of bolts, the bolts should be safety wired in pairs or in a maximum group of three (3).

(3) The double twist method may also be used for locking castellated nuts.

(4) When locking castellated nuts with safety wire, initially tighten nut to minimum specified torque. Increase the torque to align slot with hole in bolt but, do not exceed maximum specified value.

(5) The single wire method may be used on a closely spaced geometrical pattern (triangle square, circle, etc.), or where access is difficult.

(6) The single wire method is used for shear and seal wiring. Install wire so that it can easily be broken in an emergency situation. Use copper wire for locking emergency devices where it is necessary to break the wire quickly.

(7) Twist safety wire ends over a length of 0.25 to 0.5 inch (6.35 to 12.7 mm) or five (5) to six (6) twists, and bend inwards against the component to prevent snagging or injury.

D. Safetying with Cotter Pins

CAUTION: ENSURE COTTER PINS ARE THE CORRECT TYPE, AND MUST NOT BE USED MORE THAN ONCE.

DO NOT BEND SPLIT PIN HEADS.

OBSERVE INSTALLATION REQUIREMENTS REGARDING PREFERRED AND ALTERNATIVE METHODS.

NOTE: When aligning split pin hole, initially tighten nut to minimum specified torque. The torque may be increased to insert split pin but must not exceed the maximum specified value.

(I) Preferred Method (all positions).

(a) Insert cotter pin firmly into slot of nut and bend both legs around the body of the nut or hinge pin.

(2) Alternative Method (non-moving parts only).

(a) Insert cotter pin until its head is firmly in slot of nut. Bend upper leg over bolt end and shorten the leg if necessary.

(b) Bend the lower leg downwards, shortening the leg if necessary to prevent contact with the washer.

CONTROL CABLES AND PULLEYS – MAINTENANCE PRACTICES

1. Inspection/Check – Control Cables And Pulleys

A. Inspection of Control Cables

WARNING: WEAR GLOVES WHEN CHECKING CABLES FOR BROKEN WIRES.

(1) Run a soft cloth along cable length. Cloth snaging on cable is an indication of broken wires.

(2) Replace cables with indications of damage as follows:

(a) 7x7 cable – More than three (3) individual wires (six [6] wire ends visible) broken over a length of 1.0 inch (2.54 cm).

(b) 7xl9 cable – More than six (6) individual wires (twelve [12] wire ends visible) broken over a length of 1.0 inch (2.54 cm).

(c) Any wires broken near cable guides and pulleys.

(d) Cables are kinked or twisted.

(e) If corrosion is detected, release cable tension and slightly untwist cable to determine extent of corrosion. Replace cable If inner section is corroded.

(3) Control cables must be pre-stretched prior to installing final end fitting. Pre-stretch cables to loads indicated below.

(a) 1/16 inch ..... 288 + 25, - 0 pounds.

(b) 3/32 inch ..... 552 + 25, - 0 pounds.

(c) 1/8 inch ..... 1200 + 50, - 0 pounds.

(d) 5/32 inch ..... 1680 + 75, - 0 pounds.

(e) 3/16 inch ..... 2520 + 100, - 0 pounds.

(4) Control cables must never be coiled to less than minimum diameter as shown in Table 201.

Cable Diameter	Type	Coil Diameter (Inches)
1/16	7X7	10
3/32	7X7	14
1/8	7X19	19
5/32	7X19	23
3/16	7X19	29

Table 201

Cable Diameter	Type	Proof Load (Pounds)
1/16	7X7	288 (+25 to -0)
3/32	7X7	552 (+25 to -0)
1/8	7X7	1200 (+50 to -0)
5/32	7X7	1680 (+75 to -0)
3/16	7X7	2520 (+100 to -0)

Proof Load
Table 202

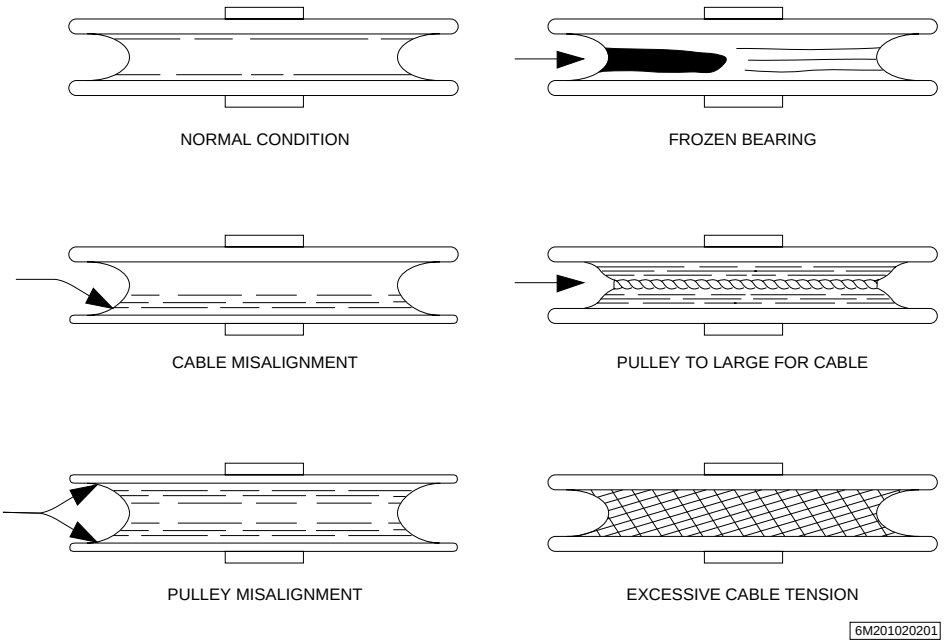
B. Inspection of Pulleys

(I) Inspect pulleys for cracks, broken edges, uneven or stiff rotation, and pulley wear patterns (see Figure 201).

(2) Periodically rotate pulleys having partial rotation during cable movement to provide a new bearing surface for cable.

(3) Check pulley alignment.

(4) Check pulley bracket and guards for damage, alignment, and security.



Control Cable Pulley Wear
Figure 201

MAINTENANCE MANUAL

RESINS AND PLASTIC PARTS – DESCRIPTION AND OPERATION

1. General

A. This section provides general information about resins and plastic parts used on the airplane.

B. Component parts of complex shape (contoured parts, panelling, etc) are often made of reinforced plastic. The reinforcement may consist of glass fiber, carbon fiber, or synthetic fibers.

C. Construction may be:

(1) Epoxy or polyester resin impregnated woven rovings and chopped strand mat (floor-boards, wing fairings, panelling).

(2) Sandwich construction of resin impregnated woven rovings and chopped strand mat with a honeycomb core (vertical stabilizer fairing and radome).

RESINS AND PLASTIC PARTS – MAINTENANCE PRACTICES

1. Preparation – Resins and Plastic Parts

WARNING: RESINS AND HARDENERS (ACCELERATORS) ARE HAZARDOUS TO SKIN AND EYES. DO NOT INHALE FUMES AS THEY ARE TOXIC AND HARMFUL TO HEALTH. WASH HANDS BEFORE EATING OR SMOKING. USE SKIN PROTECTION CREAM.

WHEN MIXING COMPONENTS, ALWAYS ADD HARDENER (ACCELERATOR) TO RESIN, NOT RESIN TO HARDENER.

A. Mixing

CAUTION: DO NOT USE RESIN/HARDENER COMPONENTS THAT HAVE EXCEEDED THEIR SHELF LIFE.

(1) Mix and apply resins when possible at ambient room temperature and a maximum relative humidity of 75%.

(2) Do not mix more than fifty (50) grams at any time.

(3) Apply mixture during pot life, before it begins to gel.

(4) Mix one hundred (100) parts (weight) epoxy resin (Part A) to twenty five (25) parts (weight) hardener (Part B).

NOTE: Resin (Part A) is a low viscosity, light yellow liquid, the hardener (Part B) is a low viscosity, yellow-brown liquid. Agitate components with a wooden spatula until a homogenous state is achieved.

(5) When employing resin for honeycomb sandwich construction, add 25% thickening agent to mixture if required.

B. Preparation of Resin/Hardener Mix as a Bonding Agent for Elastomers and Metal or Composite Plastic Components.

(1) Do not mix more than one hundred (100) grams at any time.

NOTE: Pot life is 70 min. at 77°F (25°C).

(2) Mix one hundred (100) parts (weight) epoxy resin to forty (40) parts (weight) hardener.

NOTE: Resin is a white or black paste, the hardener a yellow-brown medium viscosity liquid.

(3) Agitate components with a wooden spatula until a homogenous state is achieved.

C. Preparation of Resin/Hardener Mix as Bonding Agent for Polyester Fabric Surface Coverings.

(1) Do not mix more than one hundred (100) grams at any time

NOTE: The pot life is 135 min. at 77°F (25°C) and 85 min. at 86°F (30°C).

(2) Mix one hundred (100) parts (weight) epoxy resin to eighty (80) parts (weight) hardener.

NOTE: The resin is a cream colored, high viscosity liquid, the hardener a yellow-brown liquid.

(3) Agitate components with a wooden spatula until a homogenous state is achieved.

CONVERSION PROCEDURES AND VALUES – MAINTENANCE PRACTICES

1. Conversion Procedures

A. General Information

(1) The following tables contain information necessary to convert standard measurement units contained in SA227 Commuter Category Aircraft Maintenance Manuals to other measurements.

(a) Power Conversion Table.

1 Table 201 converts DIN HP to kW and HP; HP to DIN HP and kW; and kW to DIN HP and HP.

2 Multiply values not included on the chart by conversion factors at the top of each column to obtain desired equivalent.

(b) Weight Conversion Table.

1 Table 202 converts kp to lbs and daN; lbs to kp and daN; and daN to kp and lbs.

2 Multiply values not included on the chart by conversion factors at the top of each column to obtain desired equivalent.

(c) Pressure Conversion Table (liquid and Gas).

1 Table 203 Sheet 1 converts kp/cm^2 to psi and bar; psi to kp/cm^2 and bar; and bar to kp/cm^2 and bar. Sheet 2 converts in. Hg to psi and kp/cm^2; bar to in. Hg; kp/cm^2 to in. Hg; and psi to in. Hg.

2 Multiply values not included on the chart by conversion factor at the top of each column to obtain desired equivalent.

(d) Temperature Conversion Table.

1 Table 204 converts Celsius temperatures to Fahrenheit temperatures and Fahrenheit temperatures to Celsius temperatures.

2 Find temperature to be converted in bold lettering. Fahrenheit to celsius equivalents appear in the left hand column and Celsius to Fahrenheit in the right.

(e) Inch Conversion Table.

1 Table 205 converts inches to millimeters.

2 Multiply decimal inches by a factor of 25.40 to obtain equivalent in millimeters for values not listed on chart.

(2) Conversion Charts.

DIN HP	To: kw	To: HP	HP	To: DIN HP	To: kw	kw	To: DIN HP	To: HP
	0.735499	0.896321		1.013869	0.745699		1.359621	1.341024
1	0.736	0.986	1	1.014	0.746	1	1.360	1.341
2	1.471	1.973	2	2.028	1.491	2	2.719	2.682
3	2.206	2.959	3	3.042	2.237	3	4.079	4.023
4	2.942	3.945	4	4.055	2.983	4	5.438	5.364
5	3.677	4.932	5	5.069	3.728	5	6.798	6.705
6	4.413	5.918	6	6.083	4.474	6	8.158	8.046
7	5.148	6.904	7	7.097	5.220	7	9.517	9.387
8	5.844	7.891	8	8.111	5.966	8	10.877	10.728
9	6.619	8.877	9	9.125	6.712	9	12.237	12.069
10	7.355	9.863	10	10.139	7.457	10	13.596	13.410
11	8.090	10.850	11	11.153	8.203	11	14.956	14.751
12	8.826	11.836	12	12.166	8.948	12	16.315	16.092
13	9.561	12.822	13	13.180	9.694	13	17.675	17.433
14	10.297	13.808	14	14.194	10.440	14	19.035	18.774
15	11.032	14.795	15	15.208	11.185	15	20.394	20.115
16	11.768	15.781	16	16.222	11.931	16	21.754	21.456
17	12.503	16.767	17	17.236	12.677	17	23.114	22.797
18	13.239	17.754	18	18.250	13.423	18	24.473	24.138
19	13.974	18.740	19	19.264	14.168	19	25.883	25.479
20	14.710	19.726	20	20.277	14.913	20	27.192	26.280
21	15.445	20.713	21	21.291	15.660	21	28.552	28.161
22	16.181	21.699	22	22.305	16.405	22	29.912	29.503
23	16.916	22.685	23	23.319	17.151	23	31.271	30.884
24	17.652	23.672	24	24.333	17.897	24	32.631	32.185
25	18.387	24.658	25	25.347	18.642	25	33.991	33.526
30	22.065	29.590	30	30.416	22.371	30	40.789	40.231
35	25.742	34.521	35	35.485	26.099	35	47.587	46.986
40	29.420	39.453	40	40.555	29.828	40	54.385	53.641
45	33.097	44.384	45	45.624	33.556	45	61.183	60.346
50	36.775	49.316	50	50.694	37.285	50	67.981	67.051
60	44.130	59.179	60	60.832	44.742	60	81.577	80.461
70	51.485	69.043	70	70.971	52.199	70	95.174	93.872
80	58.840	78.906	80	80.111	59.656	80	108.770	107.282
90	66.195	88.769	90	81.248	67.113	90	122.366	120.692
100	73.550	98.632	100	101.387	74.570	100	135.962	134.102
200	147.100	197.264	200	202.774	149.140	200	271.924	268.205
300	220.650	295.896	300	304.161	223.710	300	407.886	402.307
400	294.200	394.528	400	405.548	298.280	400	543.849	536.410
500	367.750	493.161	500	506.935	372.850	500	679.811	670.512
600	441.299	591.793	600	608.321	447.419	600	815.773	804.614
700	514.849	690.425	700	709.708	521.989	700	951.735	938.717
800	588.399	789.057	800	811.095	596.559	800	1087.697	1072.819
900	661.949	887.689	900	912.482	671.129	900	1223.659	1206.921
1000	735.499	986.321	1000	1013.869	745.699	1000	1359.621	1341.024
2000	1470.998	1972.642	2000	2027.738	1491.398	2000	2719.242	2682.047
3000	2206.497	2958.963	3000	3041.607	2237.097	3000	4078.863	4023.071
4000	2941.966	3945.284	4000	4055.476	2982.796	4000	5438.485	5364.095
5000	3677.495	4931.605	5000	5069.345	3728.495	5000	6798.106	6705.119
6000	4412.994	5917.926	6000	6083.214	4474.194	6000	8157.727	8046.142
7000	5148.493	6904.247	7000	7097.083	5219.893	7000	9517.349	9387.166
8000	5883.992	7890.568	8000	8110.952	5965.592	8000	10876.969	10728.190
9000	6619.491	8876.889	9000	9124.821	6711.291	9000	12236.591	12069.213
10000	7345.990	9863.210	10000	10138.690	7459.990	10000	13596.212	13410.237

6M203000201

Power Conversion
Table 201

CONVERSION PROCEDURES AND VALUES – MAINTENANCE PRACTICES

kg	To: lbs	To: daN	lbs	To: kg	To: daN	daN	To: kg	To: lbs
	2.204622	0.980665		0.453592	0.444822		1.019716	2.248091
1	2.205	0.981	1	0.454	0.445	1	1.020	2.248
2	4.409	1.961	2	0.907	0.890	2	2.039	4.496
3	6.614	2.942	3	1.361	1.334	3	3.059	6.744
4	8.818	3.923	4	1.814	1.779	4	4.079	8.992
5	11.023	4.903	5	2.268	2.224	5	5.099	11.240
6	13.228	5.884	6	2.722	2.669	6	6.118	13.489
7	15.432	6.865	7	3.175	3.114	7	7.138	15.737
8	17.637	7.845	8	3.629	3.559	8	8.158	17.985
9	19.842	8.826	9	4.082	4.003	9	9.177	20.233
10	22.046	9.807	10	4.536	4.448	10	10.197	22.481
11	24.251	10.787	11	4.990	4.893	11	11.217	24.729
12	26.455	11.768	12	5.443	5.338	12	12.237	26.977
13	28.660	12.749	13	5.897	5.783	13	13.256	29.225
14	30.865	13.729	14	6.350	6.228	14	14.276	31.473
15	33.069	14.710	15	6.804	6.672	15	15.296	33.721
16	35.274	15.691	16	7.257	7.117	16	16.315	35.969
17	37.479	16.671	17	7.711	7.562	17	17.335	38.218
18	39.683	17.652	18	8.165	8.007	18	18.355	40.466
19	41.888	18.633	19	8.618	8.452	19	19.375	42.714
20	44.092	19.613	20	9.072	8.896	20	20.394	44.962
21	46.297	20.594	21	9.525	9.341	21	21.414	47.210
22	48.502	21.575	22	9.979	9.786	22	22.434	49.458
23	50.706	22.555	23	10.433	10.231	23	23.453	51.706
24	52.911	23.536	24	10.886	10.676	24	24.473	53.954
25	55.116	24.517	25	11.340	11.121	25	25.493	56.202
30	66.139	29.420	30	13.608	13.345	30	30.591	67.443
35	77.162	34.323	35	15.876	15.569	35	35.690	78.683
40	88.185	39.227	40	18.144	17.793	40	40.789	89.924
45	99.208	44.130	45	20.412	20.017	45	45.887	101.164
50	110.231	49.033	50	22.680	22.241	50	50.986	112.405
60	132.277	58.840	60	27.216	26.689	60	61.183	134.885
70	154.323	68.647	70	31.751	31.138	70	71.380	157.366
80	176.370	78.453	80	36.287	35.585	80	81.577	179.847
90	198.416	88.260	90	40.823	40.034	90	91.774	202.328
100	220.462	98.067	100	45.359	44.482	100	101.972	224.809
200	440.924	196.133	200	90.718	88.964	200	203.943	449.618
300	661.386	294.200	300	136.078	133.447	300	305.915	674.427
400	881.848	392.266	400	181.437	177.929	400	407.886	899.236
500	1102.310	490.333	500	226.796	222.411	500	509.858	1124.046
600	1322.772	588.399	600	272.155	266.893	600	611.830	1348.855
700	1543.234	686.466	700	317.514	311.375	700	713.801	1573.664
800	1763.696	784.532	800	3362.874	355.857	800	815.773	1798.473
900	1984.158	882.599	900	408.233	400.340	900	917.745	2023.282
1000	2204.622	980.685	1000	453.592	444.822	1000	1019.715	2248.091
2000	4409.244	1961.330	2000	907.184	889.644	2000	2039.432	4496.182
3000	6613.866	2941.995	3000	1360.776	1334.465	3000	3059.149	6744.273
4000	8818.488	3922.660	4000	1814.368	1779.287	4000	4078.865	8992.364
5000	11023.110	4903.325	5000	2267.960	2224.109	5000	5098.581	11240.455
6000	13227.732	5883.990	6000	2721.552	2668.931	6000	5118.297	13488.546
7000	15432.354	6864.655	7000	3175.144	3113.753	7000	7138.038	15736.637
8000	17636.976	7845.320	8000	3628.736	3558.574	8000	8157.730	17984.728
9000	19841.598	8825.985	9000	4082.328	4003.396	9000	9177.446	20232.819
10000	22046.220	9806.650	10000	4535.920	4448.218	10000	10197.162	22480.910

6M203000202

Weight Conversion
Table 202

CONVERSION PROCEDURES AND VALUES – MAINTENANCE PRACTICES

kg/cm^2	To: psi	To: bar	psi	To: kg/cm^2	To: bar	bar	To: kg/cm^2	To: psi
	14.223335	0.980665		0.0703069	0.0689476		1.019716	14.503768
1	14.223	0.981	1	0.070	0.069	1	1.020	14.504
2	28.447	1.961	2	0.141	0.138	2	2.039	29.008
3	42.670	2.942	3	0.211	0.207	3	3.059	43.511
4	56.893	3.923	4	0.281	0.276	4	4.079	58.015
5	71.117	4.903	5	0.352	0.345	5	5.099	72.519
6	85.340	5.884	6	0.422	0.414	6	6.118	87.023
7	99.563	6.865	7	0.492	0.483	7	7.138	101.526
8	113.787	7.845	8	0.565	0.552	8	8.158	116.030
9	128.010	8.826	9	0.633	0.621	9	9.177	130.534
10	142.233	9.807	10	0.703	0.689	10	10.197	145.038
11	156.457	10.787	11	0.773	0.758	11	11.217	159.541
12	170.680	11.768	12	0.844	0.827	12	12.237	174.045
13	184.903	12.749	13	0.914	0.896	13	13.256	188.549
14	199.127	13.729	14	0.984	0.965	14	14.276	203.053
15	213.350	14.710	15	1.055	1.034	15	15.296	217.557
16	227.573	15.691	16	1.125	1.103	16	16.315	232.060
17	241.797	16.671	17	1.195	1.172	17	17.335	246.564
18	256.020	17.652	18	1.266	1.241	18	18.355	261.068
19	270.243	18.633	19	1.336	1.310	19	19.375	275.572
20	284.467	19.613	20	1.406	1.379	20	20.394	290.075
21	298.690	20.594	21	4.476	1.448	21	21.414	304.579
22	312.913	21.575	22	4.547	1.517	22	22.434	319.083
23	327.137	22.555	23	1.617	1.586	23	23.453	333.587
24	341.360	23.536	24	1.687	1.655	24	24.473	348.090
25	355.583	24.517	25	1.758	1.724	25	25.493	362.594
30	426.700	29.420	30	2.109	2.068	30	30.591	435.113
35	497.817	34.323	35	2.461	2.413	35	35.690	507.632
40	568.933	39.227	40	2.812	2.758	40	40.789	580.151
45	640.050	44.130	45	3.164	3.103	45	45.887	652.670
50	711.167	19.033	50	3.515	3.447	50	50.986	725.188
60	853.400	58.840	60	4.218	4.137	60	61.183	870.226
70	995.633	68.647	70	4.921	4.826	70	71.380	1015.264
80	1137.867	78.459	80	5.625	5.516	80	81.577	1160.301
90	1280.100	88.260	90	6.328	6.205	90	91.774	1305.339
100	1422.334	98.067	100	7.031	6.895	100	101.972	1450.377
200	2844.667	196.133	200	14.061	13.790	200	203.943	2900.754
300	4257.001	294.200	300	21.092	20.684	300	305.915	4351.130
400	5689.334	392.266	400	28.123	27.579	400	407.886	5801.507
500	7111.668	490.333	500	35.154	34.474	500	509.858	7251.884
600	8534.001	588.399	600	42.184	41.369	600	611.830	8702.261
700	9956.335	686.466	700	49.215	48.263	700	713.801	10152.638
800	11378.668	784.532	800	56.246	55.158	800	815.773	11603.014
900	12801.002	882.599	900	63.276	62.053	900	917.745	13053.391
1000	14223.335	980.665	1000	70.307	69.948	1000	1019.715	14503.768
2000	28446.670	1961.330	2000	140.614	137.895	2000	2039.432	29007.536
3000	42670.005	2941.995	3000	210.921	206.843	3000	3059.149	43511.304
4000	56893.340	3922.660	4000	281.228	275.790	4000	4078.865	58015.072
5000	71116.675	4903.325	5000	351.535	344.738	5000	5098.581	72518.840
6000	85340.010	5883.990	6000	421.841	413.686	6000	5118.297	87022.608
7000	99563.345	6864.655	7000	492.148	482.633	7000	7138.038	101526.376
8000	113786.680	7845.320	8000	542.455	551.581	8000	8157.730	116030.144
9000	128010.015	8825.985	9000	632.762	620.528	9000	9177.446	130533.912
10000	142233.350	9806.650	10000	703.069	689.476	10000	10197.162	145037.680

6M203000203

Pressure Conversion
Table 203
(Sheet 1 of 2)

in. Hg	To: psi	To: kg/cm	To: bar	bar	To: in. Hg	kg/cm	To: in. Hg	psi	To: in. Hg
	0.491152	0.0345314	0.0338638		29.530059		28.959150		2.036028
1	0.491	0.035	0.034	1	29.530	1	28.959	1	2.036
2	0.982	0.069	0.068	2	59.060	2	57.918	2	4.072
3	1.473	0.104	0.102	3	88.590	3	86.877	3	6.108
4	1.965	0.138	0.135	4	118.120	4	115.837	4	8.144
5	2.456	0.173	0.169	5	147.650	5	144.796	5	10.180
6	2.947	0.207	0.203	6	177.180	6	173.755	6	12.216
7	3.438	0.241	0.237	7	206.710	7	202.714	7	14.252
8	3.929	0.276	0.271	8	236.240	8	231.673	8	16.288
9	4.420	0.310	0.305	9	265.771	9	260.632	9	18.324
10	4.912	0.345	0.339	10	295.301	10	289.591	10	20.360
11	5.403	0.380	0.373	11	324.831	11	318.551	11	22.396
12	5.894	0.414	0.406	12	354.361	12	347.510	12	24.432
13	6.385	0.449	0.440	13	383.891	13	376.469	13	26.468
14	6.876	0.483	0.474	14	413.421	14	405.428	14	28.504
15	7.367	0.518	0.508	15	442.951	15	434.387	15	30.540
16	7.858	0.553	0.542	16	472.481	16	463.346	16	32.576
17	8.350	0.587	0.576	17	502.011	17	492.306	17	34.612
18	8.841	0.622	0.610	18	531.541	18	521.265	18	36.649
19	9.332	0.656	0.643	19	561.071	19	550.224	19	38.685
20	9.823	0.691	0.677	20	590.601	20	579.183	20	40.721
21	10.314	0.725	0.711	21	620.131	21	608.142	21	42.757
22	10.805	0.760	0.745	22	649.661	22	637.101	22	44.793
23	11.297	0.794	0.779	23	679.191	23	666.060	23	46.289
24	11.788	0.829	0.813	24	708.721	24	695.020	24	48.865
25	12.279	0.863	0.847	25	738.251	25	723.979	25	50.901
30	14.735	1.036	1.016	30	885.902	30	868.775	30	61.081
35	17.190	1.209	1.185	35	1033.552	35	1013.570	35	71.261
40	19.646	1.381	1.355	40	1181.202	40	1158.366	40	81.441
45	22.102	1.554	1.524	45	1328.853	45	1303.162	45	91.621
50	24.558	1.727	1.693	50	1476.503	50	1447.958	50	101.801
60	29.469	2.072	2.032	60	1771.804	60	1737.549	60	122.162
70	34.381	2.417	2.370	70	2067.104	70	2027.141	70	142.522
80	39.292	2.763	2.709	80	2362.405	80	2316.732	80	162.882
90	44.204	3.108	3.048	90	2657.705	90	2606.324	90	183.243
100	49.115	3.453	3.386	100	2953.006	100	2895.915	100	203.603

6M203000204

Pressure Conversion
Table 203
(Sheet 2 of 2)

C.	C. or F.	F.	C.	C. or F.	F.	C.	C. or F.	F.	C.	C. or F.	F.
-40	-40	-40	10.6	51	123.8	71	160	320	371	700	1292
-34.4	-30	-22	11.1	52	125.6	77	170	338	377	710	1310
-28.9	-20	-4	11.7	53	127.4	82	180	356	382	720	1328
-23.3	-10	14	12.2	54	129.2	88	190	374	388	730	1346
-17.8	0	32	12.8	55	131.0	93	200	392	393	740	1364
-17.2	1	33.8	13.3	56	132.8	99	210	410	399	750	1382
-16.7	2	35.6	13.9	57	134.6	100	212	413	404	760	1400
-16.1	3	37.4	14.4	58	136.4	104	220	428	410	770	1418
-15.6	4	39.2	15.0	59	138.2	110	230	446	416	780	1436
-15	5	41.0	15.6	60	140.0	116	240	464	421	790	1454
-14.4	6	42.8	16.1	61	141.8	121	250	482	427	800	1472
-13.9	7	44.6	16.7	62	143.6	127	260	500	432	810	1490
-13.3	8	46.4	17.2	63	145.4	132	270	518	438	820	1508
-12.8	9	48.2	17.8	64	147.2	138	280	536	443	830	1526
-12.2	10	50.0	18.3	65	149.0	143	290	554	449	840	1544
-11.7	11	51.8	18.9	66	150.8	149	300	572	454	850	1562
-11.1	12	53.6	19.4	67	152.6	154	310	590	460	860	1580
-10.6	13	55.4	20.0	68	154.4	160	320	608	466	870	1598
-10.0	14	57.2	20.6	69	156.2	166	330	626	471	880	1616
-9.44	15	59.0	21.1	70	158.0	171	340	644	477	890	1634
-8.89	16	60.8	21.7	71	159.8	177	350	662	482	900	1652
-8.33	17	62.6	22.2	72	161.6	182	360	680	488	910	1670
-7.78	18	64.4	22.8	73	163.4	188	370	698	493	920	1688
-7.22	19	66.2	23.3	74	165.2	193	380	716	499	930	1706
-6.67	20	68.0	23.9	75	167.0	199	390	734	504	940	1724
-6.11	21	69.8	24.4	76	168.8	204	400	752	510	950	1742
-5.56	22	71.6	25.0	77	170.6	210	410	770	516	960	1760
-5.00	23	73.4	25.6	78	172.4	216	420	788	521	970	1778
-4.44	24	75.2	26.1	79	174.2	221	430	806	527	980	1796
-3.89	25	77.0	26.7	80	176.0	227	440	824	532	990	1814
-3.33	26	78.8	27.2	81	177.8	232	450	842	538	1000	1832
-2.78	27	80.6	27.8	82	179.6	238	460	860	543	1010	1850
-2.22	28	82.4	28.3	83	181.4	243	470	878	549	1020	1868
-1.67	29	84.2	28.9	84	183.2	249	480	896	554	1030	1886
-1.11	30	86.0	29.4	85	185.0	254	490	914	560	1040	1904
-0.56	31	87.8	30.0	86	186.8	260	500	932	566	1050	1922
0	32	89.6	30.6	87	188.5	266	510	950	571	1060	1940
0.56	33	91.4	31.1	88	190.4	271	520	968	577	1070	1958
1.11	34	93.2	31.7	89	192.2	277	530	986	582	1080	1976
1.67	35	95.0	32.2	90	194.0	282	540	1004	588	1090	1994
2.22	36	96.8	32.8	91	195.8	288	550	1022	593	1100	2012
2.78	37	98.6	33.3	92	197.6	293	560	1040	599	1110	2030
3.33	38	100.4	33.9	93	199.4	299	570	1058	604	1120	2048
3.89	39	102.2	34.4	94	201.2	304	580	1076	610	1130	2066
4.44	40	104.0	35.0	95	203.0	310	590	1094	616	1140	2084
5.00	41	105.8	35.6	96	204.8	316	600	1112	621	1150	2102
5.56	42	107.6	36.1	97	206.6	321	610	1130	627	1160	2120
6.11	43	109.4	36.7	98	208.4	327	620	1148	632	1170	2138
6.67	44	111.2	37.2	99	210.0	332	630	1166	636	1180	2156
7.22	45	113.0	37.8	100	212.0	338	640	1184	643	1190	2174
7.78	46	114.8	43	110	230	343	650	1202	649	1200	2192
8.33	47	116.6	49	120	248	349	660	1220	654	1210	2210
8.89	48	118.4	54	130	266	354	670	1238	660	1220	2228
9.44	49	120.2	60	140	284	360	680	1256	666	1230	2246
10.00	50	122.0	66	150	302	366	690	1274	671	1240	2264

6M203000205

Temperature Conversion
Table 204

in	mm	in	mm	in	mm	in	mm	in	mm
0.0001	0.0025	0.001	0.0254	0.01	0.254	0.10	2.540	1.00	25.40
0.0002	0.0050	0.002	0.0508	0.02	0.508	0.20	5.080	2.00	50.80
0.0003	0.0076	0.003	0.0762	0.03	0.762	0.30	7.620	3.00	76.20
0.0004	0.0101	0.004	0.1015	0.04	1.015	0.40	10.160	4.00	101.60
0.0005	0.0127	0.005	0.1270	0.05	1.270	0.50	12.700	5.00	127.00
0.0006	0.0152	0.006	0.1524	0.06	1.524	0.60	15.240	6.00	152.40
0.0007	0.0177	0.007	0.1778	0.07	1.778	0.70	17.780	7.00	177.80
0.0008	0.0203	0.008	0.2032	0.08	2.032	0.80	20.320	8.00	203.20
0.0009	0.0228	0.009	0.2286	0.09	2.286	0.90	22.860	9.00	228.60

6M203000206

Inch Conversion
Table 205

1. Removal/Installation – Hoses and Tubing

A. Removal/Installation – Hard Tubing and Lines

(1) Replace original hard line tubing assemblies with parts manufactured per engineering drawings. Reference Section 29-00-00, Table 201 for tubing material and material specifications.

TUBE O.D.	RECOMMENDED BEND RADII		OPTIONAL BEND RADII
	3D	4D	6D
0.125	0.380	0.500	0.750
0.188	0.560	0.750	1.130
0.250	0.750	1.000	1.500
0.313	0.940	1.250	1.880
0.375	1.130	1.500	2.250
0.438	1.310	1.750	2.630
0.500	1.500	2.000	3.000
0.625	1.880	2.500	3.750
0.750	2.250	3.000	4.500
0.875	2.630	3.500	5.250
1.000	3.000	4.000	6.000
1.125	3.380	4.500	6.750
1.250	3.750	5.000	7.500
1.375	4.130	5.500	8.250
1.500	4.500	6.000	9.000
1.625	4.880	6.500	9.750
1.750	5.250	7.000	10.50
1.875	5.630	7.500	11.25
2.000	6.000	8.000	12.00

Tube Bend Radii Data
Table 201

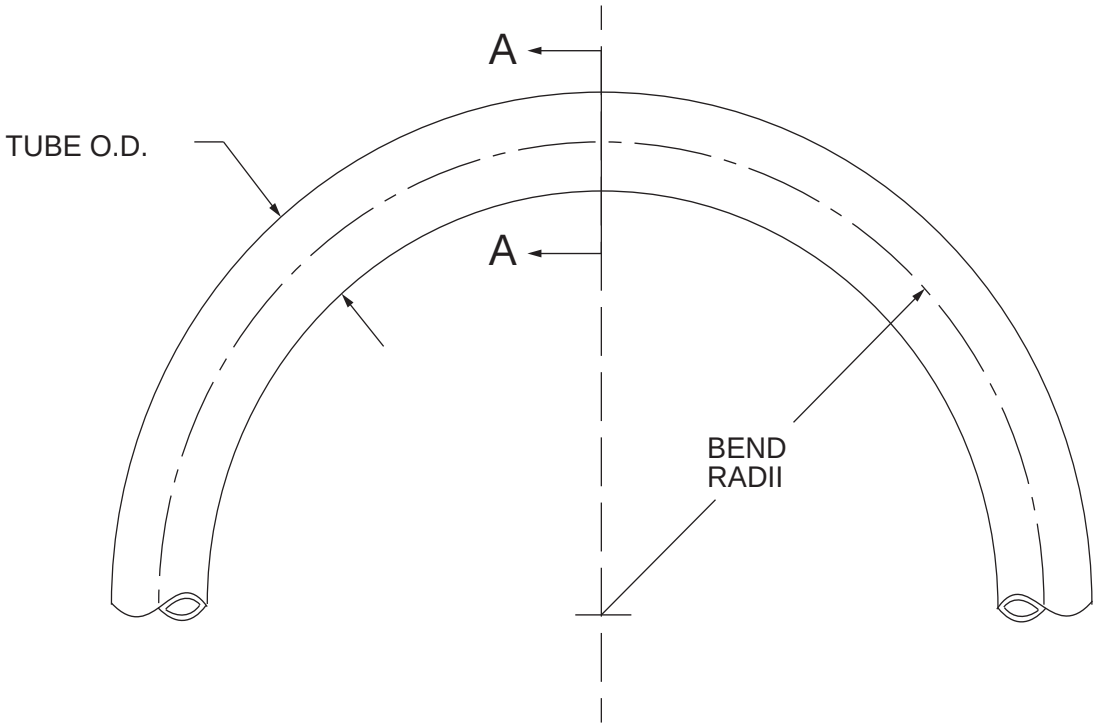
B. Bend Radius

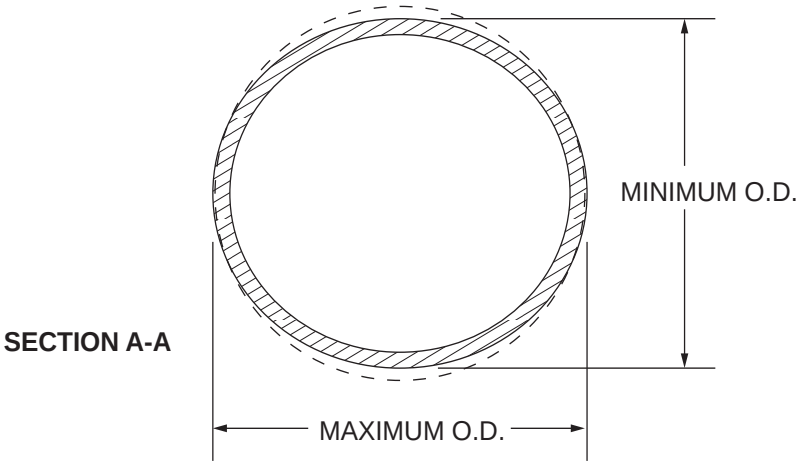
(1) Ensure bend radii of tubing is IAW the requirements of Table 201 (Figure 201).

(2) Use largest bend radii possible when forming to increase the service life of the tube.

(3) Bend radius flatness limitations (Figure 201, Section A-A).

(a) Flatness in the area of a tube bend shall be defined by the formula in Figure 201 below Section A-A.





$$\text{FLATNESS \%} = \frac{\text{MAX. O.D. - MIN. O.D.}}{\text{NOMINAL O.D. (ORIGINAL TUBE SIZE)}} \times 100$$

6M204000201

Tube Bend Radii / Flatness Formula
Figure 201

(b) When used in hydraulic systems with working pressure of 1500 psi or greater, shall not exceed flatness values of 5% for CRES or Aluminum tubing.

(c) Tube flatness for hydraulic systems with working pressure less than 1000 psi shall not exceed 10% for CRES and Aluminum tubing.

(d) Use of small bends (less than 3D) in hydraulic system tubing shall require engineering approval.

C. Squaring and Deburring Tube for MS Flareless Fittings

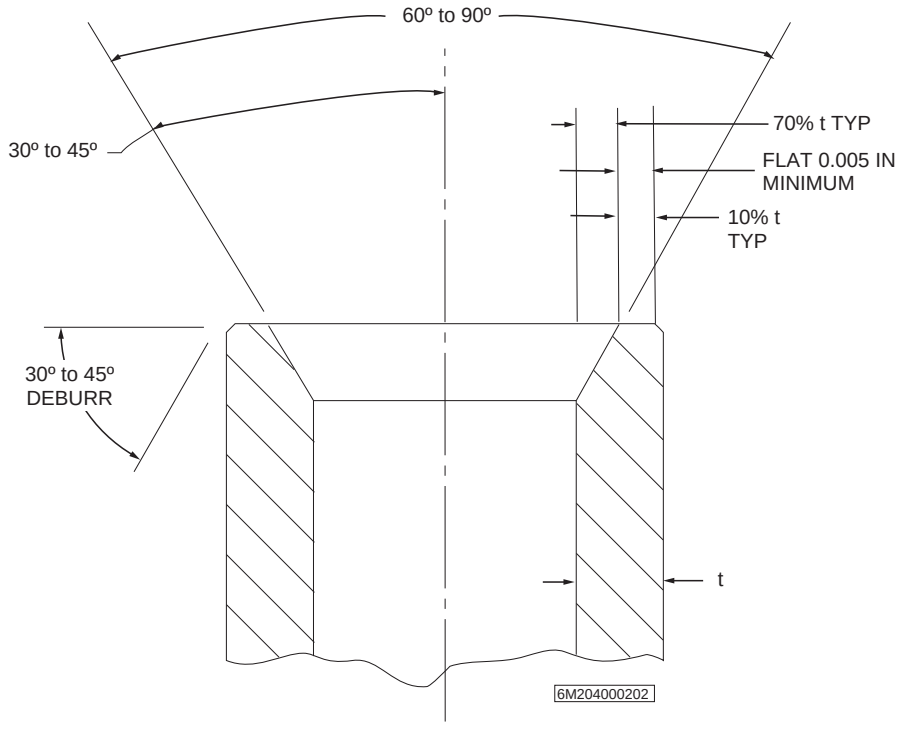
(1) Square and de-burr tube end to within one half degree (1/2°) of square to tube centerline. Chamfer inner and outer edges of the tube (see Figure 202).

(2) Ensure tubing and fitting ends to be permanently joined are clean, free of burrs, tool marks, and scratches on the tube O.D., and within roundness tolerances of the applicable tubing specification.

D. Flared Tubing

(1) Cut square, deburr interior and exterior edge and solvent clean tube ends prior to flaring.

(2) Form single flares IAW AS4330 Tubing, Flared, Standard Dimensions.



Tubing / Fitting End Chamfer Requirements
Figure 202

(3) Inspect flares in tubing for concentricity and alignment of flare with tube. Ensure flares are free of cracks and the inside surface is smooth and free from grooves, tool marks, or other surface defects.

(4) Ensure flares retain at least 80% of the original wall thickness. Superficial marks or checks on the external surface of the flare are not cause for rejection.

(5) Sleeves must contact the flared surface when pushed together by hand and be free to rotate when contacting the flare without interference at the bend radius (ref. Figure 203).

E. Tubing Connectors and Fittings

(1) Reject all fittings and/or connectors with imperfect threads, scored, pitted, or otherwise disfigured cones. Inspect the surface of all tubing before installing on aircraft.

(2) Check for cracks, gouges, dents, deep scratches, or extrusion marks. It is desirable to stock tubing in vertical racks where possible to help prevent damage to tubing.

F. Cleaning

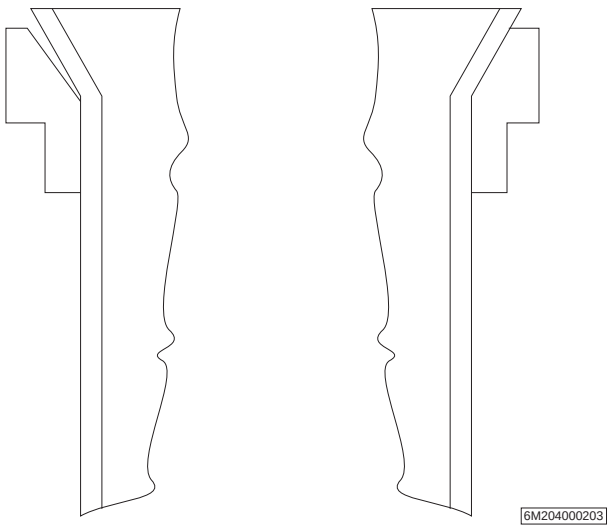
WARNING: UNDER NO CIRCUMSTANCES USE ANY CLEANING SOLUTION, OR ANY OTHER SOLUTION, INSIDE ANY PART OF OXYGEN SYSTEM WHILE IT IS INSTALLED, EXPLOSIONS AND/OR FIRES CAN RESULT.

(1) All tubing (including oxygen system) is to be cleaned after forming and flaring.

(a) Apply Klenedraw W-4166 or W-4100 soap to inside and outside of tubing.

(b) Pressure spray with clean hot water around fittings and inside tubing.

(c) Rinse with isopropyl alcohol.



Tube Flaring
Figure 203

(d) Dry with clean, dry air.

(2) After cleaning, seal all openings with thoroughly cleaned caps or plugs.

(3) Remove plugs and caps only as required to make immediate attachment to other components. When components are open, exercise extreme care to ensure no contamination enters the openings.

(4) When work is completed or temporarily stopped on a system, seal all remaining openings with clean suitable caps or plugs.

NOTE: Tools used for oxygen installations must be cleaned using same procedure as above.

G. Removal/Installation – Flexible Hose Assemblies

CAUTION: DO NOT TWIST FLEXIBLE HOSES DURING INSTALLATION.

(1) Replacement hose assemblies may be ordered by part number or locally manufactured by the operator. Flexible hose assemblies procured/manufactured by the operator, must be certified IAW the requirements specified in Chapter 20, this manual, and are to be installed and supported IAW FAA Publication AC43.13-1B (Aircraft Inspection and Repair).

CAUTION: FLEXIBLE HOSE ASSEMBLIES MAY NOT BE USED TO REPLACE HARD LINE TUBING BETWEEN BRAKE MASTER CYLINDERS AND BRAKES.

P/N Prefix	TSO-C53a				TSO-C75			
	Type A	Type B	Type C	Type D	Type II B-S/P	Type III A-S/P	Type III A-S/P-F	Type III B-S/P-F
36B	X							
36F	X							
86E				X				
94B	X							
94F			X					
111	X							
111F			X					
124						X		
130	X							
SS1000		X				X		
SS1001				X			X	
SS1008					X			
SS1009								X
SS1013							X	

TSO Certification Requirements
Table 202

(2) Replacement hoses used inside the engine compartment are seamless extruded Teflon inner tube, reinforced with one stainless steel wire braid and protected by one layer of

asbestos braid and conform to TSO-C53a or TSO-C75 (see Table 202 for type requirements). Replace hoses when they become brittle or collapse.

CAUTION: TEFLON TUBING USED IN THE ENGINE COMPARTMENT TENDS TO ASSUME A PERMANENT SET WHEN EXPOSED TO HIGH TEMPERATURES. DO NOT ATTEMPT TO STRAIGHTEN A HOSE THAT HAS BEEN IN SERVICE, AS ANY BENDING OR TWISTING MAY CAUSE WEAKENING OF THE TUBING WALL.

(3) Replacement hoses used outside the engine compartment are seamless synthetic rubber inner tubing, reinforced with one (1) fiber braid, one (1) braid of high tensile steel wire and covered with an oil resistant rubber impregnated fiber braid.

H. Removal/Installation – Clear Flexible Tubing

(1) Where clear flexible tubing is utilized (fluid drainage, etc.), replace with TYGON F-4040-A or EXCELON PVC fuel and lubricant tubing, with same diameter and wall thickness as original part.

I. Externally Swaged Repair of Hydraulic and Pneumatic Lines (Including Oxygen)

NOTE: This repair applies only to permanent joints using Permaswage fittings in the following materials and operating pressures. It is recommended that fittings and tube be of the same material. Stainless steel fittings with aluminum alloy tube should be avoided as damage may occur to the tube during torquing. In the case where limited parts are available, aluminum alloy fittings may be used with stainless steel tube, as long as an inspection program is in place to inspection for any galvanic corrosion between the contact surfaces of these dissimilar material components. Torque tightening would be governed by the aluminum alloy fittings.

(1) Tubing and Operating Pressures:

(a) Aluminum: Fitting 6061T6, AMS-QQ-A-225/8, and Tube 6061T6, AMS4083 (Up to 1500 psi) repair material for aluminum tube assemblies made of 5052-0 tubing.

(b) CRES: Fitting 21-6-9 AMS5656, and Tube 6061T6 AMS4083 (Up to 3000 psi) Fitting 21-6-9 AMS5656, and Tube 304 1/8 HD ASM5564 Class 1 or 304 HP, AMS5566 Type 1 (Up to 3000 psi).

(2) The material combinations shown in Table 203 are the only ones authorized for use in the hydraulic, pneumatic, or oxygen systems.

(3) Equipment:

(a) The equipment used shall meet the requirements for installation of the Permaswage fittings / connectors.

1 Permaswage System (Radial Swaged) - Permaswage Lightweight Tool (DLT)

2 Reference the Permaswage tool catalog for appropriate size tooling and tool manufacturing / inspection tool accessories, for fitting / connector being installed. Note: Permaswage tooling can be purchased in tooling kit variations which are tailored to the user's requirements (Refer to the Permaswage tool catalog).

Fitting Material	Tube Material	Max. Line Pressure	Tube O.D.									
			3/16	1/4	5/16	3/8	1/2	5/8	3/4	1	1 1/4	1 1/2
			-03	-04	-05	-06	-08	-10	-12	-16	-20	-24
			Tube Wall Thickness (inch)									
Alum 6061T6 AMS-QQ-A-225	Alum 6061T6 AMS4083	1500 psi		0.035	0.035	0.035	0.035					
		1000 psi		0.020		0.028	.028, 0.35	0.035	0.035	.042, .049		
		500 psi								0.035	0.035	
		100 psi										0.028
CRES 21-6-9 AMS5656	Alum 6061T6 AMS4083	3000 psi		0.035	0.042	0.049	0.064	0.065	0.095			
		1500 psi		0.035	0.035	0.035	0.035			0.065	0.065	
		1000 psi		0.020				0.035	0.035		0.065	
		500 psi		0.028	0.035	0.035	0.035	0.035	0.035	0.035	0.035	0.035
	CRES 304 1/8HD AMS5564 Class 1 or CRES 304 HP AMS5566 Type 1	3000 psi	0.016	0.020	0.020	0.028	0.035	0.042	0.058	0.065		
		1500 psi									0.049	0.065

Tubing Repair - Fitting and Tube Material
Table 203

(4) Operator Training:

(a) Operator should be trained in the use of the equipment and acceptance criteria prior to attempting the Permaswage fitting / connector repair / installation process. Permaswage offers this specialized training.

Permaswage USA – Headquarters
14800 S. Figueroa St.
Gardena, CA 90248
www.permaswage.com

(5) Materials:

(a) Eastman 910 Adhesive (or equivalent A-A-3097 Type I, Class II Adhesive)

(b) DuPont Krytox 240AC Lubricant (or equivalent MIL-PRF-27617 Type III; Grease, Fuel and Oxidizer Resistant)

(c) MIL-PRF-680 Type III, Degreasing Solvent

(6) Preparation:

CAUTION: THIS PROCESS USES TOOLS AND EQUIPMENT NOT COMPATIBLE WITH OXYGEN. WHEN USING THIS PROCESS WITH OXYGEN SYSTEMS, ENSURE LUBRICANTS, OILS, AND GREASES ARE NOT ALLOWED TO ENTER INTO OXYGEN PLUMBING, EQUIPMENT, OR FITTINGS UNLESS THEY ARE OXYGEN COMPATIBLE.

(a) Square tube end to within one half degree (1/2°) of square to the tube centerline.

(b) Deburr inside and outside of tube end using the appropriate Permaswage deburring tooling.

1 Select the appropriate size tube deburring tool, from the tube repair kit.

2 Lubricate deburr tool plug / stem only with KRYTOX 240AC.

3 Insert stem assembly into cutter end of the deburring tool. With the plunger depressed, screw the stem into the plunger until it is bottomed, finger tight (see Figure 209 view B).

4 Hold tube with one hand, depress plunger of deburring tool, and insert elastic plug into tube end. Release plunger and allow elastic plug to expand and seal tube.

5 While holding tube with one hand, rotate the knurled body in a clockwise direction, while applying slight pressure on the cutter. Continue to rotate cutter until tube is deburred.

NOTE: Excessive deburring will cause too deep a chamfer of the tube I.D. and is to be avoided. Chamfer width should generally not exceed one half (1/2) of the tubing wall thickness.

6 Without depressing the plunger, ease the tool from the tube until the first ridge of the plug is exposed. Wipe off the tube end and tool plug.

7 If tube has not been properly deburred, push tool back into tube and repeat step 3.

8 When tube has been properly deburred, pull the tool completely free of tube.

9 Remove stem subassembly from deburring tool by reversing procedure in step 3, for storage.

(c) Ensure tubing and fitting ends to be permanently joined are clean, free of burrs, tool marks, and scratches on the tube O.D., and within roundness tolerances of the applicable tubing specification.

(7) Insertion Band Marking (Figure 204):

(a) Mark an insertion band on each tube end to be permanently joined, to locate tube insertion into the permanent joint fitting.

(b) Tube insertion marking tools are shown in Figure 204, Detail B. The distance from tube end for each tube size is shown in Figure 204, Detail A.

(c) Set tool against tube end as sown in Figure 204, Detail C, for tube size to be marked, and using a D10058 or Panduit PX-0 felt tip pen, mark through the slot.

(d) Clean tube end to be swaged with methyl propyl ketone (MPK) or equivalent.

(e) Cap tube to prevent damage or contamination.

(8) Pre-Swage Assembly:

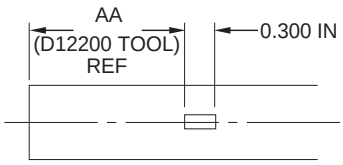
(a) Install enough lines and fittings in the aircraft so that any joint which has been swaged will not be unduly stressed by installation of additional lines.

NOTE: Do not remove fittings from their packaging until required for installation.

(b) Fasten any tube subassemblies with threaded connectors to the connecting fitting or component. Tighten B-nuts finger tight and clamp tube in place.

NOTE: All tube clamps need not be installed at this time, however, sufficient clamps shall be used to hold the tube in place. Visual inspection shall determine if the omitted clamps will fit, when installed, without excessive force.

CAUTION: DURING LOOSE INSTALLATION OF THE TUBE RUN, NEVER TWIST A LINE TO MAKE IT FIT. THIS MAY INADVERTENTLY CAUSE A TORSIONAL FORCE ON A PREVIOUSLY SWAGED FITTING WHICH MAY CAUSE THE FITTING TO WEEP OR LEAK LATER.

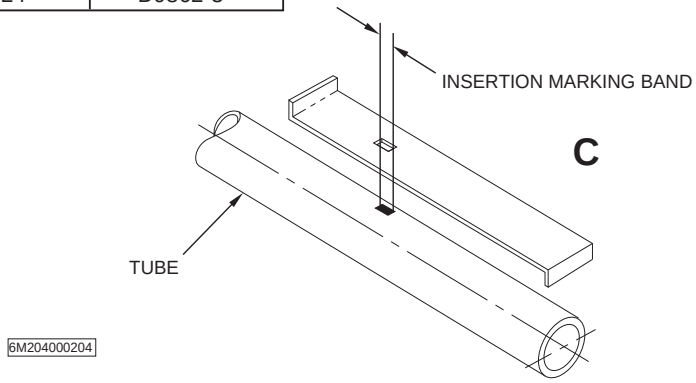


A

TUBE DIA.	TOOL D12200 AA +0.010 IN -0.000 IN
- 4	0.615 IN
- 6	0.690 IN
- 8	1.193 IN
-10	1.233 IN
-12	1.303 IN
-16	1.448 IN
-20	1.550 IN
-24	1.675 IN

B

TUBE SIZE	MARKING TOOL
- 4	D9862-1
- 5	D9862-1
- 6	D9862-1
- 8	D9862-2
-10	D9862-2
-12	D9862-2
-16	D9862-3
-20	D9862-3
-24	D9862-3



C

Insertion Band Marking
Figure 204

(c) Install the adjacent fitting-tube subassembly, inserting the tube end into the permaswage fitting end. Clamp the tube in place. Continue in this manner until the entire run of tubing (to the next threaded joint) is in place.

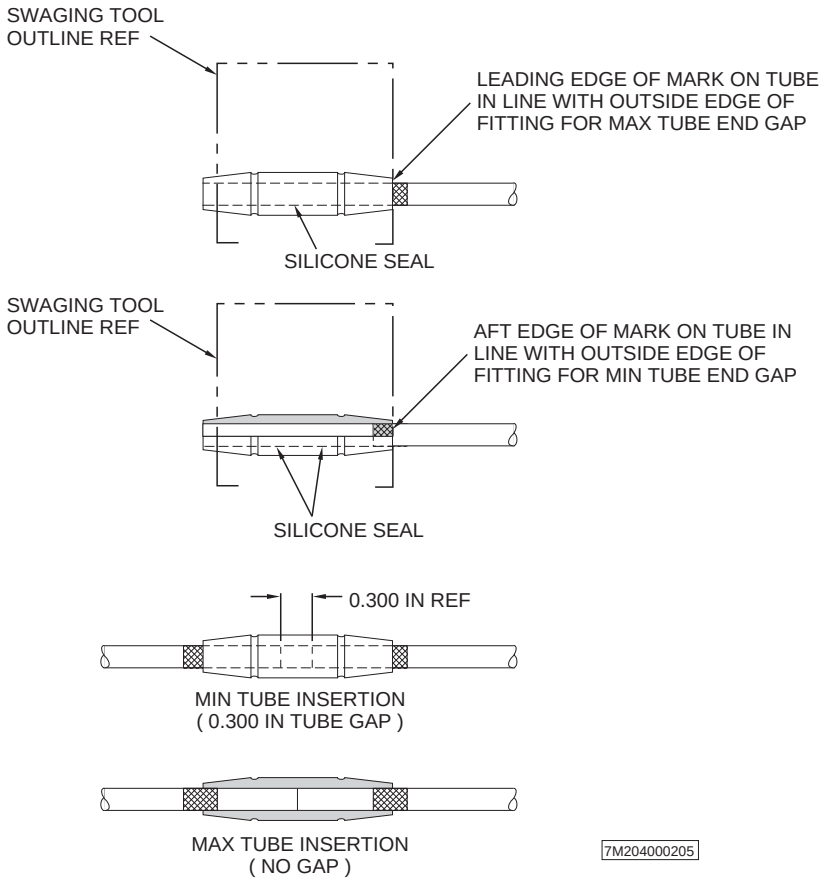
(d) In areas which contain tees and/or crosses, these lines must also be in place to the next threaded joint to ensure proper clocking of these fittings.

CAUTION: HOLD FITTINGS WITH SUITABLE WRENCHES WHEN TORQUING THE THREADED CONNECTOR TO AVOID DAMAGING THE TUBE OR JOINT.

(e) Tighten all threaded connectors. Torque IAW Chapter 91.

(f) Check the loose installation to ensure all fittings will be properly located after swaging by verifying that some portion of the insertion band is visible and some portion is covered by the fitting ends. Loosen tube clamps and adjust the tube and gaps to obtain proper fitting locations. Reclamp the tubes.

(g) Fittings shall not be swaged if they cannot be properly located.



Acceptable Insertion Limits (Union)
Figure 205

(h) Check the loose installation to ensure that adequate clearance will be maintained between other tubes, components, and structure.

(i) Swage each fitting end in turn IAW **1.**, I., (12).

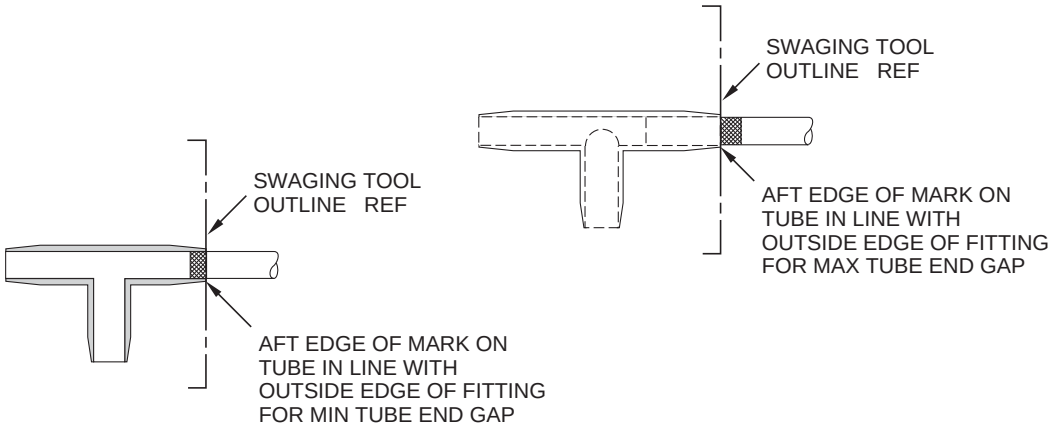
NOTE: To obtain tool clearance, it is permissible to loosen the tube clamps to allow tubes to be moved away from obstructions during the swaging operation. However, movement of the tubes shall not cause misalignment between the tube and fitting. Ensure no twisting force is applied on any previously swaged fitting.

(9) Tube/Fitting Insertion (Union) Using D12200 Tooling:

(a) Using the 0.300 inch tube end gap tolerance with union fittings, set fitting into tool (see Figure 205) with edge of tool falling within the marked band. The limits shown in Figure 205 must be complied with to provide a reliable leak free joint.

(10) Tube/Fitting Insertion (Shaped Fitting) Using D12200 Tooling:

(a) Using the 0.300 inch tube end gap tolerance with shaped fittings, set fitting into tool (see Figure 206) with edge of tool falling within the marked area of the tube.



Acceptable Insertion Limits (Shaped Fitting)
Figure 206

(11) Unacceptable Limits – Unions and Shapes:

(a) Unacceptable limits with respect to tube insertion depths into fittings to be swaged may result in the following (see Figure 207).

1 Insufficient insertion may result in a leaking joint.

2 Excessive insertion will leave insufficient depth for inserting connecting tube, or may restrict or block flow in shaped fittings.

(12) Swaging:

(a) Accomplish swaging using tooling shown in Table 204.

Tool Size	Tool Number
-4	D12204
-5	D12205
-6	D12206
-8	D12208
-10	D12210
-12	D12212
-16	D12216
-20	D12220

Swaging Tools
Table 204

(b) The material combinations shown in Table 203 are the only ones authorized for use in the hydraulic, pneumatic, or oxygen systems.

(c) Fitting to Tube Installation.

1 Select proper size tool body and die halves for tube size to be swaged (see Table 204).

2 Assemble die halves and tool required.

CAUTION: THE UPPER DIE BLOCK MUST BE LOCKED INTO POSITION BY THE LOCK LEVER, PRIOR TO INITIATING OPERATION OF THE HYDRAULIC POWER SOURCE.

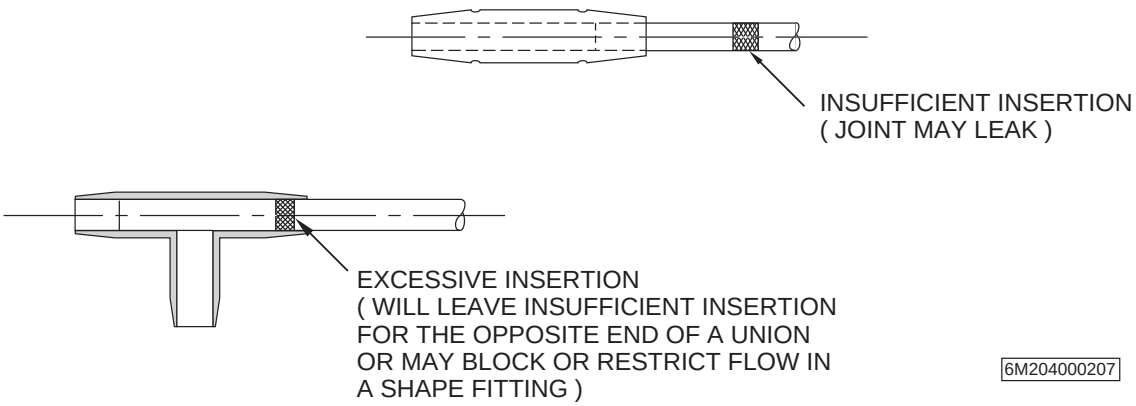
3 Connect swaging tool to a hydraulic power source supplying 5500, ± 250 psi (386.65, ± 17.575 kg/cm^2).

4 Before placing a fitting over the tube ends for swaging, ensure the 0.300 inch gap allowable at tube end is not exceeded. Do not swage fitting if gap is exceeded.

5 Insert lower die half into tool. Place the fitting into the die half (for bench swaging). Insert upper die half into tool. Lock upper die half into tool by engaging safety lock lever.

6 Insert the tube into the fitting, adjusting the tube position so that the band on the tube aligns with the tool (see Figure 205).

NOTE: Do not slide the fitting along the tube more than is necessary when positioning the tool within the locating band.



Unacceptable Insertion Limits
Figure 207

7 Activate the hydraulic power source supplying 5500, ± 250 psi (386.65, ± 17.575 kg/cm^2) to the swaging tool. The swaging tool dies will automatically swage the fitting onto the tube.

NOTE: Do not cycle the swaging tool if the upper die block is not locked in place by the lock lever.

8 When swaging action is complete, release pressure and remove upper die block from the tool and the tool from the fitting.

(13) Inspection:

(a) When using the D12200 tooling, check each completed swage joint IAW A and B dimensions in Figure 208, with the proper size swage inspection gage before continuing with the next joint.

(b) Protect fittings being swaged to tube assemblies, where one or more fitting ends remain unswaged, with suitable closures to prevent contamination until they are subsequently swaged. Do not clean unswaged ends of fittings prior to swaging.

(c) It is permissible to reswage fittings to meet the dimensional requirements of Figure 208 A and B. Reswage only with proper size tooling. Before reswaging a joint, check for adequate swage pressure, and for foreign materials between die set segments. Do not reswage a joint more than once.

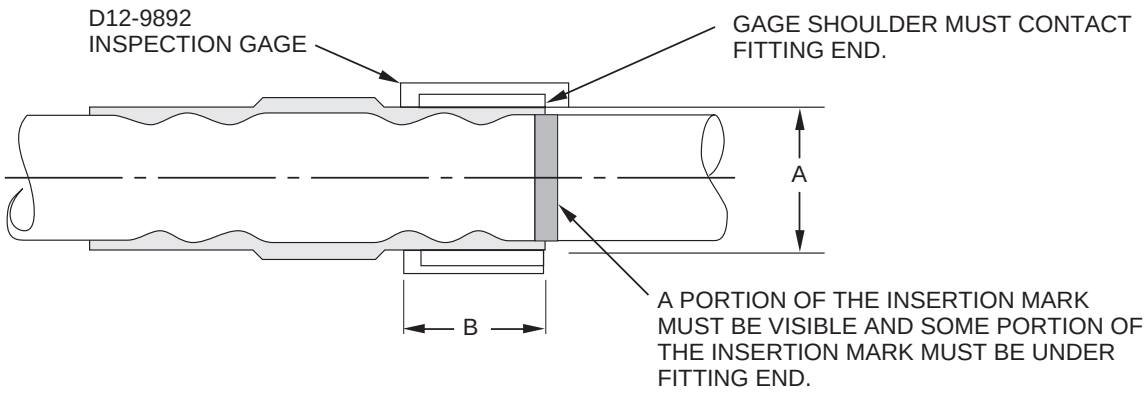
(d) When swaging aluminum fittings, it is permissible to have extruded bare aluminum splines around the outside of the fitting.

(14) Modification Of Existing Lines Using Permaswage Fittings:

NOTE: Modify tubing assemblies using the following procedures:

(a) Marking:

1 Mark tube for cutting.



TUBE DIA.	SIZE	MAXIMUM SWAGED DIA. "A" DIMENSION	MINIMUM SWAGED LENGTH "B" DIMENSION	INSPECTION GAGE
1/4	-4	0.315	0.460	D12-9892-4
5/16	-5	0.381	0.500	D12-9892-5
3/8	-6	0.447	0.530	D12-9892-6
1/2	-8	0.606	1.020	D12-9892-8
5/8	-10	0.735	1.020	D12-9892-10
3/4	-12	0.863	1.020	D12-9892-12
1	-16	1.144	1.160	D12-9892-16
1-1/4	-20	1.390	1.406	D12-9892-20
1-1/2	-24	1.680	1.420	D12-9892-24

D12200 Tooling Check
Figure 208

(b) Cutting:

1 Select appropriate chipless cutter P/N D9852, D9853, or D9872 located in tube repair kit P/N D12323-1-3456, D12324-1-812, or D12325-1-124, for the respective tube size (see Figure 209 view A).

2 Position cutter wheel clear of cutter lead opening, by backing off the drivescrew using socket wrench (see Figure 209 view A).

3 Position cutter on tube with cutter wheel at marking on tube location to be cut.

4 Tighten drivescrew until positive contact is made by cutter wheel on tube, then torque 1/4 turn (see Figure 209 view A).

5 Rotate cutter ratchet handle until there is noticeable ease of rotation. Check that cutter wheel is on tube marking made earlier.

6 Tighten drivescrew another 1/8 turn.

7 Repeat steps 5 and 6 until cut has been completed.

NOTE: Do not overtighten drivescrew during cutting. In soft tubing a large burr will result. In hard tubing the cutter wheel will experience excessive wear or breakage and result in substandard cut of the tube.

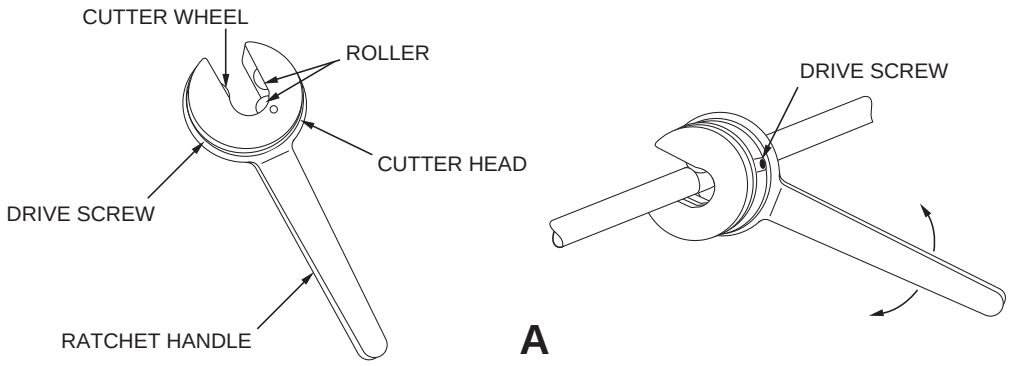
(c) Deburring:

1 Select appropriate deburring tool, D9849, D9850, or D9851 from the tube repair kit.

2 Lubricate stem only with KRYTOX 240AC.

3 Insert stem assembly into cutter end of the deburring tool. With the plunger depressed, screw the stem into the plunger until it is bottomed, finger tight (see Figure 209 view B).

4 Hold tube with one hand, depress plunger of deburring tool, and insert elastic plug into tube end. Release plunger and allow elastic plug to expand and seal tube.



Cutter/Deburr Tool Operation
Figure 209

NOTE: Excessive deburring will cause too deep a chamfer of the tube I.D. and is to be avoided. Chamfer width should generally not exceed one-half (1/2) of the tubing wall thickness.

5 While holding tube with one hand, rotate the knurled body in a clockwise direction, while applying slight pressure on the cutter. Continue to rotate cutter until tube is deburred.

6 Without depressing the plunger, ease the tool from the tube until the first ridge of the plug is exposed. Wipe off the tube end and plug.

7 If tube has not been properly deburred, push tool back into tube and repeat step 5.

8 When tube has been properly deburred, pull the tool completely free of tube.

9 Remove stem subassembly from deburring tool by reversing procedure in step 3.

(d) Marking:

1 Mark and swage tube assembly IAW **1.**, I., (6) through (13).

(e) Temporary Attachment of Permaswage Fittings:

NOTE: In certain instances, where swaging cannot be performed on the airplane, the tube assembly must be removed to swage the fitting to the tube.

1 Install tube and permaswage fitting(s) on the airplane, and snug up fittings to the connecting components. Install tubing supports and ensure the assembly is properly aligned.

2 Apply one (1), (maximum two [2]) drops of Eastman 910 (or equivalent A-A-3097 Type I, Class II Adhesive), 180° apart, where the fitting end and tube meet. This will temporarily hold the fitting to the tube at its required orientation and location.

NOTE: Do not apply excessive amount of bonding agent.

3 Allow five (5) minutes to dry.

4 Carefully remove the tube/fitting assembly from the airplane. Ensure that fitting has not been disturbed.

5 Swage IAW **1.**, I., (12).

6 Install tube on airplane.

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MAINTENANCE MANUAL

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21-50-10	MP	201	JAN 01/01	21-70-30	MP	202	JUN 01/92
21-50-10	MP	202	JUN 01/92	21-70-30	MP	203	JUN 01/92
21-50-10	MP	203	JUN 01/92	21-70-40	DO	1	JUN 01/92
21-50-10	MP	204	JUN 01/92	21-70-40	DO	2	JUN 01/92
21-60-00	DO	1	JUN 01/92	21-70-40	MP	201	JUN 01/92
21-60-00	DO	2	JUN 01/92	21-70-40	MP	202	JUN 01/92
21-60-00	DO	3	JUN 01/92	21-70-40	MP	203	JUN 01/92
21-60-00	MP	201	DEC 01/01				

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MAINTENANCE MANUAL
ENVIRONMENTAL CONTROL – DESCRIPTION AND OPERATION

1. General

A. The air conditioning system supplies cold air, hot air, and fresh air to the cabin, and cockpit. Two independent systems are provided. Each system is capable of providing the complete air conditioning requirements for the airplane.

B. Bleed air from each engine drives cooling turbines that provide cool air. Bleed air is mixed with cold air to provide hot air. Either bleed air system may be operated on the ground when the respective engine is operating. Fresh air is supplied by a blower assembly located in the nose baggage compartment.

C. Ducts within the fuselage distribute the airflow to passengers and crew. An automatic temperature control system senses and regulates the temperature within the aircraft.

D. Seven environmental control systems are covered in this chapter. The system used in your particular aircraft can be identified by consulting the model and effectivity columns located in the lower left corner.

2. Compression (21-10-00)

A. Two pressure limiting flow control valves (Fig 1, item 1) and slave valves (Fig 1, item 2) are used in conjunction with two flow limiting venturis (Fig 1, item 3).

3. Air Distribution (21-20-00)

A. The distribution system delivers air throughout the aircraft. The components of the distribution system are, duct check valves (Fig 1, item 10), sensor assembly (Fig 1, item 12), cold air ducts (Fig 1, item 13), and the conditioned air ducts (Fig 1, item 14). The fresh air system is also part of the distribution system.

4. Pressurization (21-30-00)

A. The aircraft cabin is pressurized to enhance passenger comfort. The pressurization system automatically compensates for increasing aircraft altitude by maintaining cabin altitude as near to field elevation as possible. The cabin altitude will remain at sea level until the aircraft altitude reaches approximately 17,000 feet.

B. The pressurization system is equipped with an emergency relief valve, automatic over pressurization relief, and negative pressure relief.

5. Heating (21-40-00)

A. Two electrically operated mixer valve assemblies (Fig 1, item 9) control the flow of hot air into the cabin.

6. Cooling (21-50-10)

A. Two cooling turbines (Fig 1, item 4) provide cool air for the system.

7. Temperature Control (21-60-00)

A. Cabin temperature control positions the mixing valves (Fig 1, item 9). Components of the temperature control system are an electronic controller (Fig 1, item 16), cabin air temperature sensor and blower (integral with the electronic controller unit), duct temperature sensors (Fig 1, item 12), a temperature mode selector switch (Fig 1, item 18) and an automatic temperature controller (Fig 1, item 19).

8. Moisture and Contamination Control (21-70-00)

A. Pneumatically powered anti-icing provides moisture and contamination protection for the Environmental Control System (ECS). The components of the system include the hot air controlling slave valves (Fig 1, item 4), the temperature sensing master valves (Fig 1, item 8), the supply pressure regulators (Fig 1, item 5) and the water separators (Fig 1, item 7). The system is designed to maintain the cold air duct temperature at 37°F (2.78°C).

9. Component Location (Figure 1)

A. Figure 1 may be used as a locator for various ECS components within the nacelles, wings, and center section. Left and right system installations are mirror images. Only the left installation is shown. The figure represents a theoretical view from the top of the wing, a view that is not possible on the airplane.

10. System 1 – Components and Description (Figures 1 and 2)

NOTE: Refer to Figures 1 and 2 to determine location of components. The unshaded areas indicate access panels on the lower side of the wings and center section. In using these figures to locate components, consider that they represent theoretical views from the top of the wing, a view which is not possible on the aircraft.

A. Compression

(1) Two pressure limiting flow control valves (Fig 1, item 1) and slave valves (Fig 1, item 2) are used in conjunction with two flow limiting venturis (Fig 1, item 3).

B. Distribution

(1) The distribution system routes air throughout the aircraft. Components of the distribution system are duct check valves (Fig 1, item 10), conditioned air mufflers (Fig 1, item 11), cold air ducts (Fig 1, item 13), and conditioned air ducts (Fig 1, item 14). The fresh air system is part of the distribution system.

(2) Duct check valves (Fig 2, item 5) are located in the aircraft center section and are accessible through the two inboard panels.

C. Heating

(1) Two electrically operated mixing valves (Fig 1, item 9) control the flow of tempered air into the cabin.

(2) Hot air mixing valves (Fig 2, item 4) are accessible through two inboard access panels.

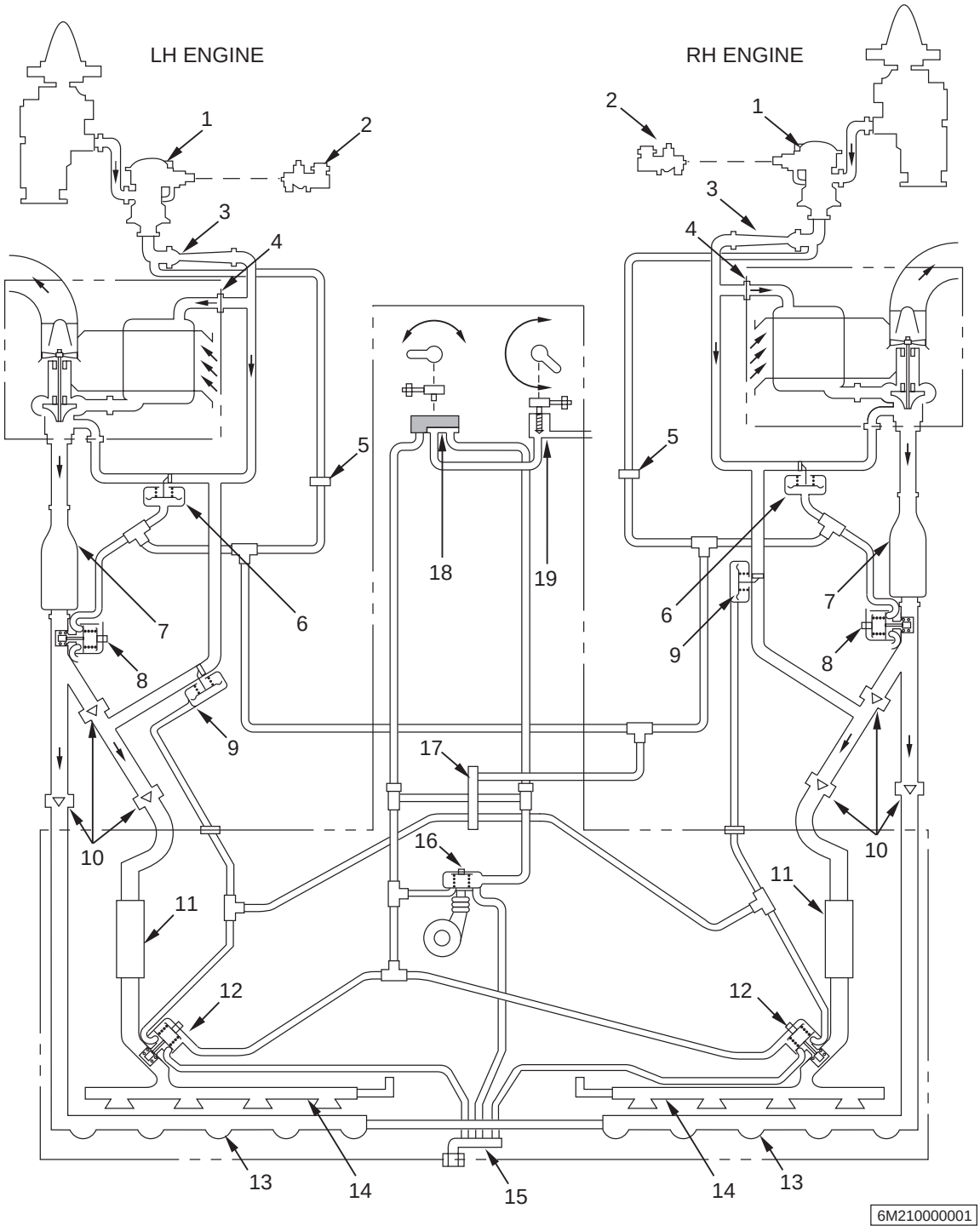
D. Cooling

(1) Two cooling turbines (Fig 1, item 4), provide cool air for the system.

(2) Cooling turbines (Fig 2, item 7) are located in the wings, and are accessible through the outboard access panel on each wing.

E. Temperature Control

(1) An electrically operated temperature control system positions the mixing valves. Components of the system are an electronic controller (Fig 1, item 16), cabin air temperature and blower (integral with the electronic controller unit), duct temperature sensors (Fig 1, item 12), temperature mode selector switch (Fig 1, item 18), automatic temperature control (Fig 1, item 19), and cabin air temperature sensor (Fig 1, item 17).

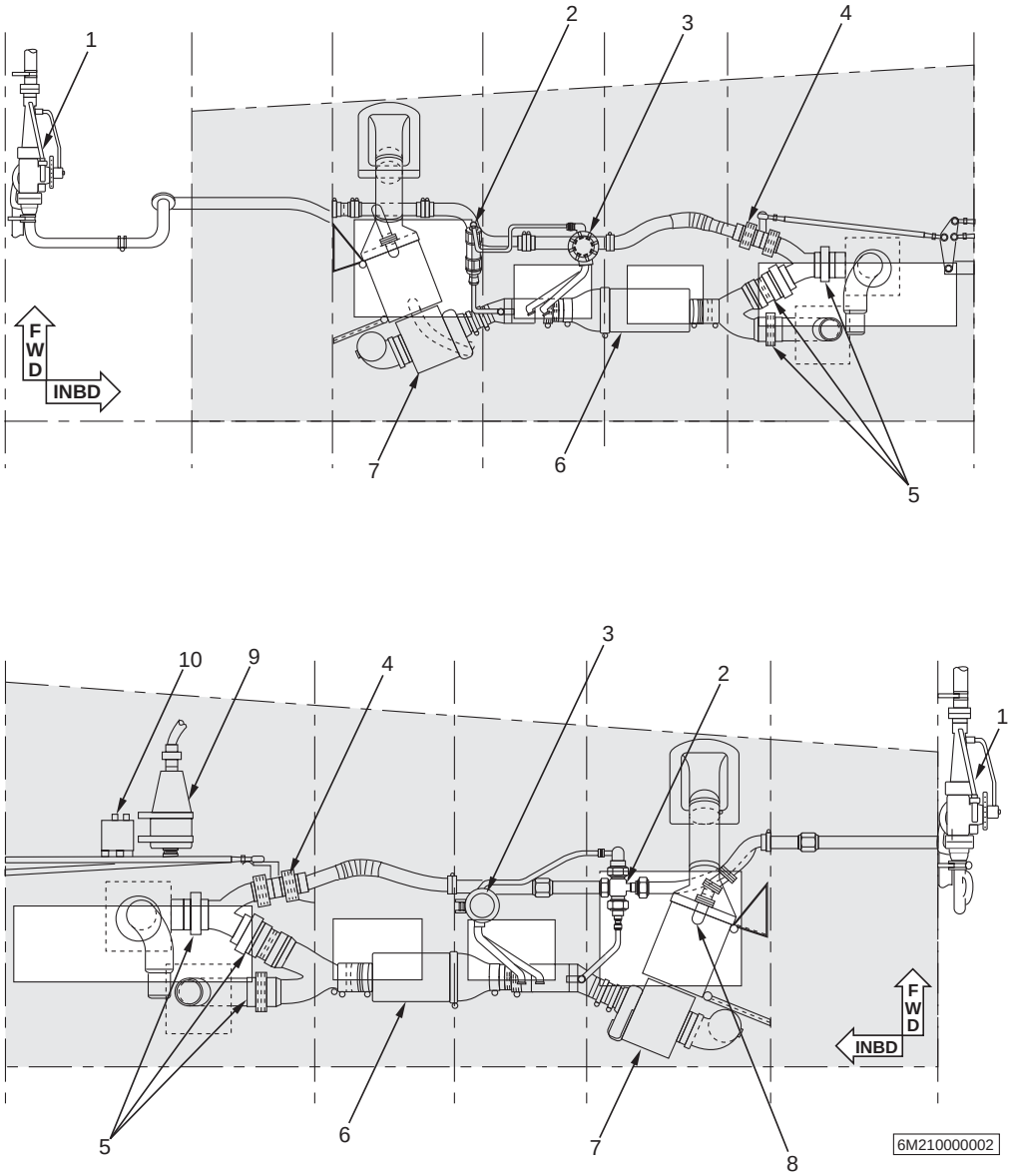


System 1 – Schematic
Figure 1

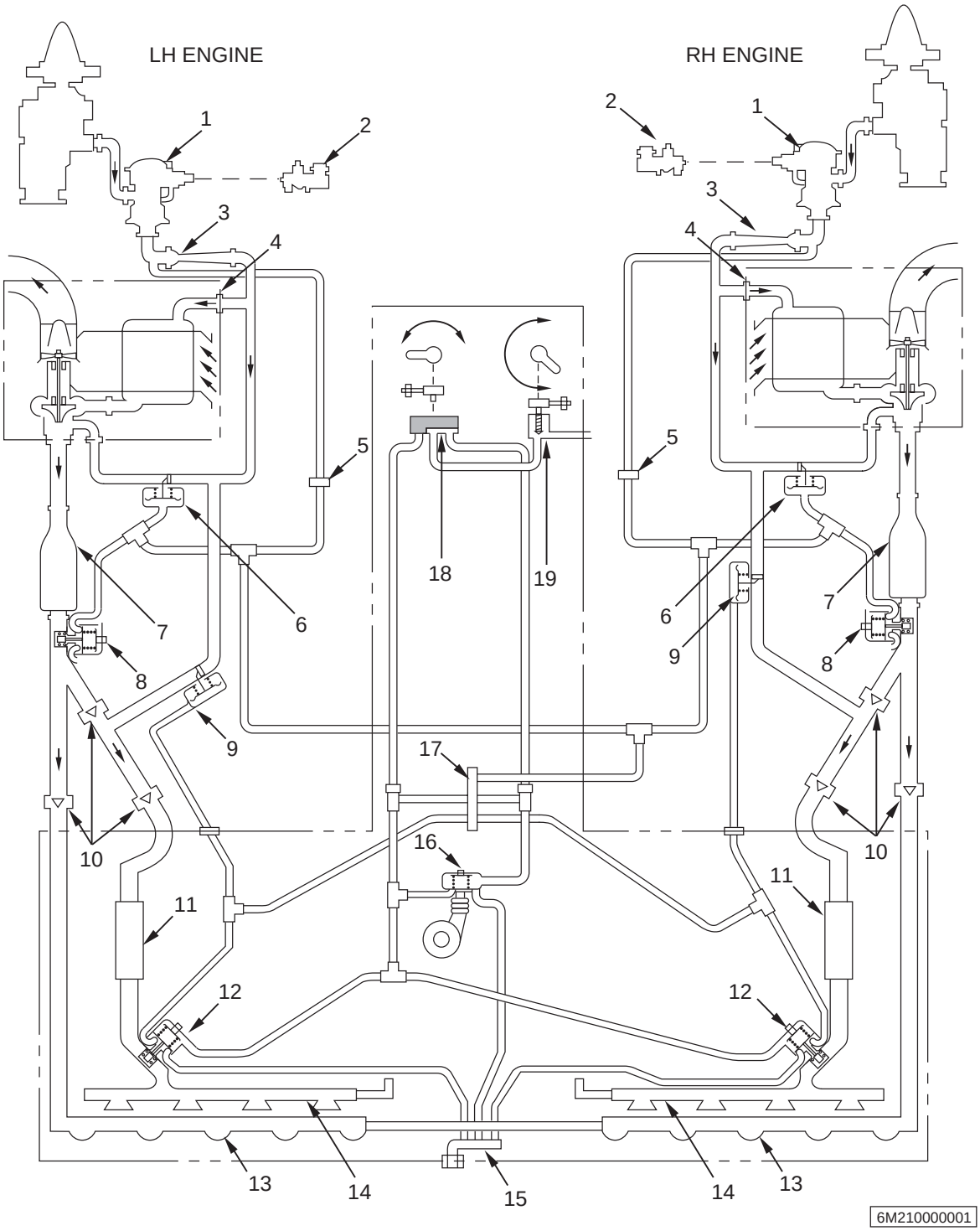
F. Moisture and Contaminant Control

(1) The differential-pressure-sensing system deices the cold air output of the cooling turbines. Components (Figure 2) of the system are hot air controlling slave valves (Fig 2, item 4), differential sensing master valves (Fig 2, item 3), and water separators (Fig 2, item 6).

(2) Hot air controlling slave valves (Fig 2, item 2) are accessible through the first (outboard) access panel of each wing. The differential sensing master valves (Fig 2, item 3) are accessible through the second access panel (moving inboard). The water separators (Fig 2, item 6) are accessible through the third access panel.



System 1 Components, LH and RH Sides
Figure 2



System 2 – Components
Figure 3

11. System 2 – Components and Description (Figures 3 and 4)

NOTE: Refer to Figures 3 and 4 to determine location of components. The unshaded areas indicate access panels on the lower side of the wings and center section. In using these figures to locate components, consider that they represent theoretical views from the top of the wing, a view which is not possible on the aircraft.

A. Compression

(1) Two pressure limiting flow control valves (Fig 3, item 1) and slave valves (Fig 3, item 2) are used in conjunction with two flow limiting venturis (Fig 3, item 3).

B. Distribution

(1) The distribution system routes air throughout the aircraft. Components of the distribution system are duct check valves (Fig 3, item 10), conditioned air mufflers (Fig 3, item 11), cold air ducts (Fig 3, item 13), and conditioned air ducts (Fig 3, item 14). The fresh air system is part of the distribution system.

(2) Duct check valves (Fig 4, item 4) are located in the aircraft center section and are accessible through the two inboard panels.

C. Heating

(1) Two electrically operated mixing valves (Fig 3, item 9) control the flow of tempered air into the cabin.

(2) Hot air mixing valves (Fig 4, item 3) are accessible through two inboard access panels.

D. Cooling

(1) Two cooling turbines (Fig 3, item 4), provide cool air for the system.

(2) Cooling turbines (Fig 4, item 7) are located in the wings, and are accessible through the outboard access panel on each wing.

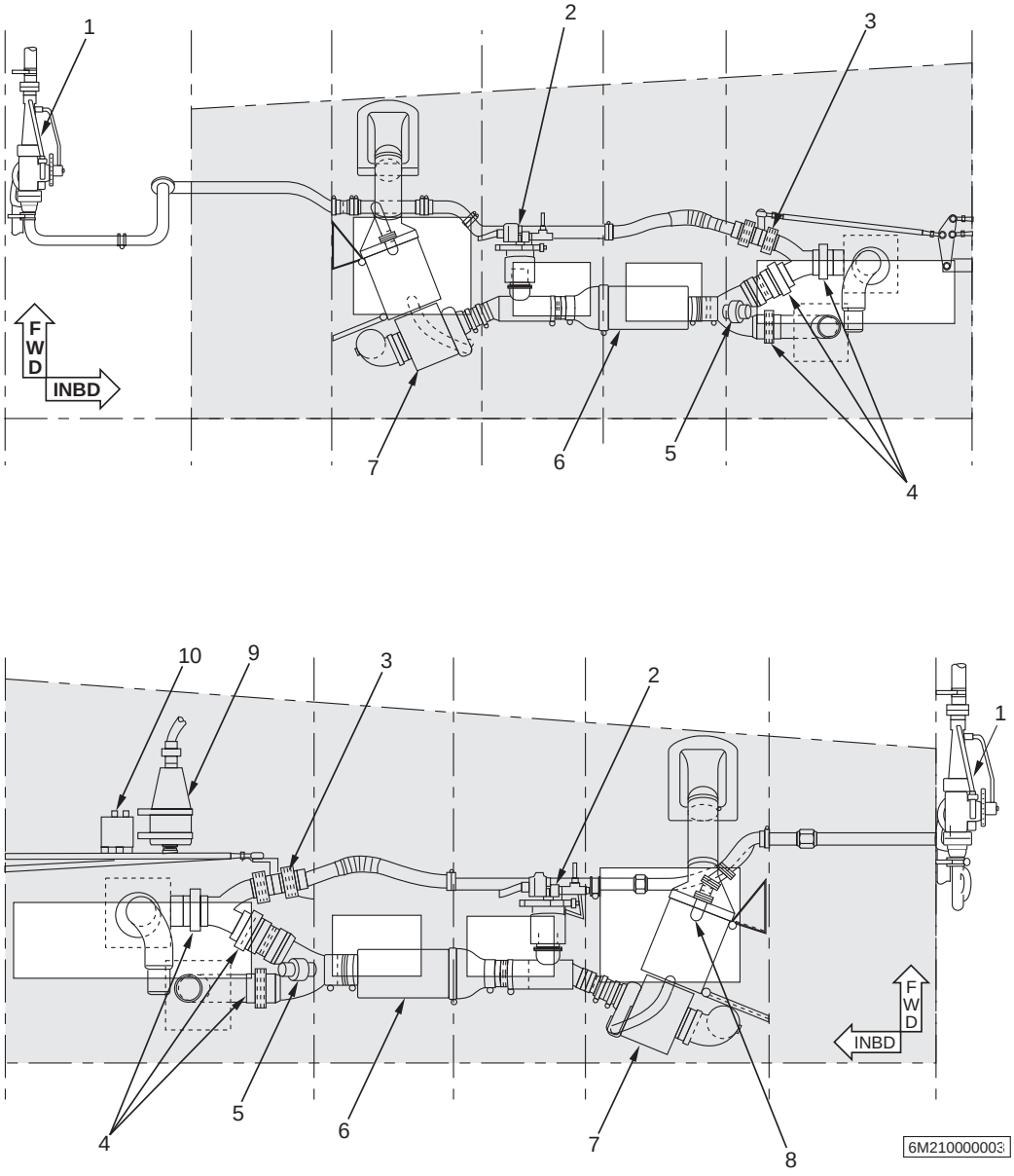
E. Temperature Control

(1) An electrically operated temperature control system positions the mixing valves. Components of the system are an electronic controller (Fig 3, item 16), cabin air temperature and blower (integral with the electronic controller unit), duct temperature sensors (Fig 3, item 12), temperature mode selector switch (Fig 3, item 18), automatic temperature control (Fig 3, item 19), and cabin air temperature sensor (Fig 3, item 17).

F. Moisture and Contaminant Control

(1) Components of the electrical powered anti-icing system are hot air conditioning slave valves (Fig 4, item 2), temperature sensing master valves (Fig 4, item 5), and water separators (Fig 4, item 6). The system maintains the cold air duct temperature at 35°F (1.67°C).

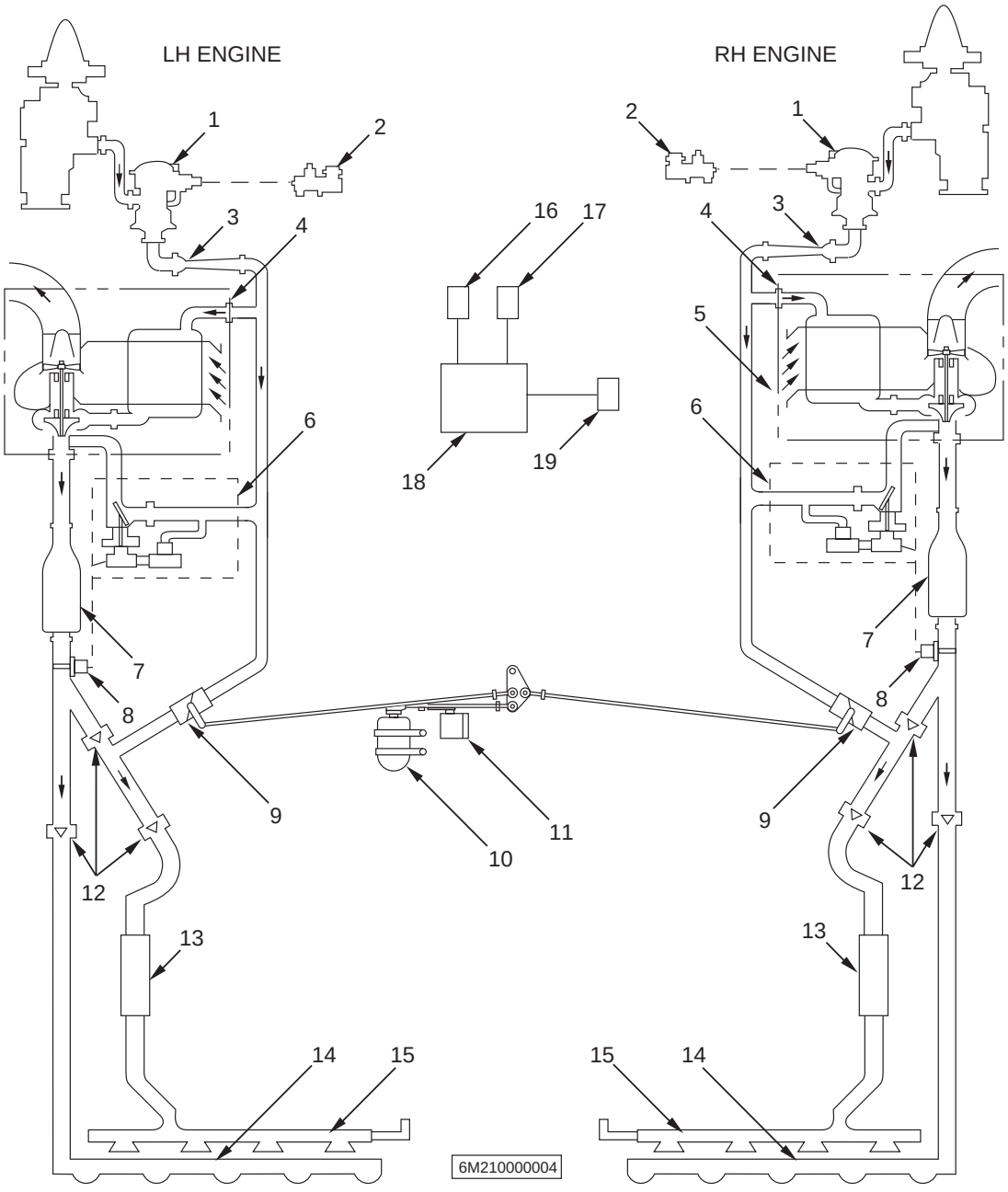
(2) The hot air controlling slave valves (Fig 4, item 2) are accessible through the second (moving inboard) panel of each wing. Water separators (Fig 4, item 6) are accessible through the third panel (moving inboard). Temperature sensing master valves (Fig 4, item 5) are accessible through the inboard access panel.



System 2 Components, LH and RH Sides
Figure 4

12. System 3 – Components and Description (Figures 5 and 6)

NOTE: Refer to Figures 5 and 6 to determine the location of components. The unshaded areas indicate access panels on the lower side of the wings and center section. In using these figures to locate components, consider that they represent theoretical views from the top of the wing, a view which is not possible on the aircraft.



System 3 – Schematic
Figure 5

A. Compression

(1) Two pressure limiting flow control valves (Fig 5, item 1) and slave valves (Fig 5, Item 2) are used in conjunction with two flow limiting venturis (Fig 5, item 3).

B. Distribution

(1) The distribution system routes air throughout the aircraft. Components of the distribution system are duct check valves (Fig 5, item 12), conditioned air mufflers (Fig 5, item 13), cold air ducts (Fig 5, item 14), conditioned air ducts (Fig 5, item 15). The fresh air system is part of the distribution system.

(2) Duct check valves (Fig 6, item 6) are located in the aircraft center section, and accessible through the inboard access panel.

C. Heating

(1) Two mixing valves (Fig 5, item 9) are connected to a common actuator (Fig 5, item 10) and a follow-up potentiometer (Fig 5, item 11). These mixing valves are used to control the flow of tempered air in the cabin.

(2) The hot air mixing valves (Fig 6, item 5) are accessible through the inboard panel. Mixing valve actuator and follow-up potentiometer (Fig 6, items 11 and 12) are located in the right center section, and are accessible through the inboard panel on the right side and the right wing-to-fuselage fairing.

D. Cooling

(1) Two cooling turbines (Fig 5, item 4) provide cool air for the system.

(2) Cooling turbines (Fig 6, item 9) are located in the wings, and are accessible through the outboard access panel on each wing.

E. Temperature Control

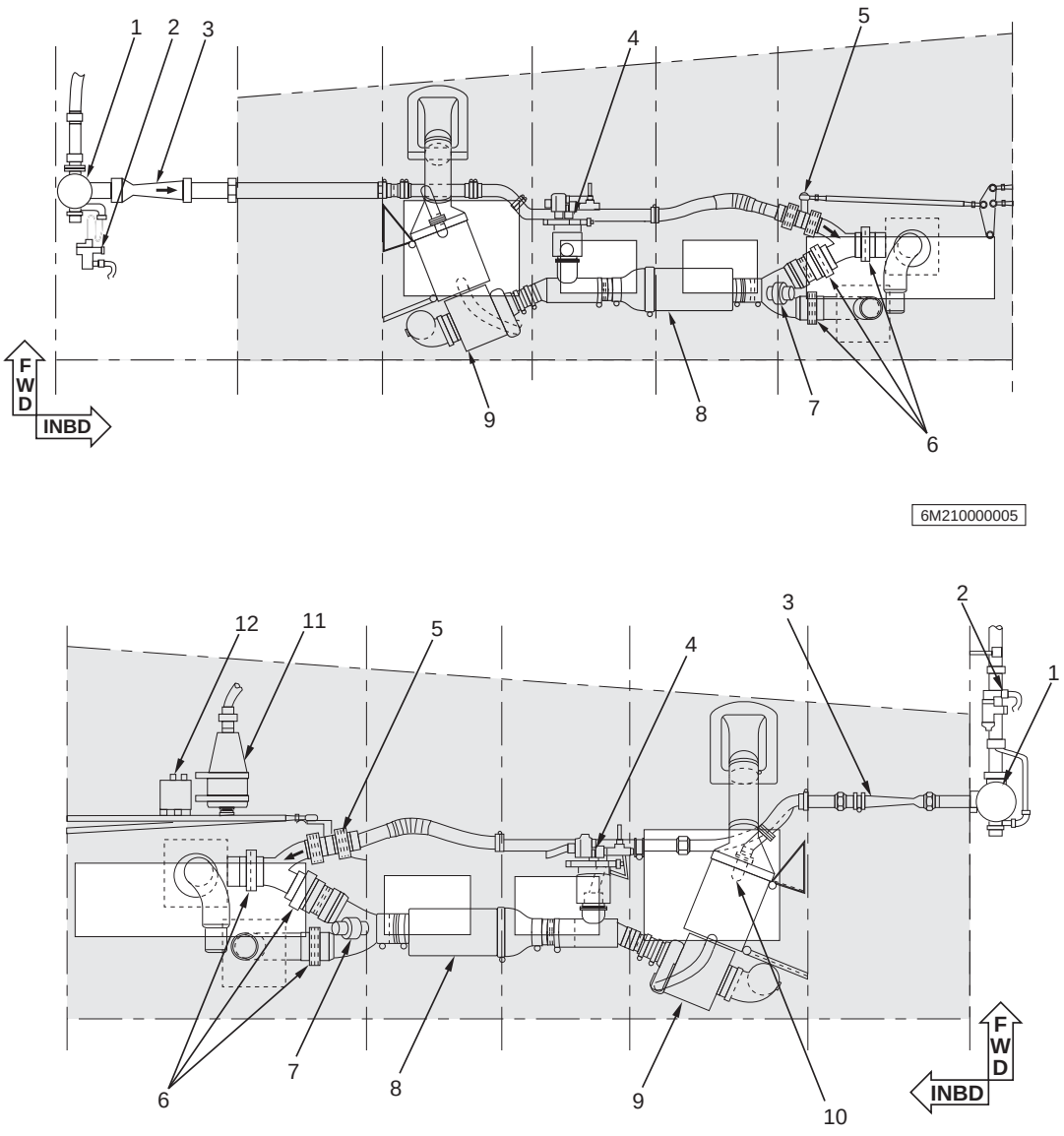
(1) An electrically operated temperature control system positions the mixing valves. Components of the system are ambient air sensor (Fig 6, item 10), mode selector switch (Fig 5, item 16), automatic temperature control (Fig 5, item 17), temperature control box (Fig 5, item 18), and cabin air temperature sensor (Fig 5, item 19).

(2) The ambient air temperature sensor (Fig 6, item 10). Is located in the outside air intake for the right cooling turbine, and is accessible through the right cooling turbine access panel.

F. Moisture and Contaminant Control

(1) Components of the electrical powered anti-icing system are hot air conditioning slave valves (Fig 6, item 5), temperature sensing master valves (Fig 6, item 7), water separators (Fig 6, item 8). The system maintains the cold air duct temperature at 35°F (1.67°C).

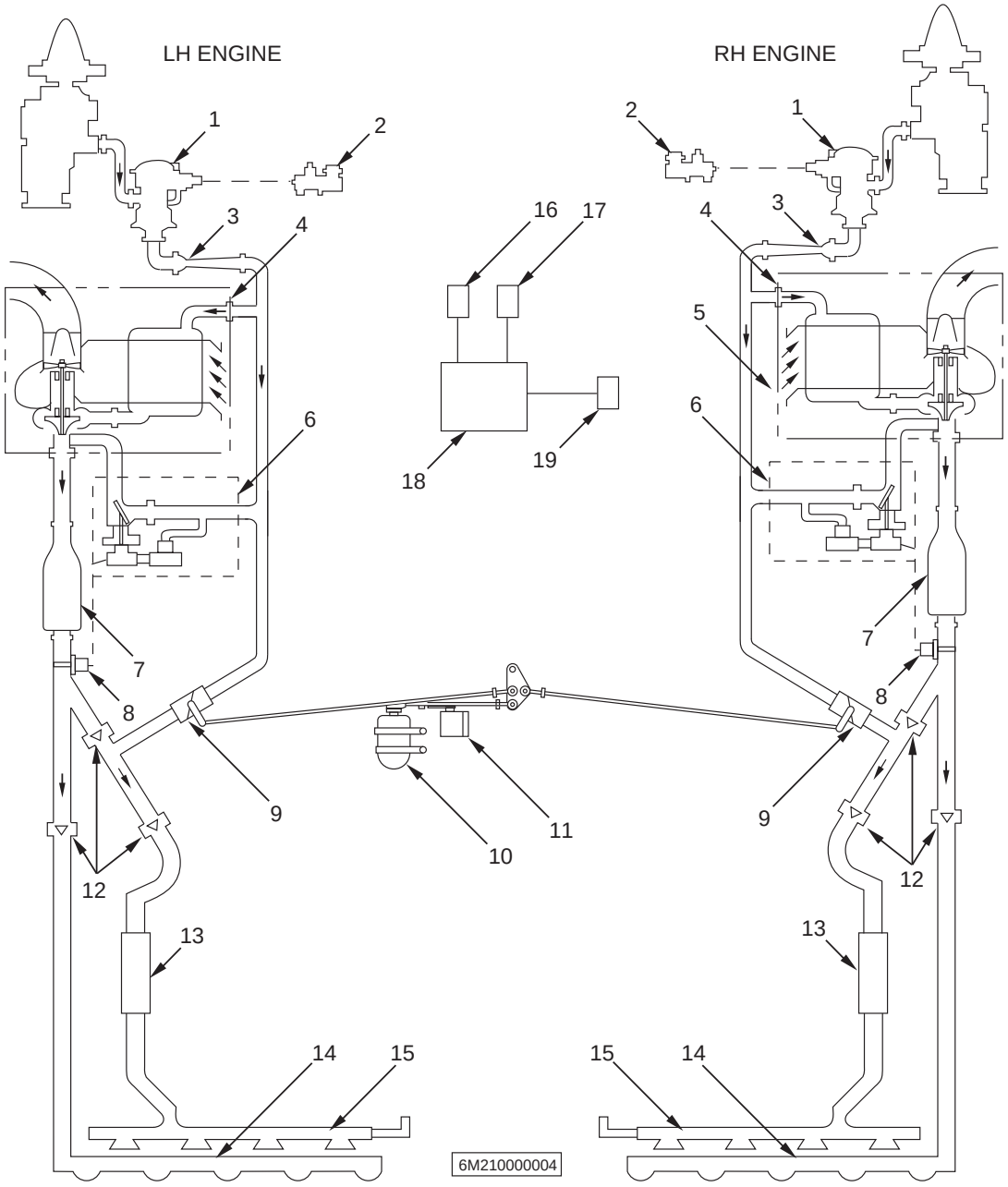
(2) The hot air controlling slave valves (Fig 6, item 4) are accessible through the outboard panel of each wing. The water separators (Fig 6, item 8) are accessible through the third panel (moving inboard). The temperature sensing master valves (Fig 6, item 7) are accessible through the two inboard panels.



System 3 Components, LH and RH Sides
Figure 6

13. System 4 – Components and Description (Figures 7 and 8)

NOTE: Refer to Figures 7 and 8 to determine the location of components. The unshaded areas indicate access panels on the lower side of the wing and center section. In using these figures to locate components, consider that they represent theoretical views from the top of the wing, a view which is not possible on the aircraft.



System 4 – Schematic
Figure 7

A. Compression

(1) Two pressure limiting flow control valves (Fig 7, item 1) and slave valves (Fig 7, Item 2) are used in conjunction with two flow limiting venturis (Fig 7, item 3).

B. Distribution

(1) The distribution system routes air throughout the aircraft. Components of the distribution system are duct check valves (Fig 7, item 12), conditioned air mufflers (Fig 7, item 13), cold air ducts (Fig 7, item 14), and conditioned air ducts (Fig 7, item 15). The fresh air system is part of the distribution system.

(2) Duct check valves (Fig 8, item 6) are located in the aircraft center section. Accessible through the inboard access panel.

C. Heating

(1) Two mixing valves (Fig 7, item 9) are connected to a common actuator (Fig 7, item 10) and a follow-up potentiometer (Fig 7, item 11). These mixing valves are used to control the flow of tempered air in the cabin.

(2) The hot air mixing valves (Fig 8, item 5) are accessible through the inboard panel. Mixing valve actuator and follow-up potentiometer (Fig 8, items 11 and 12) are located in the right center section, and are accessible through the inboard panel on the right side and the right wing-to-fuselage fairing.

D. Cooling

(1) Two cooling turbines (Fig 7, item 4) provide cool air for the system.

(2) Cooling turbines (Fig 8, item 9) are located in the wings, and are accessible through the outboard access panel on each wing.

E. Temperature Control

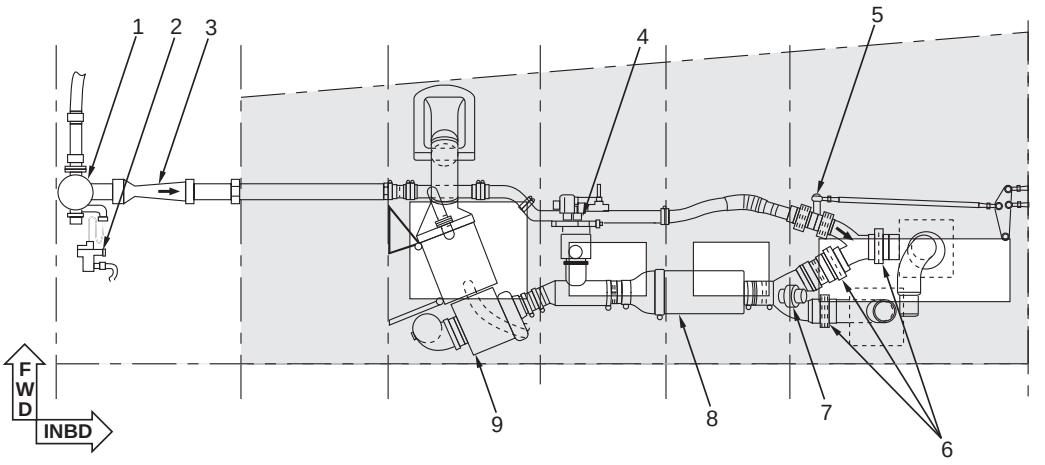
(1) An electrically operated temperature control system positions the mixing valves. Components of the system are ambient air sensor (Fig 7, item 5), mode selector switch (Fig 7, item 16), automatic temperature control (Fig 7, item 17), temperature control box (Fig 7, item 18), and cabin air temperature sensor (Fig 7, item 19).

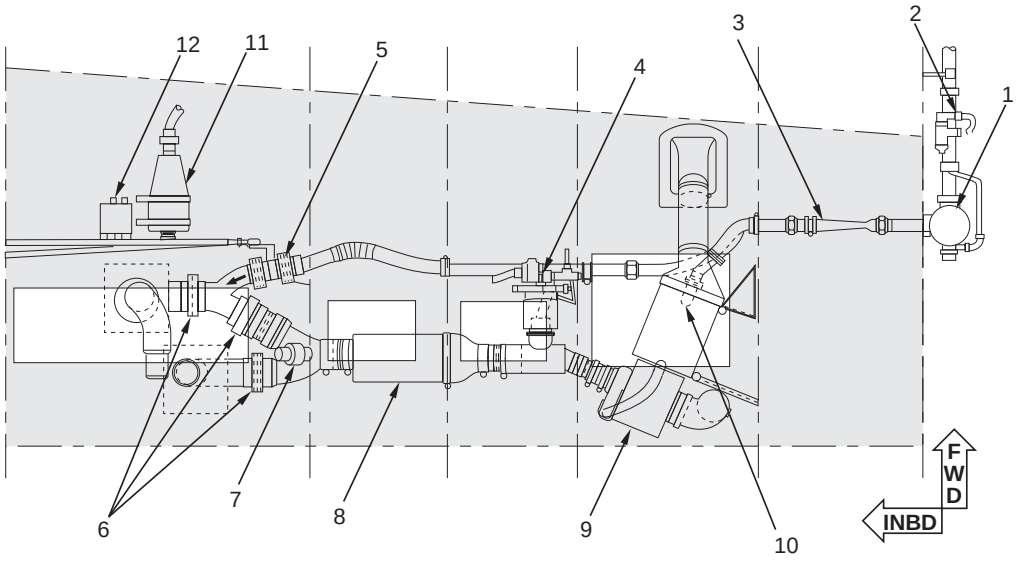
(2) The ambient air temperature sensor (Fig 8, item 10). Is located in the outside air intake for the right cooling turbine, and is accessible through the right cooling turbine access panel.

F. Moisture and Contaminant Control

(1) Components of the electrical powered anti-icing system are hot air conditioning slave valves (Fig 8, item 5), temperature sensing master valves (Fig 8, item 7), and water separators (Fig 8, item 8). The system maintains the cold air duct temperature at 35°F (1.67°C).

(2) The hot air controlling slave valves (Fig 8, item 4) are accessible through the second panel (moving inboard) of each wing. The water separators (Fig 8, item 8) are accessible through the third panel (moving inboard). The temperature sensing master valves (Fig 8, item 7) are accessible through the inboard panel.

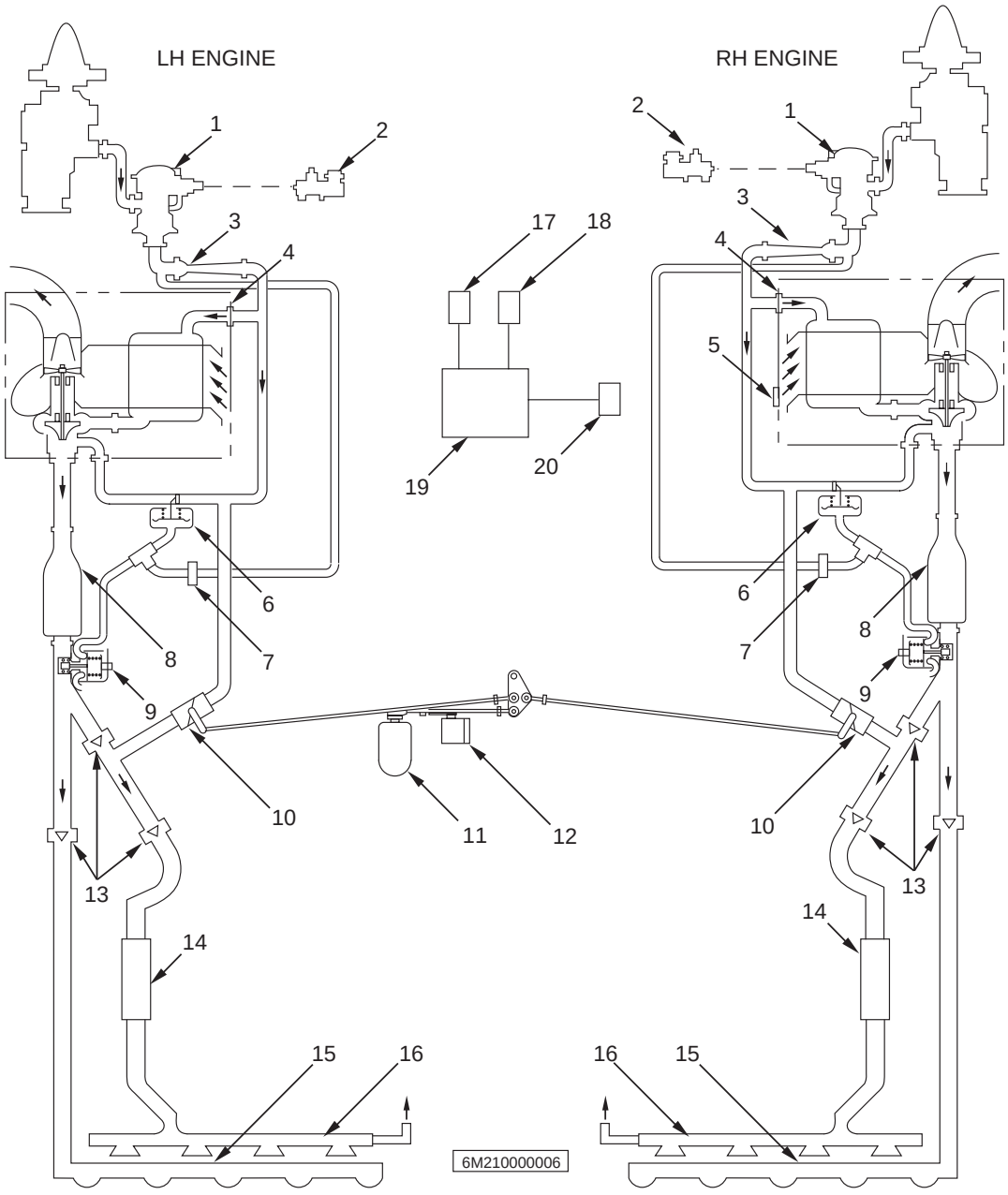




System 4 Components, LH and RH Sides
Figure 8

14. System 5 – Components and Description (Figures 9 and 10)

NOTE: Refer to Figures 9 and 10 to determine the location of components. The unshaded areas indicate access panels on the lower side of the wings and center section. In using these figures to locate components, consider that they represent theoretical views from the top of the wing, a view which is not possible on the aircraft.



System 5 – Schematic
Figure 9

A. Compression

(1) Two pressure limiting flow control valves (Fig 9, item 1) and slave valves (Fig 9, Item 2) are used in conjunction with two flow limiting venturis (Fig 9, item 3).

B. Distribution

(1) The distribution system routes air throughout the aircraft. Components of the distribution system are duct check valves (Fig 9, item 13), conditioned air mufflers (Fig 9, item 14), cold air ducts (Fig 9, item 15), and conditioned air ducts (Fig 9, item 16). The fresh air system is part of the distribution system.

(2) Duct check valves (Fig 10, item 7) are located in the aircraft center section. Accessible through the inboard access panel.

C. Heating

(1) Two mixing valves (Fig 9, item 10) are connected to a common actuator (Fig 9, item 11) and a follow-up potentiometer (Fig 9, item 12). These mixing valves are used to control the flow of tempered air in the cabin.

(2) The hot air mixing valves (Fig 10, item 6) are accessible through the inboard panel. Mixing valve actuator and follow-up potentiometer (Fig 10, items 12 and 13) are located in the right center section, and are accessible through the inboard panel on the right side and the right wing-to-fuselage fairing.

D. Cooling

(1) Two cooling turbines (Fig 9, item 4) provide cool air for the system.

(2) Cooling turbines (Fig 10, item 10) are located in the wings, and are accessible through the outboard access panel on each wing.

E. Temperature Control

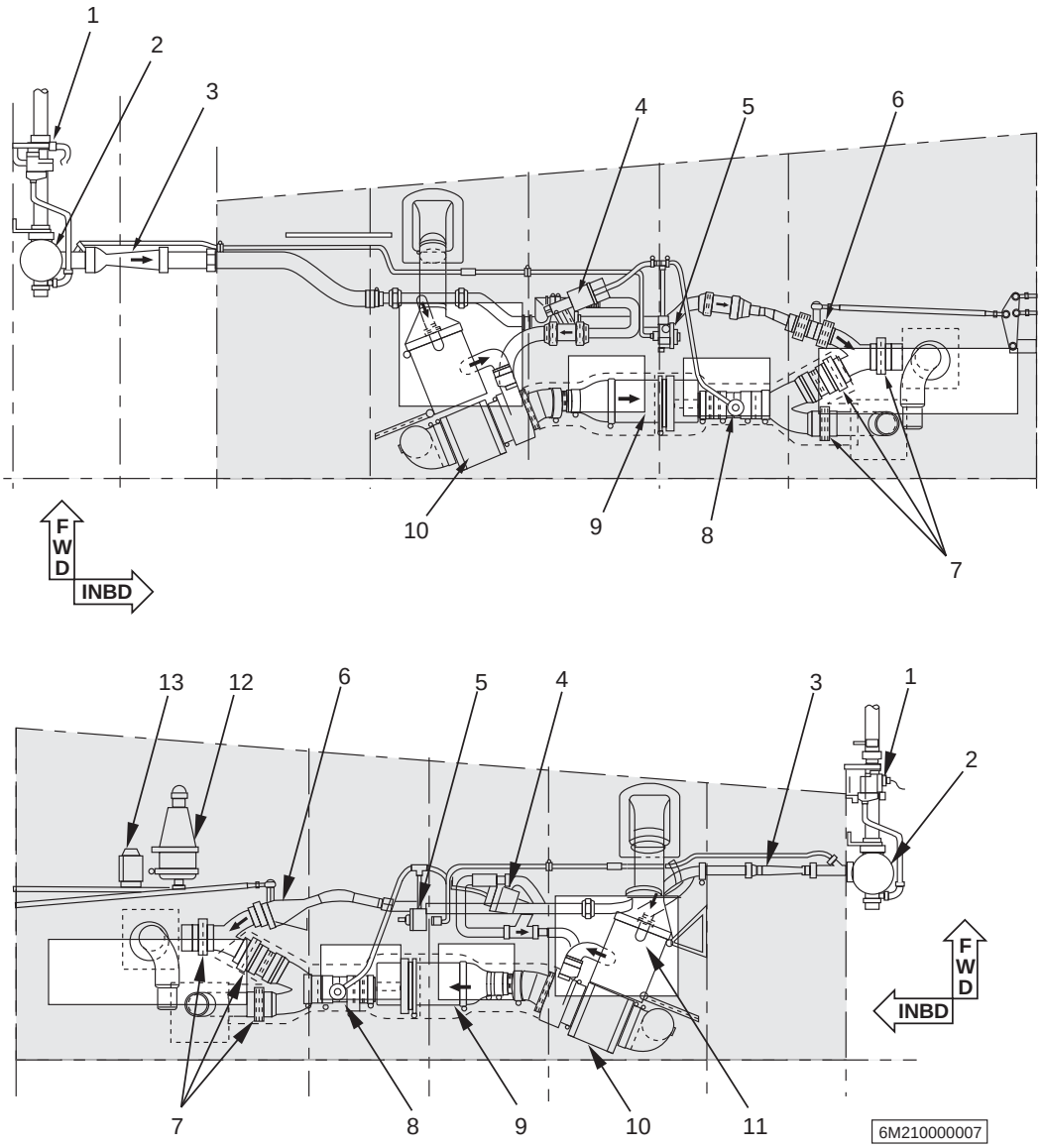
(1) An electrically operated temperature control system positions the mixing valves. Components of the system are ambient air sensor (Fig 9, item 5), mode selector switch (Fig 9, item 17), automatic temperature control (Fig 9, item 18), temperature control box (Fig 9, item 19), and cabin air temperature sensor (Fig 9, item 20).

(2) The ambient air temperature sensor (Fig 10, item 11). Is located in the outside air intake for the right cooling turbine, and is accessible through the right cooling turbine access panel.

F. Moisture and Contaminant Control

(1) Components of the electrical powered anti-icing system are hot air conditioning slave valves (Fig 10, item 6), temperature sensing master valves (Fig 10, item 11), and water separators (Fig 10, item 10). The system maintains the cold air duct temperature at 35°F (1.67°C).

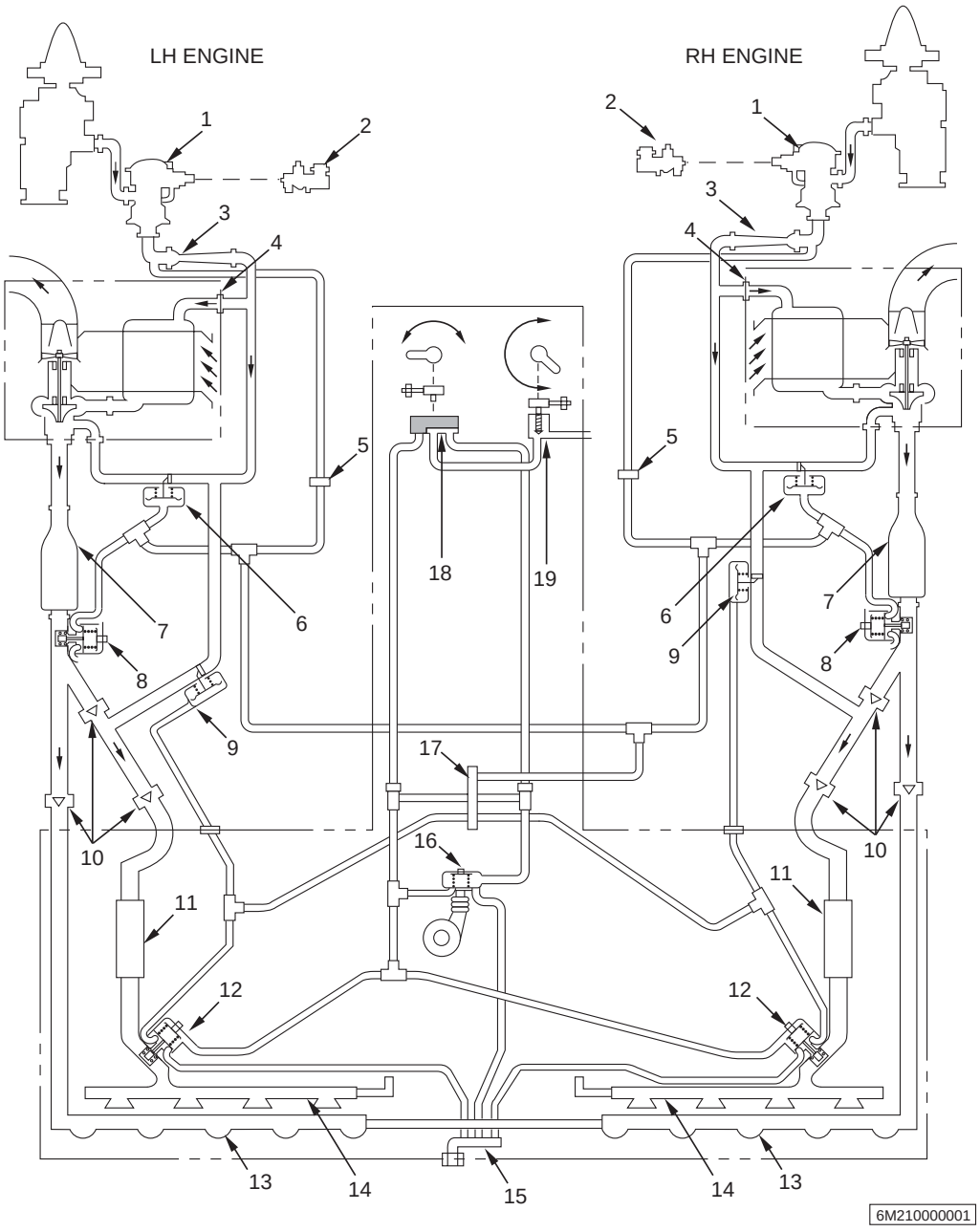
(2) The hot air controlling slave valves (Fig 10, item 6) are accessible through the second panel (moving inboard) of each wing. The water separators (Fig 10, item 9) are accessible through the third panel (moving inboard). The temperature sensing master valves (Fig 10, item 8) are accessible through the inboard panel.



System 5 Components, LH and RH Sides
Figure 10

15. System 6 – Components and Description (Figures 11 and 12)

NOTE: Refer to Figures 11 and 12 to determine the location of components. The unshaded areas indicate access panels on the lower side of the wings and center section. In using these figures to locate components, consider that they represent theoretical views from the top of the wing, a view which is not possible on the aircraft.



System 6 – Schematic
Figure 11

A. Compression

(1) Two pressure limiting flow control valves (Fig 11, item 1) and slave valves (Fig 11, item 2) are used in conjunction with two flow limiting venturis (Fig 11, item 3).

B. Distribution

(1) The distribution system routes air throughout the aircraft. Components of the distribution system are duct check valves (Fig 11, item 10), conditioned air mufflers (Fig 11, item 11), cold air ducts (Fig 11, item 13), and conditioned air ducts (Fig 11, item 14). The fresh air system is part of the distribution system.

(2) Duct check valves (Fig 12, item 7) are located in the aircraft center section and are accessible through the two inboard panels.

C. Heating

(1) Two electrically operated mixing valves (Fig 12, item 5) control the flow of tempered air into the cabin.

(2) Hot air mixing valves (Fig 12, item 6) are accessible through two inboard access panels.

D. Cooling

(1) Two cooling turbines (Fig 11, item 4), provide cool air for the system.

(2) Cooling turbines (Fig 12, item 10) are located in the wings, and are accessible through the outboard access panel on each wing.

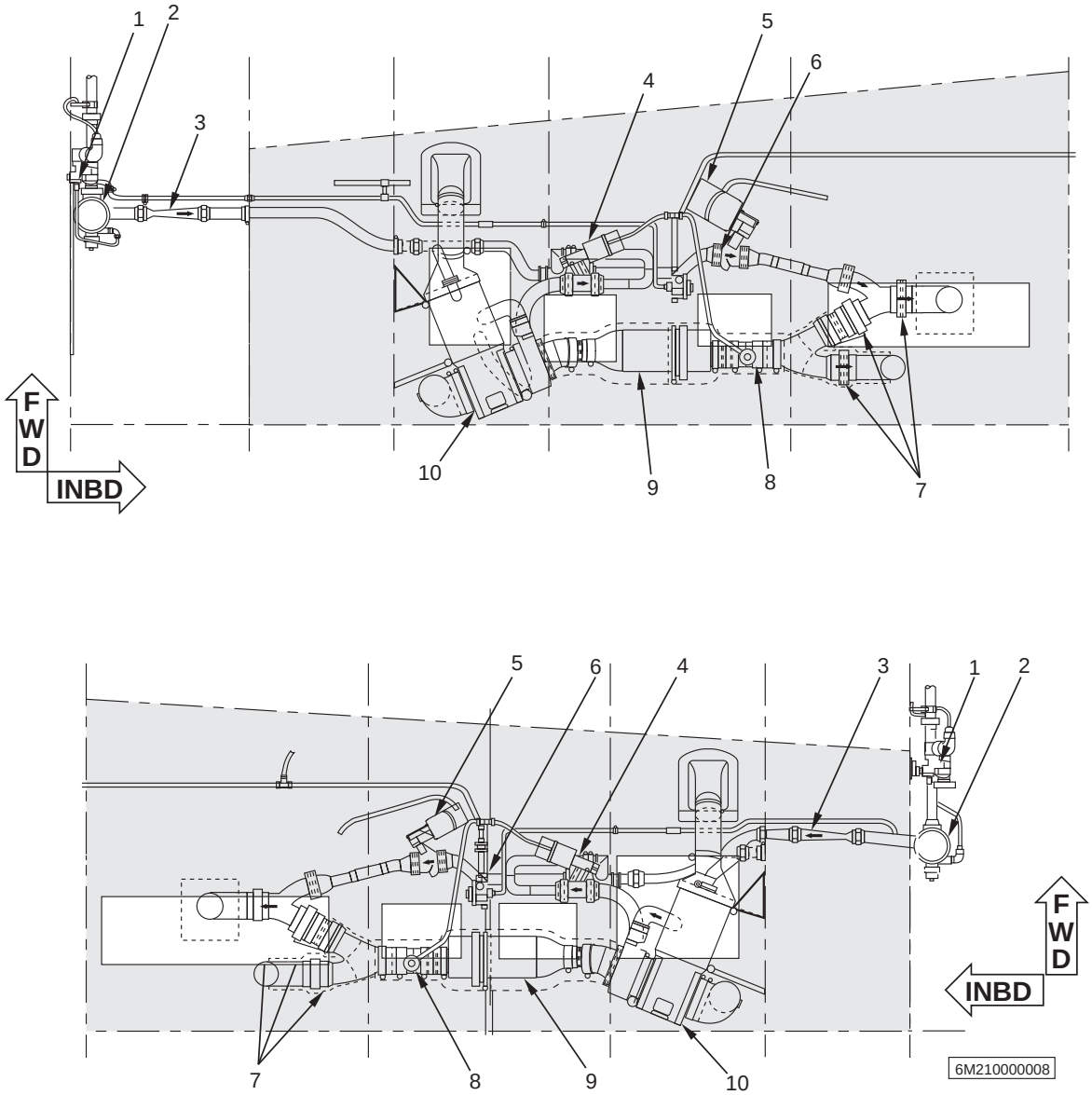
E. Temperature Control

(1) An electrically operated temperature control system positions the mixing valves. Components of the system are an electronic controller (Fig 11, item 16), cabin air temperature and blower (integral with the electronic controller unit), duct temperature sensors (Fig 11, item 12), temperature mode selector switch (Fig 11, item 18), automatic temperature control (Fig 11, item 19), and cabin air temperature sensor (Fig 11, item 17).

F. Moisture and Contaminant Control

(1) Components of the electrical powered anti-icing system are hot air conditioning slave valves (Fig 12, item 4), temperature sensing master valves (Fig 12, item 8), and water separators (Fig 12, item 9). The system maintains the cold air duct temperature at 35°F (1.67°C).

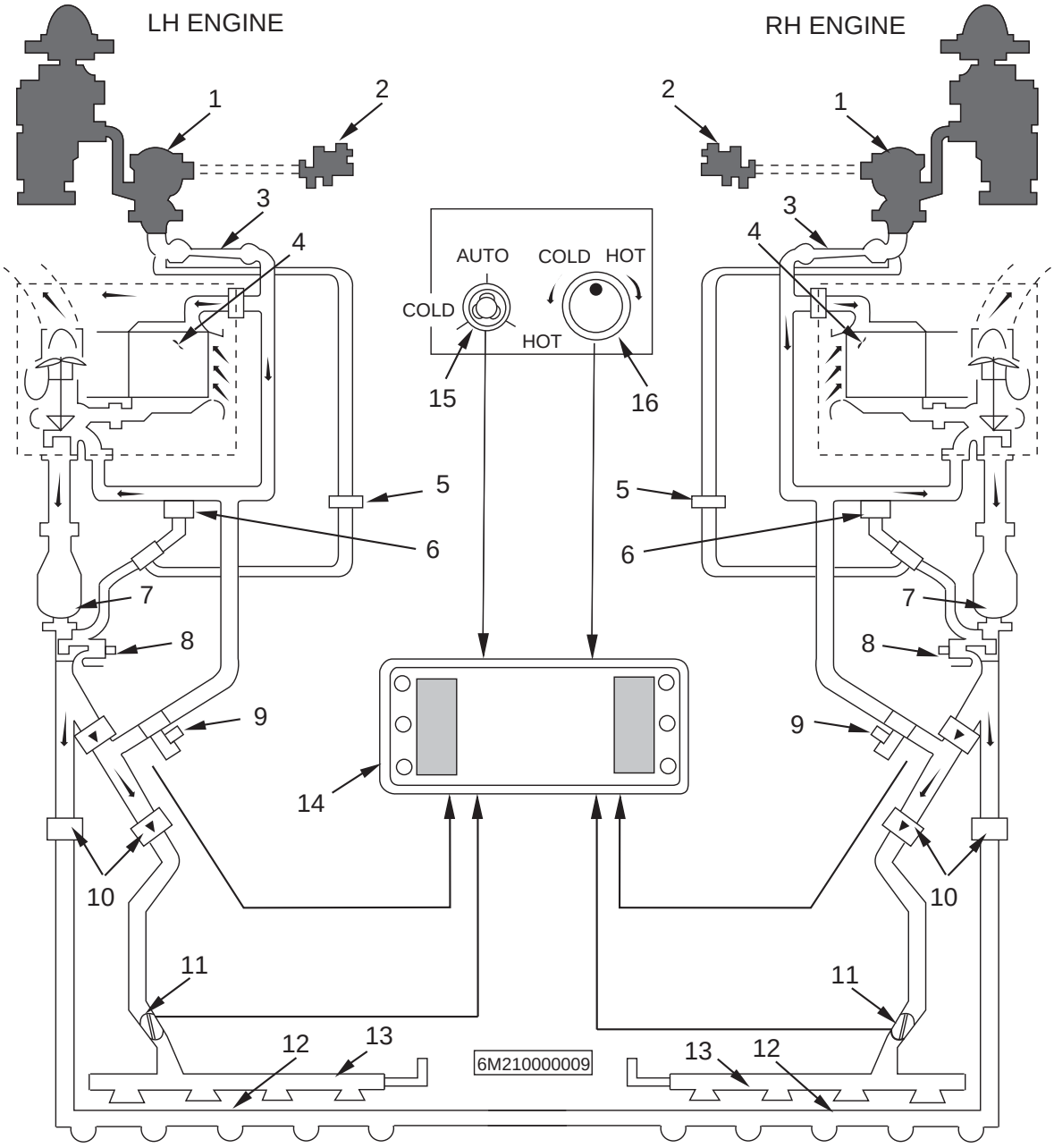
(2) The hot air controlling slave valves (Fig 12, item 4) are accessible through the second panel (moving inboard) of each wing. The water separators (Fig 12, item 9) are accessible through the third panel (moving inboard). The temperature sensing master valves (Fig 12, item 8) are accessible through the inboard panel.



System 6 Components, LH and RH Sides
Figure 12

16. System 7 – Components and Description (Figures 13 and 14)

NOTE: Refer to Figures 13 and 14 to determine the location of components. The unshaded areas indicate access panels on the lower side of the wings and center section. In using these figures to locate components, consider that they represent theoretical views from the top of the wing, a view which is not possible on the aircraft.



System 7 – Schematic
Figure 13

A. Compression

(1) Two pressure limiting flow control valves (Fig 13, item 1) and slave valves (Fig 13, item 2) are used in conjunction with two flow limiting venturis (Fig 13, item 3).

B. Distribution

(1) The distribution system routes air throughout the aircraft. Components of the distribution system are duct check valves (Fig 13, item 10), cold air ducts (Fig 13, item 12), and conditioned air ducts (Fig 13, item 13). The fresh air system is part of the distribution system.

(2) Duct check valves (Fig 15, item 7) are located in the aircraft center section and are accessible through the two inboard panels.

C. Heating

(1) Two electrically operated mixing valves (Fig 13, item 9) control the flow of tempered air into the cabin.

(2) Hot air mixing valves (Fig 13, item 9) are accessible through two inboard access panels.

D. Cooling

(1) Two cooling turbines (Fig 13, item 4), provide cool air for the system.

(2) Cooling turbines (Fig 14, item 10) are located in the wings, and are accessible through the outboard access panel on each wing.

E. Temperature Control

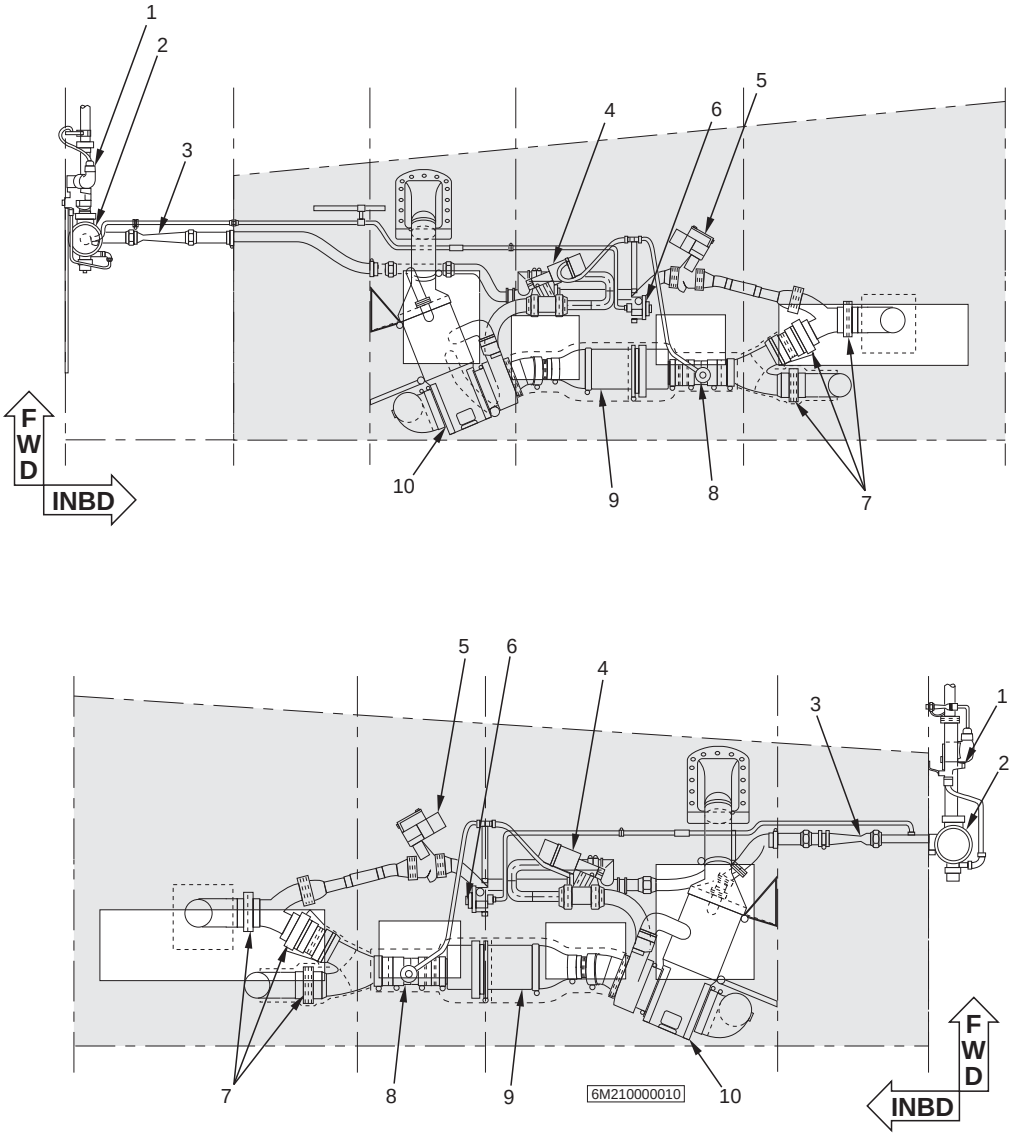
(1) A dual electric cabin temperature control system positions the mixing valves (Fig 13, item 9). Components of the system are; electronic controller (Fig 13, item 14), cabin air temperature sensor and blower, duct temperature sensor (Fig 13, item 11), temperature mode selector switch (Fig 13, item 15), and auto temperature control (Fig 13, item 16).

NOTE: The cabin air temperature sensor and blower are integral with the electronic controller unit.

F. Moisture and Contaminant Control

(1) Components of the electrically powered anti-icing system are; hot air conditioning slave valves (Fig 14, item 4), temperature sensing master valves (Fig 14, item 8), supply pressure regulators (Fig 14, item 6), and water separators (Fig 14, item 9). The system maintains the cold air duct temperature at 37°F (2.78°C).

(2) The hot air controlling slave valves (Fig 14, item 4) and the water separators (Fig 14, item 9) are accessible through the second panel (moving inboard). The supply pressure regulators (Fig 14, item 6) and the temperature sensing master valves (item 8) are accessible through the third panel (moving inboard).



System 7 Components, LH and RH Sides
Figure 14

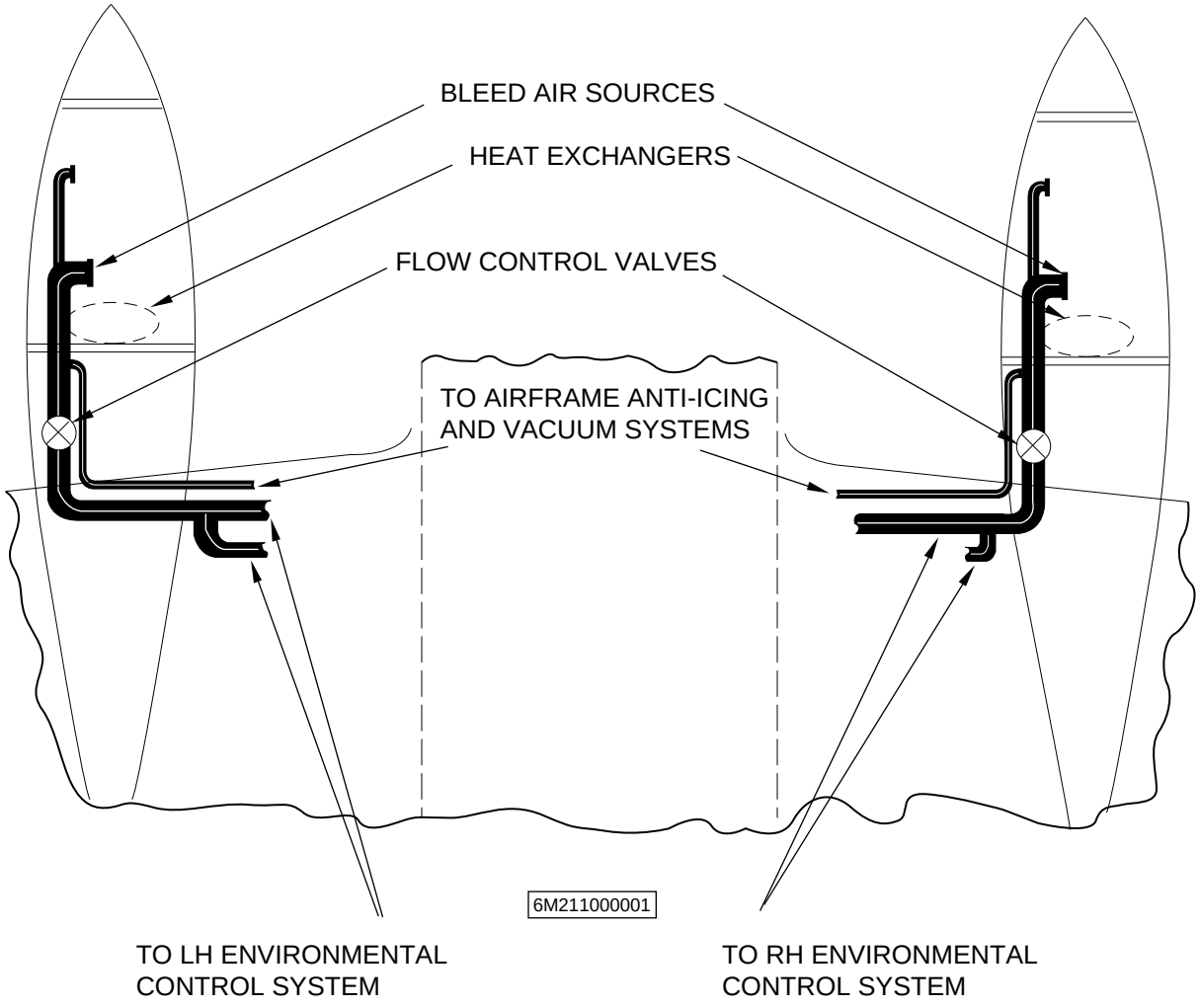
COMPRESSION – DESCRIPTION AND OPERATION

1. General

A. Air extracted from a port on the left side of each engine diffuser is used in the environmental control system (ECS), airframe anti-icing system and vacuum system. A heat exchanger installed in each engine aspirator lowers the temperature of the bleed air approximately 100°F (37.8°C).

B. Aft of the firewall the bleed air tubing divides into two tubes. The smaller tube supplies air for airframe anti-icing and vacuum systems. A check valve is installed to prevent reverse flow when the engine is not operating. The large tube is routed to the flow control valve for use in the ECS.

C. The control valve regulates bleed air flow. Since the extraction of bleed air will cause a loss of engine power, the amount of air extracted must be carefully regulated. The flow control valve is calibrated to extract a preset amount of bleed air from the engine under all operating conditions.



Compression
Figure 1

D. SA226 aircraft use a duel bleed air system. The dual control arrangement provides a maximum and a minimum setting for bleed air extraction, enabling increased cruse performance.

(1) Two electronically operated flow controls valves per system are controlled from a switch in the cockpit (HI OFF LO).

E. When the cockpit bleed air switch is moved to OFF, the flow control valve closes to stop bleed airflow to the ECS. The bleed air supplied to the airframe anti-icing and vacuum system is neither controlled nor affected by the flow control valve.

F. When the AWI system arming switch is ON and the power levers are advanced 0.125 inch (3.175 mm) or more beyond flight idle, electrical power is applied to the bleed air valve. This allows all available air to be routed through the engine for maximum performance. If the AWI arming switch is turned OFF or if the power levels are positioned to 0.125 inch (3.175 mm) or less beyond flight idle, the bleed air valve will open.

COMPRESSION – FAULT ISOLATION

1. Troubleshooting – Compression

PROBLEM:

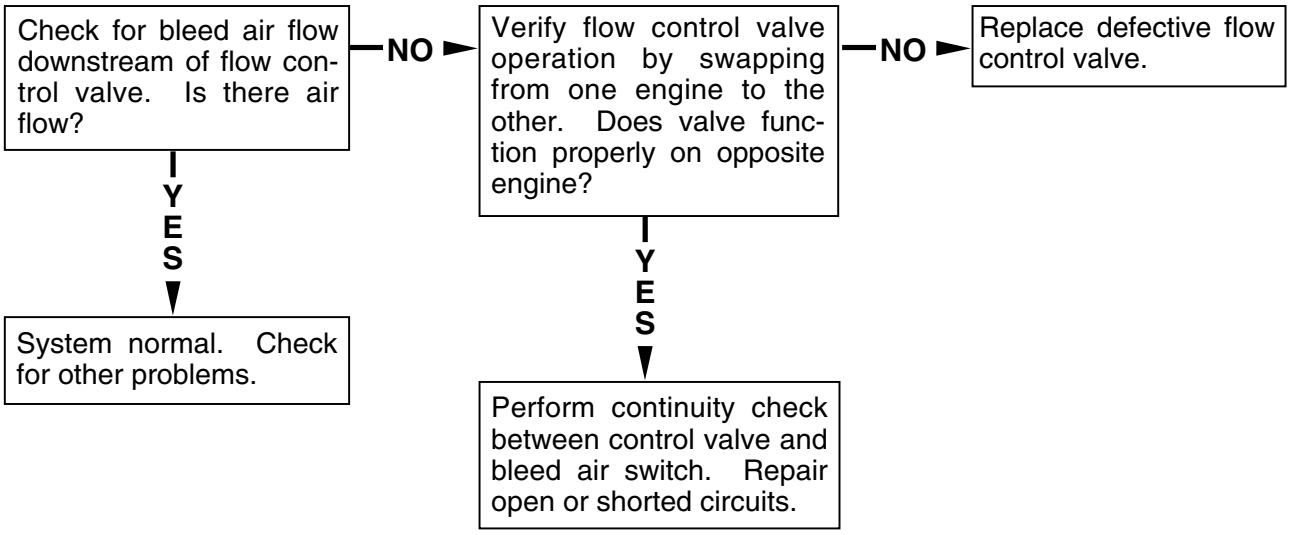
NO DROP IN TORQUE WHEN BLEED AIR IS TURNED ON.

PREREQUISITES:

Electrical Power ON, CAWI Arm Switch OFF, Bleed Air Switch ON.

NOTE: No drop in torque when bleed air is turned on may be accompanied by high engine oil temperature, engine fire warning light, and low or fluctuating vacuum or deice pressure.

START:



6M211000101

END

COMPRESSION – FAULT ISOLATION

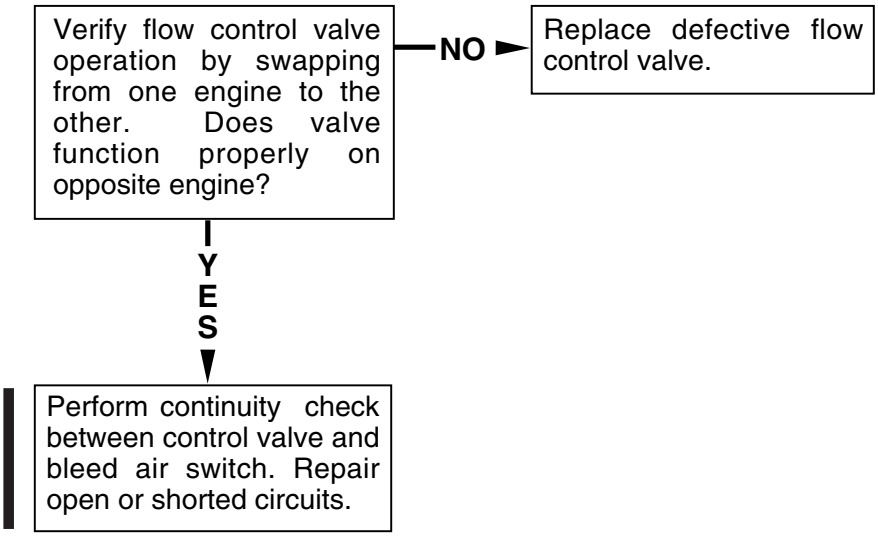
PROBLEM:

EXCESSIVE DROP IN TORQUE WHEN BLEED AIR IS TURNED ON.

PREREQUISITES:

Electrical Power ON, CAWI Arm Switch OFF, Bleed Air Switch ON.

START:



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END

BLEED AIR FORWARD OF FIREWALL – DESCRIPTION AND OPERATION

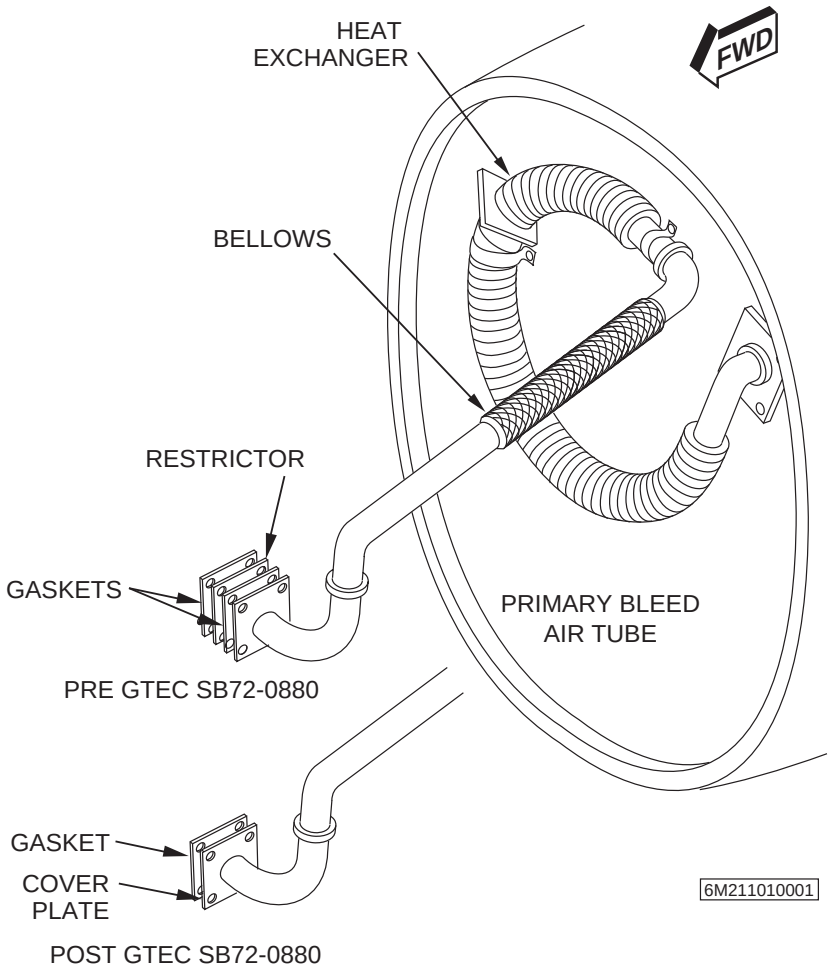
1. General

A. Bleed air is extracted from the engine compressor through a bleed air port on the left side of the engine diffuser section.

B. A bleed air line channels the air from the bleed air port to a heat exchanger. On aircraft that have not complied with GTEC SB27-0880; a restrictor plate is installed between the bleed air port and the primary bleed air tube to limit the amount of air extracted from the engine should a massive bleed air leak occur. On aircraft that have complied with GTEC SB27-0880, the restrictor plate and one gasket is removed leaving one gasket and cover plate.

C. A bellows installed in the bleed air line absorbs engine vibration.

D. A heat exchanger installed in the aspirator area lowers the bleed air temperature to approximately 100°F (37.8°C). Ambient air enters the oil cooler scoop, passes through the oil cooler and over the heat exchanger coils. The air exits the aircraft through the tail pipe along with the engine exhaust gases.



Bleed Air
Forward of Firewall
Figure 1

BLEED AIR FORWARD OF FIREWALL – FAULT ISOLATION

2. Troubleshooting – Bleed Air Forward of Firewall

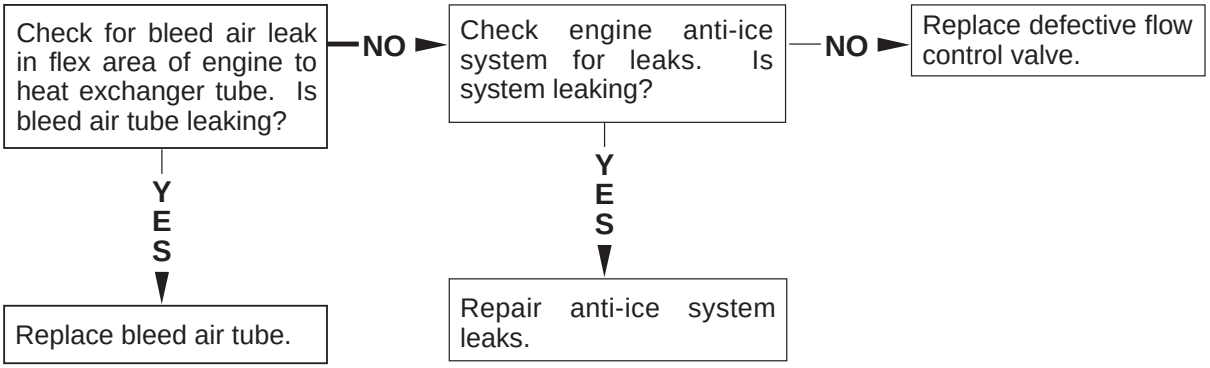
PROBLEM:

SLIGHT DROP IN ENGINE TORQUE WITH AIR ON, EXCESSIVE DROP IN TORQUE WITH BLEED AIR OFF.

PREREQUISITES:

Electrical Power ON, CAWI Arm Switch OFF, Bleed Air Switch as required.

START:



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END

BLEED AIR FORWARD OF FIREWALL – FAULT ISOLATION

PROBLEM:

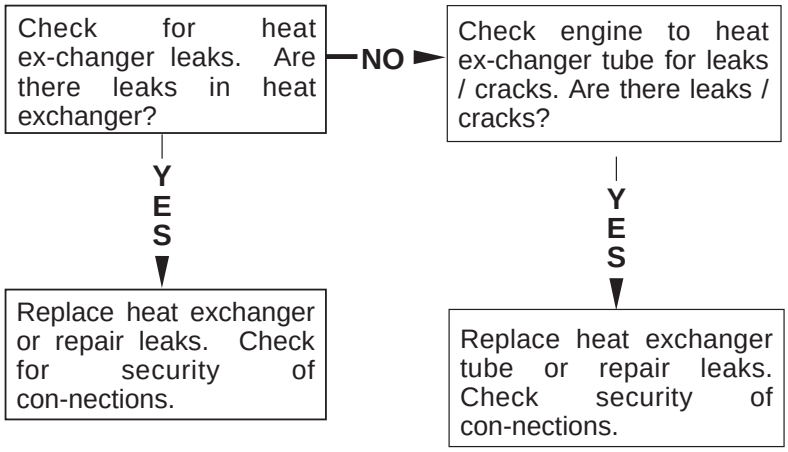
HIGH ENGINE OIL TEMPERATURE DURING GROUND IDLE OR ENGINE FIRE WARNING LIGHT ON.

PREREQUISITES:

Electrical Power Applied, CAWI Arm Switch OFF, Bleed Air Switch ON.

NOTE: Bleed Air System leaks may manifest as high engine oil temperature or trigger an engine fire warning light. If repair or replacement of leaking bleed air components does not eliminate problem, or if no leaks are found in the Bleed Air System, trouble may be caused by Engine Oil System or Fire Warning System.

START:



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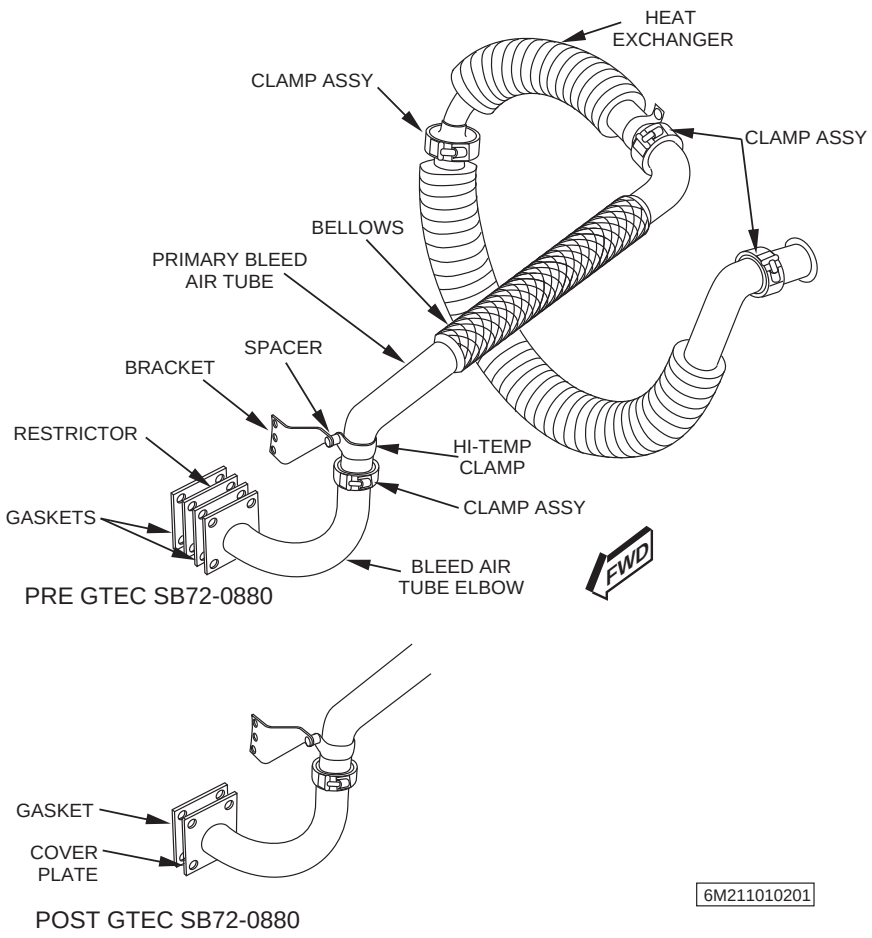


Primary Bleed Air Tube
Figure 201

1. Removal/Installation – Bleed Air Forward of Firewall

A. Removal – Primary Bleed Air Tube (Figure 201)

(1) Open left or right engine cowling.

(2) Locate primary bleed air tube on left side of engine. Tube runs parallel with upper truss from mid engine aft to aspirator.

(3) Disconnect tube at aspirator by removing clamp assembly.

(4) Disconnect bleed air tube elbow from engine diffuser. Elbow is secured with four (4) nuts and washers.

(5) Temporarily install a protective cover over the bleed port and bleed air gasket in engine diffuser with previously removed hardware.

(6) Disconnect hi-temp clamp from engine bracket. Retain clamp and hardware for installation.

B. Installation – Primary Bleed Air Tube (Figure 201)

(1) Coat diffuser port studs with Fel-Pro C5-A Anti-Seize compound or equivalent.

(2) Pre GTEC SB27-0880, position bleed air tube elbow onto engine diffuser bleed port studs with gaskets and restrictor. Stack up with gasket against port, restrictor plate, gasket, and elbow. Secure elbow with washers and nuts. Post GTEC SB27-0880, position bleed air tube elbow onto engine diffuser bleed port studs with gasket and cover plate. Stack up with gasket against port, cover plate, and elbow. Secure elbow with washers and nuts.

(3) Install hi-temp clamp with spacer. Secure to engine bracket with bolt, washers, and nut. Position bolt with head forward.

CAUTION: SIDE LOADS ON THE BELLOWS WILL RESULT IN PREMATURE CRACKING AND RESULTANT LOSS OF BLEED AIR FROM THE ENGINE.

(4) Apply a liberal coat of anti-seize compound (Fel-Pro C5-A or equivalent) to clamp assy and secure aft end of primary bleed air tube to heat exchanger.

(5) Perform bleed air leak check.

(6) Close applicable engine cowling.

2. Inspection/Check – Bleed Air Forward of Firewall

A. Inspect heat exchanger for the following:

(1) Any noticeable fin damage.

(2) More than six (6) loose fins per assembly.

(3) Any flaws, dents, or nicks in unfinished portion of tube.

(4) Any air leaks.

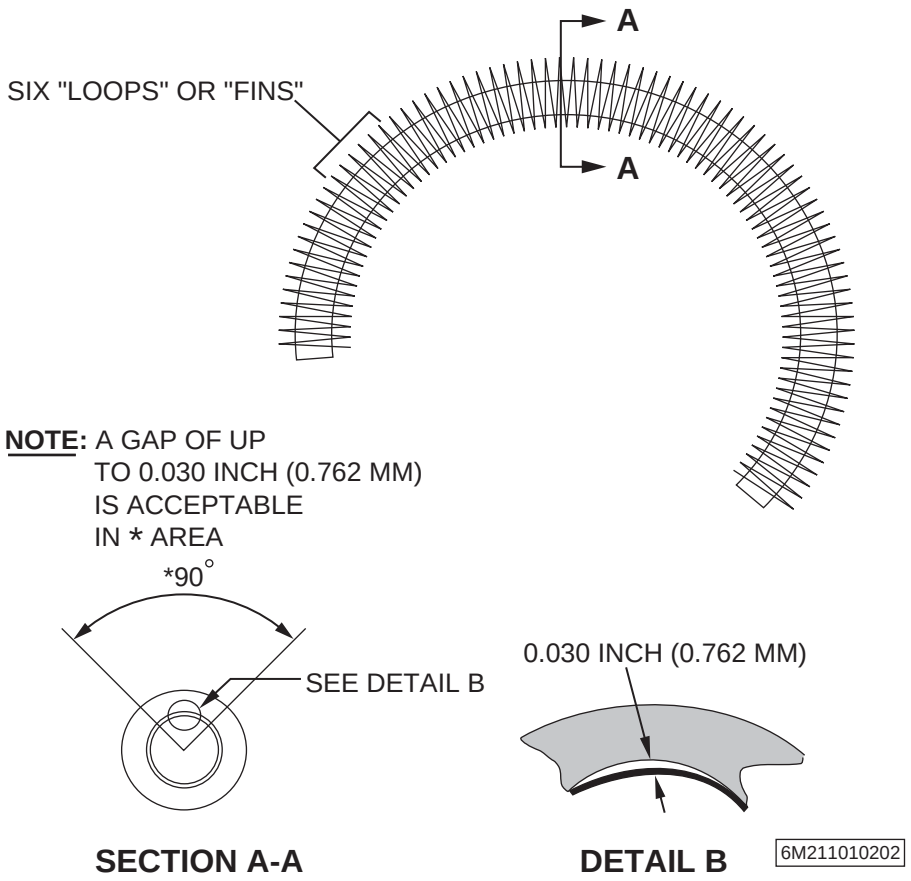
B. Replace heat exchanger if:

(1) More than six (6) fins (loops) of helical fin are loose.

(2) More than six (6) fins of the helical fin are loose.

(3) More than six (6) fins exhibit detachment from tube in excess of that shown in figure 202.

(4) Cracks over 0.250 inch exist in fin which may cause loss of fin material in more than six (6) fins.



Heat Exchanger
Figure 202

3. Inspection/Check – Primary Bleed Air Tube Pressure Test

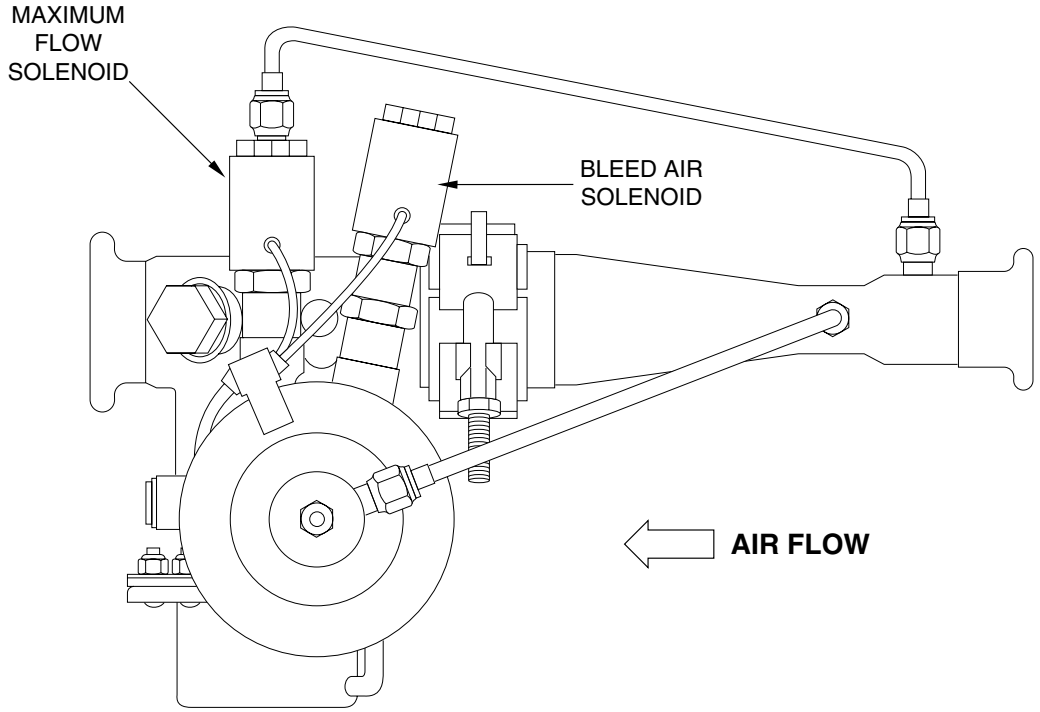
A. Refer to Service Bulletin 226-21-022 for primary bleed air tube pressure test procedure and required tooling.

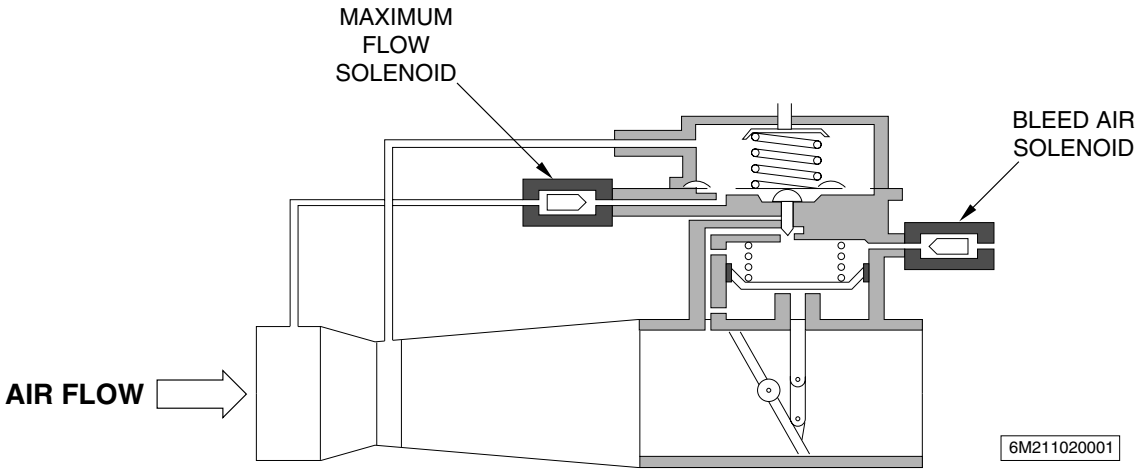
FLOW CONTROL VALVE – DESCRIPTION AND OPERATIONS

1. General

A. A flow control valve installed in each engine nacelle regulates the amount of bleed air extracted from the engine.

(1) A butterfly inside the flow control valve, governed by sensing pressure upstream of the butterfly valve, regulates the airflow. A maximum flow function allows the flow control valve to provide more air flow for ground cooling.





Flow Control Valve
Figure 1

FLOW CONTROL VALVE – DESCRIPTION AND OPERATIONS

CAUTION: WITH THE FLOW CONTROL VALVE IN THE MAXIMUM FLOW POSITION, ENGINE SPEED ABOVE 80% RPM MAY DAMAGE THE COOLING TURBINE.

B. The maximum flow function is electrically controlled by two switches. The bleed air switch has two positions, ON and OFF; the max air switch has two positions, NORM and MAX.

(1) With the bleed air switch ON and the max flow switch NORM, the butterfly in the flow control valve regulates airflow. With the bleed air switch OFF, electrical power is applied to the bleed air solenoid causing the flow control valve to go fully close.

(3) With the bleed air switch ON and the max air switch max, electrical power is applied to the max flow solenoid causing the flow control valve to open for maximum airflow.

C. At effectivity T 249-257, AT 025-042, TC 213-221, and those aircraft retrofitted by Service Bulletin 21-006, the max flow electrical circuitry is routed through the 90% contacts of the speed switch.

(1) Above 90% RPM, a relay will open to prevent operation in the max flow mode. This is a last chance safety feature, and should not be depended upon in normal operation. Above 80% RPM, move the max flow switch to the NORM position.

FLOW CONTROL VALVE – FAULT ISOLATION

1. Troubleshooting – Flow Control Valve

PROBLEM:

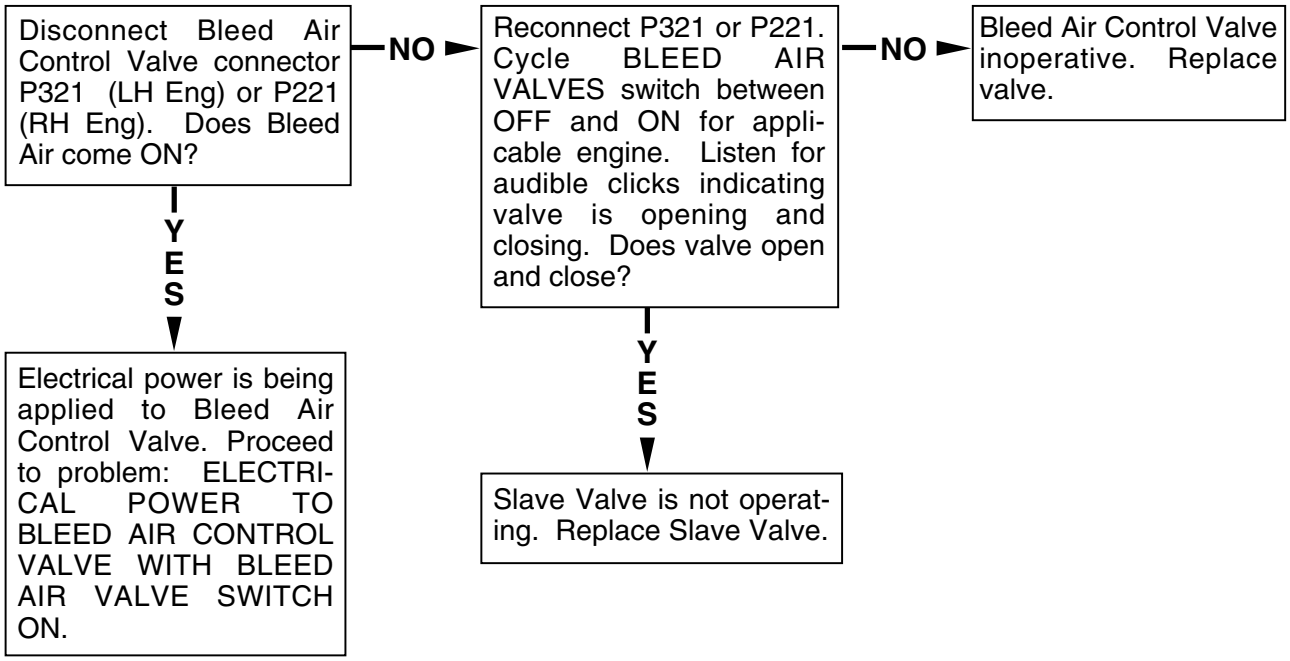
BLEED AIR WILL NOT COME ON.

PREREQUISITES:

Electrical Power ON, CAWI Arm Switch OFF, BLEED AIR VALVES Switch ON, applicable engine operating.

NOTE: When troubleshooting bleed air system, note that the bleed air valve CLOSES when electrical power is applied (BLEED AIR VALVES Switch OFF) and OPENS when power is terminated (BLEED AIR VALVES Switch ON).

START:



END

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FLOW CONTROL VALVE – FAULT ISOLATION

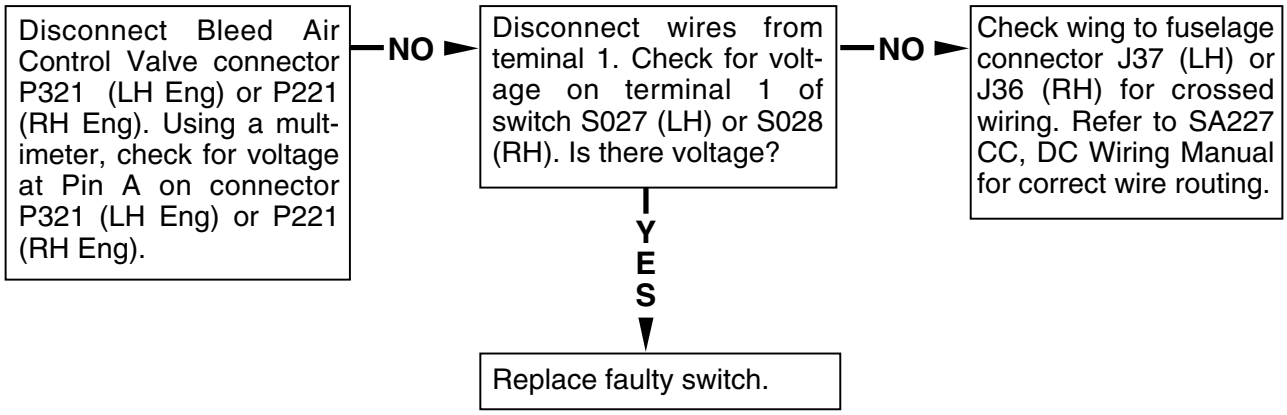
PROBLEM:

ELECTRICAL POWER TO BLEED AIR CONTROL VALVE WITH BLEED AIR SWITCH ON.

PREREQUISITES:

Electrical Power ON, CAWI Arm Switch OFF, BLEED AIR VALVES Switch ON.

START:



END

6M211020102

FLOW CONTROL VALVE – MAINTENANCE PRACTICES

1. Adjustment/Test – Flow Control Valve

A. Functional Test

(1) Move aircraft to a run up area.

(2) Operate both engines IAW AFM.

(3) Position bleed air switches OFF.

(a) Advance engines to takeoff power.

CAUTION: DO NOT ALLOW ITT TO EXCEED 923°C.

(b) Note torque obtained.

(4) Position bleed air switches ON.

(a) Check max flow switches NORM.

(b) Note torque obtained.

CONDITION	SYMPTOM	CAUSE
Bleed air OFF	Torques are approximately equal.	System normal
Bleed air ON	Torque drop 90-140 foot-pounds.	System normal
	Torque drop outside parameters.	Malfunctioning flow control valve or leak downstream of valve.
	Less than normal drop of one torque.	Bleed air leak upstream of flow control valve.

Torque Check
Table 201

NOTE: The above data is based on an operable torque indicating system. If any component of the system is malfunctioning, the functional check is not valid.

FLOW CONTROL VALVE – DESCRIPTION AND OPERATIONS

1. General

A. The flow control valve regulates the amount of bleed air extracted from the engines and, if required, stops the flow of bleed air. The valve consists of three primary components.

(1) Control Valve

(a) The electrically actuated control valve allows the slave to extract an exact amount of air from the engine. A spring-loaded valve controls the pressure to position the slave regulating valve. The spring tension is present and nonadjustable.

(b) When electrical power is applied to the valve, the actuator retracts and control pressure is vented to atmosphere. The slave valve closes, blocking the bleed airflow.

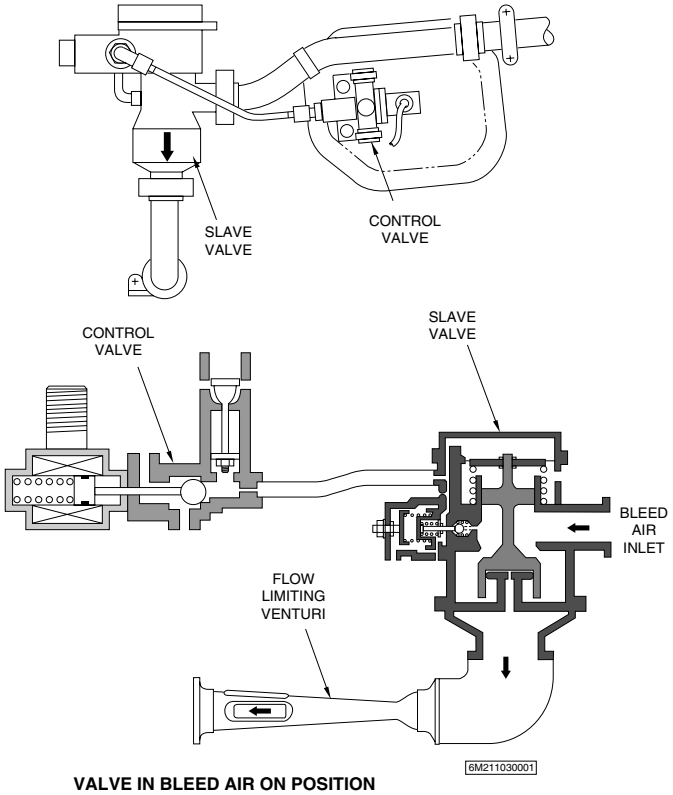
(2) Slave Valve

(a) A balanced poppet regulates the amount of air extracted from the engine and, if desired, stop the flow of air from the engine. The slave valve extracts air from the engine at the rate determined by the control valve.

(b) When the bleed air switch is OFF, the control pressure equalizes with the ambient pressure and the slave valve stops airflow.

(3) Flow Limiting Venturi

(a) The maximum amount of air extracted from the engine is limited by the size of the throat of the venturi. A venturi is installed downstream of each slave valve.



Flow Control Valve
Figure 1

AIR DISTRIBUTION – DESCRIPTION AND OPERATIONS

1. General

A. The distribution system consists of ducts, tubes, hoses, and lines that route and distribute air for the enviromental control system (ECS). Beginning at the engine, bleed air is routed through stainless steel tubing to the flow control valve. Bleed air leaving the flow control valve is routed through stainless steel tubing to the cooling turbine. Air exiting the cooling turbine is routed to the fuselage area through aluminum tubing and ducts.

B. Aluminum ducts distribute cold air and conditioned air throughout the fuselage. Flexible wire supported hoses (sceet) route conditioned and cold air to the cockpit. Sceet hoses are also used as couplings to join tubing and ducts.

C. A blower motor and duct system are supply fresh air to the cockpit and cabin. A check valve is installed in the fresh air duct to prevent air leakage when the cabin is pressurized.

FRESH AIR DISTRIBUTION – DESCRIPTION AND OPERATIONS

1. General

A. A blower motor and associated distribution plumbing supplies fresh air to the cockpit. The blower is located under the left nose baggage compartment floorboard. A check valve prevents the escape of cabin pressurization in flight.

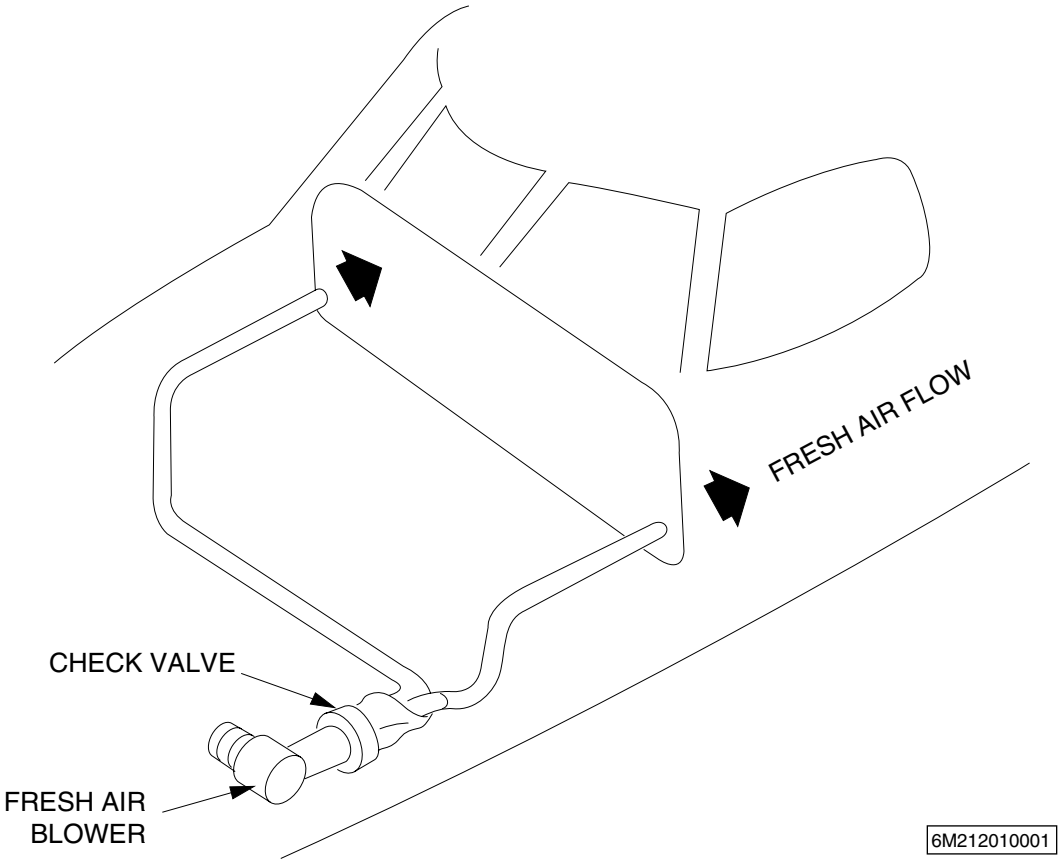
B. Aircraft T 201-248, AT 001-024, and TC 201-211 are equipped with a fresh air fan switch connected directly to the fresh air fan relay.

CAUTION: IF THE SWITCH IS LEFT ON IN FLIGHT, OVERHEATING AND BLOWER DAMAGE MAY RESULT.

C. Aircraft T 249-278, AT 025-061, and TC 212-232 have the blower motor circuitry wired through the nose gear interlock switch to prevent fan operation while airborne.

NOTE: Refer to SA226 Wiring Manual for electrical circuit information.

D. Aircraft T 279 and up, AT 062 and up, TC 233 and up, and aircraft retrofitted IAW SB 21-012 have a three position switch (NORM-OFF-OVERRIDE) and an override circuit. The override circuit permits fan operation while airborne.



Fresh Air Distribution Figure 1

FRESH AIR DISTRIBUTION – FAULT ISOLATION

1. Troubleshooting – Fresh Air Distribution

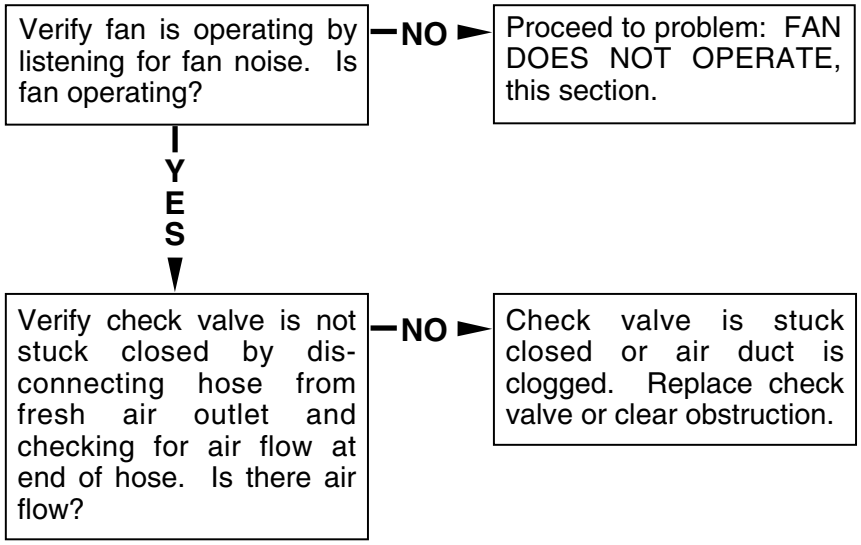
PROBLEM:

NO AIR FROM FRESH AIR OUTLETS.

PREREQUISITES:

Electrical Power Applied, Fresh Air Fan Circuit Breaker In, FRESH AIR switch in OVERRIDE.

START:



END

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FRESH AIR DISTRIBUTION – FAULT ISOLATION

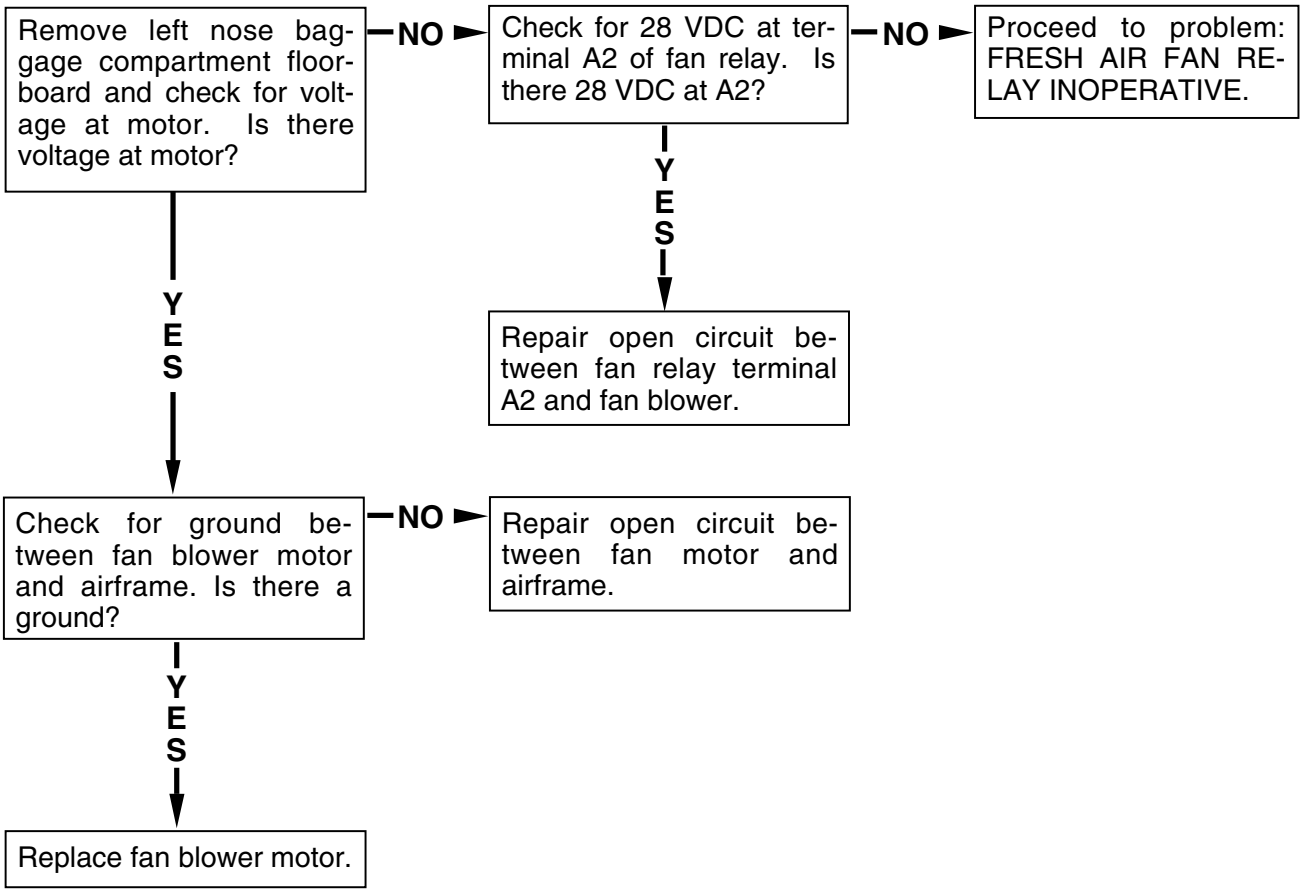
PROBLEM:

FAN DOES NOT OPERATE.

PREREQUISITES:

Electrical Power ON, Fresh Air Fan Circuit Breaker IN, FRESH AIR switch in OVERRIDE.

START:



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END

FRESH AIR DISTRIBUTION – FAULT ISOLATION

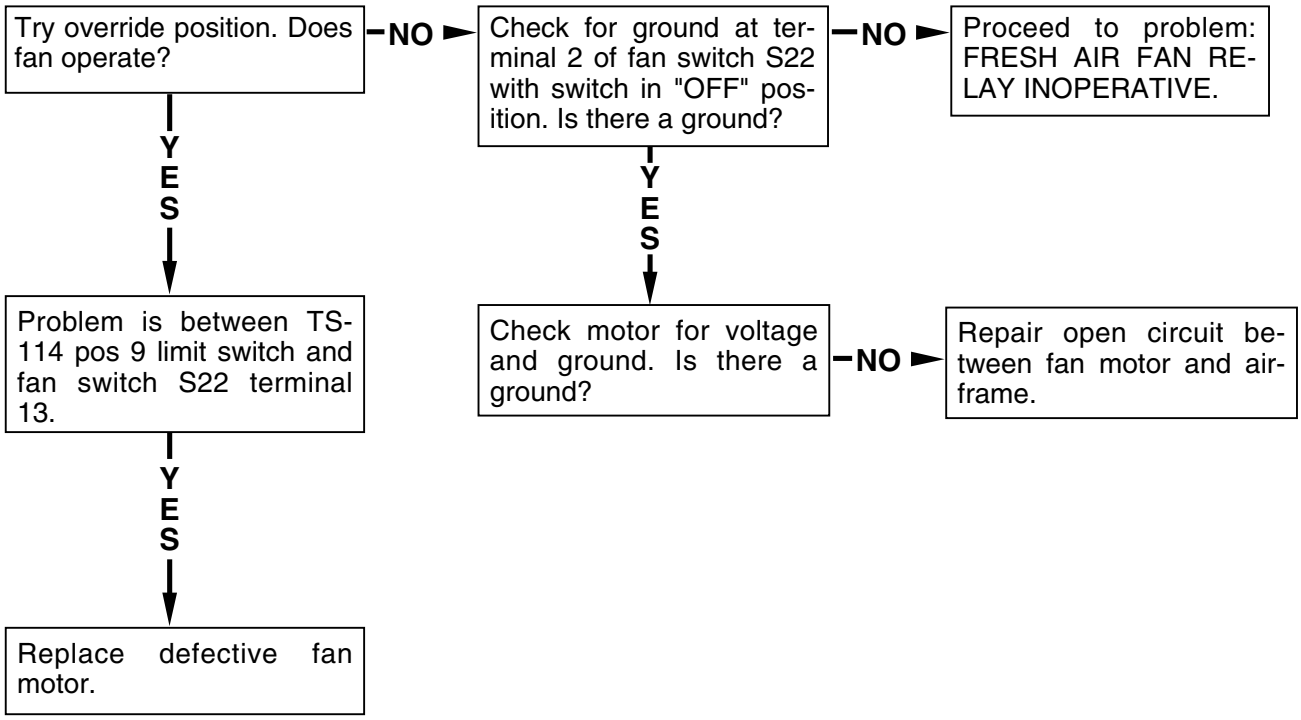
PROBLEM:

FRESH AIR FAN INOPERATIVE WHEN SWITCH IS PLACED IN OVERRIDE.

PREREQUISITES:

Electrical Power ON, Fresh Air Fan Circuit Breaker IN, FRESH AIR switch in NORM.

START:



END

6M212010103

FRESH AIR DISTRIBUTION – FAULT ISOLATION

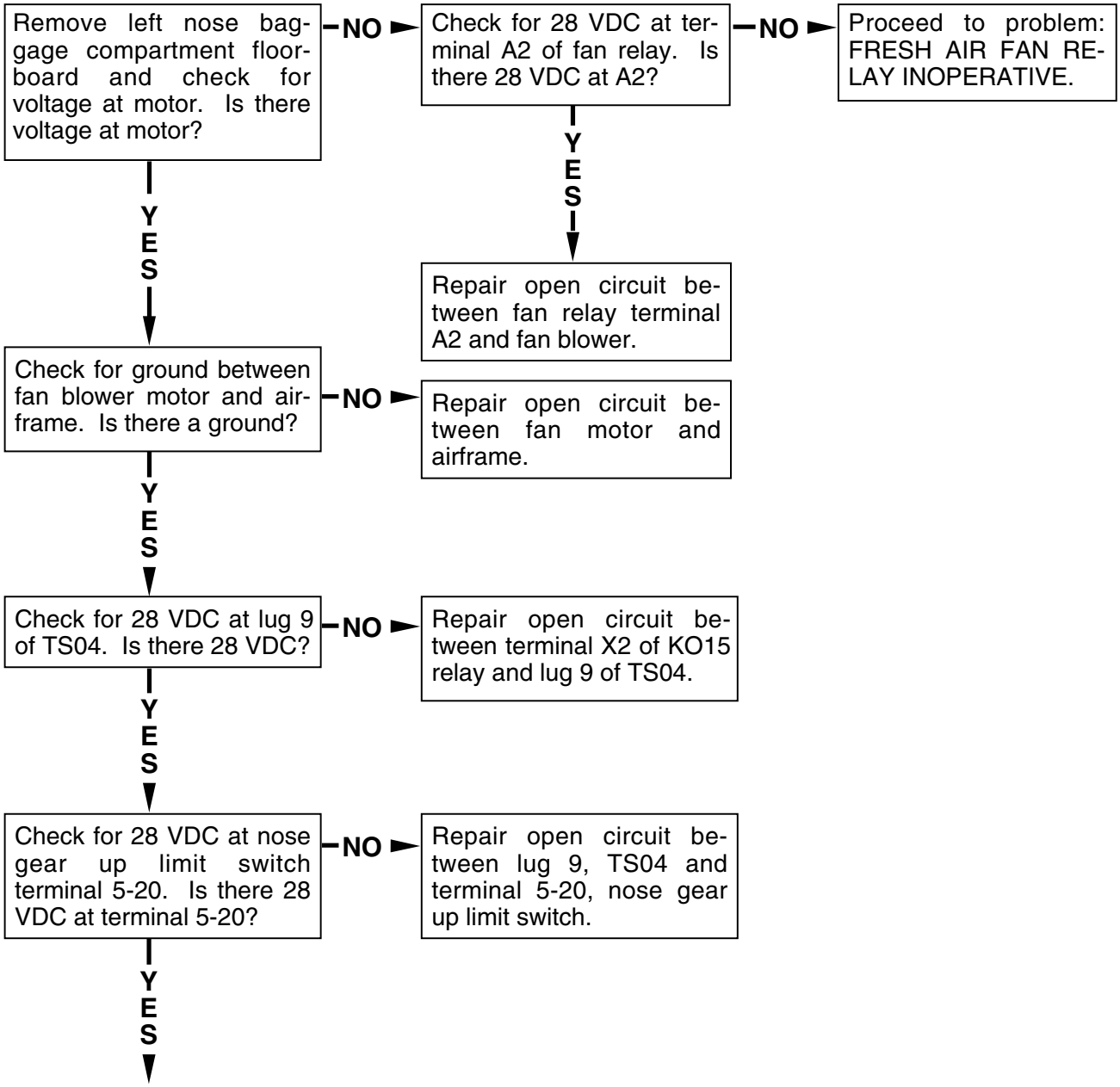
PROBLEM:

FRESH AIR FAN INOPERATIVE WHEN SWITCH IS PLACED IN NORM.

PREREQUISITES:

Electrical Power ON, Fresh Air Fan Circuit Breaker IN, FRESH AIR switch in NORM.

START:



CONTINUE

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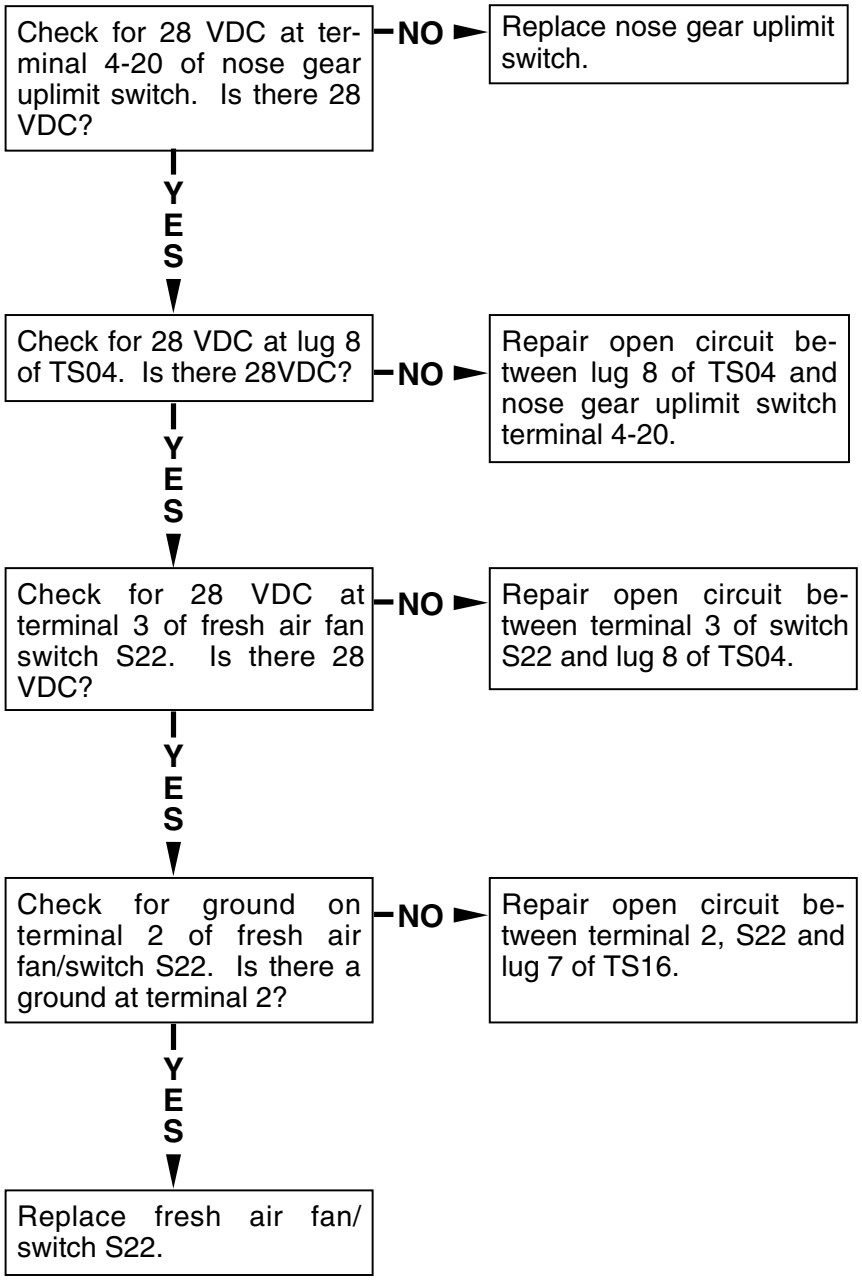

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6M212010104A

FRESH AIR DISTRIBUTION – FAULT ISOLATION

PROBLEM:

FRESH AIR FAN RELAY INOPERATIVE.

PREREQUISITES:

Electrical Power ON, Fresh Air Fan Circuit Breaker IN, FRESH AIR switch in OVERRIDE.

START:

Check for 28 VDC at fresh air fan circuit breaker. Is there 28 VDC?

NO → Replace fresh air fan circuit breaker.

YES

Check for 28 VDC at terminal A1 of K15 fan relay. Is there 28 VDC?

NO → Repair open circuit between terminal A1 of K15 relay and fan circuit breaker.

YES

Check for 28 VDC at terminal X1 of fan relay. Is there 28 VDC?

NO → Repair open circuit between terminal X1 and A1 of fan relay.

YES

Check for 28 VDC at terminal X2 of fan relay. Is there 28 VDC?

NO → Replace K15 fan relay.

YES

Check for 28 VDC at terminal 9 of TS04. Is there 28 VDC?

NO → Repair open circuit between X2 of K15 relay and terminal 9 of TS04.

YES

CONTINUE

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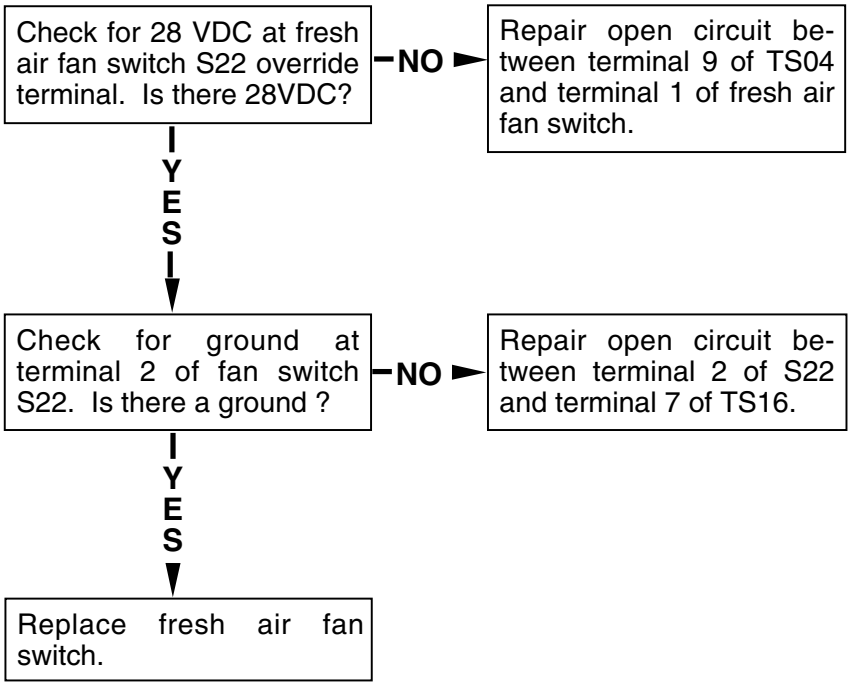

END

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FRESH AIR DISTRIBUTION – FAULT ISOLATION

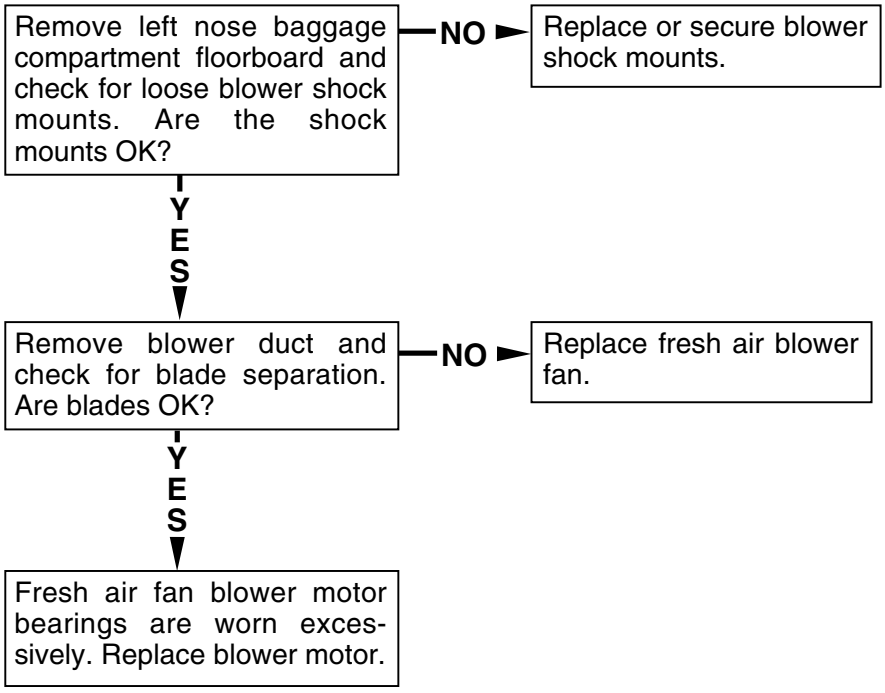
PROBLEM:

LOAD NOISE AND VIBRATION WHEN FRESH AIR FAN IS ON.

PREREQUISITES:

Electrical Power ON, Fresh Air Fan Circuit Breaker IN, FRESH AIR switch in OVERRIDE.

START:



END

6M212010106

WING BLEED AIR DISTRIBUTION – DESCRIPTION AND OPERATIONS

1. General

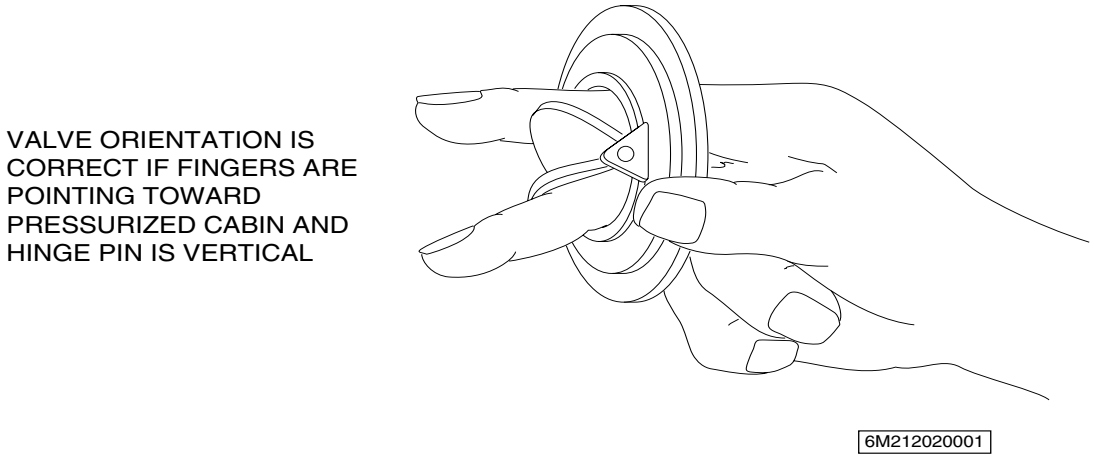
A. The wing distribution system routes air from the engine into the fuselage. The system is consists of bleed air distribution plumbing, conditioned air distribution plumbing, cold air distribution plumbing, and duct check valves.

B. Distribution Duct Check Valves

(1) The distribution duct check valves are spring loaded, flapper type check valves installed in the duct system to prevent a back flow of pressurized cabin air (see Figure 1).

(2) Pressurized air forces the valve to open in one direction, but air pressure applied from the opposite direction closes the valve.

(3) Three check valves are used in each side of the heating and air conditioning system. One check valve is installed in the fresh air fan system. Each check valve is located within a two piece coupling secured to the ducting with a clamp.



Duct Check Valve Installation
Figure 1

WING BLEED AIR DISTRIBUTION – MAINTENANCE PRACTICES

1. General – Wing Bleed Air Distribution

A. Bleed Air Distribution Plumbing

(1) Plumbing used to route bleed air is made of 304ST stainless steel (MIL-T-8504). Replacement tubes may be fabricated locally; however, locally manufactured tubes should be pressure checked to 200 psi (14.06 kg/cm^2) after manufacture. No leakage is allowed.

(2) Bleed air tubing aft of the firewall should be wrapped with insulation (pink type III Owens Corning, or equivalent). The insulation should be secured using duct tape (Permacel P-12, or equivalent). Insulation may terminate before reaching the coupling at each end of the tubes.

(3) Bleed air coupling may be accomplished by threaded fastening (B-nuts) or by slip coupling clamps. All threaded fastenings exposed to bleed air should have the threads coated with a high temperature anti-seize (Fel-Pro C5-A, or equivalent) at the time of assembly. Coupling clamps should be torqued 35 to 45 inch-pounds (3.95 to 5.08 N·m) at of assembly.

B. Conditioned Air and Cold Air Distribution Plumbing

(1) Tubing and ducts used to route conditioned and cold air are made of 5052-0 aluminum (WW-T-700/4). Replacement tubes and ducts may be fabricated locally.

(2) Coupling between conditioned and cold air ducts is accomplished by slip coupling clamps or flexible hose (sceet) and continuous clamps. Torque slip coupling clamps should be 35 to 45 inch-pounds (3.95 to 5.08 N·m). Torque continuous clamps no more than 25 inch-pounds (2.82 N·m) to prevent collapsing the aluminum duct.

(3) Wrapped conditioned air and cold air ducts and tubes with insulation (pink type III Owens Corning, or equivalent). Secure insulation using duct tape (Permacel P-12, or equivalent). Insulation may terminate before reaching the coupling at each end of the tube.

2. Removal/Installation – Wing Bleed Air Distribution

A. Removal – Duct Check Valves

(1) Remove clamp securing coupling halves. Loosen the connections on either side of the coupling to provide enough free play to remove the valve.

(2) Separate coupling halves and remove the check valve.

B. Installation – Duct Check Valves

(1) Separate coupling halves apart and insert the valve. The valve must be oriented as shown in Figure 1. Locate the hinge pin vertically in the duct.

(2) Install the clamp and torque 45 inch-pounds (5.08 N·m).

3. Inspection/Check – Wing Bleed Air Distribution

A. Inspection – Duct Check Valves

(1) Remove and inspect duct check valves in the heating and cooling system.

(2) Check that valves operate freely.

(3) Check for signs of corrosion. Clean or replace any sticky or corroded valve.

(4) Check couplings and clamps for security or evidence of leakage.

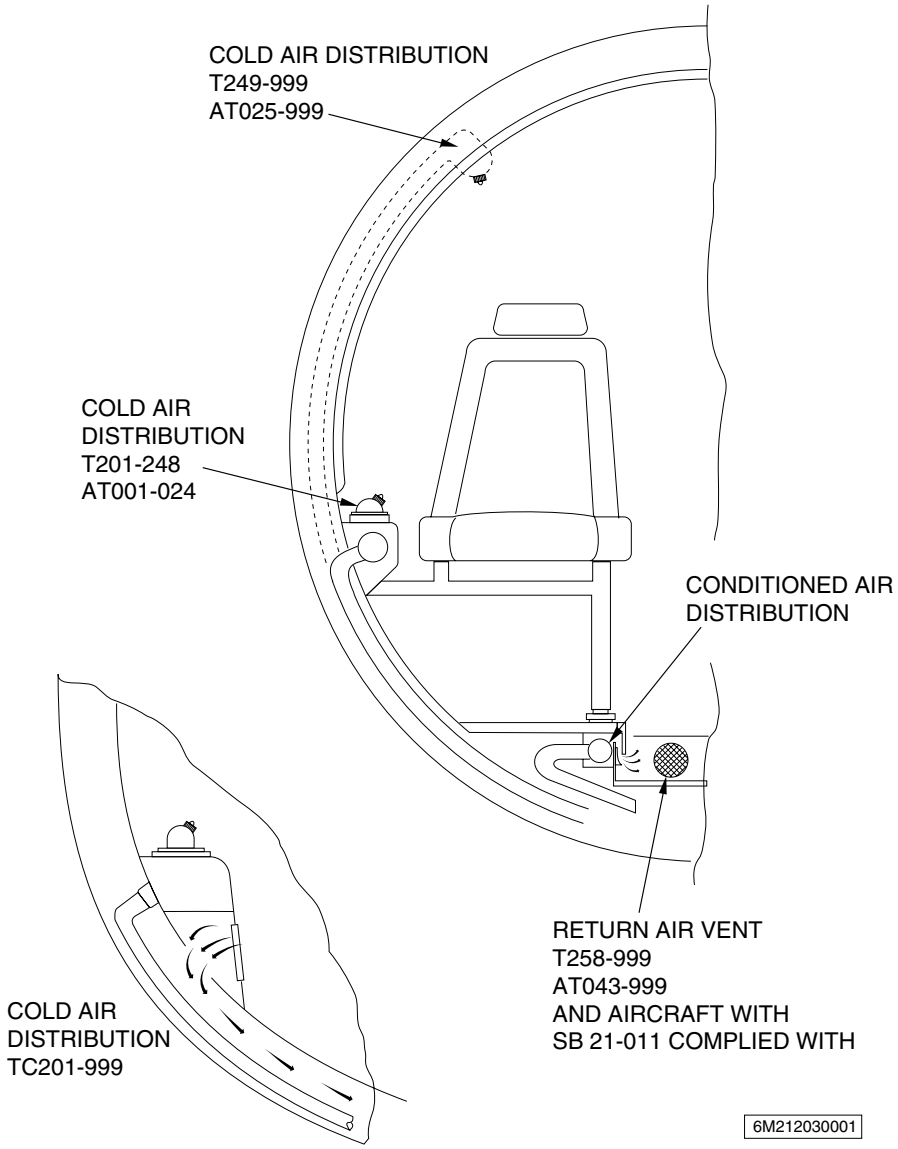
CABIN AIR DISTRIBUTION – DESCRIPTION AND OPERATIONS

1. General

A. Air distribution through the cabin is accomplished by full length ducts for conditioned air, and individual eyeball outlets for cold air.

B. Conditioned Air Distribution

(1) Conditioned air enters the aircraft just forward of the main spar. A muffler is installed just aft of the conditioned air inlet to quiet air noise. After leaving the muffler, conditioned air is routed (Figure 1) to a duct below the inboard attachment for each passenger seat. A deflector shield directs the conditioned airflow downward. A separate line routes air forward to the cockpit.



Cabin Air Distribution System
Figure 1

C. Cold Air Distribution

(1) Cold air distribution is accomplished through individually controllable eyeball outlets. Early T and AT aircraft are equipped with cold air eyeballs installed on a waist level duct. Later T and AT aircraft are equipped with cold air eyeball outlets installed overhead (Figure 1). A separate line routes cold air to the cockpit.

D. Cabin Air Return

(1) Cabin air introducted by the ECS travel through the aircraft and exits through the outflow valve. T and AT aircraft are equipped with a return vent located at the aft end of the passenger isle (Figure 1).

NOTE: Vents can be retrofitted to earlier aircraft through SB 21-011. Air enters the vents, passes under the aft baggage compartment, and exit through the outflow valve.

(2) TC return airflow is through vents along the side of the fuselage. The cold air distribution duct is divided into an upper and lower section with each section being separated from the other. The backside of the lower section is open behind the interior panel. Air enters through the vents, travels behind the interior panels and underneath the floorboards aft to the outflow valve.

MAINTENANCE MANUAL

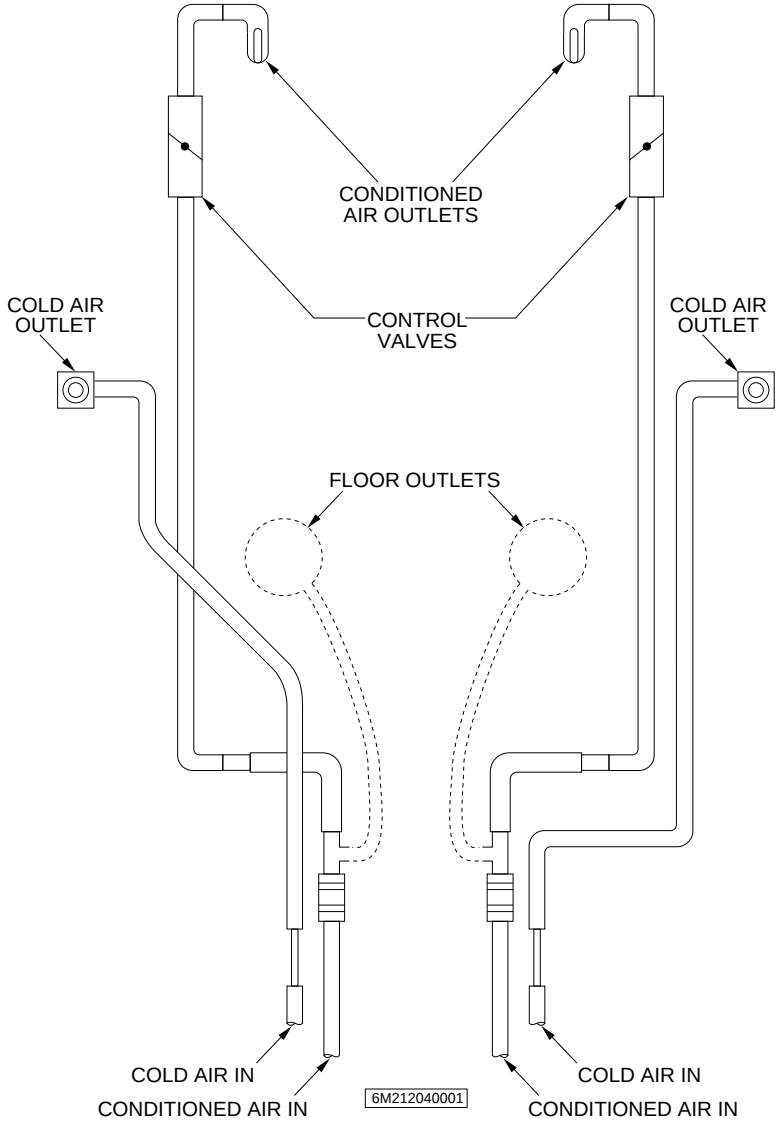
COCKPIT AIR DISTRIBUTION – DESCRIPTION AND OPERATIONS

1. General

A. The cockpit air distribution system consists of a cold air outlet, for each pilot and conditioned air outlets above the rudder pedals.

(1) Each conditioned air outlet is controlled by a variable butterfly valve to allow each pilot to select the amount of conditioned air desired.

(2) Earlier aircraft were equipped with two additional floor level conditioned air outlets (Figure 1).



Cockpit Air Distribution
Figure 1

COCKPIT AIR DISTRIBUTION – DESCRIPTION AND OPERATIONS

B. Aluminum ducts and tubes and wire reinforced flexible hose (sceet) are used to supply conditioned and cold air to the cockpit. Ducts are routed through the side console to the respective termination points.

MAINTENANCE MANUAL

COCKPIT AIR DISTRIBUTION – MAINTENANCE PRACTICES

1. Removal/Installation – Cockpit Air Distribution

A. Removal – Sceet Hose and Plastic Hose

(1) Remove continuous clamps or cable ties attaching sceet or plastic hose to aluminum duct.

(2) Remove sceet or plastic hose from duct.

B. Installation – Sceet Hose and Plastic Hose

(1) Install sceet or plastic hose onto aluminum ducts.

NOTE: Plastic hose screws onto ducts.

(2) Secure ends with continuous clamps or cable ties as required. Torque continuous clamps to 25 inch-pounds (2.82 N·m).

(3) Cover aluminum ducts, wtes, tees, and clamps with thermofit RT-876, or wrap with high temperature electrical tape LW401B12.

(4) Cover sceet or plastic hoses with thermofit RT-876, or wrap with AW272-9 fire sleeving.

2. Adjustment/Test – Cockpit Air Distribution

A. Functional Test – Cockpit Air Ducts

NOTE: New sceet hose has an oily preservative coating inside. Hot air causes it to vaporize and generate smoke. Use this procedure to burn off the vapor after installing new sceet hose before returning aircraft to service.

(1) Open escape hatches to provide cross draft in cabin.

(2) Set temperature control on the copilots switch panel to HOT (manual).

(3) Ground run aircraft for 15 to 20 minutes or until residue burns off.

(4) Check conditioned air and cold air metal fittings behind side console panels to prevent electrical short.

(a) Cover with thermofit RT-876, or wrap with high temperature electrical tape LW401B12.

(b) Fill any openings with RTV so that no exposed metal remains.

(c) Secure with cable ties.

(d) Cover sceet hoses with thermofit tubing RT-876, or wrap with AE272-9 fire sleeving.

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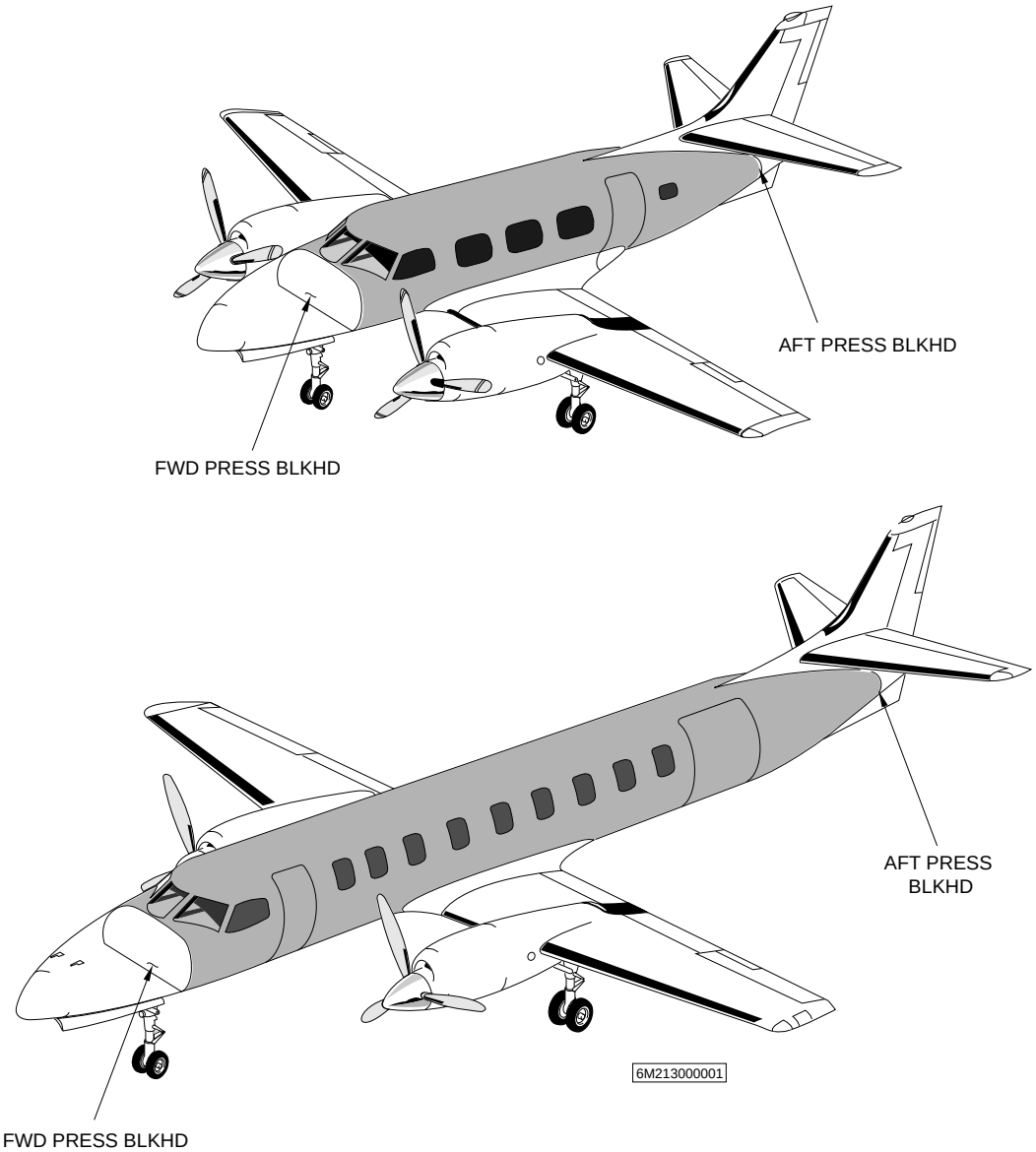
MAINTENANCE MANUAL

PRESSURIZATION – DESCRIPTION AND OPERATIONS

1. General

A. The pressurization system is designed to maintain aircraft cabin pressure at any selected pressure altitude equal to or below aircraft altitude.

B. In normal operation, the system controls cabin pressure changes and rate at which pressure changes take place.

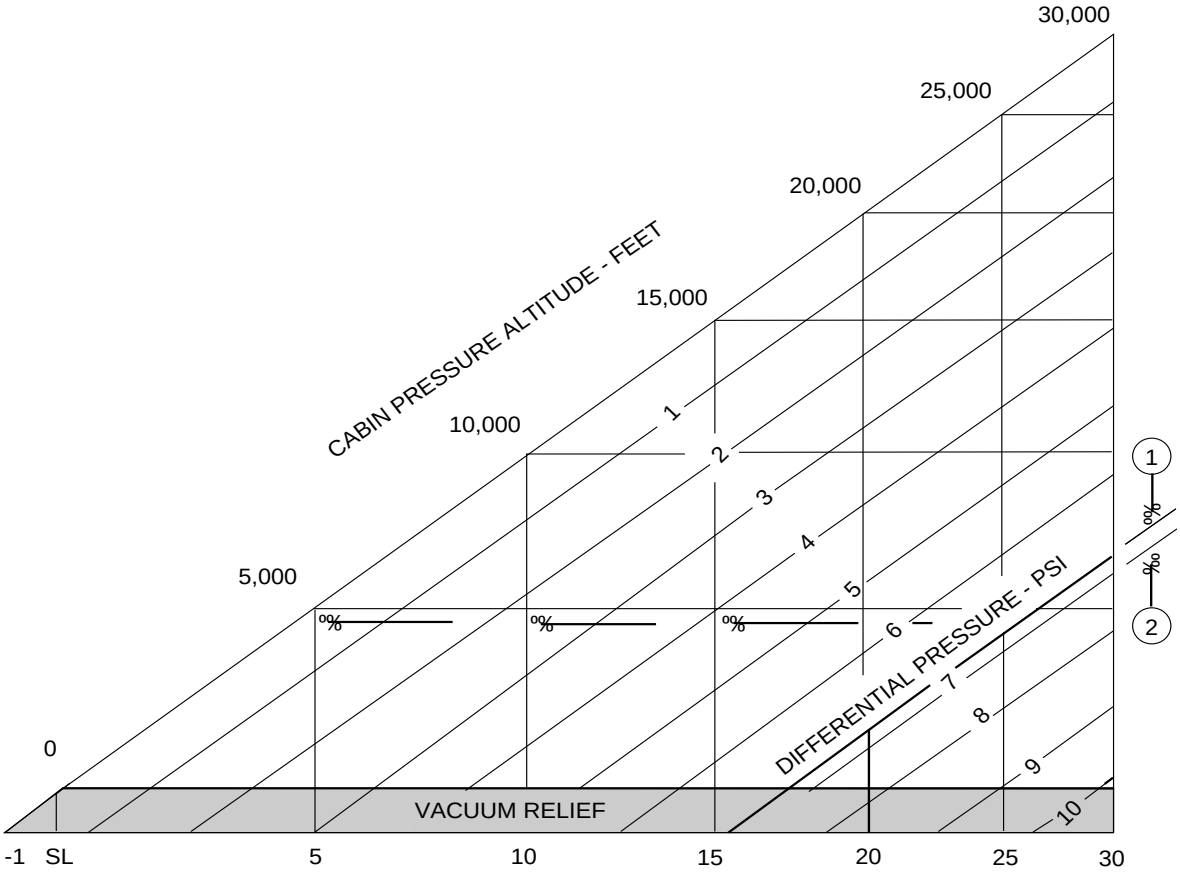


Pressurized Volume
Figure 1

PRESSURIZATION – DESCRIPTION AND OPERATIONS

C. Safety features prevent the cabin from exceeding maximum pressurization limit and/or decreasing to a negative pressure (cabin pressure less than field pressure). An emergency "dump" valve, located on the FWD pressure bulkhead, is used to expel excess cabin pressure.

D. The entire fuselage, with exception of the nose baggage compartment, is pressurized (see Figure 1). An outflow valve is installed to control cabin air pressure. The outflow valve is located on the aft pressure bulkhead. Normal airflow through the aircraft is rearward and out the outflow valve.



(1) NORMAL DIFFERENTIAL RANGE (7.00 PSI)

(2) MAXIMUM DIFFERENTIAL RANGE (7.15 - 7.40 PSI)

EXAMPLE
GIVEN: AIRCRAFT ALTITUDE = 25,000 FEET
DIFFERENTIAL PRESSURE = 7 PSI

OBTAIN: CABIN ALTITUDE = 4,600 FEET

6M213000002

Operating Ranges
Figure 2

PRESSURIZATION – DESCRIPTION AND OPERATIONS

2. Definitions

A. Aircraft Pressure Altitude – The pressure altitude at which aircraft is operating. This altitude is indicated on aircraft altimeters when set at 29.92 inches Hg and on the differential pressure needle (see Figure 3) of the cabin differential pressure indicator.

B. Cabin Pressure Altitude – The pressure altitude within the pressurized portion of the aircraft. Cabin pressure altitude will correspond with reading of the cabin altitude needle (see Figure 3) of the cabin differential pressure indicator.

C. Isobaric Range – The range cabin pressurization control system is capable of maintaining cabin pressure at level selected on the pressure controller. During operation in this range, passengers and crew will experience no change in pressure as aircraft climbs or descends.

D. Normal Differential Range – After pressurization control system has reached its limits of maintaining the cabin at selected elevation, cabin pressure changes as necessary to maintain maximum pressure differential of 7 psi. Pressure will be maintained until the aircraft descends to a level where differential between aircraft altitude and cabin altitude is less than 7 psi.

E. Maximum Differential Range – If cabin pressurization control system malfunctions, a built-in safety function incorporated in the outflow valve opens the valve when pressure differential between atmosphere and cabin reaches 7.3 +0.1 / -0.15 psi.

F. Rate of Change – The rate, calibrated in feet per minute, of cabin pressure changes. This rate is controlled by the pressure controller. The minimum value is 50 feet per minute; the maximum is approximately 2,000 feet per minute. A cabin rate of climb indicator indicates the rate of change.

G. Vacuum Relief – If aircraft were to take off from a high elevation field and land at a low elevation field, the possibility exists of the cabin being at a lower pressure than ambient pressure of the landing field. The vacuum relief portion of the cabin outflow valve prevents this by opening the valve any time cabin pressure is lower than ambient pressure.

3. Indicators and Controls

A. Cabin Rate of Climb Indicator – Changes in rate of cabin pressurization or depressurization are indicated by the cabin rate of climb indicator. In the example shown in Figure 3, the cabin is climbing (pressure decreasing) at a rate of 1,650 feet per minute.

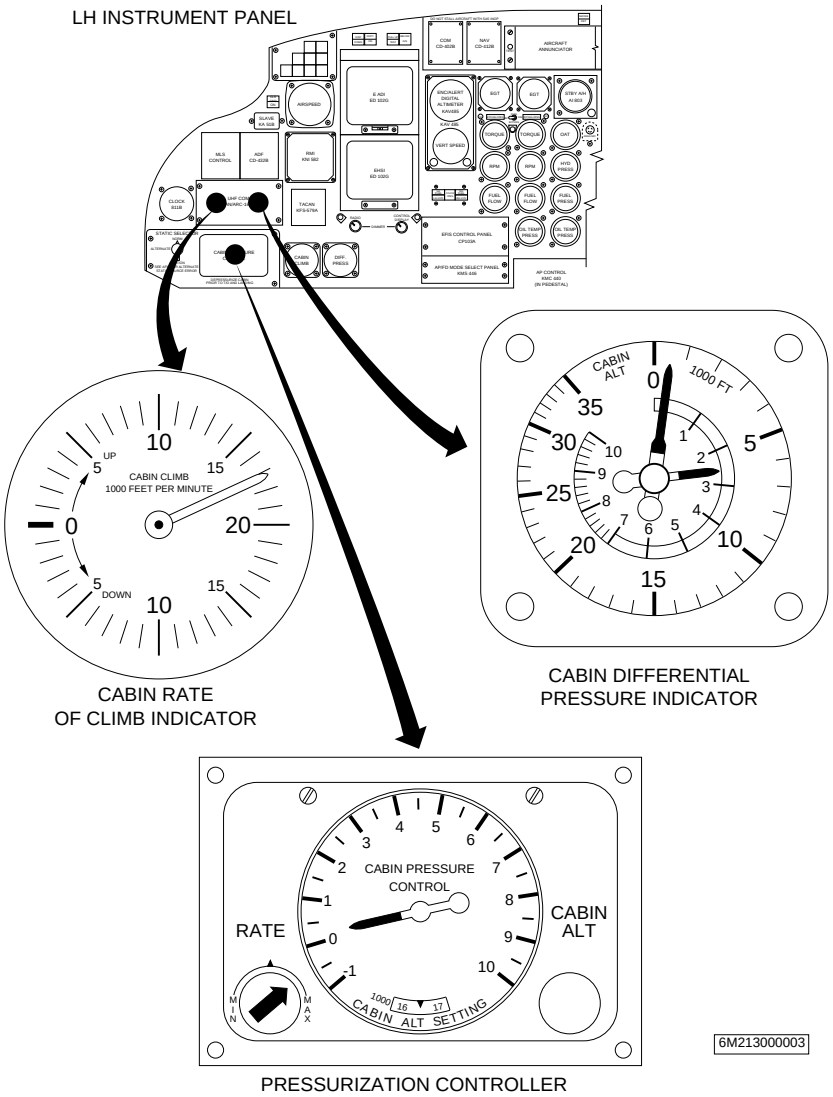
B. Cabin Differential Pressure Indicator – Cabin altitude and cabin differential pressure are displayed on a common indicator. The outer scale indicates cabin altitude while the inner scale indicates cabin pressure relative to pressure outside the aircraft. The cabin altitude scale is marked in 1,000 foot increments from 0 to 35,000 feet. The pressure differential scale is marked from 0 to 10 psi in 1 psi increments.

(1) During normal pressurized flight, cabin altitude is maintained at less than 10,000 feet. If aircraft is operating unpressurized, cabin altitude will be approximately the same as aircraft altitude.

(2) Setting pressurization controller for a sea level cabin and then climbing aircraft to 16,800 feet will cause a cabin pressure differential indication of 7.0 psi and a cabin altitude of sea level.

(3) Setting pressurization controller to 25,000 feet aircraft altitude, will provide a cabin altitude of 4,600 feet and a cabin differential pressure of 7.0 psi once the aircraft reaches the set altitude.

(4) The relationship between aircraft altitude and cabin altitude at maximum differential pressure can be seen by comparing the scales on the controller.



Indicators and Controls
Figure 3

C. Pressurization Controller – The pressurization controller establishes a control reference pressure based on pilot input, existing cabin pressure, and atmospheric pressure. Pilot input is provided by the rate control knob and cabin altitude selector knob.

(1) Reference pressure is maintained by the isobaric control mechanism within the controller, when operating at less than maximum pressure differential. When maximum cabin differential pressure of 7.0 psi is reached, the controller operates in the differential control mode. Desired cabin altitude is set using the knob on the right side of the controller while observing the indicator needle.

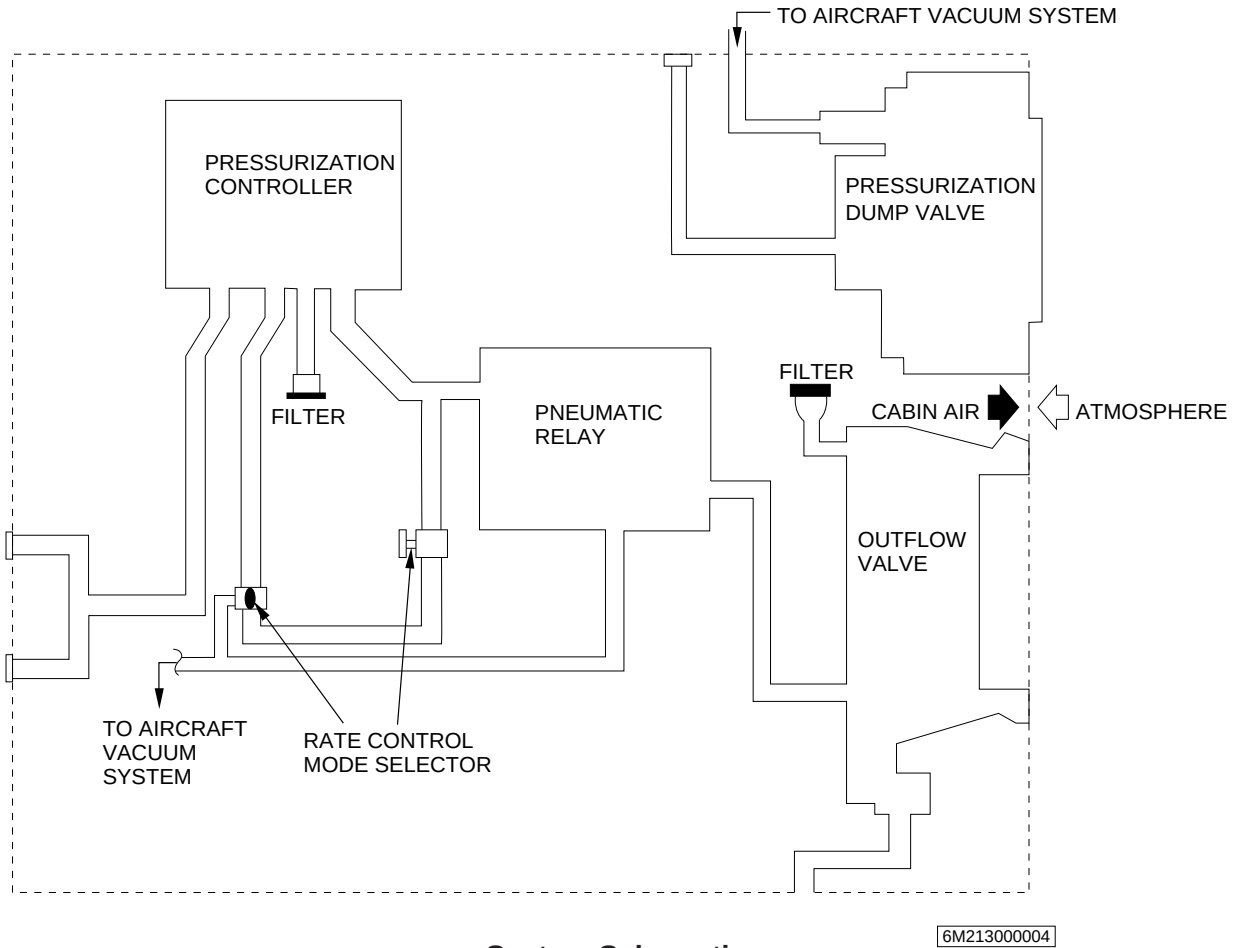
(2) Referring to Figure 3, the desired cabin altitude is set for approximately 200 feet. The small window at the bottom of the dial displays the corresponding aircraft altitude where maximum differential pressure (7.0 psi) is obtained.

(3) The rate control knob, on the left side of the controller, is used to select the rate at which cabin pressure changes.

(4) Referring to Figure 3, the rate control is set to MAX position of 2,000 feet per minute. Setting the control to minimum position produces a cabin altitude rate change of 50 feet per minute. The rate setting affects increasing and decreasing cabin altitudes.

4. System Operation

A. A constant amount of air is supplied to the cabin by the bleed air system. The pressurization system controls rate and amount of air allowed to exit through the outflow valve. When aircraft altitude is at or near cabin altitude selected on the pressure controller, the pressurization system allows air entering the cabin to escape. As the aircraft ascends, the pressurization system restricts outflow of air to maintain the originally selected pressure in the cabin.



System Schematic
Figure 4

PRESSURIZATION – DESCRIPTION AND OPERATIONS

B. When the pressurization control has reached the system pressure limit 7 psi, the cabin altitude can no longer be held constant as the aircraft continues climbing. As the aircraft ascends and descends, the pressure within the cabin changes as necessary to maintain 7 psi differential (Normal Differential Range).

C. The normal differential range will be reached at 16,800 feet pressure altitude when a sea level cabin altitude has been selected. The pressurization system then maintains pressure differential at seven (7) psi.

D. As the aircraft ascends, while in normal differential range, ambient pressure decreases and the pressurization controller causes cabin altitude to decrease (the outflow valve moved farther toward full open) as necessary to maintain the seven (7) psi differential.

E. As the aircraft descends, while in the maximum differential range, the pressure controller causes cabin pressure to increase (the outflow valve moves toward the closed position) as necessary to maintain 7 psi differential.

F. If the aircraft continues descent to an altitude where the differential between selected cabin altitude and airplane pressure altitude is less than 7 psi, the pressure controller opens the outflow valve and reduces pressure differential as necessary to maintain the selected cabin altitude (Isobaric Range).

G. When the aircraft descends to the pressure altitude selected on the pressure controller, there will be no pressure differential. If the aircraft continues to descend below the altitude selected on the pressure controller, the cabin remains unpressurized and the cabin altitude will be the same as the airplane altitude.

MAINTENANCE MANUAL

PRESSURIZATION – DESCRIPTION AND OPERATIONS

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PRESSURIZATION – FAULT ISOLATION

1. Troubleshooting – Pressurization

A. Before troubleshooting the pressurization system, a thorough search of previous maintenance activities should be made. Special attention should be paid to the following types of maintenance activities.

(1) Maintenance involving removal of the rear pressure bulkhead:

(a) Loose or kinked connections to outflow valve.

(b) Physical damage to valve or mounting.

(2) Maintenance behind instrument panel, near pressurization controller:

(a) Lines pulled out of the back of the controller.

(b) Lines kinked or punctured.

(c) Lines on the back of the suction indicator are part of the system and should be checked.

(3) Maintenance in the left console:

(a) Lines kinked, damaged or disconnected from the pressurization mode selector or manual rate control.

(b) If the left console overlay was removed, check the pressurization mode selector for proper indexing.

PRESSURIZATION – FAULT ISOLATION

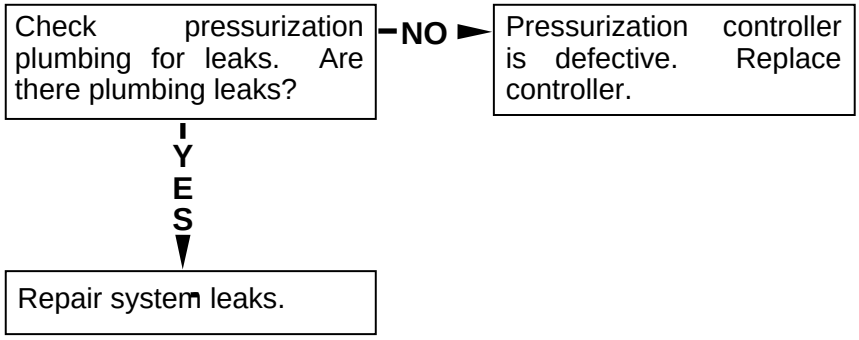
PROBLEM:

AIRCRAFT WILL NOT PRESSURIZE IN AUTOMATIC MODE.

PREREQUISITES:

Aircraft electrical power ON, both engines OPERATING and AUTO selected on pressurization controller.

START:



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PRESSURIZATION – FAULT ISOLATION

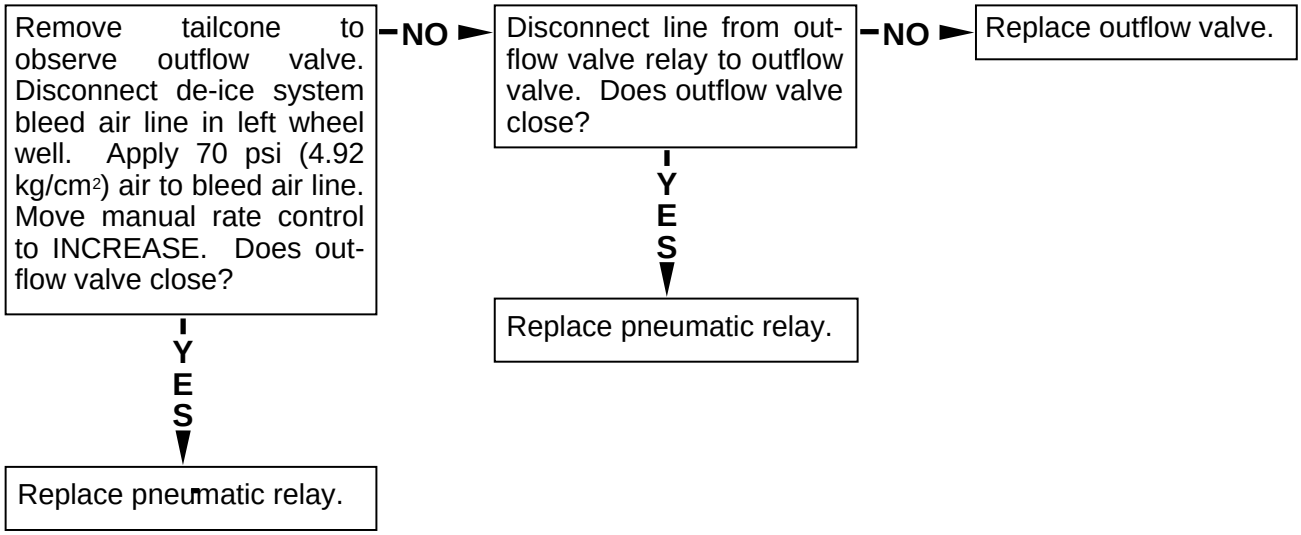
PROBLEM:

AIRCRAFT WILL NOT PRESSURIZE IN AUTOMATIC OR MANUAL MODE.

PREREQUISITES:

Aircraft electrical power ON, both engines OPERATING and MANUAL selected on pressurization controller.

START:



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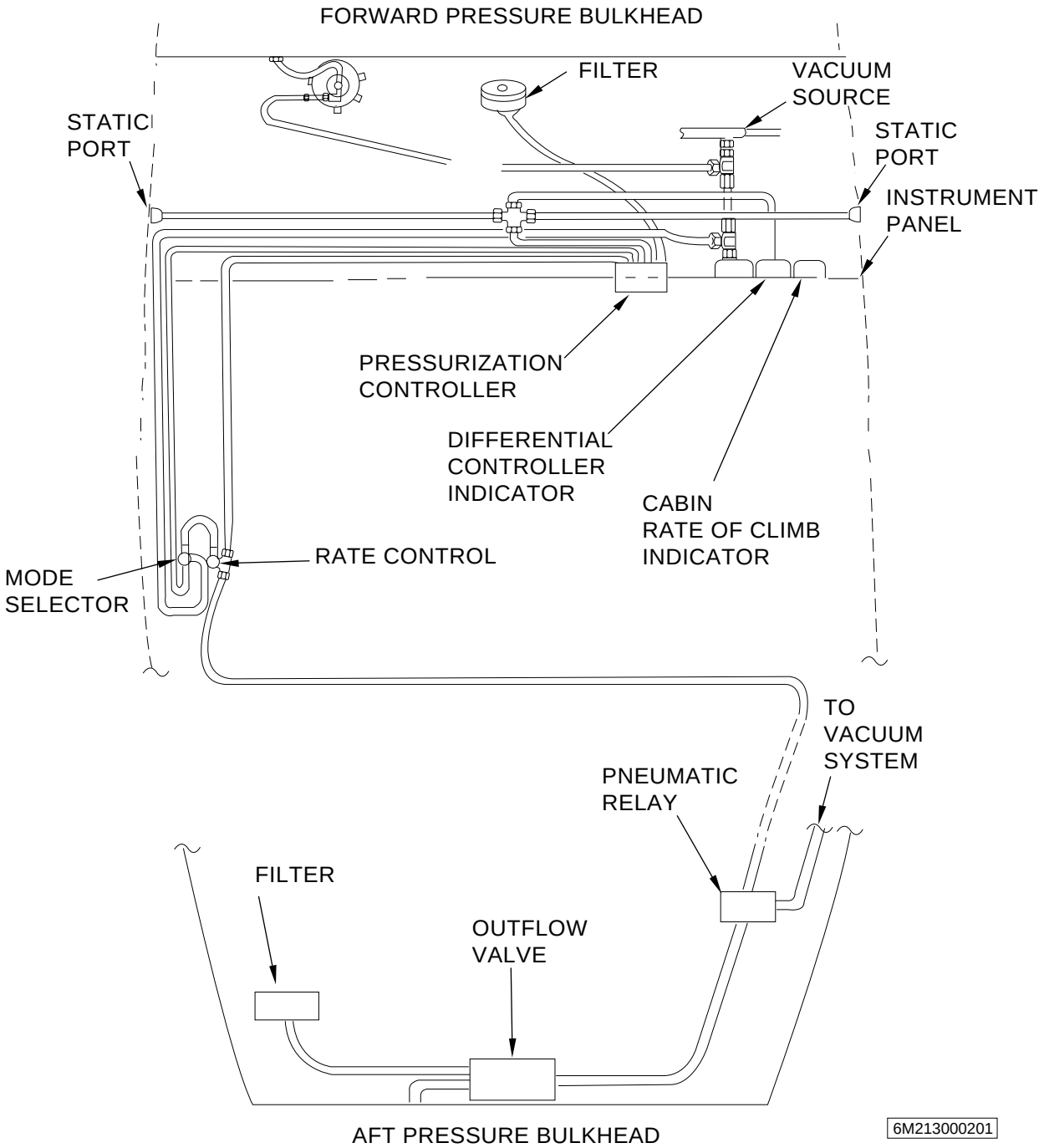
PRESSURIZATION – MAINTENANCE PRACTICES.



Component Illustration
Figure 201

MAINTENANCE MANUAL

PRESSURIZATION – MAINTENANCE PRACTICES

1. Adjustment/Test – Pressurization

A. Pressurization Test

(1) Move aircraft to a suitable run up area.

(2) Start engines in accordance with the AFM.

(3) Open CABIN PRESS DUMP CB on left console.

(4) Position cabin pressure dump switch to NORM.

(5) Position both BLEED AIR switches ON.

(6) Position pressurization mode selector to AUTO.

(7) Rotate cabin altitude control to position cabin altitude pointer at the six (6) o'clock position.

(8) As the airplane begins to pressurize, position RATE selector to MIN.

(9) Verify rate of climb is 50 feet per minute down.

(10) Position RATE selector to MAX.

(11) Verify rate of climb is 2000 feet per minute down.

(12) When cabin differential pressure reaches approximate two (2) to three (3) psi, reposition altitude selector to 2000 feet.

(13) Verify cabin rate of climb indicator indicates a climb.

(14) Position the RATE selector to MIN.

(15) Verify rate of climb indicator indicates approximately 50 feet per minute climbing. Position rate control to MAX; the rate of climb should indicate approximately 2000 feet per minute up.

(16) When differential pressure indicator approaches zero, reposition pressurization mode selector to MANUAL.

(17) Close manual rate control.

(18) Verify cabin rate of climb indicator indicates a descent.

(19) Slowly open manual rate control.

(20) Verify that rate of climb indicator, indicates a climb.

(21) When cabin pressure reaches 0.5 psi or less, reset the CABIN PRESS DUMP circuit breaker.

NOTE: A load noise caused by rushing air may be heard behind the instrument panel when dump valve opens and the cabin rate of climb may peg on the up side.

(22) When the pressure is completely dumped, return the pressurization mode selector to AUTO.

B. Static Pressurization Test

NOTE: Aircraft may be pressurized without electrical power. The dump valve cannot be operated nor tested if electrical power is not applied to the aircraft.

(1) Position both bleed air switches ON.

(2) Position cabin altitude control knob to a setting below zero; the pointer should be position at approximately six (6) o'clock.

(3) Position pressurization mode selector to AUTO.

(4) As the aircraft begins to pressurize, move cabin pressure RATE selector to MIN.

(5) Verify rate of climb indicates approximately 50 feet per minute down.

(6) Position cabin pressure control RATE selector to MAX.

(7) Verify rate of climb indicates approximately 2,000 feet per minute down.

(8) As cabin differential pressure approaches two (2) to three (3) psi, position ALTITUDE selector to 2,000 feet.

(9) Verify cabin rate of climb indicator indicates a climb.

(10) Position pressurization controller RATE control to MIN.

(11) Verify rate of climb indicates approximately 50 feet per minute up.

(12) Position pressurization control RATE control to MAX.

(13) Verify rate of climb indicates approximately 2,000 feet per minute up.

(14) As cabin differential pressure approaches zero, reposition the pressurization mode selector to MANUAL.

(15) Ensure manual rate control is completely closed (clockwise).

(16) Verify cabin rate of climb indicator indicates a descent.

(17) Slowly open manual rate control.

(18) Verify rate of climb indicator indicates a climb.

(19) When cabin pressure reaches 0.5 psi or less, reset the CABIN PRESS DUMP circuit breaker.

NOTE: A loud noise caused by rushing air may be heard behind the instrument panel when dump valve opens and the cabin rate of climb may peg on the up side.

(20) When the pressure is completely dumped, return the pressurization mode selector to AUTO.

C. Test Procedure

WARNING: MAXIMUM DIFFERENTIAL PRESSURE MAY CAUSE INNER EAR PAIN AND DISCOMFORT. IF INDIVIDUALS ARE EXPERIENCING DISCOMFORT, PRESSURIZATION OPERATIONS SHOULD BE STOPPED IMMEDIATELY AND THE PERSON ALLOWED TO LEAVE THE AIRPLANE.

(1) Standard Test Gauge set up procedure.

(a) Open left nose baggage compartment door.

(b) Locate one of the blanked bulkhead unions on FWD pressure bulkhead that feeds through to pressure vessel. Remove cap from blanked union.

(c) Attach necessary plumbing to permit temporary mounting of a calibrated pressure gauge in the upper aft corner of the left baggage door frame.

(d) Make connections between bulkhead union and a calibrated test gauge of 20 psi minimum capacity.

(e) Secure test gauge to left baggage door frame with tape, clamp or other device. Position test gauge so that it can be monitored from the pilots seat.

(2) Absolute Pressure Test Gauge set up procedure.

(a) Position a calibrated test gauge of 20 psi minimum capacity on top of the glareshield. Position gauge so that it can be monitored from the pilots seat.

(3) Prior to beginning test, record ambient pressure of absolute pressure test gauge.

NOTE: Test gauges are installed to verify pressurization control indicators. Rely on test gauges to determine actual pressures. Cabin pressure (actual) is determined by subtracting original ambient pressure recorded in step **1.**, C., (3) from observed pressure indicated on absolute pressure test gauge during pressurization test.

(4) Start engines IAW AFM.

(5) Position pressurization mode selector to manual.

(6) Adjust manual rate control full clockwise.

(7) Position both bleed air switches ON.

(8) Verify cabin rate of climb needle indicates a descent.

(9) If the rate of pressurization is too fast, causing ear discomfort for occupants, the manual rate control may be adjusted to proved a slower rate of pressurization.

(10) Monitor pressure differential indicator and test gauges.

PRESSURIZATION – MAINTENANCE PRACTICES

CAUTION: DO NOT EXCEED 7.5 PSI PRESSURE DIFFERENTIAL. THE AIRCRAFT STRUCTURE IS NOT STRESSED FOR GREATER PRESSURES.

(11) When the cabin pressure has reached seven (7) psi as indicated by the test gauge, conduct interior leak check as follows:

(a) Position left bleed air switch OFF.

(b) Hold smoke generator near eyeball outlets.

(c) Check that smoke is not drawn into the eyeball.

(d) Position left bleed air switch ON and right bleed air switch OFF.

(e) Repeat steps **1.**, C., (11), (b) through (11), (c).

(f) When leak check is complete, position both bleed air switches to ON.

(g) Slowly open the manual rate control and decrease cabin pressure.

(12) Adjust the control to provide desired depressurization rate.

(13) When the cabin differential pressure and test pressure indicators indicate zero, shut down engines IAW the AFM.

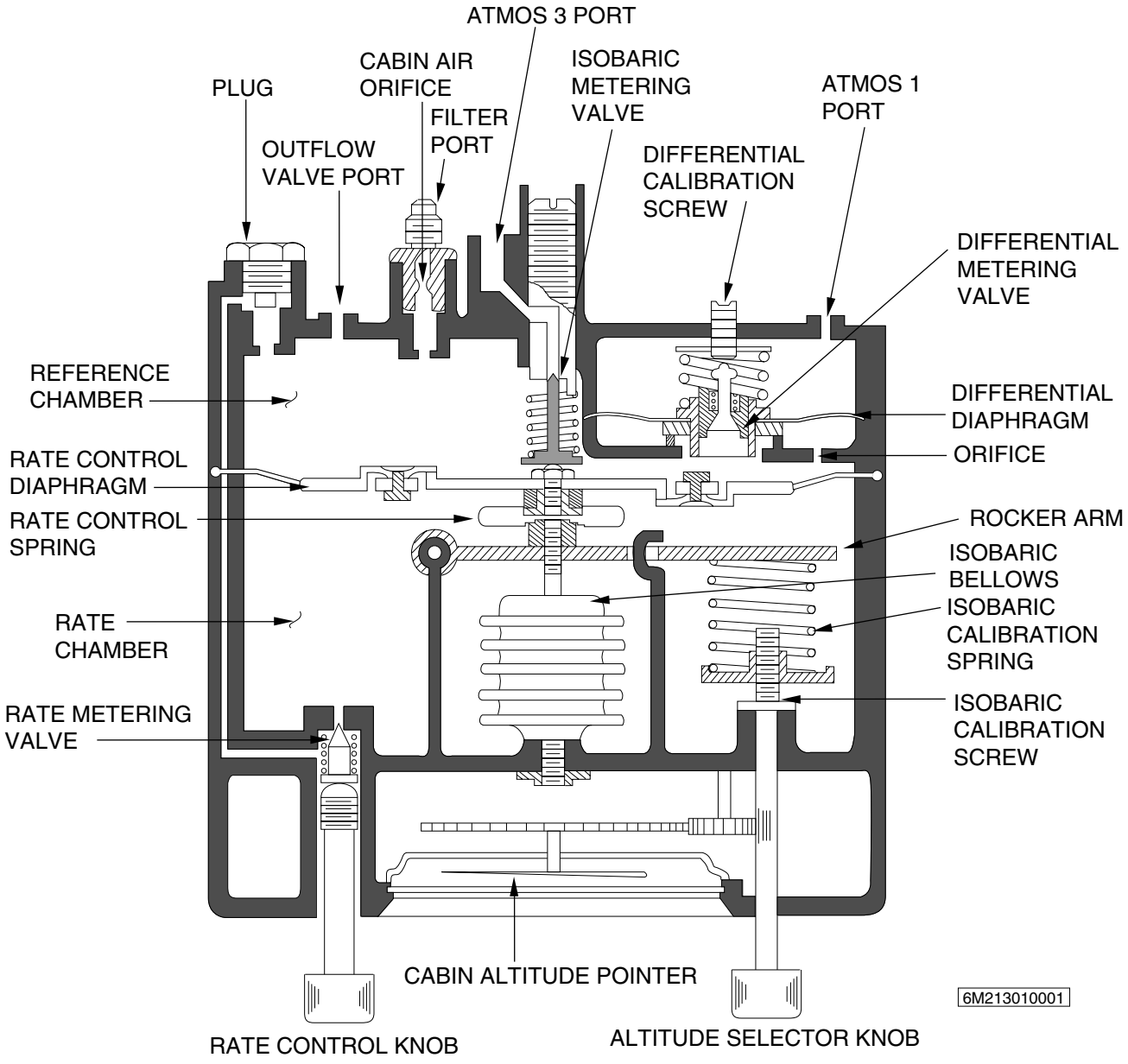
(14) Remove test gauges and plumbing.

(15) Secure bulkhead union blanking plug.

PRESSURIZATION CONTROLLER AND FILTER – DESCRIPTION AND OPERATION

1. General

A. When the pressurization system is in the AUTO mode, the cabin pressurization controller controls the outflow valve to vary the pressure within the cabin. The pilot, sets the field elevation and rate of pressurization change. The controller will maintain pressure within the aircraft at field elevation until the maximum pressure differential (seven psi) between cabin and ambient is reached. After the maximum differential is reached, the controller will position the outflow valve so as to maintain but not exceed seven (7) psi.



Pressurization Controller
Figure 1

PRESSURIZATION CONTROLLER AND FILTER – MAINTENANCE PRACTICES

1 Removal/Installation – Pressurization Controller and Filter

A. Removal - External Filter (See Figure 201).

(1) Locate filter

NOTE: Filter may have been moved during avionics installation.

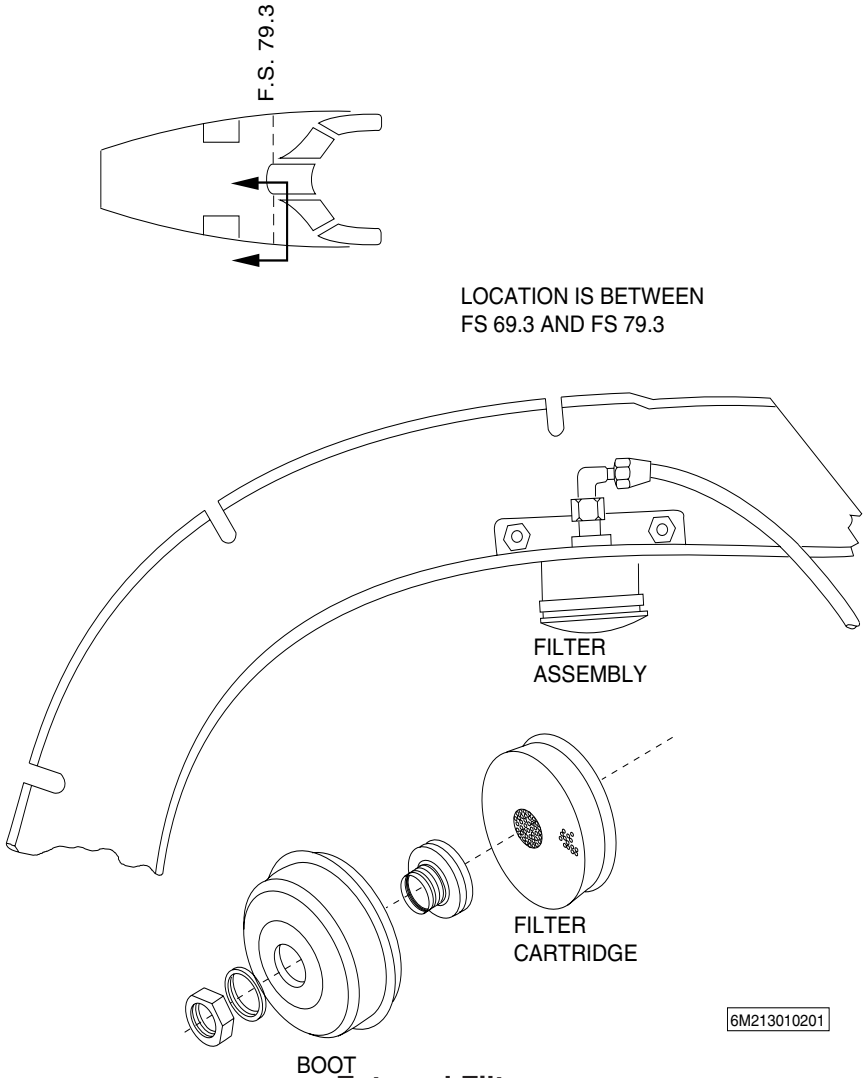
(2) Disconnect filter line and remove filter.

(3) Push filter cartridge out of boot and discard cartridge.

B. Installation - External Filter (Figure 201)

(1) Install new cartridge into boot.

(2) Install filter in aircraft and reconnect line.



External Filter
Figure 201

PRESSURIZATION CONTROLLER AND FILTER – MAINTENANCE PRACTICES

C. Removal – Internal Filter (See Figure 202).

(1) Remove plug from filter port

(2) Remove outer screen, gauze, and inner screen.

(3) Clean screens with isopropyl alcohol, or replace screens if they are defective.

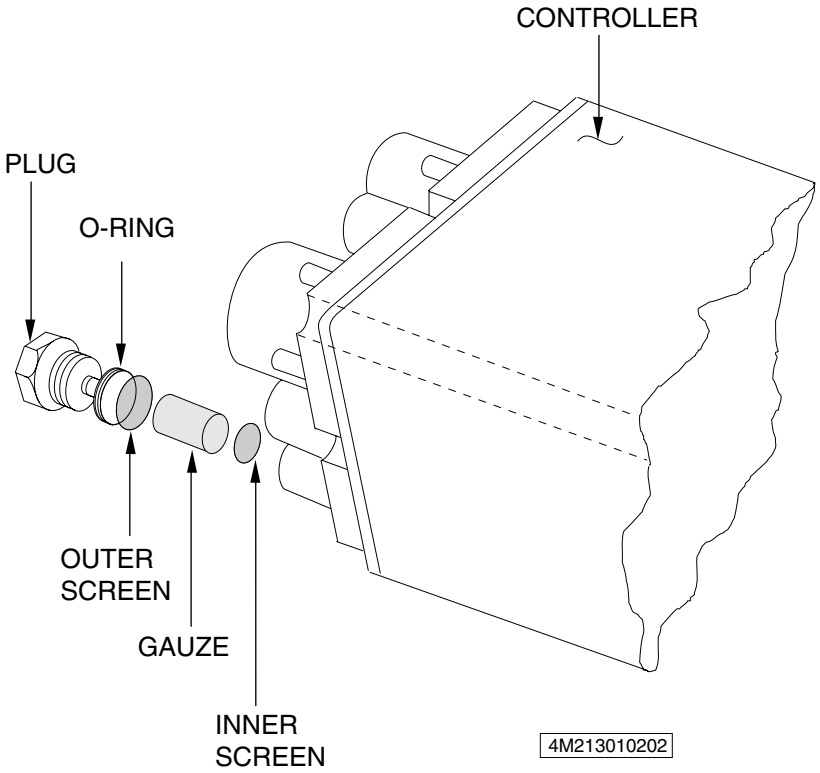
D. Installation - Internal Filter (See Figure 202).

(1) Install inner (smaller diameter) screen.

(2) Fold new gauze (12 ply, 1.5 X 3.0 inches) evenly and insert uniformly so that recess is filled but not tightly packed.

(3) Install outer (large diameter) screen.

(4) Install plug.



Internal Filter
Figure 202

E. Removal – Pressurization Controller (Figure 203).

(1) Gain access to controller.

NOTE: Depending upon avionics installation, access may be gained by removing the glareshield or by removing avionics equipment.

(2) Tag and remove lines to controller. Cap open lines.

(3) Remove screws securing controller to instrument panel. Remove controller.

(4) Remove plastic fittings from back of controller.

F. Installation – Pressurization Controller

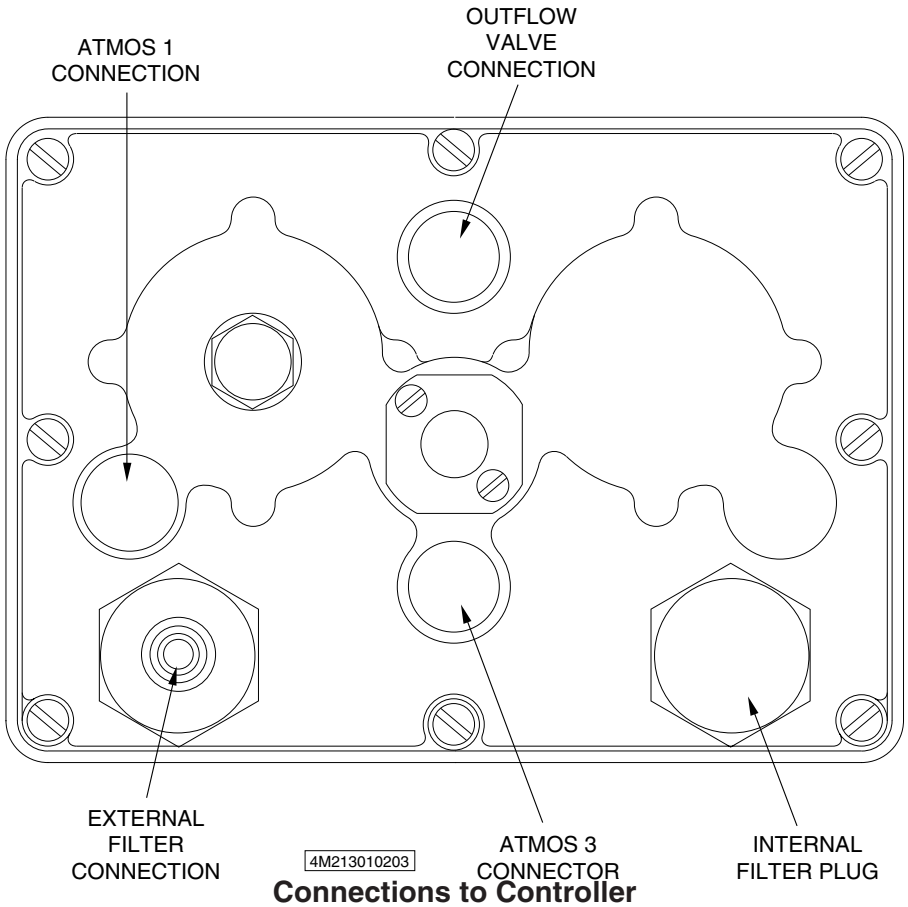
(1) Install previously removed fittings in new controller.

(2) Position controller in place and secure with screws.

(3) Remove caps and install lines on controller.

(4) Install removed avionics and glareshield.

(5) Perform pressurization system functional check.



Connections to Controller
Figure 203

2. Inspection/Check – Pressurization Controller and Filter

A. Inspection – Pressurization Controller and Filter

(1) Inspect external filter element cartridge for dirt and damage. If required install new filter element cartridge.

(2) Inspect internal filter element for security and cleanliness. Clean or replace as required.

PRESSURIZATION DUMP VALVE – DESCRIPTION AND OPERATION

1. General

A. The pressurization dump valve is used to manually depressurize the cabin should a malfunction of the pressurization system occur. The dump valve, located on the left side of the forward pressure bulkhead, is connected to a static source, the vacuum system, and the dump valve electrical circuitry.

2. Operation

A. When the aircraft is on the ground, and the dump switch is in NORMAL, power is routed through the RH landing gear squat switch, to the vacuum valve. When electrical power is applied, the valve opens and evacuates the reference chamber of the dump valve. As the relative pressure of the chamber decreases with respect to the outside atmosphere, the poppet valve return spring retracts, moving the outflow valve cover plate away from the outflow valve diaphragm, and allowing cabin pressure to escape.

B. On take off the electrical circuit supplying power to the vacuum valve disconnects at the RH squat switch. When electrical power is terminated, the vacuum valve closes and air enters through the atmospheric connection, slowly equalizing pressure between the reference chamber and outside ambient pressure. The valve closes slowly preventing a pressure dump after take off.

C. In flight, the pilot may dump cabin pressure by switching the Cabin Pressure Dump Switch to DUMP causing the vacuum valve to open and reduce the pressure inside the reference chamber. When the pressure inside the vacuum valve falls below ambient pressure, the valve opens and allows cabin pressure to escape.

D. The dump valve also contains an internal over pressurization relief valve that is forced open by cabin pressure if the difference between cabin and ambient pressure exceeds 7.30 plus or minus 0.10 psi. When open, the valve allows the escape of cabin pressure through the atmospheric connection.

E. Electrical power for the valve may be obtained from either bus.

PRESSURIZATION DUMP VALVE – MAINTENANCE PRACTICE

1. Removal/Installation – Pressurization Dump Valve

A. Removal – Pressurization Dump Valve

(1) Remove instrument panel glareshield, avionics and instruments as necessary to gain access to dump valve.

(2) Disconnect and cap lines to dump valve. Disconnect electrical connector.

(3) Remove screws securing transition duct to pressure bulkhead mounting duct.

CAUTION: WHEN REMOVING DUMP VALVE BE CAREFUL NOT TO DAMAGE ADJACENT COMPONENTS.

(4) Withdraw valve and assembly into aircraft.

B. Installation – Pressurization Dump Valve

(1) Bond new valve to duct using adhesive sealant (92-018 Dow Corning or equivalent).

(2) Install screws finger tight and safety.

(3) Coat inside of pressure bulkhead mounting duct with sealant (PR 1422A or equivalent) and install duct and valve assembly.

(4) Install fasteners securing duct.

(5) Connect lines and electrical connector to valve.

(6) Install all components removed for access.

(7) Install glareshield.

(8) Perform pressurization dump valve check.

2. Adjustment/Test – Pressurization Dump Valve

A. Operational Test

(1) Move aircraft to run up area and operate engines IAW AFM.

(2) Turn both bleed air switches ON.

(3) Position pressurization mode selector to MANUAL.

(4) Position cabin pressurization manual control to CLOSED (clockwise).

(5) Open the CABIN PRESS DUMP circuit breaker.

(6) As soon as the aircraft begins pressurizing, reset CABIN PRESS DUMP circuit breaker.

(7) Listen for a loud rush of air escaping into the baggage compartment and watch for any movement of the cabin rate of climb indicator, indicating dump valve has opened.

NOTE: If the aircraft does not begin to pressurize, advance the speed levers to a position corresponding to 80% RPM. Do not allow the aircraft to pressurize to more than 0.5 psi (as indicated on the cabin differential pressure gauge) before resetting the CABIN PRESS DUMP circuit breaker.

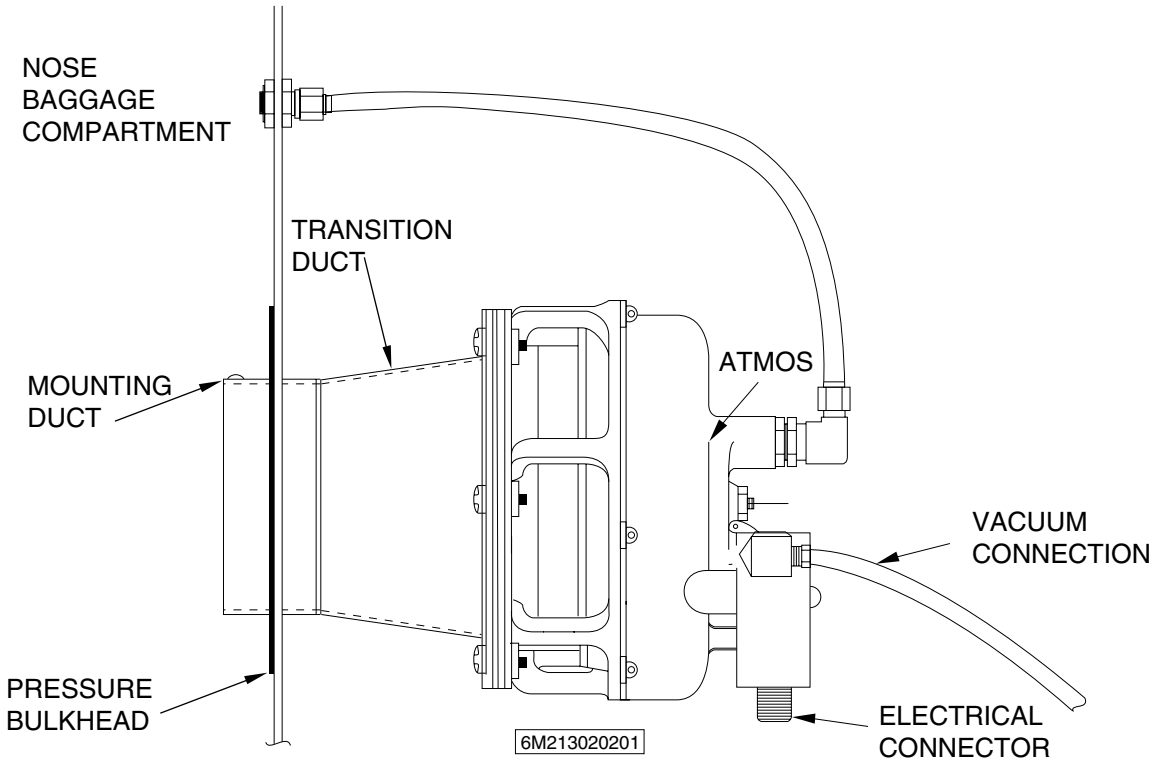
PRESSURIZATION DUMP VALVE – MAINTENANCE PRACTICE

1. Inspection/Check – Pressurization Dump Valve

A. Inspect pressurization dump valve for security of attachments and connections. The poppet valve and seat may be cleaned using NRG plus #678.

CAUTION: RAISE VALVE POPPET WITH HANDS PLACED AT BOTH SIDES OF POPPET AS NEARLY 180° APART AS POSSIBLE. RAISING POPPET FROM ONLY ONE SIDE MAY BREAK VALVE SHAFT. DO NOT USE COMPRESSED AIR TO CLEAN ANY PART OF VALVE OR DAMAGE TO VALVE DIAPHRAGM MAY RESULT.

B. NRG plus #678 is available from, National Colloid Corporation, P.O. Box 293, Garden Grove, CA 92642.



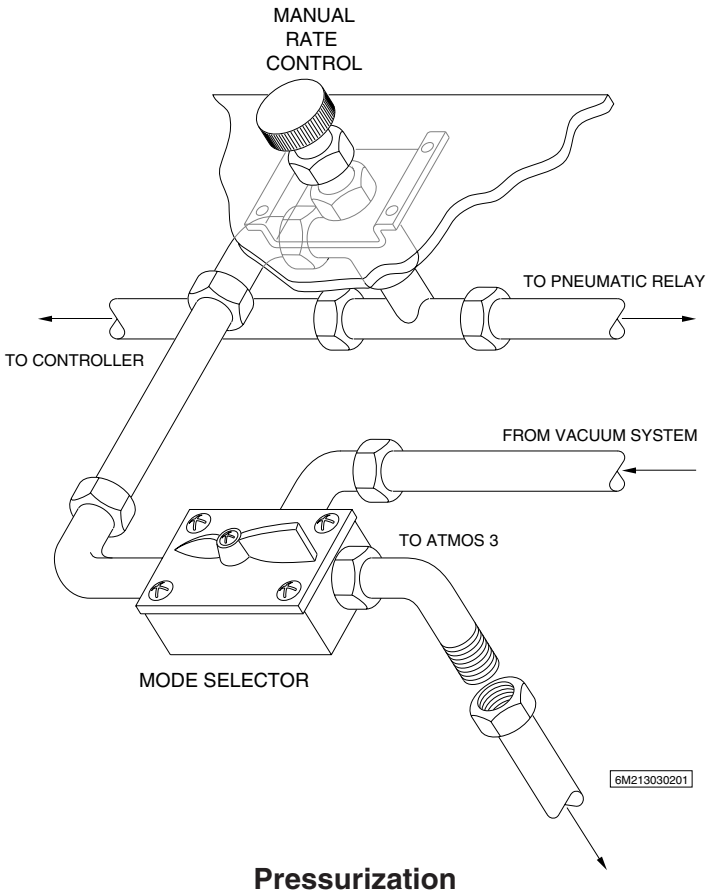
Dump Valve
Figure 201

PRESSURIZATION CONTROL – DESCRIPTION AND OPERATION

1. General

A. The pressurization control consist of a pressurization mode selector and a manual rate control. The manual pressurization rate control controls the amount of vacuum supplied to the outflow valve. Opening the valve will supply more vacuum to cause the outflow valve to open, and decrease cabin pressure. Closing the valve will supply less vacuum to cause the outflow valve to close and increase cabin pressure.

B. The manual rate control is a very coarse adjustment. When using the manual system for ground pressurization checkout, small adjustments must be made to the manual rate control and adequate time must be allowed for their effect.



Pressurization Control
Figure 201

1. Removal/Installation – Pressurization Control

CAUTION: DURING MAINTENANCE INVOLVING REMOVAL OF THE SIDE PANEL OVERLAYS OR THE HANDLE ON THE PRESSURIZATION MODE SELECTOR, MAKE SURE THE VALVE INDEX MARKS ARE ORIENTED EXACTLY AS SHOWN IN FIGURE 201. IF THE HANDLE IS INSTALLED WITH THE VALVE INCORRECTLY INDEXED, THE AIRCRAFT WILL NOT PRESSURIZE.

A. Removal – Pressurization Mode Selector

(1) Remove valve handle retaining screw and remove handle.

(2) Remove manual rate control knob retaining screw and remove knob.

(3) Remove overlay panel retaining screws and aft access door from left hand side console.

(4) Remove four screws securing valve to console.

(5) Push valve down through console opening.

NOTE: Tag lines attached to valve before disconnecting.

(6) Disconnect lines and remove valve; cap open lines.

B. Installation – Pressurization Mode Selector

NOTE: Selectors on new valves may be hard to turn. Due to drying out of the valve lubricant. The valve may be loosened by securing the valve and pressing against the center screw in the handle. Turning the handle several times will loosen the selector. The valve may be disassembled, old lubricant cleared off, and lubricated using a small quantity of LUBE 3 (P/N of Teledyne Republic Manufacturing).

When installing a new valve, remove fittings from old valve for use in new valve.

(1) Install fittings in new valve.

(2) Remove dust caps and install lines to valve.

(3) Position valve in console and secure.

CAUTION: DURING MAINTENANCE INVOLVING REMOVAL OF SIDE PANEL OVERLAYS OR PRESSURIZATION MODE SELECTOR, ENSURE VALVE INDEX MARKS ARE ORIENTED AS SHOWN IN FIGURE 201. IF HANDLE IS INSTALLED WITH VALVE INCORRECTLY INDEXED, AIRCRAFT WILL NOT PRESSURIZE.

(4) Install overlay and secure.

(5) Install knob on rate control and secure.

(6) Install access door.

C. Removal – Manual Rate Control

(1) Remove rate control knob retaining screw and remove knob.

(2) Remove mode selector handle retaining screw and remove handle.

(3) Remove overlay panel and aft access door from left side console.

(4) Remove screws and nuts securing valve bracket to console.

(5) Lower valve down into console.

(6) Tag lines attached to valve and remove lines.

(7) Cap open lines. Remove valve and bracket assembly.

(8) Unscrew valve gland nut and remove bracket.

D. Installation – Manual Rate Control

(1) Remove gland nut from new valve.

(2) Screw jam nut down (toward valve).

(3) Install bracket on valve and install gland nut. Tighten gland nut finger tight plus one half to three quarters turn.

(4) Install knob on valve and check for slight resistance as valve is opened or closed.

(5) Remove knob.

(6) Tighten jam nut against bracket.

(7) Remove caps and install lines to valve.

(8) Secure bracket to console using original hardware.

(9) Install left hand console overlay and access door.

CAUTION: DURING MAINTENANCE INVOLVING REMOVAL OF SIDE PANEL OVERLAYS OR PRESSURIZATION MODE SELECTOR, ENSURE VALVE INDEX MARKS ARE ORIENTED AS SHOWN IN FIGURE 201. IF HANDLE IS INSTALLED WITH VALVE INCORRECTLY INDEXED, AIRCRAFT WILL NOT PRESSURIZE.

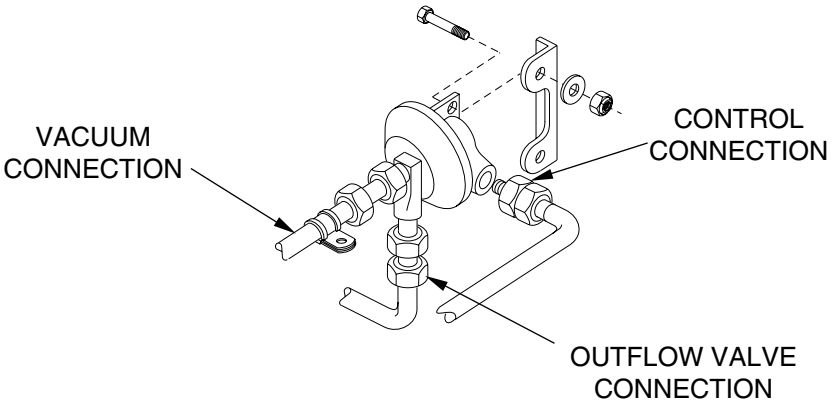
(10) Install mode selector handle and secure with screw.

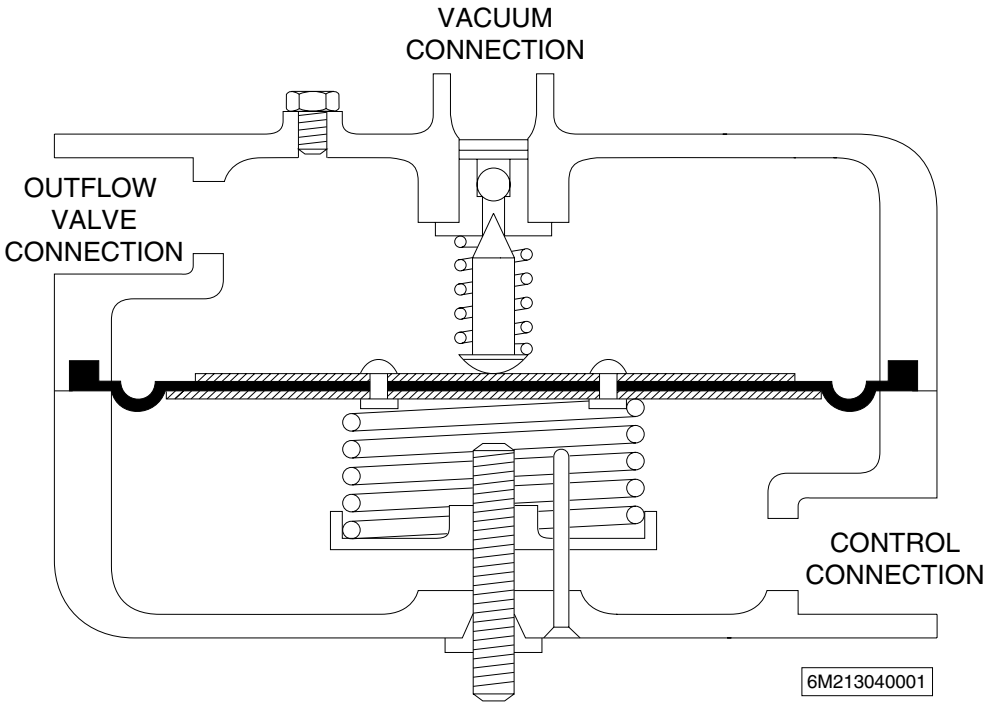
(11) Install knob on rate control and secure with screw.

PNEUMATIC RELAY – DESCRIPTION AND OPERATION

1. General

A. The pneumatic relay is used to speed up the reaction time of the outflow valve to commands given by the pressurization control system. Small changes in pressure to the control connection will produce larger but corresponding changes in pressure to the outflow valve connection. The relay is located in the aft fuselage area at F.S. 372 for SA226 – T aircraft and at F.S. 631 on SA226 – AT and TC aircraft.





Pneumatic Relay
Figure 1

PNEUMATIC RELAY – MAINTENANCE PRACTICES

1. Removal/Installation – Pneumatic Relay

A. Removal – Pneumatic Relay

(1) Remove aft compartment bulkhead to gain access to relay.

(2) Tag and remove lines; cap open lines.

(3) Remove two bolts securing relay to bracket and remove relay.

(4) Remove plastic fittings from relay if new relay is to be installed.

B. Installation – Pneumatic Relay

(1) Install fittings in new relay if new relay is being installed.

(2) Position relay to bracket and secure using original hardware.

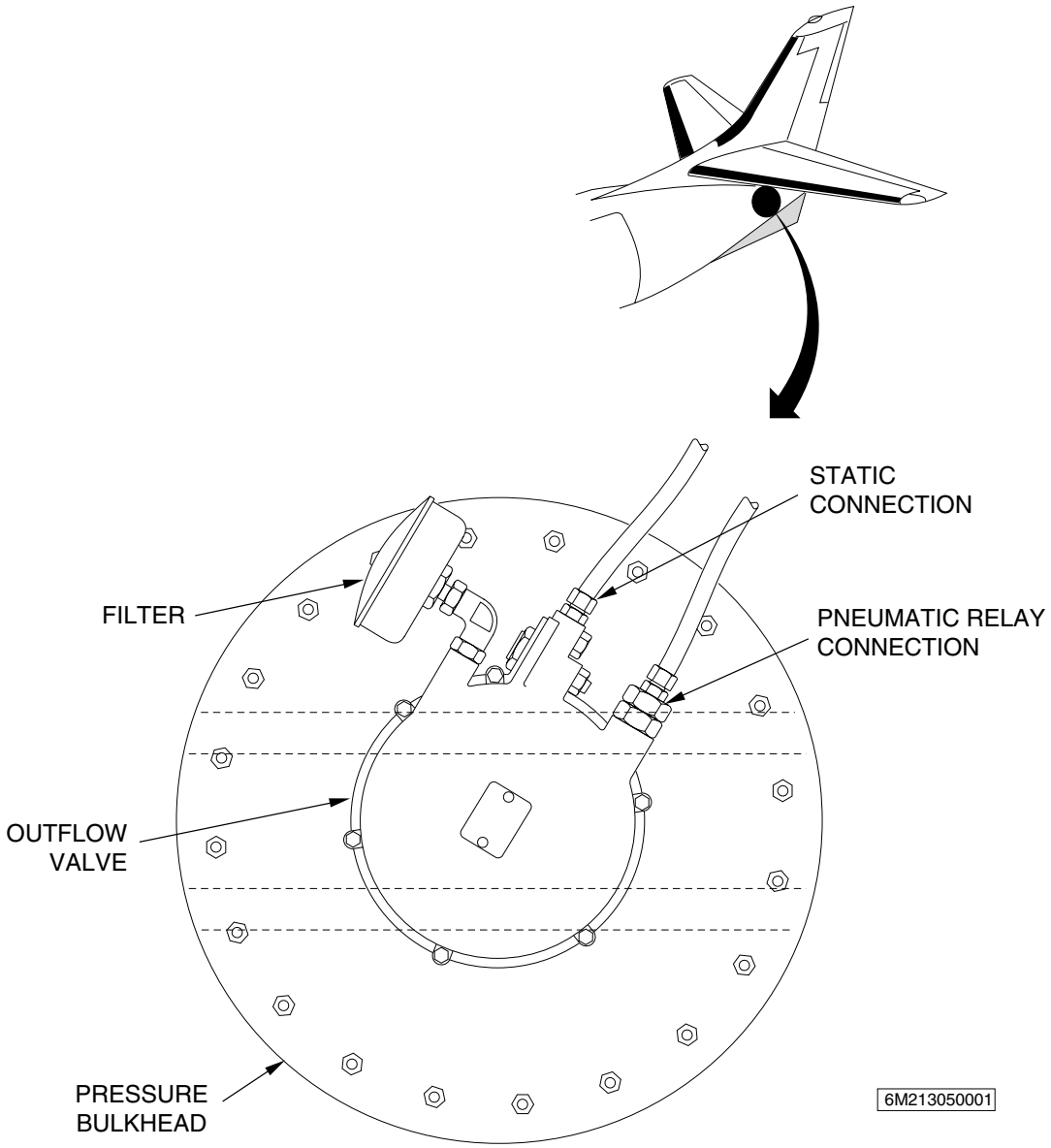
(3) Remove caps from lines and connect to relay.

(4) Install aft bulkhead.

OUTFLOW VALVE – DESCRIPTION AND OPERATION

1. General

A. The outflow valve, installed on the aft pressure bulkhead, controls the flow of air out of the aircraft pressure vessel. The valve responds to pressure commands supplied by the pressurization control system through the pneumatic relay. If the maximum differential pressure between the aircraft cabin and ambient exceeds 7.15 psi, the valve will open regardless of the command being supplied by the pressure control system. Automatic vacuum relief capability is incorporated into the outflow valve. If the pressure in the cabin is lower than the ambient pressure, the outflow valve will open to equalize the pressures.



Outflow Valve
Figure 1

OUTFLOW VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – Outflow Valve

A. Removal – Outflow Valve

(1) Remove tailcone. If applicable, remove JATO bottle and rack.

(2) Remove bolts securing aft pressure bulkhead and tilt bulkhead aft.

(3) Remove and cap lines to valve.

(4) Separate handshake disconnects in tail light wiring and remove bulkhead.

(5) For JATO equipped aircraft, turn bulkhead 180° and secure. Do not allow bulkhead weight to strain JATO wires.

(6) Remove eight bolts securing valve and remove.

(7) Remove plastic fittings and air filter from valve.

B. Installation – Outflow Valve

(1) Install original fittings in valve.

(2) Install air filter.

NOTE: Examine condition of gasket. If gasket is not torn or chipped it may be reused.

(3) Position valve in place on bulkhead and secure with bolts.

NOTE: Longer bolts go in holes that pass through the stiffening channels on aft side of bulkhead.

(4) Torque bolts evenly no more than 30 inch-pounds.

(5) Apply a bead of sealer (Coast Pro-Seal 890A or equivalent) around the edge of the outflow valve base where it contacts the bulkhead.

(6) Remove caps and connect lines to valve.

(7) Connect tail light wiring.

(8) Position bulkhead in place and secure with original hardware.

NOTE: Longer bolts go in holes that pass through the stiffening channels on aft side of bulkhead.

(9) If applicable, install JATO bottle and rack.

(10) Install tailcone. Move aircraft to a run up area and perform an operational check of the system.

2. Inspection/Check – Outflow Valve

A. Remove and inspect the rear pressure bulkhead/outflow valve assembly. Inspect the outflow valve poppet and seat for dirt or cigarette smoke residue.

CAUTION: RAISE VALVE POPPET WITH HANDS PLACED AT BOTH SIDES OF POPPET AS NEARLY 180° APART AS POSSIBLE. RAISING POPPET FROM ONLY ONE SIDE MAY BREAK VALVE SHAFT. DO NOT USE COMPRESSED AIR TO CLEAN ANY PART OF VALVE, OR DAMAGE TO VALVE DIAPHRAGM MAY RESULT.

B. Remove and replace the filter element attached to the outflow valve. The filter and element assembly are available under part number 135257-4. The element is available separately under part number CU79030 (P/N of Mine Safety Appliance Company, Pittsburgh, PA).

MZ AEROSPACE
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MAINTENANCE MANUAL

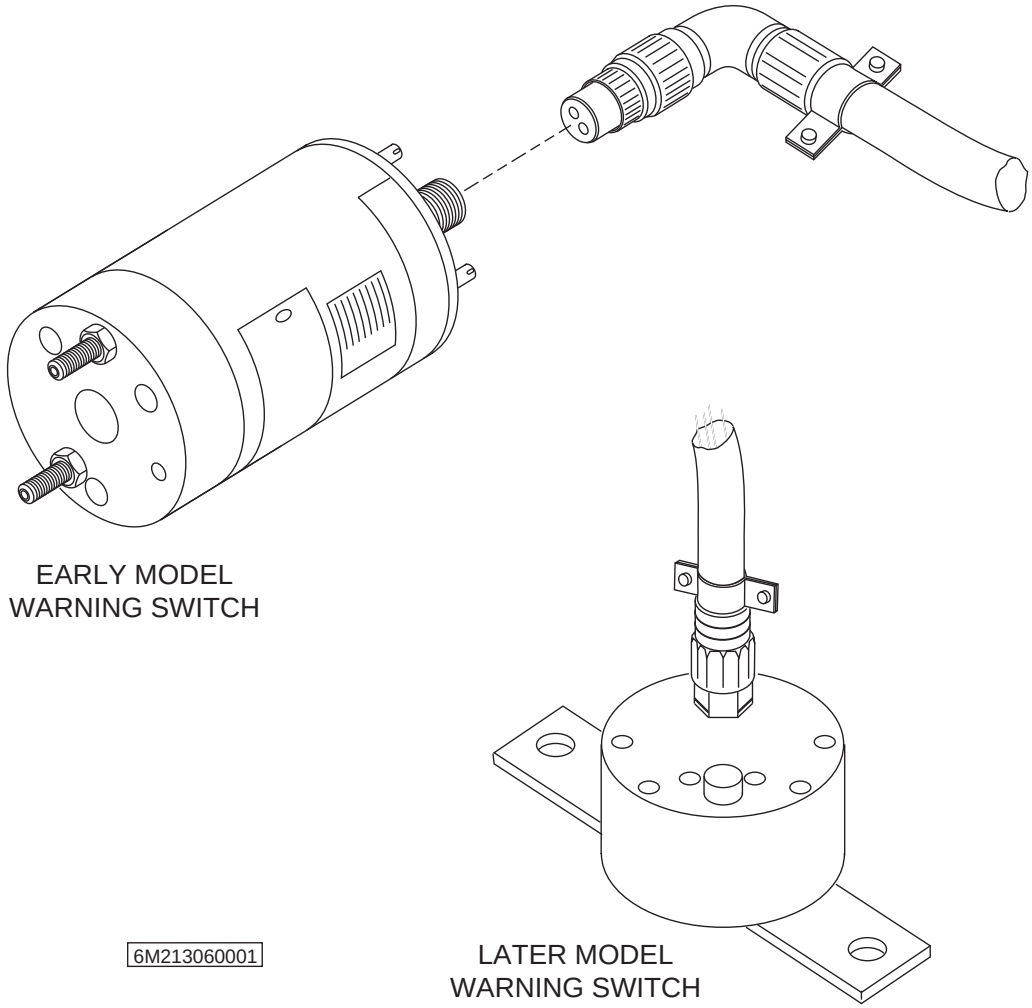
CABIN ALTITUDE WARNING – DESCRIPTION AND OPERATION

1. General

A. Cabin altitude warning is used to alert the pilot that cabin altitude has exceeded 11,000 feet. Above this altitude, supplemental oxygen is required.

B. Warning light illumination altitude is approximate. The cabin altitude warning switch trip point is set between 10,800 and 11,200 ft. ascending altitude. The trip point is set assuming an equivalent barometric pressure of 29.92 inches Hg at sea level.

C. If the pressurization system is functioning normally, the cabin altitude will be equivalent to 7000 ft. when the aircraft is at 30,000 ft. Cabin altitude should never exceed 11,000 ft. at any operating altitude and illumination of the cabin altitude warning light usually indicates a problem with the pressurization system.



Cabin Altitude Warning System
Figure 1

CABIN ALTITUDE WARNING – MAINTENANCE PRACTICES

1. Removal/Installation - Cabin Altitude Warning

A. Removal - Cabin Altitude Waning Switch

(1) Gain access to the cabin altitude warning switch located on the pressure bulkhead forward of pilot's rudder pedals.

(2) Disconnect altitude warning switch electrical connector.

(3) Remove altitude warning switch fasteners. Remove switch.

B. Installation - Cabin Altitude Waning Switch

(1) Position altitude warning switch in place. Secure using previously removed hardware.

(2) Connect electrical connector.

2. Inspection/Check - Cabin Altitude Warning Operational Test

Either 2.A. System Flight Test or 2.B. System Ground Test procedure may be used to satisfy the Cabin Altitude Warning System Operational Test requirement in Chapter 5 of the Aircraft Maintenance Manual.

A. System Flight Test Procedure

(1) Cabin altitude warning system must be in standard configuration during flight test. While complying with crew oxygen requirements per AFM, perform flight following AFM Normal Procedures and attain level 12,000 feet altitude.

(2) Prior to performing test don oxygen mask(s).

(3) Cabin pressurization mode selector – turn to manual mode.

(4) Cabin pressurization manual control knob - turn as required to control pressurization or depressurization rate.

(5) Check that cabin altitude warning light illuminates when cabin altitude is between 10,800 and 11,200 feet pressure altitude and extinguishes before cabin descends below 10,800 feet pressure altitude.

(6) If cabin altitude warning indication is not within tolerance, isolate and correct discrepancy prior to further flight. Repeat in-flight operational check to verify corrective action.

B. System Ground Test Procedure

(1) If necessary, construct a test chamber of sufficient size to contain the cabin altitude warning pressure switch (see Figure 1 for an example).

(2) Without disconnecting wiring from pressure switch, remove fasteners securing cabin altitude warning switch to bulkhead forward of pilot rudder pedals. It may be necessary to cut tie wraps from harness to allow sufficient wire length to place switch into test chamber.

(3) Place pressure switch into test chamber, install end cap and seal wiring cutout with plumbers putty or equivalent.

(4) Apply external electrical power to the aircraft (Ref 24-40-00).

(5) Connect pitot static tester vacuum hose to test chamber fitting. Apply vacuum to the test chamber and check for leaks. Correct any leaks before proceeding.

(6) While observing the altitude on pitot static tester, slowly increase the vacuum on test chamber until CABIN ALT annunciator illuminates. The annunciator warning light must illuminate between 10,800 and 11,200 feet altitude.

(7) While observing the altitude on pitot static tester, slowly decrease the vacuum on test chamber until CABIN ALT annunciator extinguishes. The annunciator warning light must extinguish before altitude descends below 10,800 feet.

(8) If installed switch does not meet these limits, replace pressure switch and perform test on new switch until a correctly operating switch is connected to the aircraft.

(9) Disconnect external electrical power from aircraft (Ref 24-40-00).

(10) Disconnect pitot static tester vacuum hose from test chamber fitting. Remove pressure switch from test chamber.

(11) Secure pressure switch to bulkhead with fasteners and secure wiring to harness as necessary.

C. Pressure Switch Ground Test Procedure

Use this procedure to test cabin altitude pressure switch setting only. This test is performed without switch connected to aircraft wiring and DOES NOT satisfy the Cabin Altitude Warning System Operational Test requirement in Chapter 5 of the Aircraft Maintenance Manual.

(1) If necessary, construct a test chamber of sufficient size to contain the cabin altitude warning pressure switch (see Figure 1 for an example).

(2) Place pressure switch into test chamber with wires connected to the two switch receptacle pins necessary to check for continuity routed through wiring cutout to outside of test chamber (Refer to Aircraft Wiring Manual, Section 21-30-07). Connect an ohmmeter to the two wires.

(3) Install test chamber end cap and seal wiring cutout with plumbers putty or equivalent.

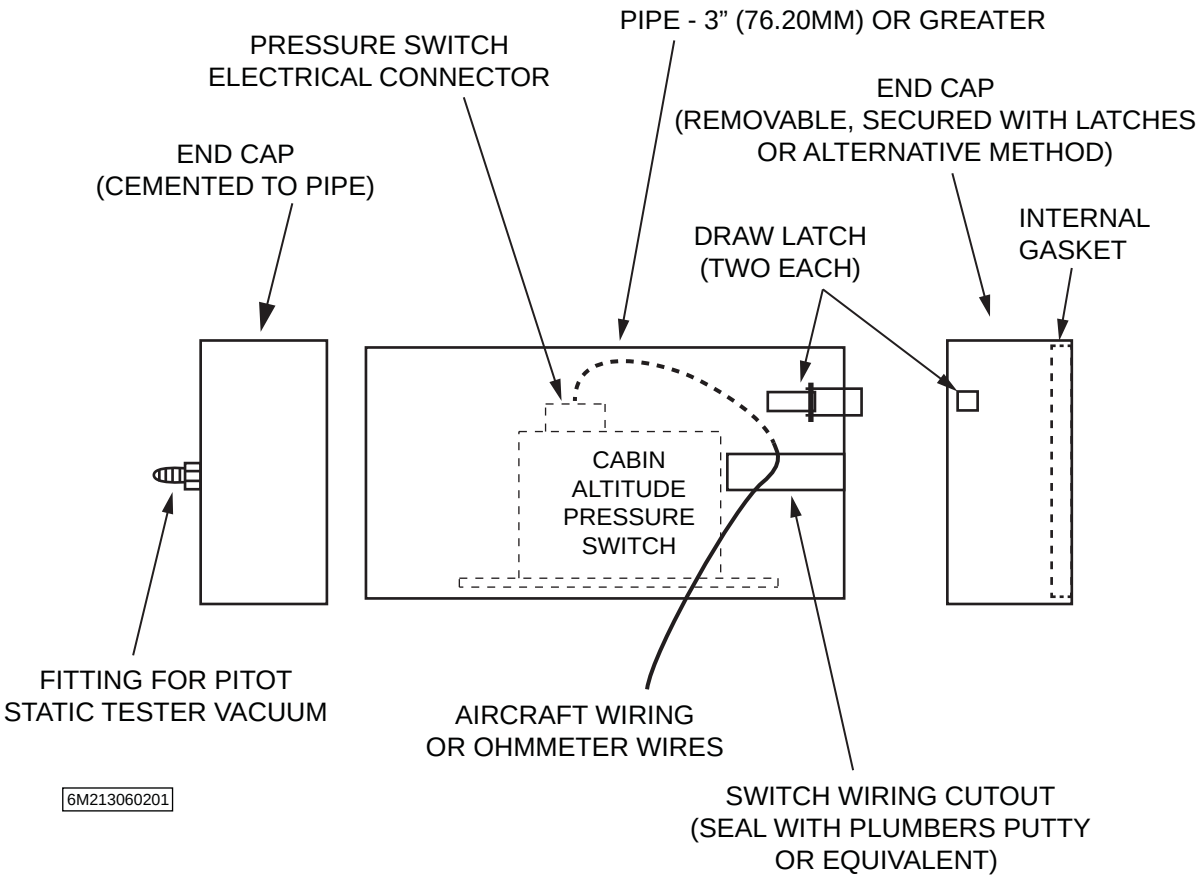
(4) Connect pitot static tester vacuum hose to test chamber fitting. Apply vacuum to the test chamber and check for leaks. Correct any leaks before proceeding.

(5) While observing the altitude on pitot static tester, slowly increase the vacuum on test chamber until ohmmeter indicates continuity. The ohmmeter must indicate continuity between 10,800 and 11,200 feet altitude.

(6) While observing the altitude on pitot static tester, slowly decrease the vacuum on test chamber until ohmmeter indicates loss of continuity. The ohmmeter must indicate loss of continuity before altitude descends below 10,800 feet.

(7) Disconnect pitot static tester vacuum hose from test chamber fitting.

(8) Remove pressure switch from test chamber.



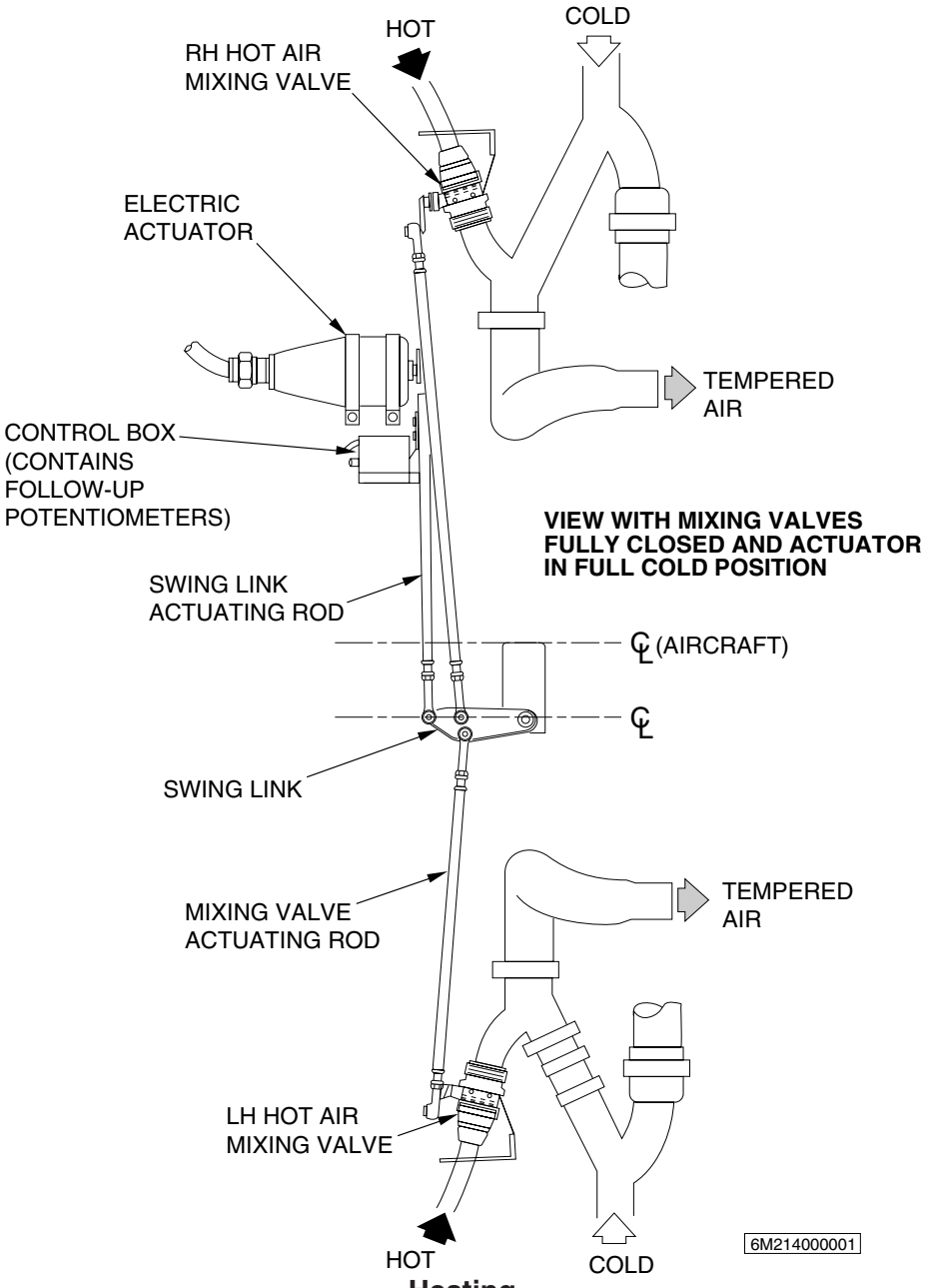
Example of Cabin Altitude Warning System Test Chamber
Figure 201

HEATING – DESCRIPTION AND OPERATION

1. General

A. Bleed air from the engine, mixed with cold air from the air conditioning turbine, is used as tempered air for heating the cockpit and cabin. A hot air mixing valve controls the flow of engine bleed air into the cold air duct. Both mixing valves are mechanically interconnected and operated by a common actuator. The actuator receives signals from the temperature control circuit and positions the mixing valves accordingly.

B. Refer to the effectivity columns in the Table of Contents for identification of your heating system.



Heating
Figure 1

1. Adjustment/Test – Heating

A. Mixing Valve Adjustment

NOTE: If the mixing valve system is disassembled or suspected of being out of adjustment, the following procedure may be used to adjust the system:

(1) Move the mode selector switch to the COLD position and allow the actuator to move to the internal stop in the full cold position.

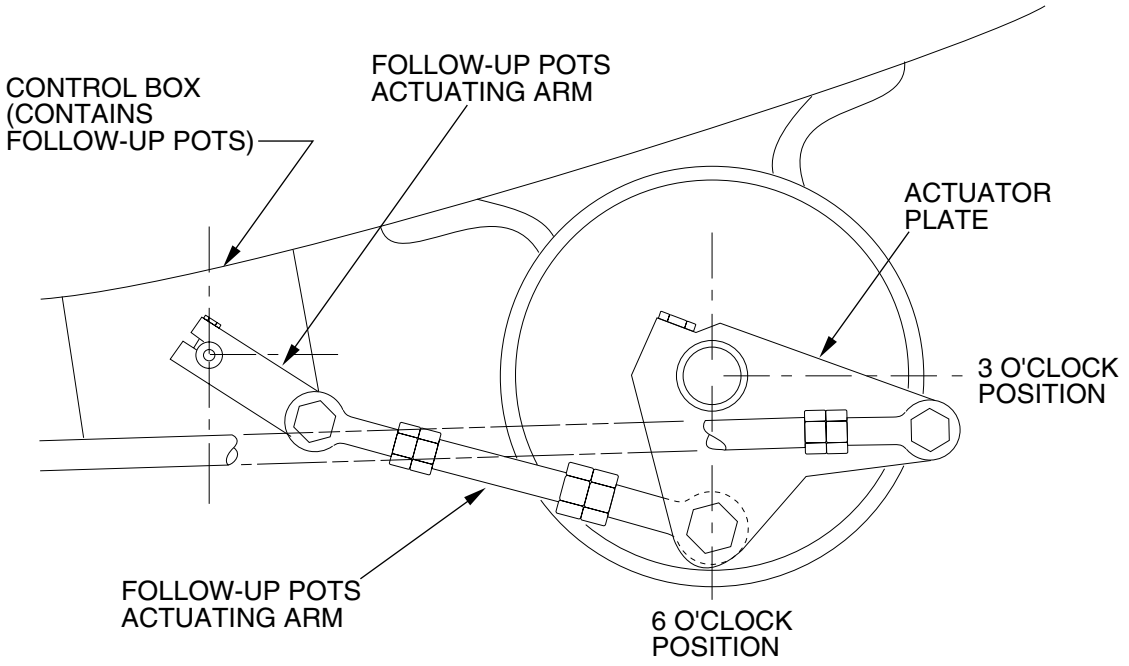
(2) Disconnect the mixing valve actuating rods.

(3) Position each mixing valve to the full closed condition.

NOTE: The full closed condition may be determined by visual inspection. It can be more accurately determined by disconnecting a duct upstream of the mixing valve and applying air pressure to the duct.

(4) Adjust the valve to stop the flow of air.

(5) If necessary, reindex the serrated arm to position the arm straight up and down with the valve closed (see Figure 201).



Mixing Valve Actuator Figure 201

(6) Adjust each mixing valve actuating rod so that the center swing link is adjusted parallel to the center line of the aircraft (see Figure 201).

NOTE: Do not tighten actuating rod attaching hardware or jam nuts at this point.

(7) With the actuator in the full cold position, index the actuator plate as shown in Figure 201.

(8) The actuating rod attach point for the follow-up pots is approximately at the six o'clock position.

(9) The actuating rod attach point for the center swing link is between the three and four o'clock positions.

(10) Disconnect the actuator arm on the follow-up potentiometers.

(11) Adjust the potentiometers to a resistance of 100 ohms.

(12) Attach and index the actuating arm between the four and five o'clock position (see Figure 201).

(13) Adjust and attach the follow-up pot actuating rod as shown in Figure 201.

(14) Adjust and attach the swing link actuating rod to the actuator plate.

(15) Verify swing link is parallel to the center line of the aircraft (Figure 1).

(16) The actuating arms on the mixing valves are straight down and the mixing valves are full closed.

(17) The actuator plate is in the six o'clock position (Figure 201).

(18) Actuating arm for the follow-up pots is between four and five o'clock (Figure 201).

(19) When all adjustments are correct, tighten and safety all attaching hardware and tighten all actuating rod jam nuts.

HEATING – DESCRIPTION AND OPERATION

1. General

A. The heating system consists of two hot air mixing valves controlled by the temperature control system. Each valve controls the introduction of hot air into the cold air duct.

1. Troubleshooting

A. The following troubleshooting procedures affect only those aircraft that have complied with SB 21-013.

HEATING – FAULT ISOLATION

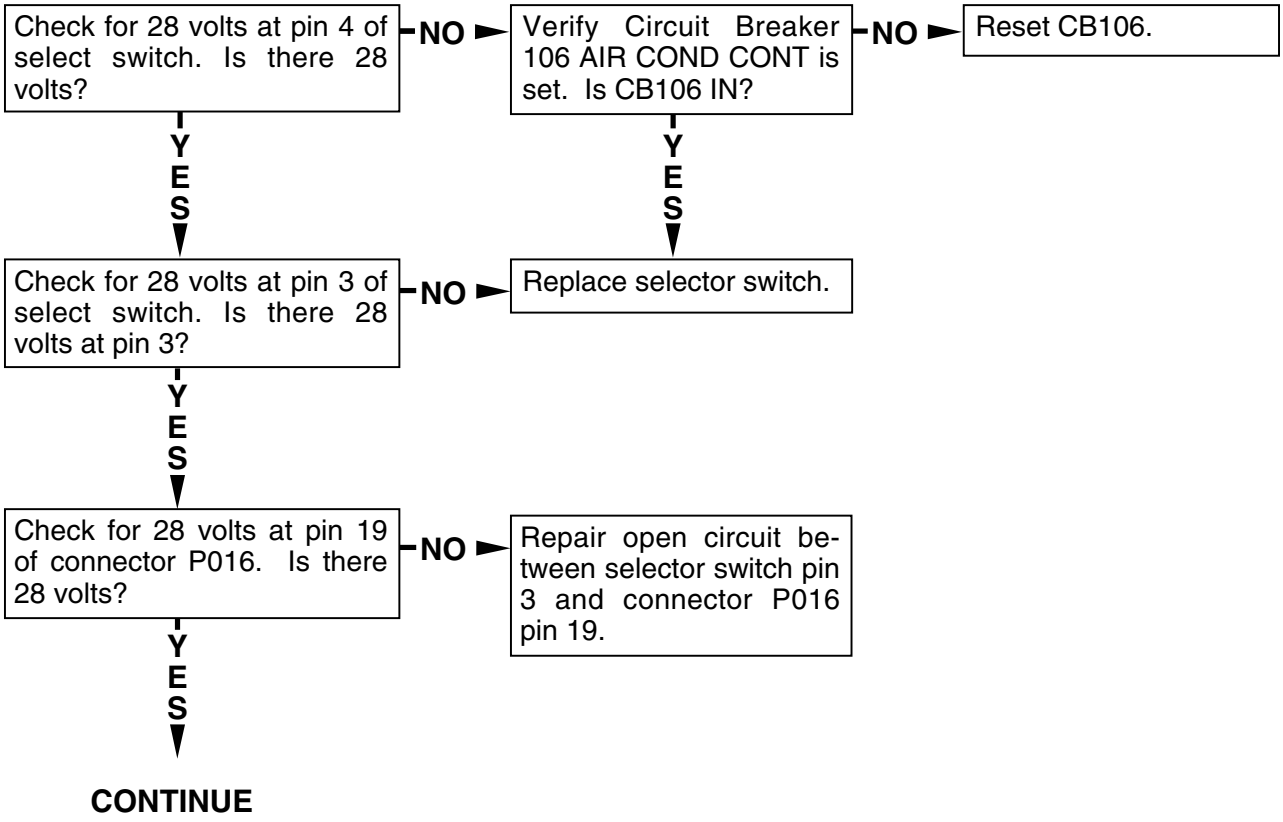
PROBLEM:

NO HOT AIR IN MANUAL POSITION (LEFT OR RIGHT ENGINE).

PREREQUISITES:

Electrical Power Applied, CAWI ARM Switch OFF, BLEED AIR VALVES Switch ON, applicable engine operating, TEMP MODE SELECT SWITCH in HOT.

START:



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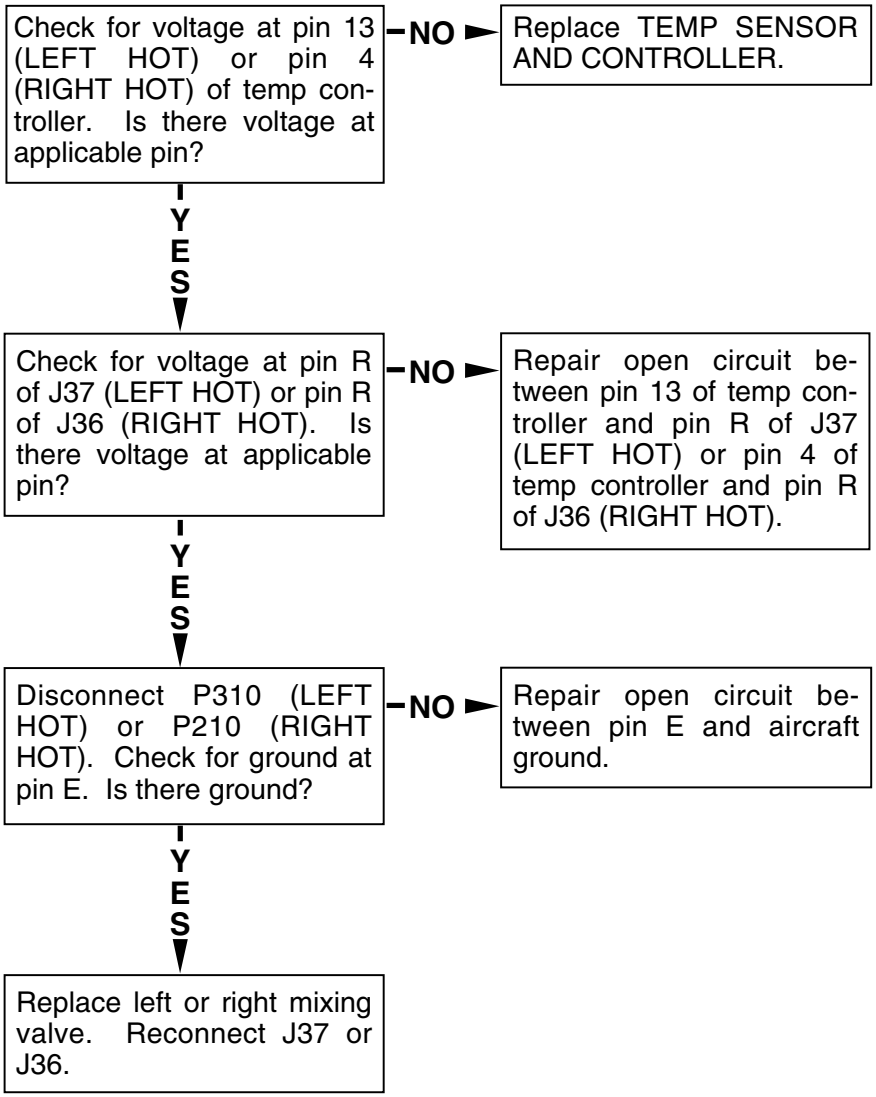

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HEATING – FAULT ISOLATION

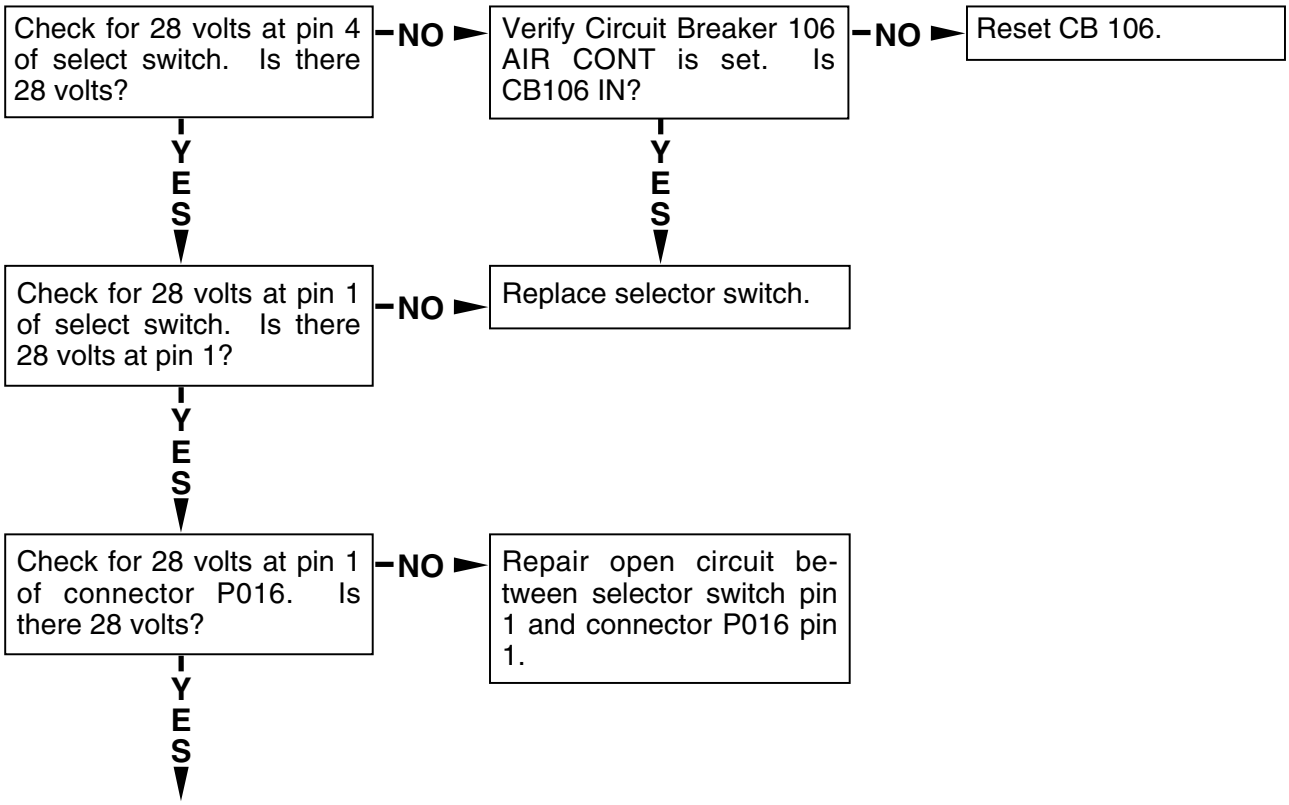
PROBLEM:

NO HOT AIR IN AUTO POSITION (LEFT OR RIGHT ENGINE).

PREREQUISITES:

Electrical Power Applied, CAWI ARM Switch OFF, BLEED AIR VALVES Switch ON, applicable engine operating, TEMP MODE SELECT SWITCH in AUTO.

START:



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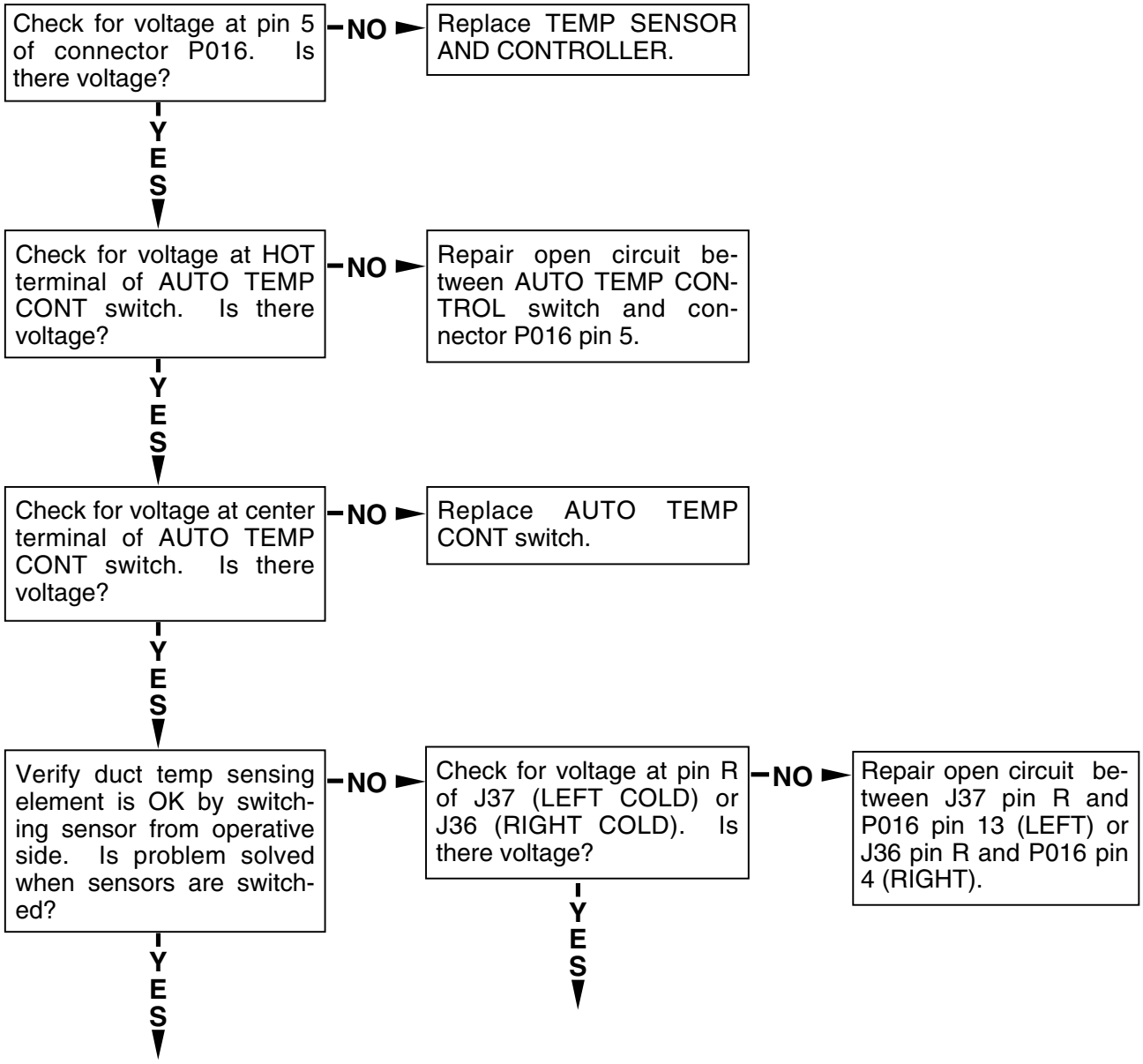

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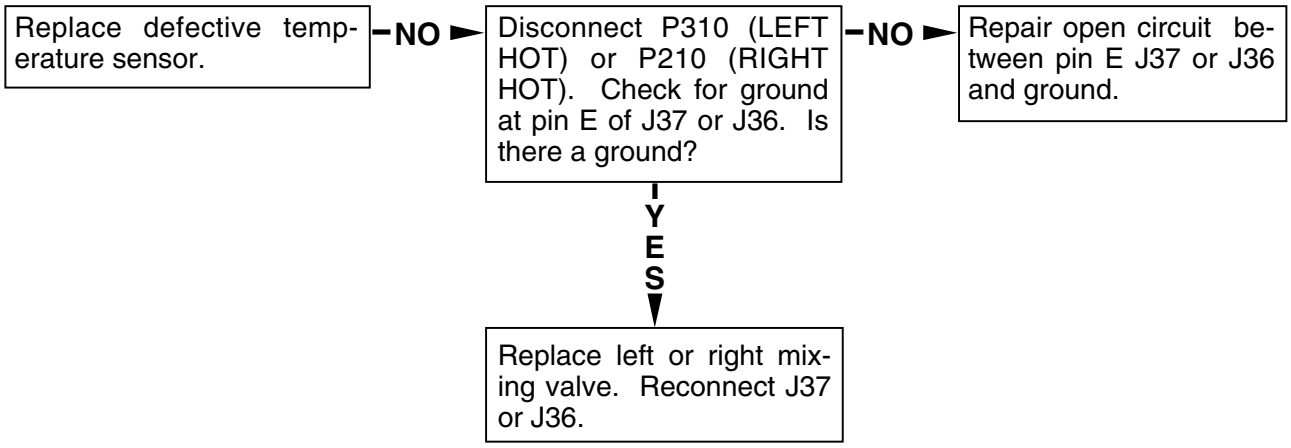

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1. Removal/Installation – Heating System

A. Removal – Hot Air Mixing Valve

(1) Gain access to valve by removing center section ECS access panels.

(2) Remove clamp securing supply line hose and remove hose.

CAUTION: REMOVE HOSE CAREFULLY OR DAMAGE TO FITTING MOUNTING BOSS MAY OCCUR. IF NECESSARY, SPLIT HOSE WITH RAZOR BLADE TO REMOVE.

(3) Loosen clamps securing valve to inlet and outlet tubing. Remove clamps and carefully pry out valve.

B. Installation – Hot Air Mixing Valve

(1) Inspect metal gaskets. Replace if necessary.

(2) Position valve in place. Ensure that gaskets are properly positioned. Install clamps and torque 35-40 inch-pounds. Install supply line and secure with clamp.

(3) Apply 45-60 psi air pressure to the upstream side of either ECS pressure regulator.

(4) Move the mode selector to MANUAL.

(5) Vary the temperature control from COLD to HOT.

(6) Verify mixing valve moves from closed to open with a smooth steady motion.

(7) Reconnect pressure regulator and install access panels.

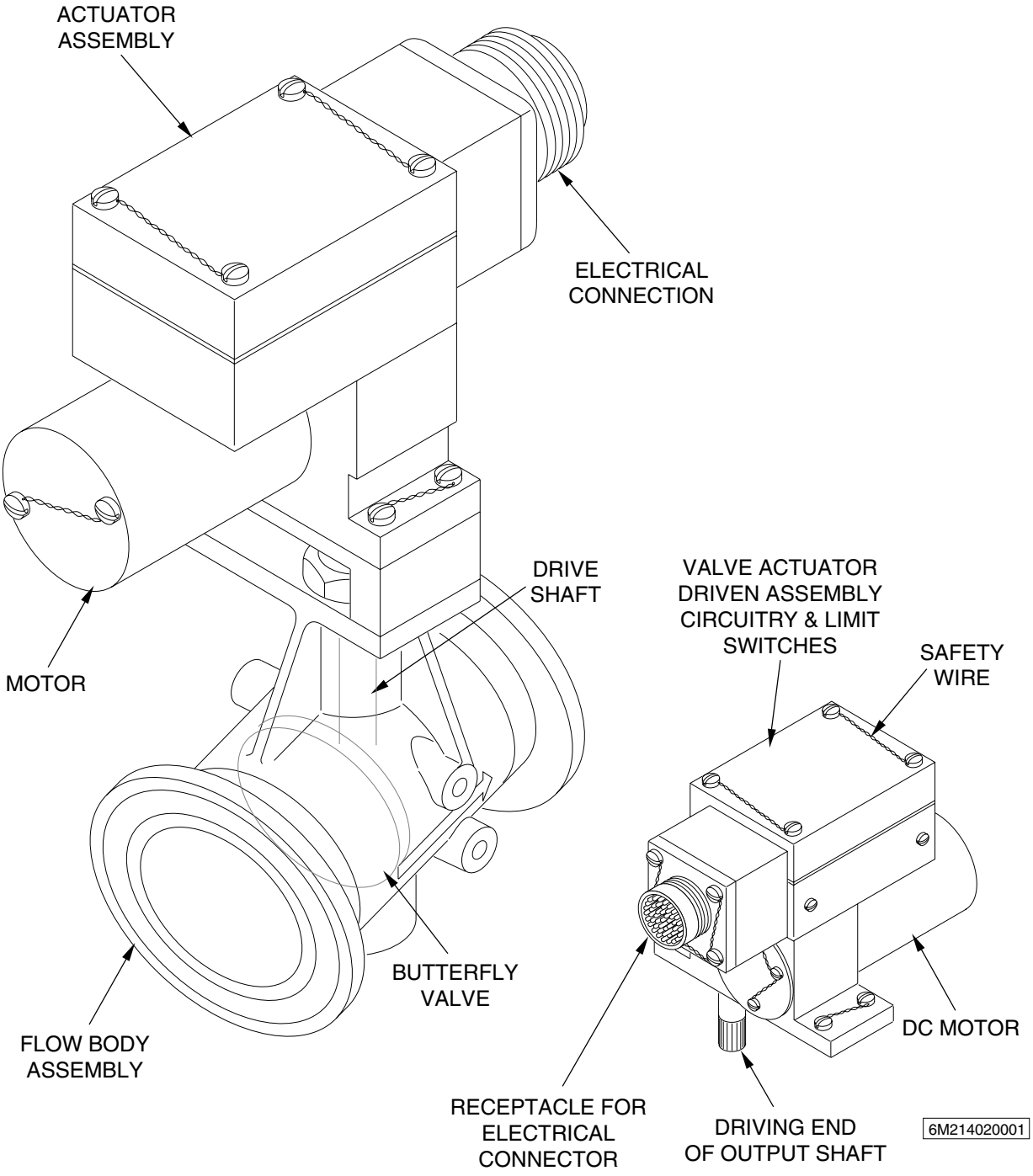
2. Inspection/Check – Heating System

A. The operation of the valve may be checked by supplying approximately 45-60 psi air pressure to the upstream side of either ECS pressure regulator. Move the mode selector to MANUAL and vary the temperature control from COLD to HOT. The mixing valves should move near full travel and with a smooth steady motion. Watch each valve for evidence of sticking or jerky movement.

HEATING – DESCRIPTION AND OPERATION

1. General

A. The heating system consists of two hot air mixing valves controlled by the temperature control system. Each valve controls the introduction of hot air into the cold air duct.



Hot Air Mixing Valve
Figure 1

HEATING – FAULT ISOLATION

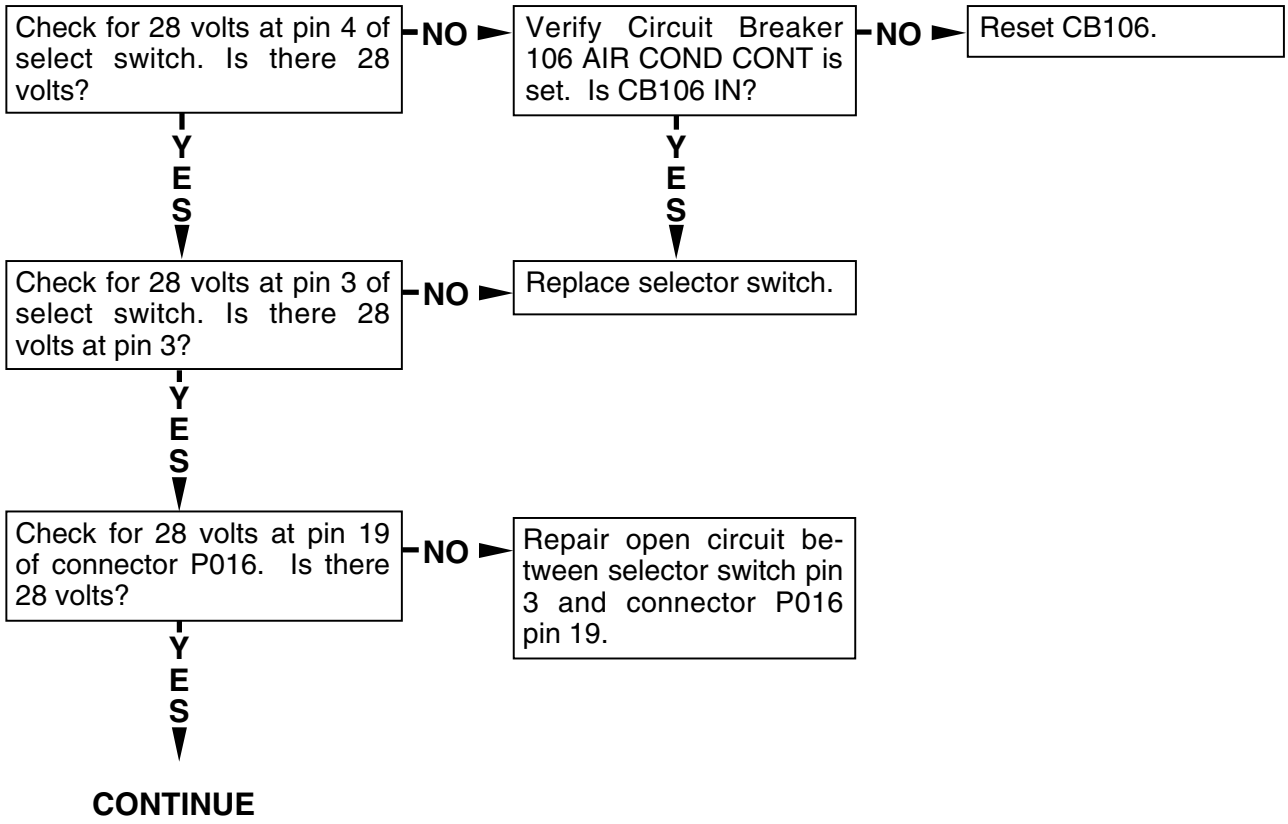
PROBLEM:

NO HOT AIR IN MANUAL POSITION (LEFT OR RIGHT ENGINE).

PREREQUISITES:

Electrical Power Applied, CAWI ARM Switch OFF, BLEED AIR VALVES Switch ON, applicable engine operating, TEMP MODE SELECT SWITCH in HOT.

START:



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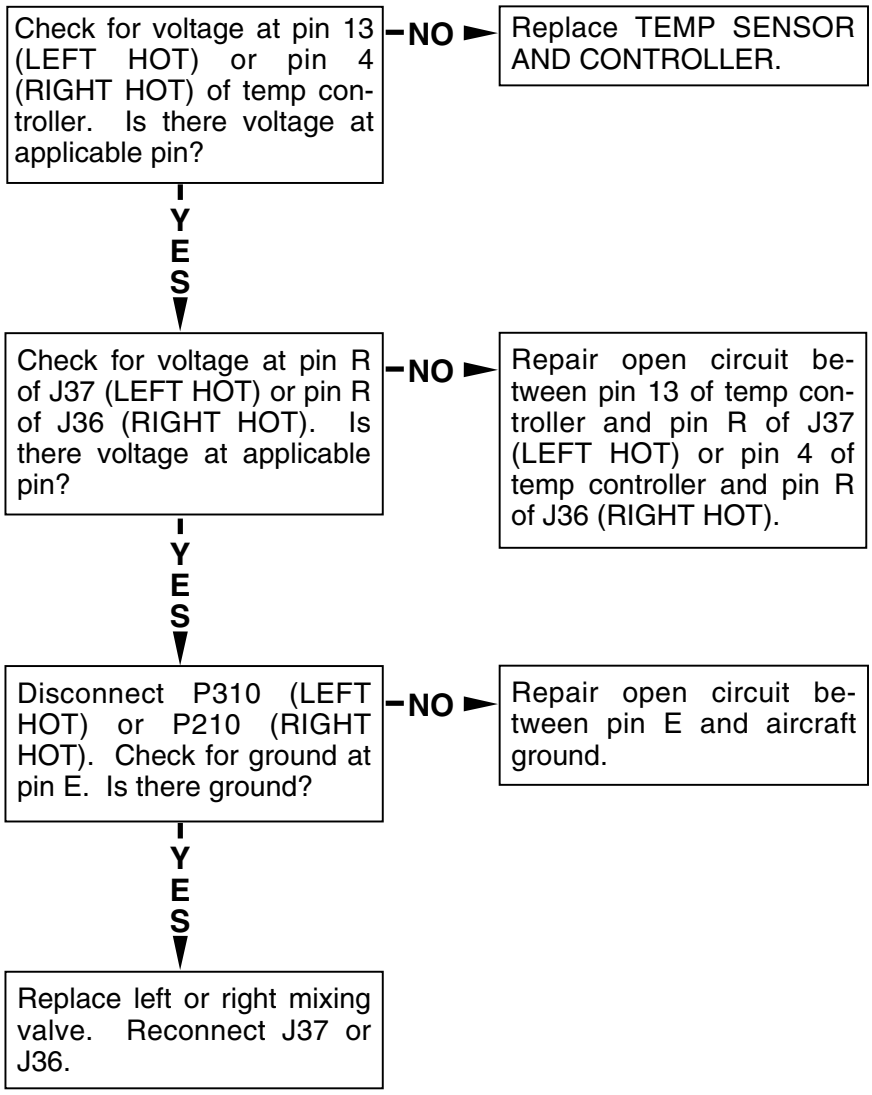

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HEATING – FAULT ISOLATION

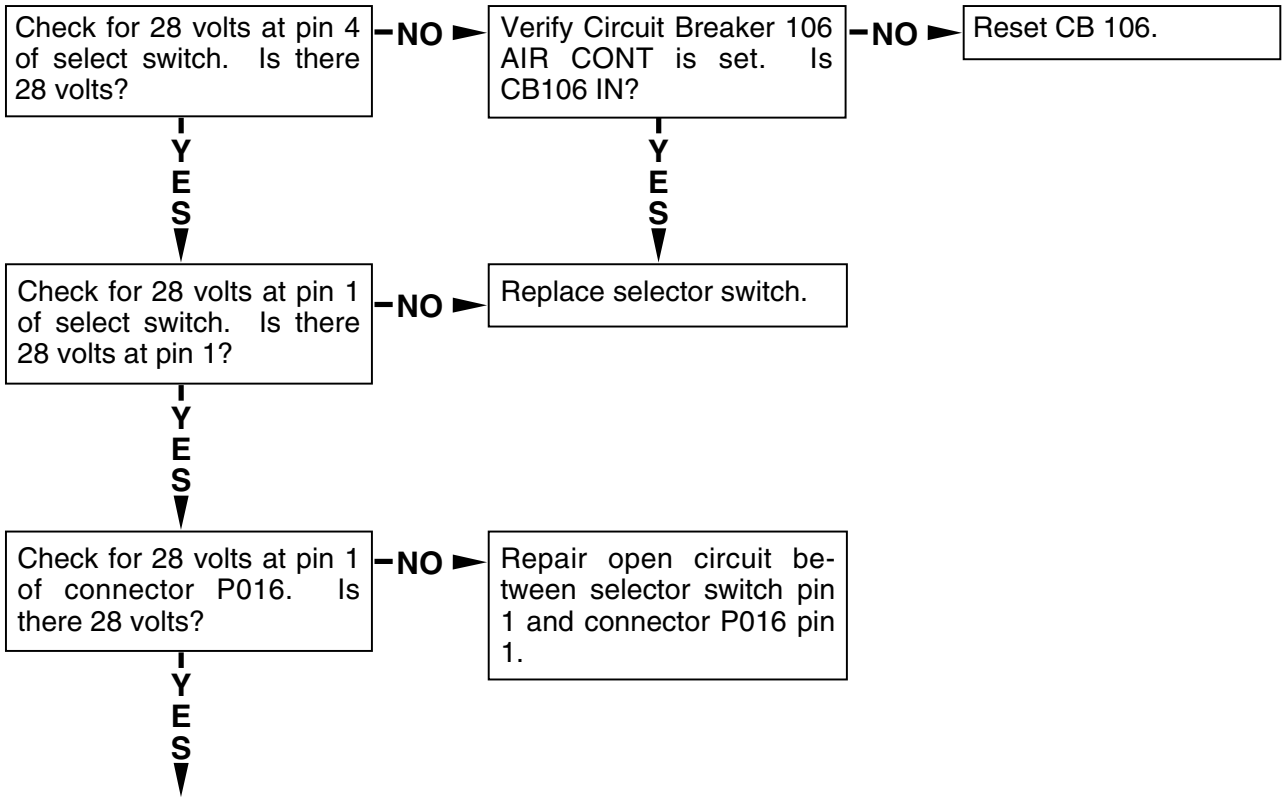
PROBLEM:

NO HOT AIR IN AUTO POSITION (LEFT OR RIGHT ENGINE).

PREREQUISITES:

Electrical Power Applied, CAWI ARM Switch OFF, BLEED AIR VALVES Switch ON, applicable engine operating, TEMP MODE SELECT SWITCH in AUTO.

START:



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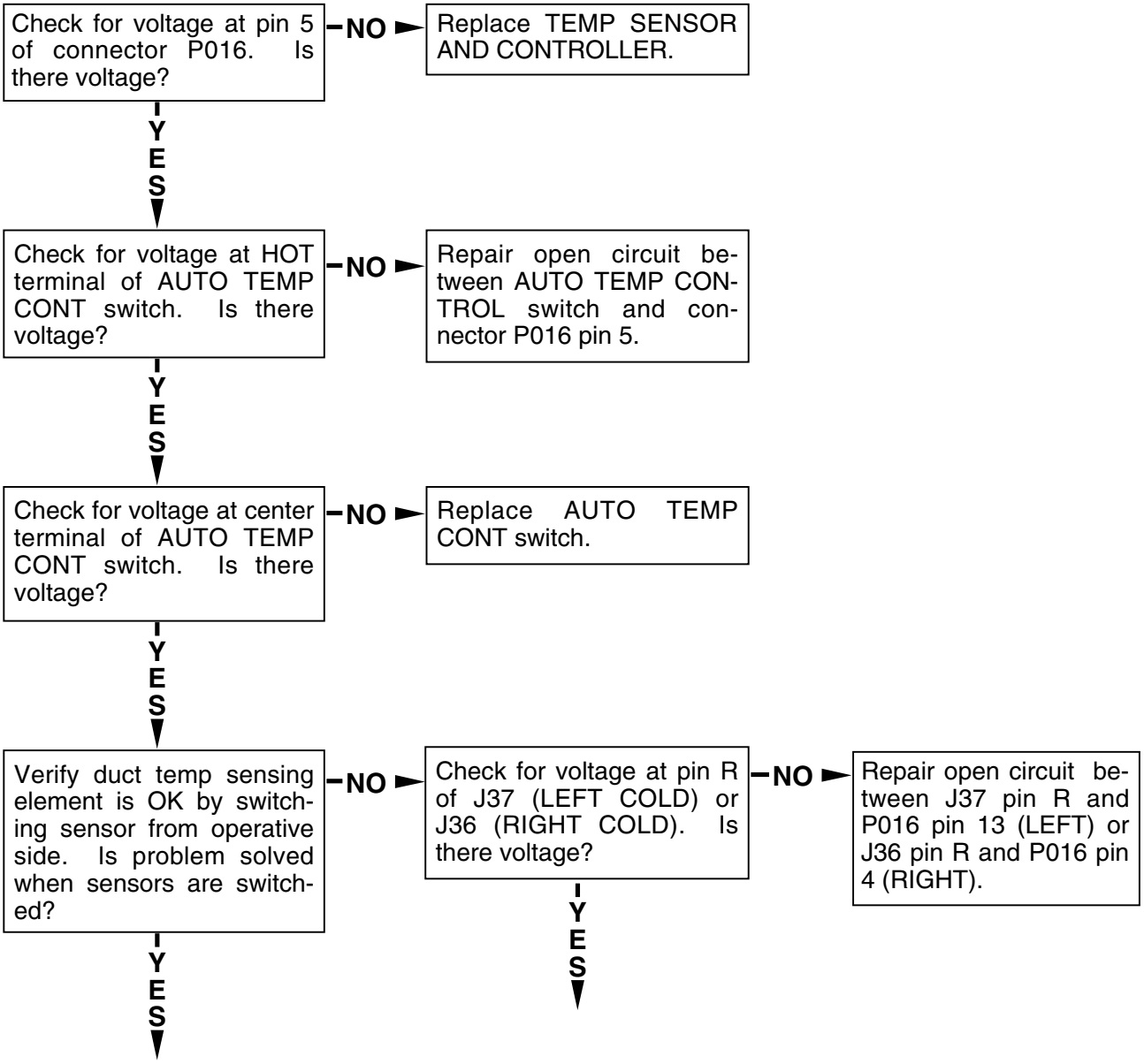

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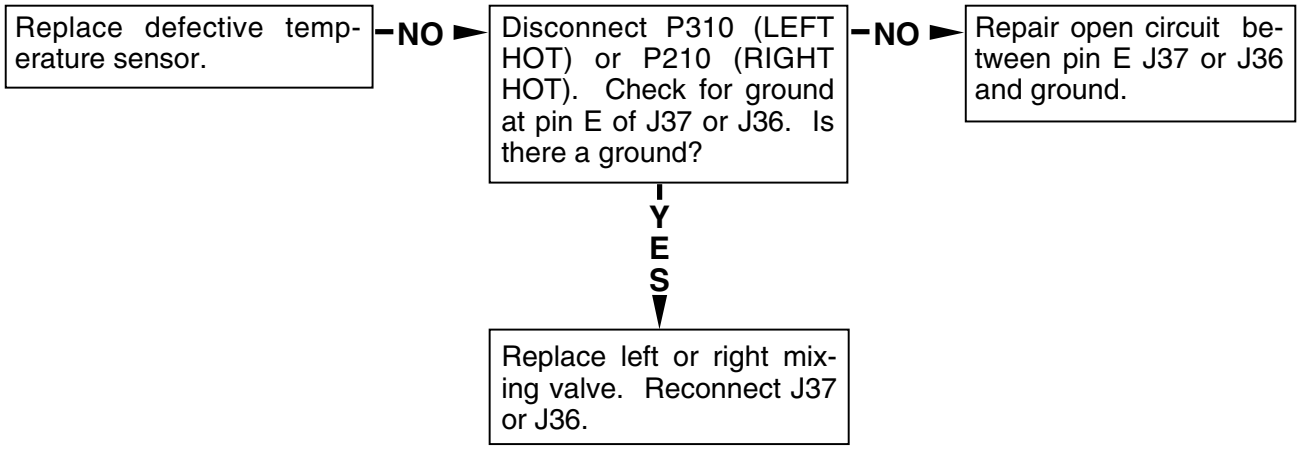

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1. Removal/Installation – Heating

A. Removal – Hot Air Mixing Valve

(1) Gain access to valve by removing center section ECS access panels.

(2) Disconnect electrical connector from valve.

(3) Loosen clamps securing valve to inlet and outlet tubing. Remove clamps and carefully remove valve.

B. Installation – Hot Air Mixing Valve

(1) Inspect metal gaskets for cracks, corrosion, distortion, and general condition. Replace if necessary.

(2) Position valve in place. Ensure that gaskets are properly positioned.

(3) Install clamps and torque 35-40 inch-pounds (3.95 to 4.52 N°m).

(4) Install electrical connector and safety.

(5) Move the mode selector to AUTO.

(6) Vary the auto temperature control from COLD to HOT.

(7) Verify mixing valve moves from closed to open with a smooth steady motion.

(8) Run engine and check for operation of valve and bleed air leaks.

(9) Install access panels.

2. Adjustment/Test – Hot Air Mixing Valve

A. Operational Test – Hot Air Mixing Valve

(1) Apply 28 VDC to pin A of actuator electrical connector.

(2) Move temperature control mode switch to HOT.

(3) Verify butterfly valve moves to open position in a smooth steady motion.

(4) Apply 28 VDC to pin B of actuator electrical connector.

(5) Move temperature control mode switch to COLD.

(6) Verify butterfly valve moves to closed position in a smooth steady motion.

AN ELBIT SYSTEMS OF AMERICA, LLC, COMPANY

MAINTENANCE MANUAL

COOLING – DESCRIPTION AND OPERATION

1. General

A. A cooling turbine is installed near the nacelle in each wing center section. Bleed air from the flow control valve drives the turbines, cools and pressurizes the cabin. Downstream from the turbine, other systems remove moisture from the cooled air and ensure that the ducts and water separator do not accumulate ice.

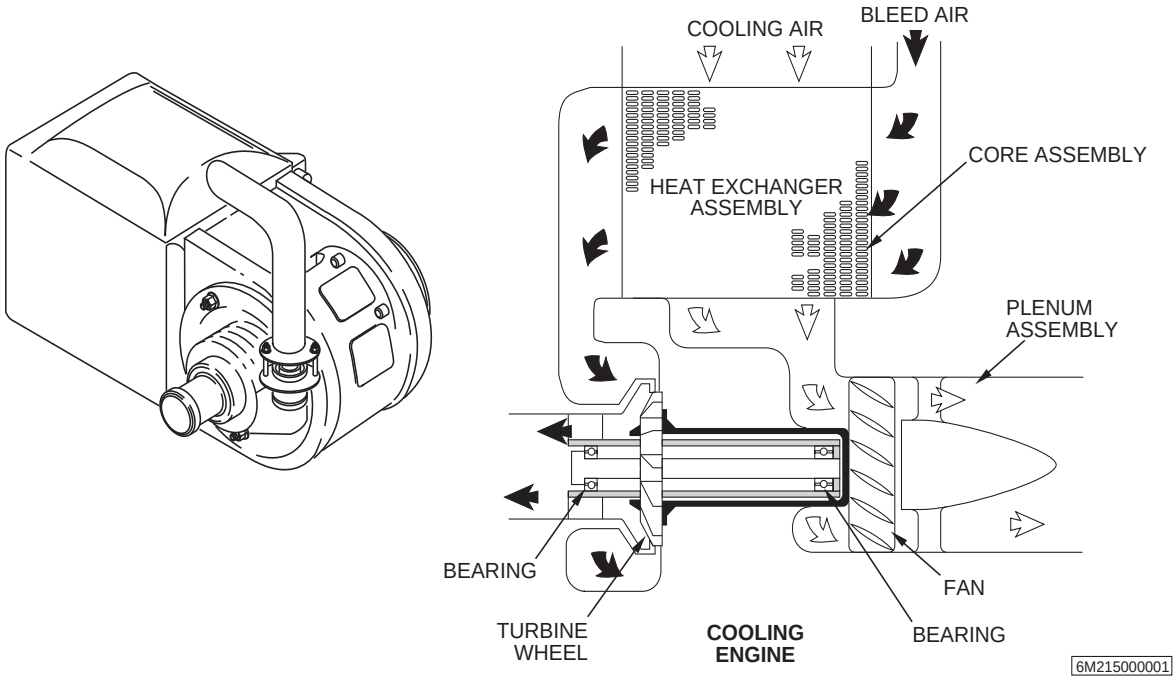
B. Air enters the turbine and passes through an air-to-air heat exchanger (see Figure 1). The cooling turbine inlet scoop and outlet duct are located in the wing near the turbine. During flight, ram air enters the scoop and cools the bleed air. The inlet transition duct inside the wing features a spring loaded door to supply air to the exchanger in the event of the inlet scoop becoming blocked.

C. Bleed air passes through the heat exchanger and strikes the turbine wheel, causing the wheel to rotate. The turbine wheel is attached to a common shaft with the heat exchanger fan. As the bleed air causes rotation of the turbine wheel, energy is expended and the air is allowed to rapidly decrease in pressure, with a resultant increase in velocity. The effect of these changes is a considerable lowering of the air temperature.

D. The rotation of the heat exchanger fan draws air through the inlet scoop, over the heat exchanger coils, and overboard through the outlet. The air passing over the heat exchanger coils further cools the bleed air.

E. A sump and wick assembly are used to provide lubrication to the turbine wheel and fan wheel bearings. The oil system must be replenished periodically (reference MAINTENANCE PRACTICES).

F. Two different cooling turbine assemblies are used. Refer to the effectivity columns in the Table of Contents for identification of your cooling system.



Cooling Turbine Assembly
Figure 1

COOLING – MAINTENANCE PRACTICES

1. Servicing – Cooling

A. Every 300 hours plus or minus 30 hours (Figure 201):

(1) Remove turbine unit from aircraft.

(2) Clean ram air inlet side of heat exchanger of all debris.

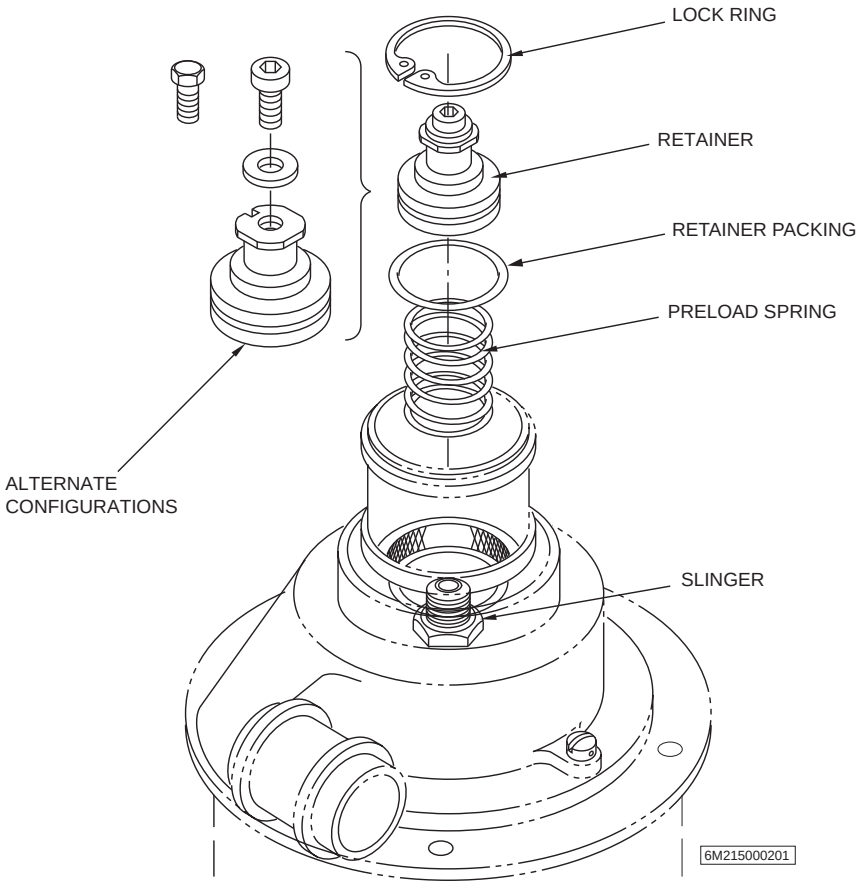
(3) Position turbine with discharge port up and remove snap ring.

(4) Using pliers with tape covered jaws, carefully rock retainer out of turbine, and remove preload spring.

CAUTION: A NEW PACKING (P/N S8057-115) MUST BE USED EACH TIME RETAINER IS REMOVED. UPON REMOVAL, CAREFULLY REMOVE ANY PACKING CHIPS FROM BEARING AND GROOVES BEFORE PROCEEDING. COMPLETE CLEANLINESS OF OIL, BENCH, AND TOOLS IS MANDATORY. TURBINE MUST BE KEPT COVERED EXCEPT WHILE BEING REWORKED.

(5) Add enough oil to cover slinger.

NOTE: Allow soak period of one hour for wicks and washers to absorb oil. Keep cavity full during soak period.



Cooling Turbine Lubrication
Figure 201

(6) Recommended oils:

(a) L-245X (F. E. Anderson Oil Company, New York, NY).

(b) Andersol-L-L-401-D (Lehigh Chemical Products Co., Chestertown, MD).

(c) Oils conforming to MIL-L-23699, Type II are acceptable (see under B. below).

(7) Turn turbine over and allow any surplus oil to drain.

(8) Replace retainer packing (P/N S8057-115).

(9) Install preload spring, retainer, and snap ring. Make sure retainer is installed with index notch up. Torque retainer screw 25 to 30 inch-pounds (2.82 to 3.39 N·m).

(10) Install turbine in aircraft.

B. Oil Conforming to MIL-L-23699, Type II:

Aeroshell Turbine Oil 500

Brayco 899S

British Petroleum Enerjet 52 or 523

Caltex RPM Jet Engine Oil 5

Castrol 205

Chevron Jet Oil No. 5

Exxon Turbo Oil 25

Exxon Turbo Oil 2380

Exxon Turbo Oil 5251

Hatcol 3211

Mobil Jet II

Mobil Jet Oil XRM-193A3

Monsanto Skylube 450

Royco 899

Signal Astro Turbine 25

Sinclair Turbo S Type II

Stauffer Jet II

Texaco or Caltex 5180, 7388, 7730 and Starjet 5

Texaco Syn Lube Oil 4508

Turbonycoil II

COOLING – DESCRIPTION AND OPERATION

1. General

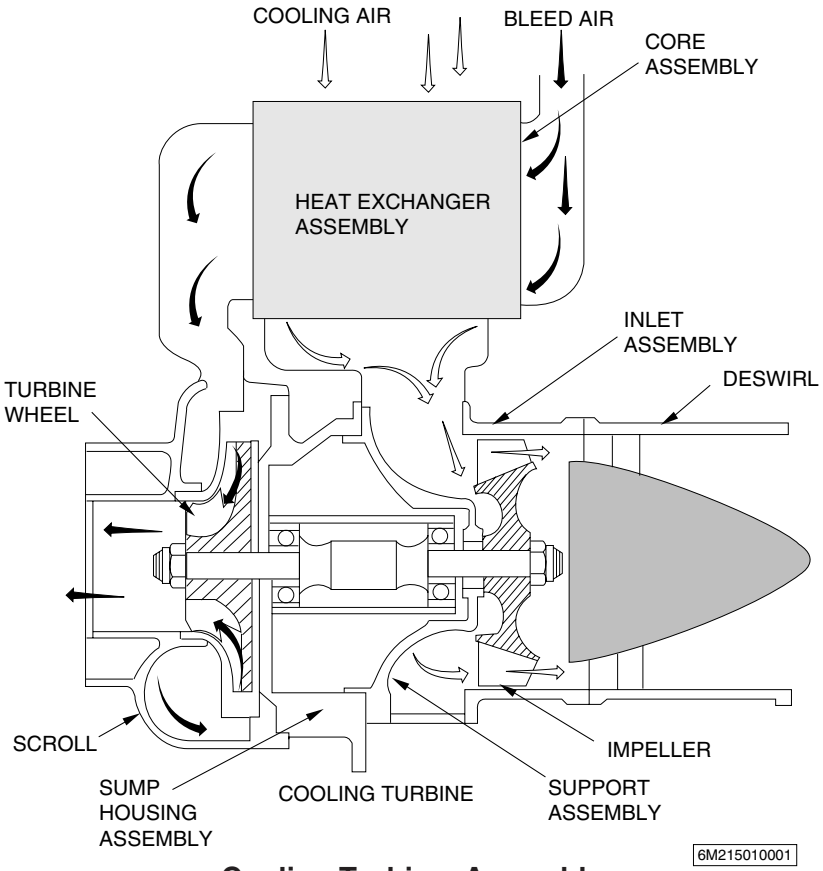
A. A cooling turbine is installed near the nacelle in each wing center section. Bleed air from by the flow control valve drives the turbine, cools and pressurizes the cabin. Downstream from the turbine, other systems remove moisture from the cooled air and ensure that the ducts and water separator do not accumulate ice.

B. Air enters the turbine and passes through an air-to-air heat exchanger (see Figure 1). The cooling turbine inlet scoop and outlet duct are located in the wing near the turbine. During flight, ram air enters the scoop and cools the bleed air. The inlet transition duct inside the wing features a spring loaded door to supply air to the exchanger in the event of the inlet scoop becoming blocked.

C. Bleed air passes through the heat exchanger and strikes the turbine wheel, causing the wheel to rotate. The turbine wheel is attached to a common shaft with the impeller. As the bleed air causes rotation of the turbine wheel, energy is expanded and the air is allowed to rapidly decrease in pressure, with a resultant increase in velocity. The effect of these changes is a considerable lowering of the air temperature.

D. The rotation of the impeller draws air through the inlet scoop, over the heat exchanger coils, and overboard through the outlet. The air passing over the heat exchanger coils further cools the bleed air.

E. A sump housing assembly contains oil to lubricate the turbine wheel and impeller bearings. The oil system must be replenished periodically (reference Maintenance Practices).



Cooling Turbine Assembly
Figure 1

COOLING – MAINTENANCE PRACTICES

1. Servicing – Cooling

A. Inspect the turbine for security of installation. Check the hose and lines attached to the turbine for condition and security. P/N's 204755-4-1, 204755-4-3, 204755-4-4, and 204755-4-5 must be serviced.

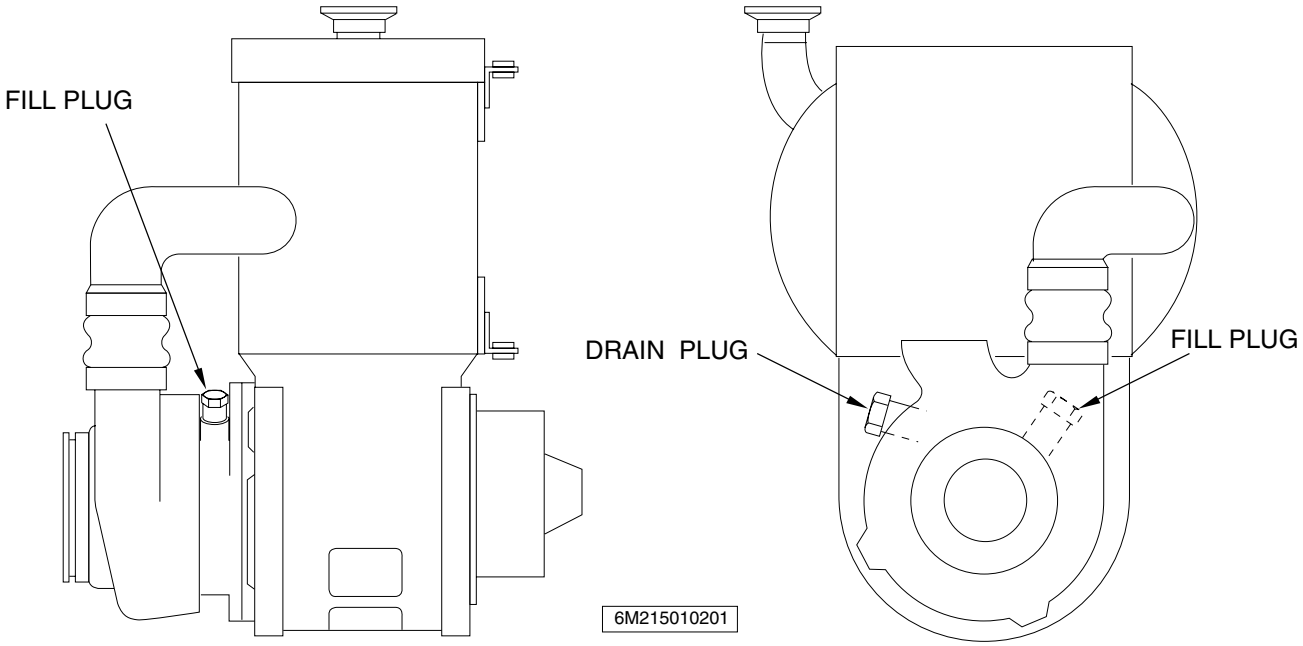
B. Service units modified to P/N 204755-4-6 IAW Service Bulletin 5-2234 as follows:

(1) Remove the turbine oil fill plug (see Figure 201).

(2) Add 40 cc of oil conforming to MIL-L-23699 to turbine oil fill port.

(3) Wait 20 to 30 minutes and remove the turbine oil drain plug and allow oil to drain from oil port until oil stops dripping.

(4) Install the turbine oil fill and drain plugs. Torque drain plugs 35 to 40 inch-pounds (3.95 to 4.52 N·m) and safety.



Cooling Turbine Lubrication
Figure 201

C. Oil Conforming to MIL-L-23699:

Aeroshell Turbine Oil 500

Brayco 899S

British Petroleum Enerjet 52 or 523

Caltex RPM Jet Engine Oil 5

Castrol 205

Chevron Jet Oil No. 5

Exxon Turbo Oil 25

Exxon Turbo Oil 2380

Exxon Turbo Oil 5251

Hatcol 3211

Mobil Jet II

Mobil Jet Oil XRM-193A3

Monsanto Skylube 450

Royco 899

Signal Astro Turbine 25

Sinclair Turbo S Type II

Stauffer Jet II Texaco or Caltex 5180, 7388, 7730 and Starjet 5

Texaco Syn Lube Oil 4508

Turbonycoil II

2. Removal/Installation – Cooling

A. Removal – Cooling Turbine (Figure 202)

(1) Remove two (2) outboard lower wing panels (item 1).

(2) Remove 3 screws attaching forward end of heat exchanger to transition duct, (one (1) on top and two (2) on bottom, (item 2).

(3) Remove one (1) clamp attaching forward end of transition duct to sceet hose (item 3).

(4) Remove transition duct (item 4).

(5) Remove clamps at ends of wrapped tube above intake duct sceet hose and remove tube (item 5).

(6) Remove screw attaching slave valve tube to turbine (item 6).

(7) Remove clamp attaching tube to outboard end of slave valve and remove tube (item 7).

(8) Remove one (1) bolt and washer attaching heat exchanger to forward bracket (item 8).

(9) Remove two (2) bolts and washers attaching heat exchanger to aft bracket (item 9).

(10) Remove clamp attaching exhaust duct sceet hose to turbine (item 10).

(11) Cut safety wire and loosen three (3) screws attaching turbine to water separator duct (item 11).

(12) Slide water separator duct away from turbine.

(13) Twist turbine 45 degrees inboard and down to remove.

B. Installation – Cooling Turbine (Figure 202)

(1) Position turbine. Secure water separator duct to turbine with three (3) screws and safety wire (item 11).

(2) Connect turbine to exhaust duct sceet hose and tighten clamp (item 10).

(3) Install one (1) bolt and washer to forward bracket and exchanger (item 8).

(4) Install two (2) bolts and washers to aft bracket and heat exchanger (item 9).

(5) Install tube above intake duct sceet hose using clamps on ends of tube. Torque clamps 25 to 45 inch-pounds (item 5).

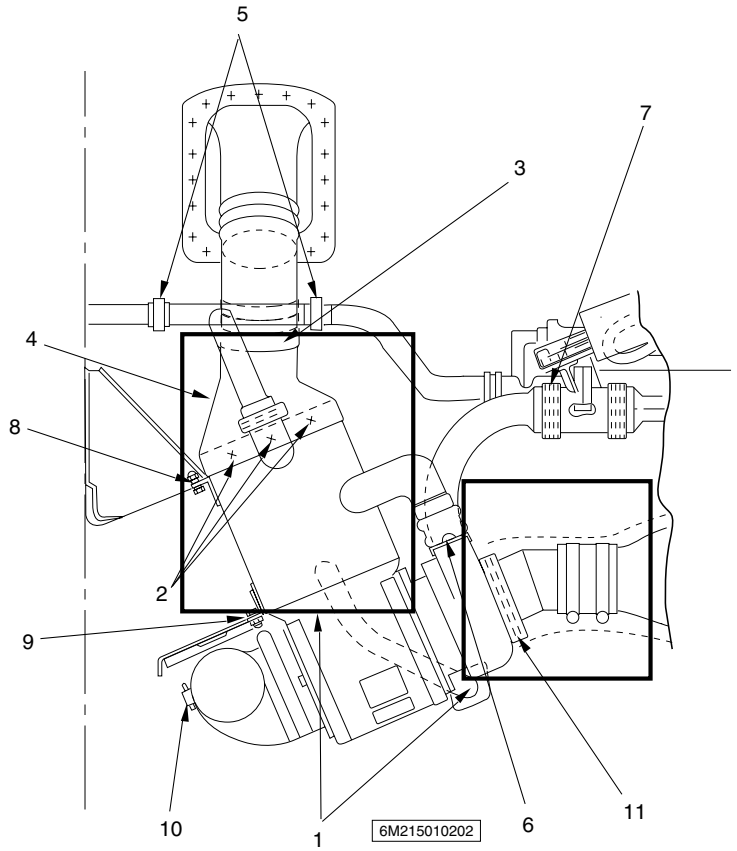
(6) Install transition duct to forward end of heat exchanger with three (3) screws, (one [1] on top and two [2] on bottom, item 2).

(7) Connect forward end of transition duct to intake duct sceet hose. Tighten clamp (item 3).

(8) Connect slave valve tube to turbine with screw (item 6).

(9) Connect tube with clamp to outboard end of slave valve. Torque clamp 45 to 65 inch-pounds (item 7).

(10) Install access panels (item 1).



LH Cooling Turbine Removal/Installation Figure 202

TEMPERATURE CONTROL – DESCRIPTION AND OPERATION

1. General

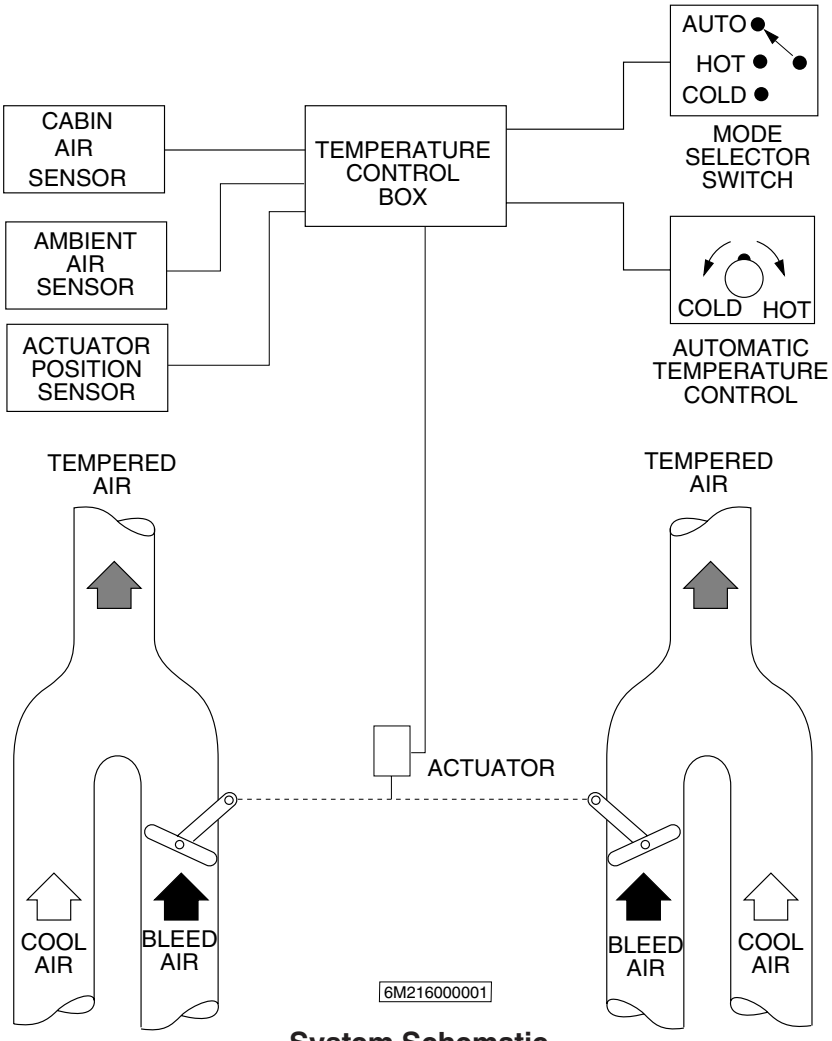
A. The electrically powered temperature control system maintain the cabin temperature at pilot-selected levels. The system may be operated in automatic or manual mode.

(1) In the automatic mode, the pilot selects a temperature level he wishes to maintain within the aircraft. The temperature control system will monitor the cabin temperature and the outside air temperature, and adjust the amount of conditioned air introduced into the cabin.

(2) In the manual mode, the pilot controls the amount of conditioned air supplied to the cabin. The principal system components are as follows.

B. Three different control systems are used. Refer to the effectivity columns in the Table of Contents for identification of your temperature control system.

C. The cabin air temperature indicating system consists of a cockpit indicator and a temperature sensor located midway in the cabin area. Changes in cabin air temperature changes the resistance of the temperature sensor.



System Schematic
Figure 1

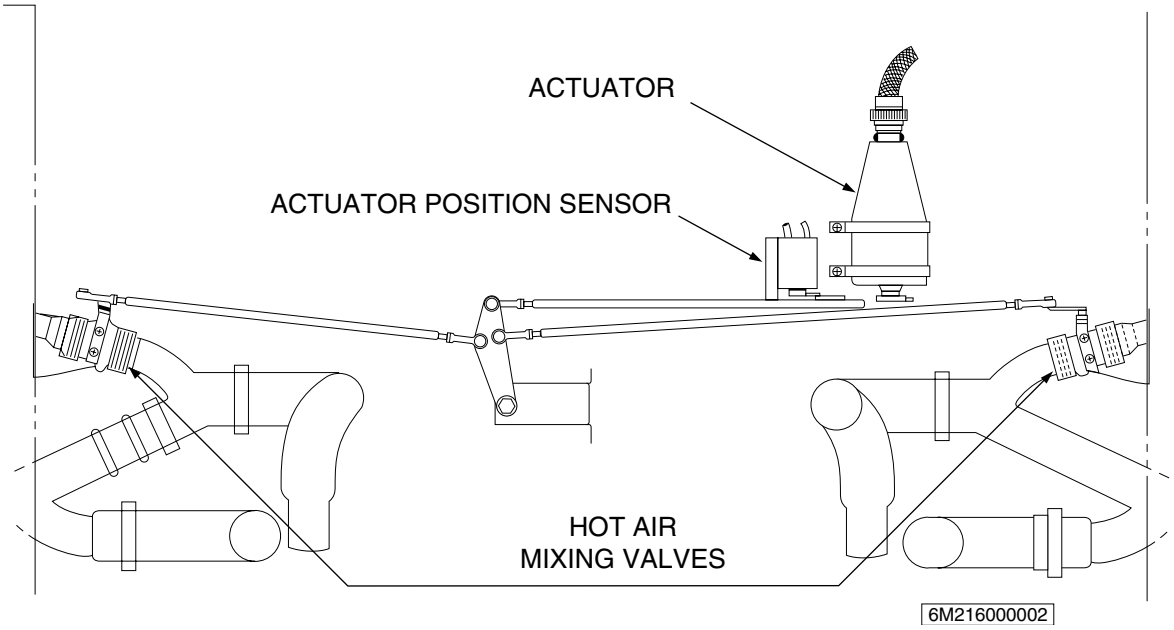
TEMPERATURE CONTROL – DESCRIPTION AND OPERATION

(1) Hot Air Mixing Valves and Actuator

(a) The hot air mixing valves (see Figure 2) are installed in each wing center section. The mixing valves are mechanically connected to a common actuator. The electrically powered actuator receives commands from the cockpit installed temperature control box.

(2) Actuator Position Sensor

(a) The actuator position sensor (see Figure 2) is mechanically linked to and driven by the hot air mixing valve actuator. Inside the position sensor are two interconnected potentiometers whose resistance is varied with movement of the mixing valve actuator. The potentiometers form part of the electrical circuitry of the temperature control system.



Mixing Valve and Actuator Figure 2

(3) Ambient Air Temperature Sensor

(a) The ambient air temperature sensor (see Figure 3) is installed in the inlet of the right hand cooling turbine. Resistance of the sensor changes with air temperature.

(4) Cabin Air Temperature Sensor

(a) The cabin air temperature sensor (see Figure 3) contains two temperature sensitive resistors and a ducting fan. The fan is used to blow cabin air across the two resistors. The two (2) resistors, are part of the temperature control system electrical circuitry.

(5) Temperature System Controls

(a) A mode selector switch and automatic temperature control are installed on the right hand switch panel. The mode selector has four positions, OFF, AUTO, HOT, and COLD.

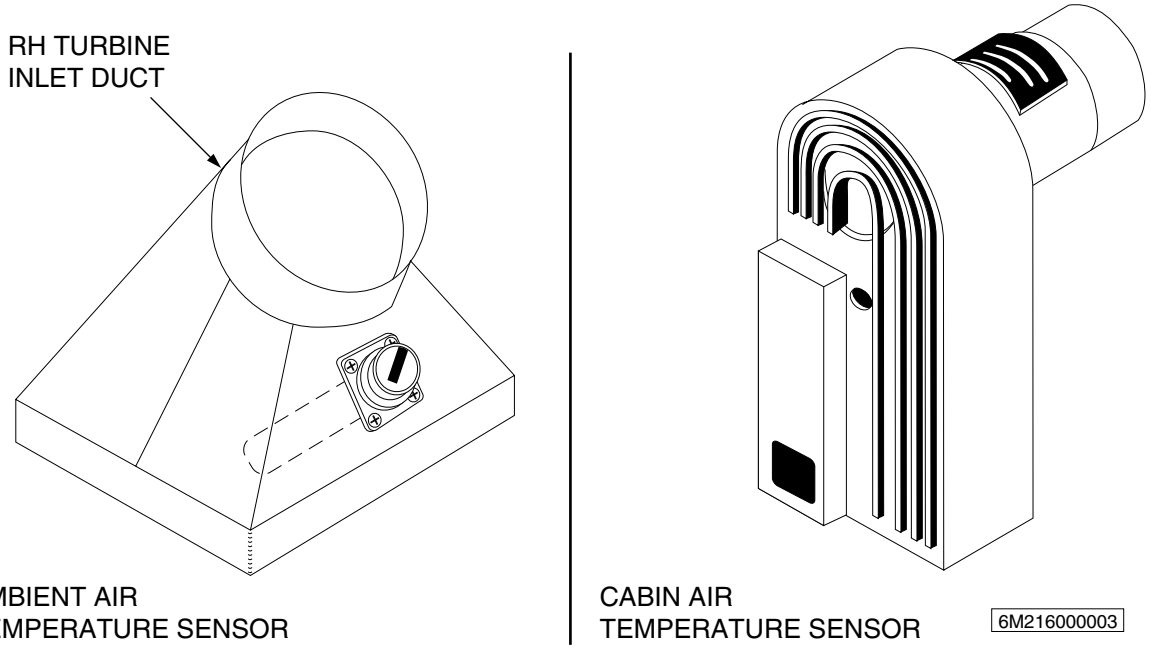
(b) In the OFF position, all power to the temperature control circuit is terminated.

(c) Placing the control in the HOT position will move the hot air mixing valves to the maximum open position.

(d) Moving the control to the COLD position will move the mixing valves to the full closed position.

(e) In the AUTO position, power is applied to the temperature control box and associated circuitry.

(f) The automatic temperature control, located adjacent to the mode selector, controls the cabin temperature when the system is operating in the automatic mode.



Temperature Sensors
Figure 3

1. Adjustment/Test – Temperature Control

A. Functional Test – Temperature Control

NOTE: The following procedure may be used to functional test the temperature control system:

(1) Ensure that battery switch(es) are ON.

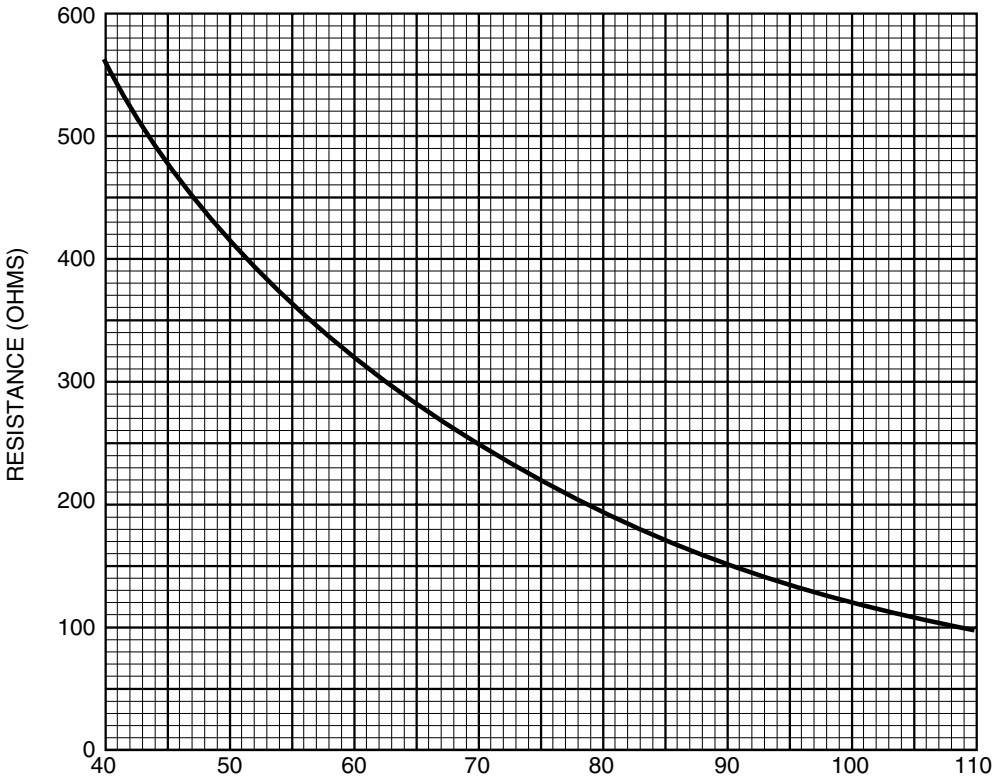
(2) Position mode selector switch to COLD.

(3) Move automatic temperature control to full COLD.

(4) Allow time for actuator to reach full travel.

(5) When actuator has stopped, turn battery switch(es) OFF and remove electrical connector from temperature control box.

CABIN AIR TEMPERATURE SENSOR



REFERENCE: RESISTANCE @ 75°F = 220 ± 6 OHMS

EXAMPLE

GIVEN: TEMPERATURE = 65°F
OBTAIN: RESISTANCE = 280 OHMS

6M216000201

Resistance vs Temperature
Figure 201

(6) Conduct the following tests:

TEST POINT	CONDITIONS	INDICATION
Pin G to ground	Mode selector to AUTO Battery switch(es) ON	28 VDC.
Pin D to ground	Mode selector to HOT Battery switch(es) ON	28 VDC.
Pin B to ground	Mode selector to COLD Battery switch(es) ON	28 VDC.
Pin M to ground	None	0 ohms to ground.
Pin N to ground	Battery switch(es) ON	28 VDC.
Pin I to Pin H	Determine temperature, see Figure 201	Resistance of cabin temp sensor (T1).
Pin F to ground	Determine temperature, see Figure 201	Resistance of cabin temp sensor (T2).
Pin J to ground	Determine temperature, see Figure 202	100 ohms plus the resistance of the ambient air temp sensor.
Pin C to ground	None	Infinity.
Pin A to ground	None	Field resis-tance of motor (10.2 plus 3% ohms).
Pin E to Pin L	Automatic temp control full COLD	400 ohms.
	Move control slowly to HOT	Resistance should slowly drop to 100 ohms.

Temperature Control Electrical Test
Table 201

(7) Replace electrical connector on temperature control box.

(8) Ensure that battery switch(es) are ON.

(9) Move mode selector switch to HOT.

(10) Check automatic temperature control in full HOT. Allow time for actuator to reach full travel.

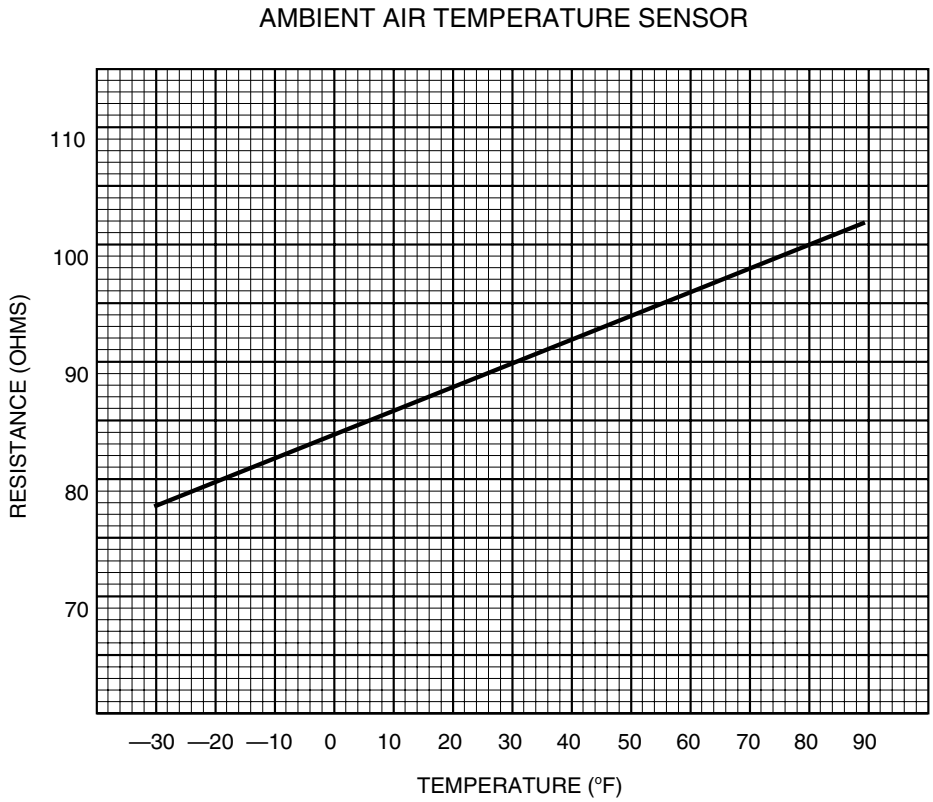
(11) When actuator has stopped, turn battery switch(es) OFF and disconnect electrical connector from temperature control box.

(12) Conduct the following test:

TEST POINT	CONDITIONS	INDICATION
Pin J to ground	Determine temperature, see Figure 202	240 ohms plus the resistance of the ambient air temp sensor.
Pin C to ground	None	Field resistance of motor (10.2 plus 3% ohms).
Pin E to Pin L	Automatic temp control full HOT	150 ohms
	Move control slowly to COLD	Resistance should slowly increase to 450.

Temperature Control Electrical Test
Table 202

(13) Replace electrical connector on temperature control box. Move mode selector switch to OFF.



Resistance vs Temperature
Figure 202

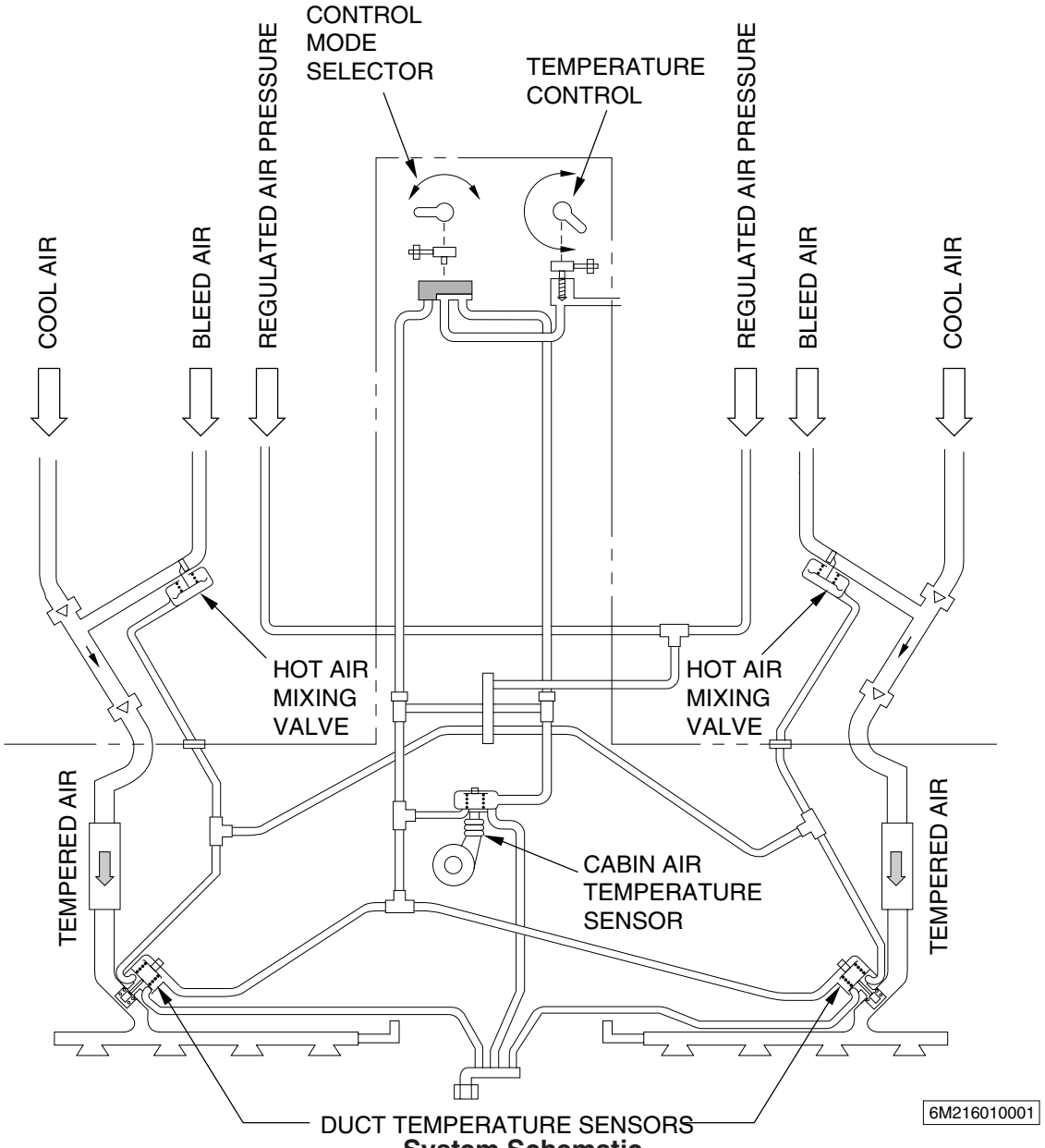
TEMPERATURE CONTROL – DESCRIPTION AND OPERATION

1. General

A. The cabin temperature control system uses regulated air pressure to control and maintain the cabin temperature at pilot-selected levels. The system may be operated in either, AUTO or MANUAL.

(1) In the AUTO mode, the pilot selects a temperature level he wishes to maintain. The temperature control system will monitor the cabin temperature and the conditioned air duct temperature, and adjust the amount of conditioned (hot) air introduced into the cabin.

(2) In the MANUAL mode, the pilot controls the amount of conditioned air supplied to the cabin.

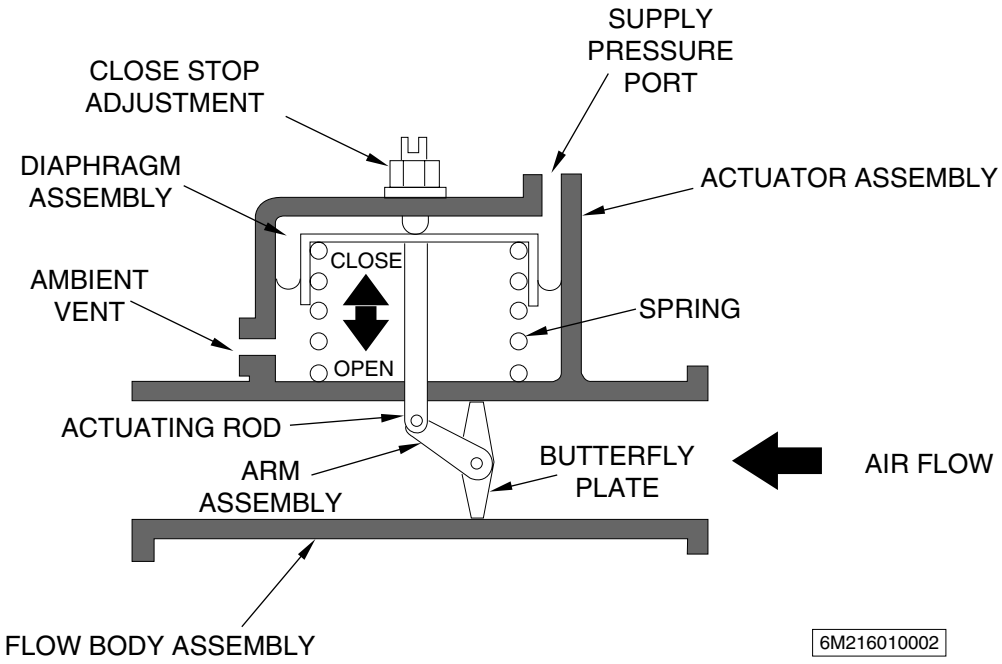


System Schematic
Figure 1

B. The temperature control system consists of:

(1) Hot Air Mixing Valves

(a) The mixing valve assembly (Figure 2) contains a butterfly plate spring-loaded to the closed position. The plate is mechanically linked to an actuator assembly. Inside the actuator is a diaphragm and pressure chamber. Air pressure displaces the diaphragm causing the butterfly plate to open. An internal spring determines the amount of air pressure that necessary to open the valve.



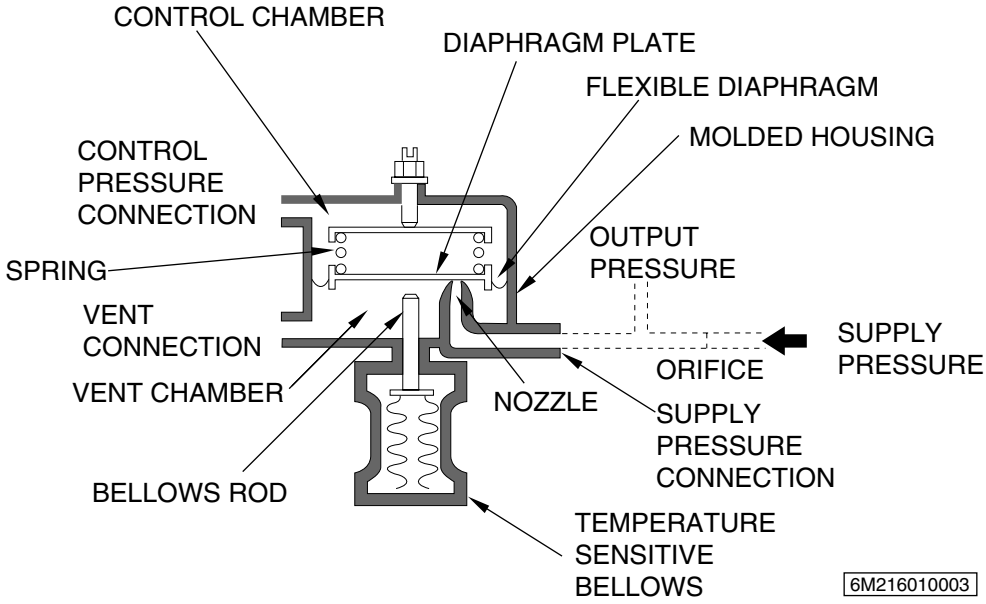
Hot Air Mixing Valve Figure 2

(2) Temperature Sensor

(a) The temperature sensor (Figure 3) consists of a temperature sensitive bellows, a control chamber and vent chamber separated by a flexible diaphragm plate, and a supply pressure controlling nozzle. An internal spring and adjustment normally preload the diaphragm plate against the supply pressure nozzle, blocking the flow through the nozzle.

(b) The temperature sensitive bellows exerts an opposing force to the internal spring and attempts to lift the diaphragm plate and open the pressure controlling nozzle. Different temperature range bellows are installed, depending upon the valve's application.

(c) The diaphragm plate separates the vent chamber and the control chamber. Pressure in the control chamber is used to change the response of the sensor. If the pressure in the control chamber is increased above ambient pressure, the result will be the same as increasing the spring preload pressure. The temperature at which the sensor will begin to vent supply pressure will be increased.



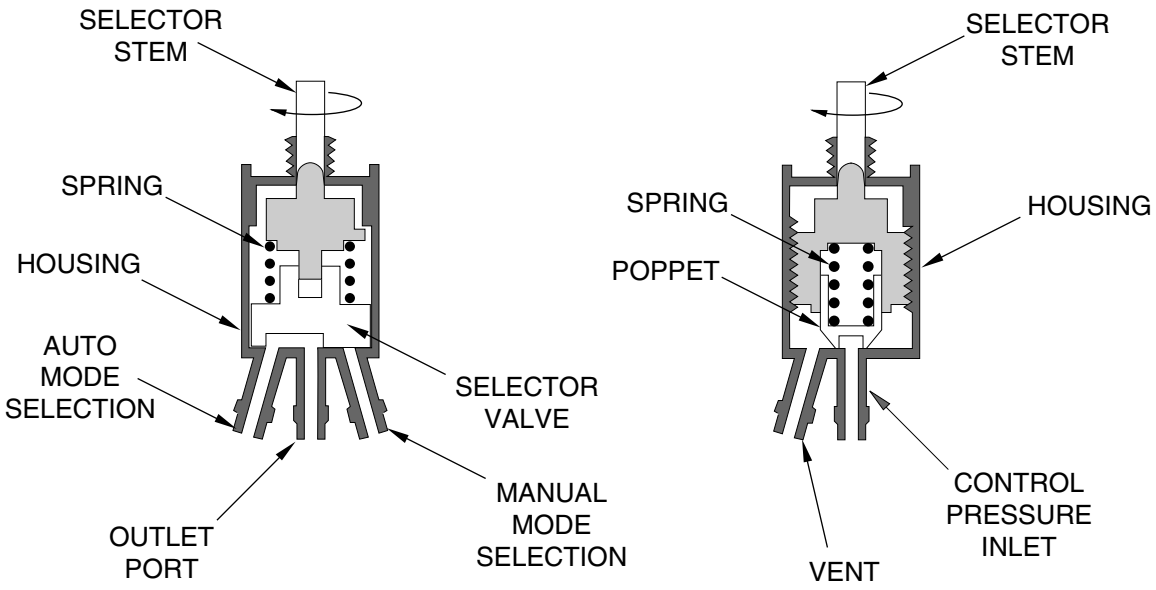
Temperature Sensor
Figure 3

(3) System Controls

(a) The controls available to the pilot are the control mode selector and the temperature control.

(b) The control mode selector selects either of two input pressures and routes the selected pressure to the outlet port. As shown in Figure 4, a selector valve interconnects two ports while blocking the third. The selector has two positions, AUTO and MANUAL.

(c) The outlet port of the selector is connected to the temperature control. The temperature control is a needle type regulating valve. Pressure from the mode selector is directed to the control pressure inlet port of the temperature selector. An overboard vent is supplied to the other side of the temperature control. As the control is rotated toward COLD, more of the supply pressure is vented overboard. Rotating the control toward HOT reduces the amount of pressure vented overboard.



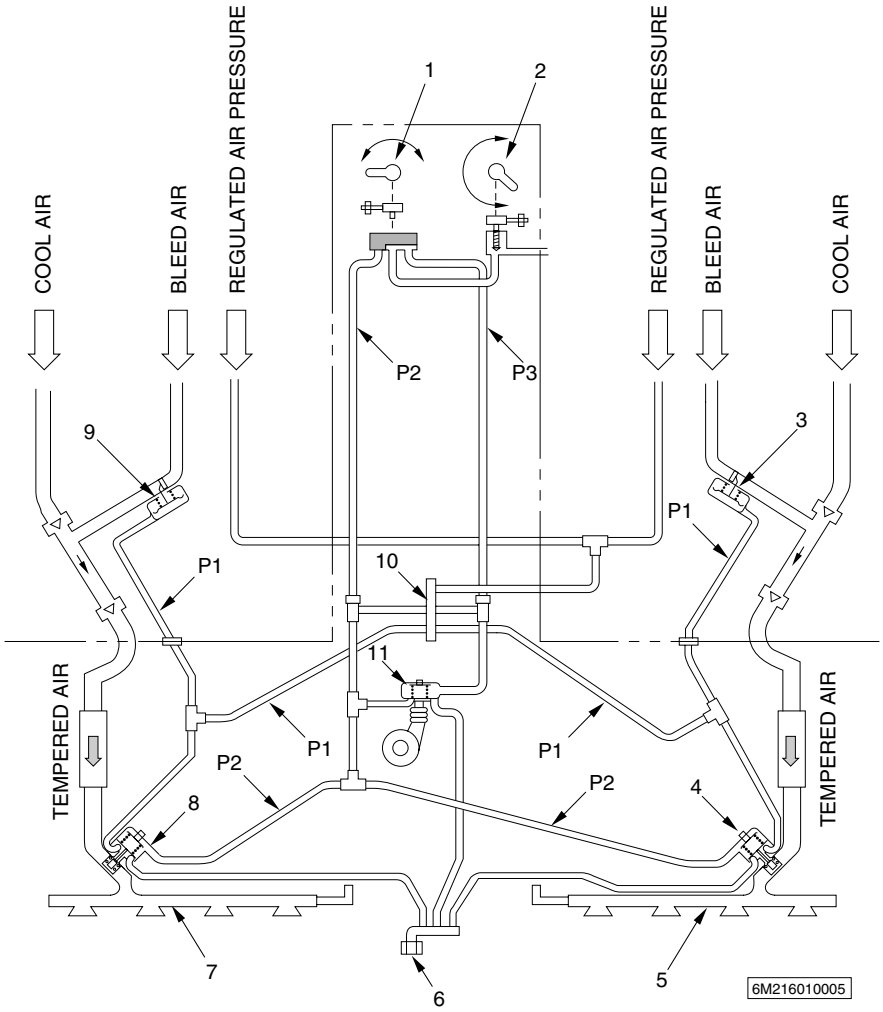
System Controls
Figure 4

TEMPERATURE CONTROL – DESCRIPTION AND OPERATION

C. Bleed air extracted from each engine is routed to a regulator located in each wing section. The regulator reduces the bleed air pressure to 16.4 plus or minus 1 psi. The regulated air pressure is routed to a tee fitting to provide regulated air pressure to the anti-ice system and the temperature control system. The temperature control system regulated pressure is routed to a manifold located below the right hand cabin floorboards, aft of the rear wing spar.

D. The regulated pressure is divided into three circuits (see Figure 5).

E. Part of the air pressure is routed to the hot air mixing valves and the duct temperature sensors. This pressure is referred to as P1. Air pressure is also routed to the control mode selector, the cabin temperature sensor, and the duct temperature sensors. This pressure is referred to as P2. Pressure to the control mode selector and the control chamber of the cabin temperature sensor is referred to as P3.



1. CONTROL MODE SELECTOR
2. TEMPERATURE CONTROL
3. RH HOT AIR MIXING VALVE
4. RH DUCT TEMPERATURE SENSOR
5. RH FLOOR OUTLETS
6. VENT
7. LH FLOOR OUTLETS
8. LH DUCT TEMPERATURE SENSOR
9. LH HOT AIR MIXING VALVE
10. PRESSURE MANIFOLD
11. CABIN TEMPERATURE SENSOR

Control Schematic
Figure 5

F. P1 is part of a circuit comprised of the respective hot air mixing valve and duct temperature sensor. The mixing valve will be full open (hot) when P1 is at or near supply pressure (16 psi). As P1 decreases, the mixing valve will close until the valve cracking pressure is reached (6.5 - 7 psi), then the valve will be fully closed. The duct temperature sensor controls the mixing valve by varying P1 pressure. If the sensor is venting most of P1 pressure, the mixing valve will be closed. As the sensor begins to vent less, P1 pressure increases causing the mixing valve to open.

G. The duct temperature sensor will eventually achieve a balanced state. The force exerted by the bellows will exactly oppose the force exerted by the control pressure. When this state is reached, P1 will be approximately stable, the mixing valve butterfly will be stopped in the desired position, and the temperature sensor will be venting as necessary. If the temperature within the duct changes, the temperature sensor will no longer be balanced and will begin to adjust P1 pressure (by additional venting or restricting) as necessary to again achieve balance. If the pressure supplied to the control chamber changes, the balance of the temperature sensor is upset and the sensor must adjust P1 to position the mixing valve to the necessary position.

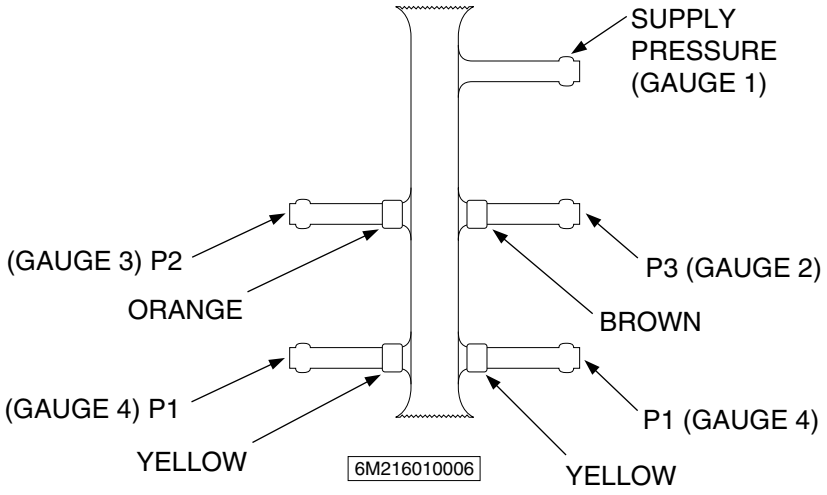
H. The pressure supplied to the control chamber of the duct temperature sensor is called P2. The origin of this pressure is the supply pressure manifold. P2 is affected by the cabin temperature sensor and the position of the temperature control. The cabin temperature sensor is identical in function to the duct temperature sensor, the exceptions being in the calibration settings because of the different temperature environments in which the valves are located. The control mode selector selects whether P2 pressure will be controlled automatically or manually.

I. In the MANUAL position, the selector routes P2 pressure to the temperature control. When the temperature control is moved toward HOT, less P2 pressure is vented overboard which causes the control pressure in each duct temperature sensor to increase. The duct temperature sensor will open the mixing valve farther to increase the temperature within the duct and allow the sensor to reach a balance state. If the temperature control is moved toward COLD, more P2 pressure is vented overboard which will result in the mixing valves moving toward closed.

J. Pressure from the pressure manifold is also routed to the opposite side of the control mode selector and to the control chamber of the cabin temperature sensor. This pressure is referred to as P3. With the control mode selector in MANUAL, P3 is deadheaded at the selector and the control chamber pressure of the cabin temperature sensor becomes supply pressure. With the control chamber pressure this high, the cabin temperature sensor can be considered effectively removed from the sensor would only open and vent P2 pressure if the cabin temperature was very high (in excess of 90°F).

K. When the control mode selector is positioned in AUTO, P2 pressure is deadheaded at the selector and P3 pressure is routed to the temperature control. The cabin temperature control chamber pressure is now controlled by the temperature control. P2 pressure is now being controlled entirely by the cabin temperature sensor. The cabin temperature sensor, like the duct temperature sensor, will seek to balance the force applied by the pressure in its control chamber and the force applied by the temperature sensitive bellows. If the temperature in the cabin changes, the force applied by the bellows will change and result in a different P2 pressure being applied to the duct temperature sensors. The mixing valves will be positioned as necessary to again balance the cabin temperature sensor. If the temperature control is moved toward the HOT position, the P3 pressure will increase causing the cabin air temperature sensor to increase P2 pressure. The increase in P2 pressure will cause the duct temperature sensor to increase the P1 pressure and thereby open the mixing valve farther to supply more hot air.

L. If the temperature control is moved toward COLD, the P3 pressure will decrease, the P2 pressure will decrease, and the P1 pressure will decrease causing the mixing valves to close.



Pressure Manifold
Figure 6

TEMPERATURE CONTROL – DESCRIPTION AND OPERATION

1. General

A. The temperature control system is electrically powered and designed to maintain cabin temperature at pilot selected levels. The pilot may select AUTO, HOT, or COLD positions with the temperature mode control.

(1) When the pilot selects AUTO and a desired temperature, the temperature control system will automatically regulate and maintain the aircraft at the selected temperature.

(2) When the pilot selects HOT or COLD, the automatic temperature control unit is bypassed and the admission of conditioned air is adjusted in accordance with the selection. The principle components of the system are as follows:

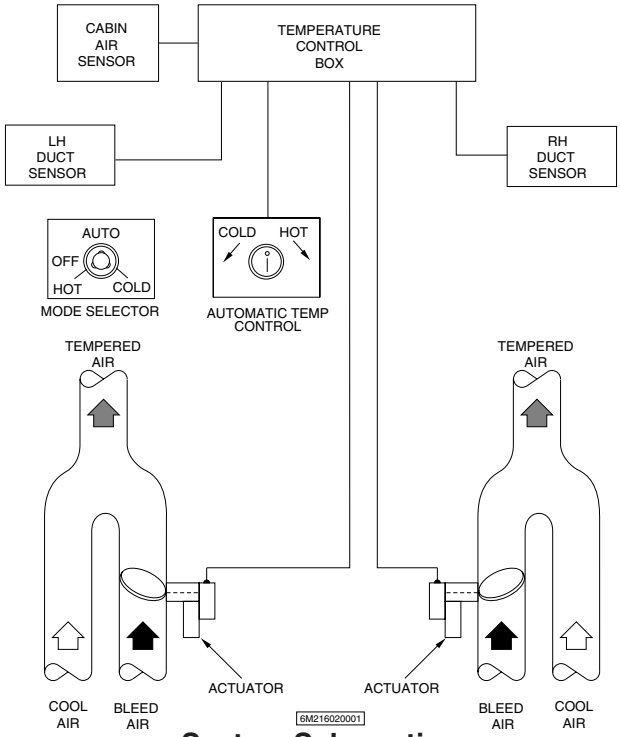
B. The temperature control system consists of:

(1) Temperature Controller Assembly

(a) The temperature controller assembly located at F. S. 255 (LH) on SA226 – T and F. S. 339 (RH) on SA226 – AT and TC contains a blower, a temperature sensor, and necessary electronic circuitry to properly regulate the hot air mixer valve. Input to controller from cabin duct sensors, cockpit temperature mode selector, and cabin air temperature sensor cause the temperature controller to energize the mixer valve actuator. The mixer valve actuator rotates the mixer valve to admit the correct amount of conditioned air to the cabin.

(2) Hot Air Mixing Valve

(a) The hot air mixing valves (see Figure 1) are installed in each respective wing section. The mixing valves are mechanically connected to an actuator. The actuators, when energized by the temperature controller, position the butterfly plate to regulate conditioned air flow.



System Schematic
Figure 1

(3) Temperature System Controls

(a) A mode selector switch and automatic temperature control are installed on the right hand switch panel. The mode selector has four positions; OFF, AUTO, HOT, and COLD.

(b) In the OFF position, all electrical power to the temperature control circuit is terminated.

(c) Placing the control in the HOT position will move the hot air mixing valves to the maximum open position.

(d) Moving the control to the COLD position will move the mixing valves to the full closed position.

(e) In the AUTO position, power is applied to the temperature control box and associated circuitry. The automatic temperature control, located adjacent to the mode selector, controls the cabin temperature when the system is operating in the automatic mode.

(4) Duct Temperature Sensor Assembly

(a) A duct temperature sensor assembly consisting of a sensor mount assembly and a duct sensing element is installed in the heater ducts (RH and LH) at F. S. 171 on SA226 – T and F. S. 281 on SA226 – AT and TC. The duct sensors monitor and detect changes in the temperature of conditioned air in the ducts.

SYSTEM CONTROLS – DESCRIPTION AND OPERATION

1. General

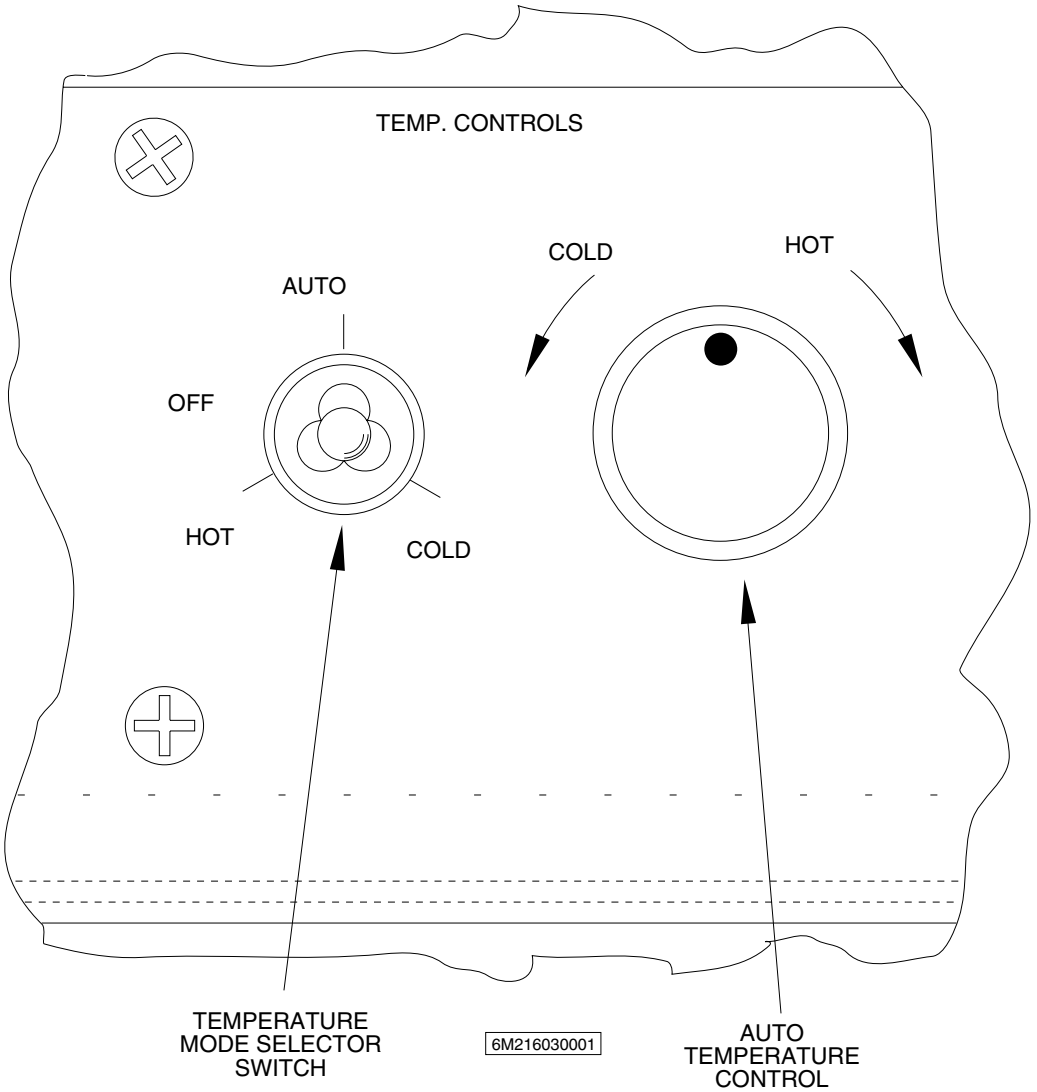
A. A mode selector switch and an automatic temperature control are installed on the right hand switch panel to control cabin temperature. The temperature mode selector switch is used to select one of four positions; AUTO, HOT, COLD, and OFF.

(a) In the OFF position, all electrical power to the temperature control circuit is terminated.

(b) Selecting HOT will position the hot air mixing valves to maximum open.

(b) Selecting COLD will position the mixing valves to full closed.

(d) In the AUTO position, power is applied to the temperature control box and associated circuitry. The automatic temperature control, located adjacent to the mode selector, controls the cabin temperature when the system is in the automatic mode.



Temperature Controls Right Switch Panel Figure 1

1. Adjustment/Test – System Controls

A. Control Mode Selector Test

(1) Place mode selector switch in COLD. Check pin #18 at cabin temperature controller connector for 28 VDC.

(2) Place mode selector switch in HOT. Check pin #19 at cabin temperature controller connector for 28 VDC.

(3) Place mode selector switch in AUTO. Check pin #1 at cabin temperature controller connector for 28 VDC.

B. Auto Temperature Control Test

(1) Connect ohmmeter between pins #4 and #6 at cabin temperature controller connector.

(2) Place control fully CCW COLD. Ohmmeter should read maximum 100 ohms.

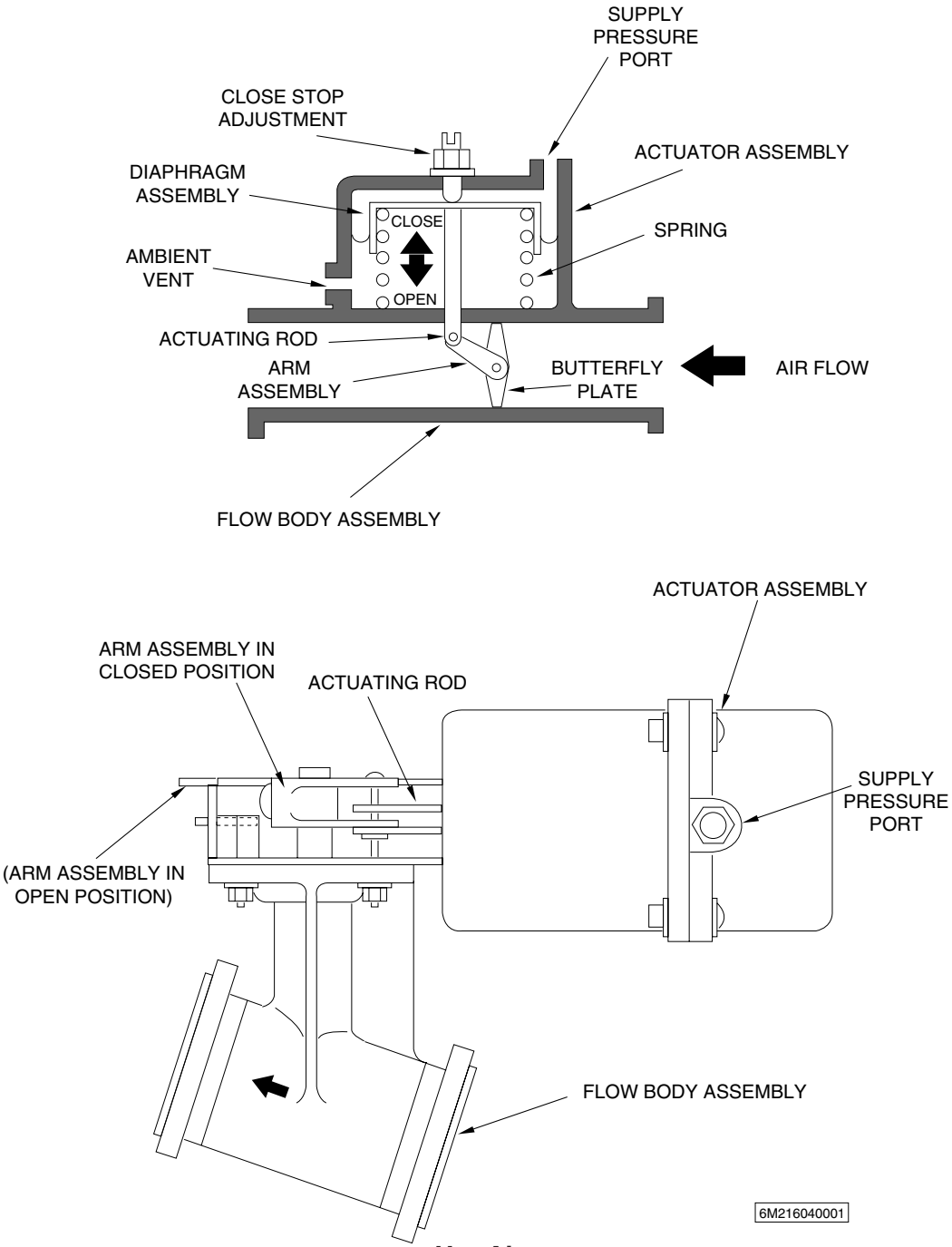
(3) Place control fully CW HOT. Ohmmeter should read 0 ohms.

(4) When rotating control knob CCW from full HOT to full COLD, observe that the ohmmeter rises smoothly from zero to 100 ohms.

HOT AIR MIXING VALVE – DESCRIPTION AND OPERATION

1. General

A. The hot air mixing valve consists of a butterfly plate spring loaded closed. The plate is mechanically linked to an actuator assembly. The actuator assembly consists of a diaphragm and a pressure chamber. Air pressure displaces the diaphragm and causes the butterfly to open.



Hot Air mixing Valve Figure 1

HOT AIR MIXING VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – Hot Air Mixing Valve

A. Removal – Hot Air Mixing Valve

(1) Gain access to valve by removing center section ECS access panels.

(2) Remove clamp securing supply line hose and remove hose.

CAUTION: REMOVE HOSE CAREFULLY OR DAMAGE TO FITTING MOUNTING BOSS MAY OCCUR. IF NECESSARY, SPLIT HOSE WITH RAZOR BLADE TO REMOVE.

(3) Loosen the two clamps securing valve to inlet and outlet tubing. Remove clamps and carefully pry out valve.

B. Installation – Hot Air Mixing Valve

(1) Inspect metal gasket. Replace if necessary.

(2) Position valve in place. Ensure gaskets are properly positioned.

(3) Install clamps and torque to 35 - 40 inch-pounds.

(4) Install supply line and secure with clamp.

(5) Apply 45 - 60 psi air pressure to the upstream side of either ECS pressure regulator.

(6) Move the mode selector to MANUAL.

(7) Vary the temperature control from COLD to HOT.

(8) Verifly mixing valve moves from closed to open in a smooth, steady motion.

(9) Reconnect pressure regulator and install access panels.

2. Adjustment/Test – Hot Air Mixing Valve

A. The operation of the valve may be checked as follows:

(1) Apply 28 VDC to pin A of actuator electrical connector.

(2) Move temperature control mode switch to HOT.

(3) Verify butterfly valve moves to open position in a smooth steady motion.

(4) Apply 28 VDC to pin B of actuator electrical connector.

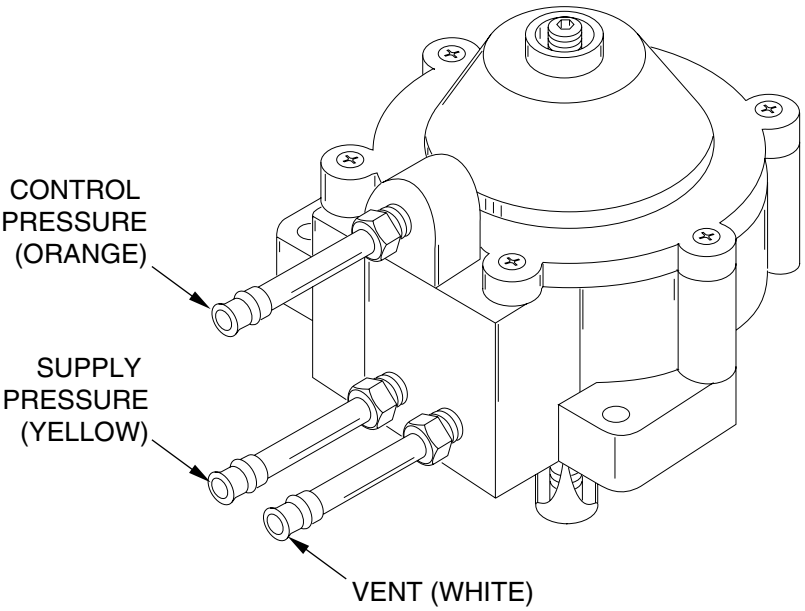
(5) Move temperature control mode switch to COLD.

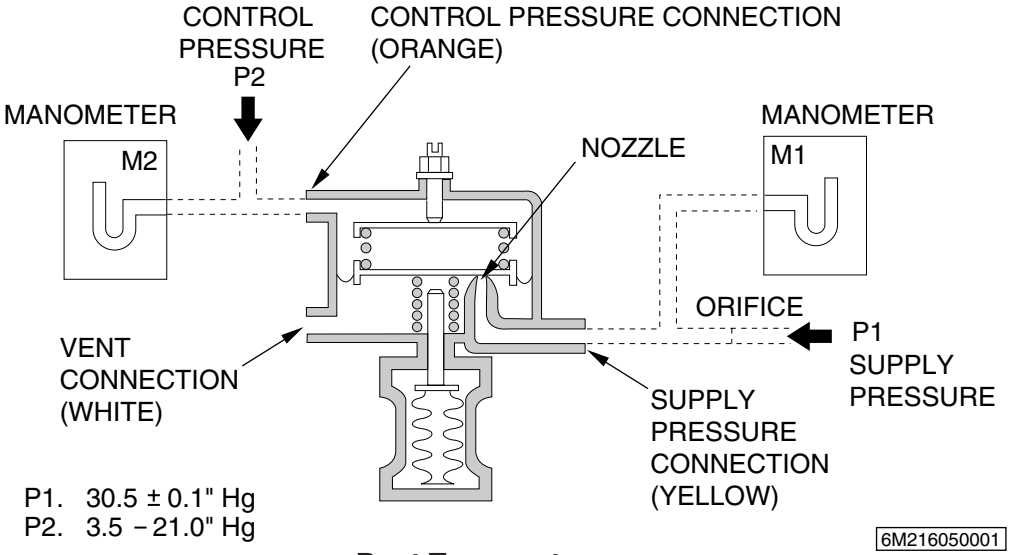
(6) Verify butterfly valve moves to closed position in a smooth steady motion.

DUCT TEMPERATURE SENSOR – DESCRIPTION AND OPERATION

1. General

A. A duct temperature sensor is installed in each conditioned air duct under the cabin floorboards at F. S. 197.25 in the SA226 – T aircraft, and at F. S. 320.88 in the SA226 – AT and TC aircraft. The sensor is used to vary P1 pressure to control the hot air mixing valve. Control pressure (P2) is supplied to the sensor by either the cabin temperature sensor or the temperature control.





Duct Temperature Sensor
Figure 1

DUCT TEMPERATURE SENSOR – MAINTENANCE PRACTICES

1. Removal/Installation –- Duct Temperature Sensor

A. Removal – Duct Temperature Sensor

(1) Gain access to sensor by removing respective cabin floorboard.

(2) Remove and tag the three (3) hoses connected to sensor.

CAUTION: REMOVE HOSES CAREFULLY OR DAMAGE TO FITTING MOUNTING BOSS MAY OCCUR. IF NECESSARY, SPLIT HOSE WITH RAZOR BLADE TO REMOVE.

(3) Remove three (3) screws and nuts securing sensor to duct and remove sensor.

B. Installation – Duct Temperature Sensor

(1) Position sensor in place and secure with original hardware.

(2) Install three (3) hoses and secure with clamps.

(3) Move aircraft to run up area and operate engines IAW AFM. Functionally check the operation of the temperature control system.

(4) Install cabin floorboard.

2. Inspection/Check – Duct Temperature Sensor

A. Required Equipment

(1) Two sources of regulated air pressure.

(2) Two mercury manometers or sensitive pressure gauges.

(3) One 0.0292 inch orifice.

B. Procedure

(1) Conduct the following test with sensor removed from aircraft:

(a) Connect a 30.5 plus or minus 0.1 inches Hg source of air pressure through the 0.0292 orifice to one port of a tee fitting. Connect one port of the tee fitting to the yellow fitting on the sensor, and the final port of the tee fitting to one of the manometers (M1).

(b) Connect regulated source of air pressure to a tee fitting. Connect one port of tee to manometer (M2). Connect final port of the tee to the control (orange) connection on the sensor.

(c) Apply a control pressure (orange) of 3.5 plus or minus 0.1 inches Hg to the sensor. Verify the output indicated on M1 is 3.3 plus or minus 2 inches Hg.

(d) Increase the control pressure to 21 inches Hg. Verify the output indicated on M1 is a minimum of 28 inches Hg with an ambient temperature 60° - 70°F.

NOTE: Some air leakage by the bellows mounting flange may be detected. Vent leakage is due to surface irregularity and is not critical to the sensor's function.

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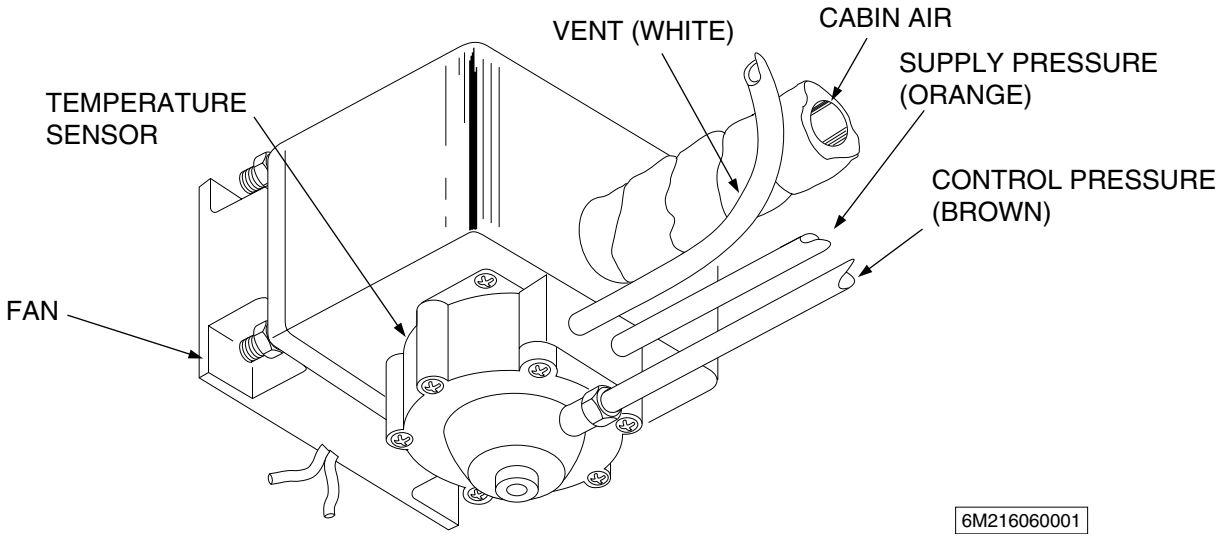
CABIN AIR TEMPERATURE SENSOR – DESCRIPTION AND OPERATION

1. General

A. A cabin air temperature sensor is installed below the right hand cabin floorboards at F. S. 243.19 in the SA226 – T aircraft, and at F. S. 332.27 in the SA226 – AT and TC aircraft. Cabin air is drawn from a vent in the overhead panel in the passenger compartment. A duct extending behind the right hand cabin furnishing routes cabin air to the sensor. A fan, located adjacent to the sensor, draws air through the duct and across the sensor. The sensor and fan are installed on a shock-mounted metal box.



Cabin Air
Temperature Sensor
Figure 1

CABIN AIR TEMPERATURE SENSOR – MAINTENANCE PRACTICES

1. Removal/Installation – Cabin Air Temperature Sensor

A. Removal – Cabin Air Temperature Sensor

(1) Gain access to sensor assembly by removing the respective right hand cabin floorboard.

(2) Disconnect and tag the three (3) hoses connected to sensor.

CAUTION: REMOVE HOSES CAREFULLY OR DAMAGE TO FITTING MOUNTING BOSS MAY OCCUR. IF NECESSARY, SPLIT HOSE WITH RAZOR BLADE TO REMOVE.

(3) Remove cabin air supply hose from sensor assembly. Tag electrical wires to fan and cut at existing splices.

(4) Remove four (4) screws and washers on the forward side of the fuselage frame and remove sensor assembly.

(5) Disassemble sensor assembly as required.

B. Installation – Cabin Air Temperature Sensor

(1) Position sensor assembly in place against fuselage frame and secure with original hardware.

(2) Install the three (3) hoses to the sensor. Install the cabin air supply line and secure using a tie strap.

(3) Connect the wires to the fan using permanent splices. (Ref to the Wiring Manual).

(4) Move the aircraft to run up area and run the engines IAW AFM. Functionally check the operation of the temperature control system.

(5) Install the cabin floorboards.

2. Inspection/Check – Cabin Air Temperature Sensor

A. Cabin sensor assembly may be check when removed from the aircraft as follows:

(1) Required Equipment:

(a) Two (2) sources of regulated air pressure.

(b) Two (2) mercury manometers or sensitive pressure gauges.

(c) One (1) 0.0292 inch orifice.

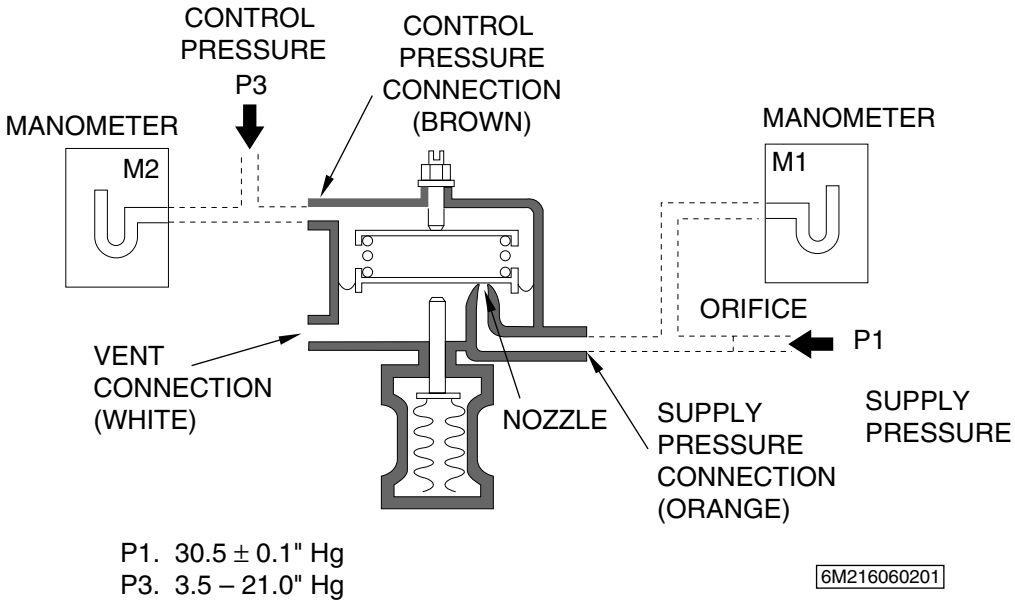
(2) Procedure.

(a) Connect a 30.5 ± 0.1 inch Hg source of air pressure, through the 0.0292 orifice, to one port of a tee fitting. Connect one port of tee to the orange fitting on the sensor, and the final port of tee to one of the manometers (M1).

(b) Connect a regulated source of air pressure to a tee. Connect one port of tee to manometer (M2). Connect the final port of the tee to control (brown) connection on the sensor.

(c) Apply a control pressure (brown) of 3.5 ± 0.1 inches Hg. Verify indicated pressure on M1 is 19.8 ± 2 inches Hg with ambient temperature of 60° ± 1°F. With an ambient temperature of 70°F, the output will be approximately 8 to 10 inches Hg. lower.

(d) Increase the control pressure to 21 inches Hg. Verify indicated pressure on M1 should be a minimum of 28 inches Hg.



Connections for Cabin Sensor Check Figure 201

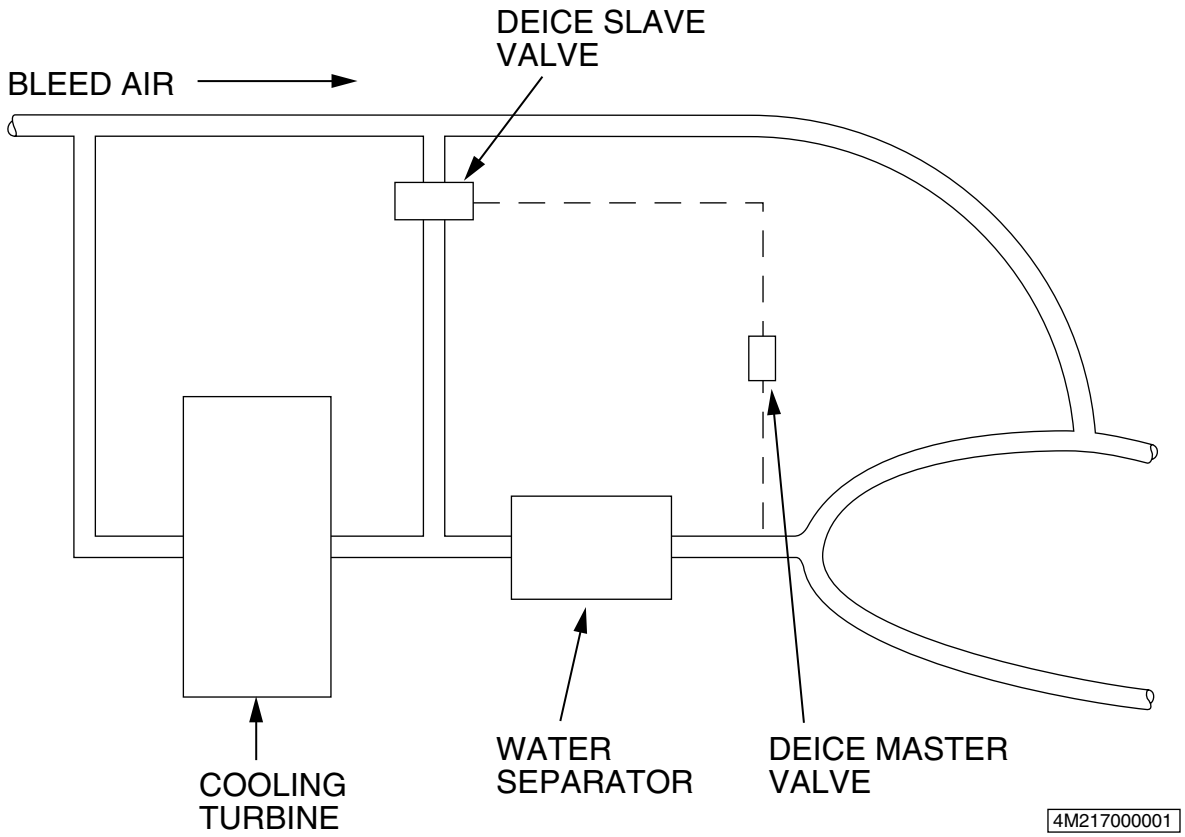
MOISTURE AND CONTAMINANT CONTROL – DESCRIPTION AND OPERATION

1. General

A. Moisture and other contaminants are removed from engine supplied bleed air before the air enters the cabin. Three different systems have been employed to accomplish this function, however, the basic theory of operation is the same for all three.

B. A water separator is installed downstream of the cooling turbine to remove water from the air. A deicing system is installed to prevent blocking of the water separator.

C. The deicing system consists of a slave valve that controls the introduction of hot bleed air into the cold air duct and a master valve that controls the operation of the slave valve. The master valve may sense the outlet air temperature of the water separator or the pressure differential across an ice collecting screen. In either case, the master valve will control the slave valve as necessary to prevent freezing inside the water separator.



Moisture and Contaminant Control Basic System Figure 1

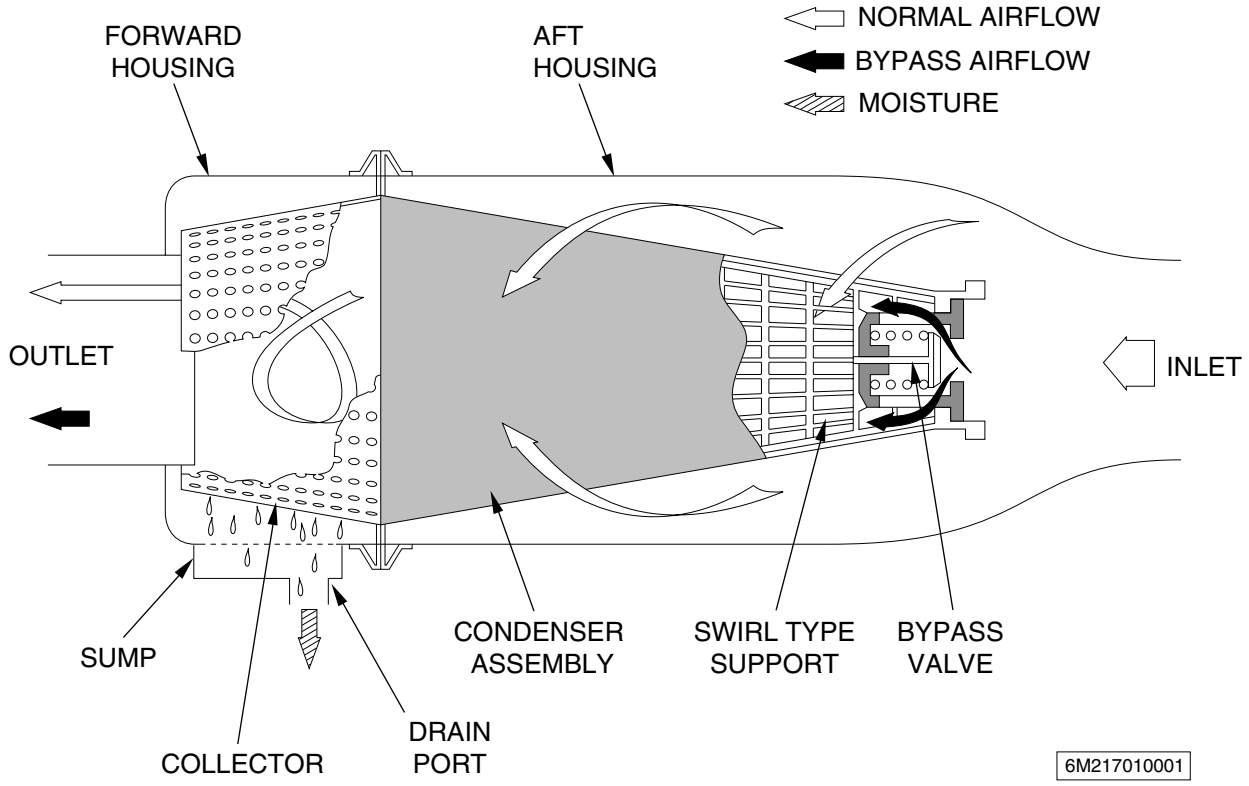
WATER SEPARATOR – DESCRIPTION AND OPERATION

1. General

A. A water separator is installed downstream of each turbine outlet to remove trapped moisture from the cold air supply.

B. Air entering the separator is forced to travel in a circular motion and through a fiberglass sock. The air travels in a circular motion through the center of the separator until centrifugal action forces the water droplets outboard through the holes of the collector. Water accumulates in the sump of the water separator and is discharged overboard through a drain port.

C. A bypass valve allows free flow of air should the water separator sock become clogged or so soiled as to impede air flow. When the pressure differential across the water separator reaches 4.6 psi, the bypass valve opens and air flows through the water separator. With the bypass valve open, the relative airflow to the cabin will remain the same but the moisture removing action of the separator is bypassed.



Water Separator
Figure 1

WATER SEPARATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Water Separator

A. Removal – Water Separator

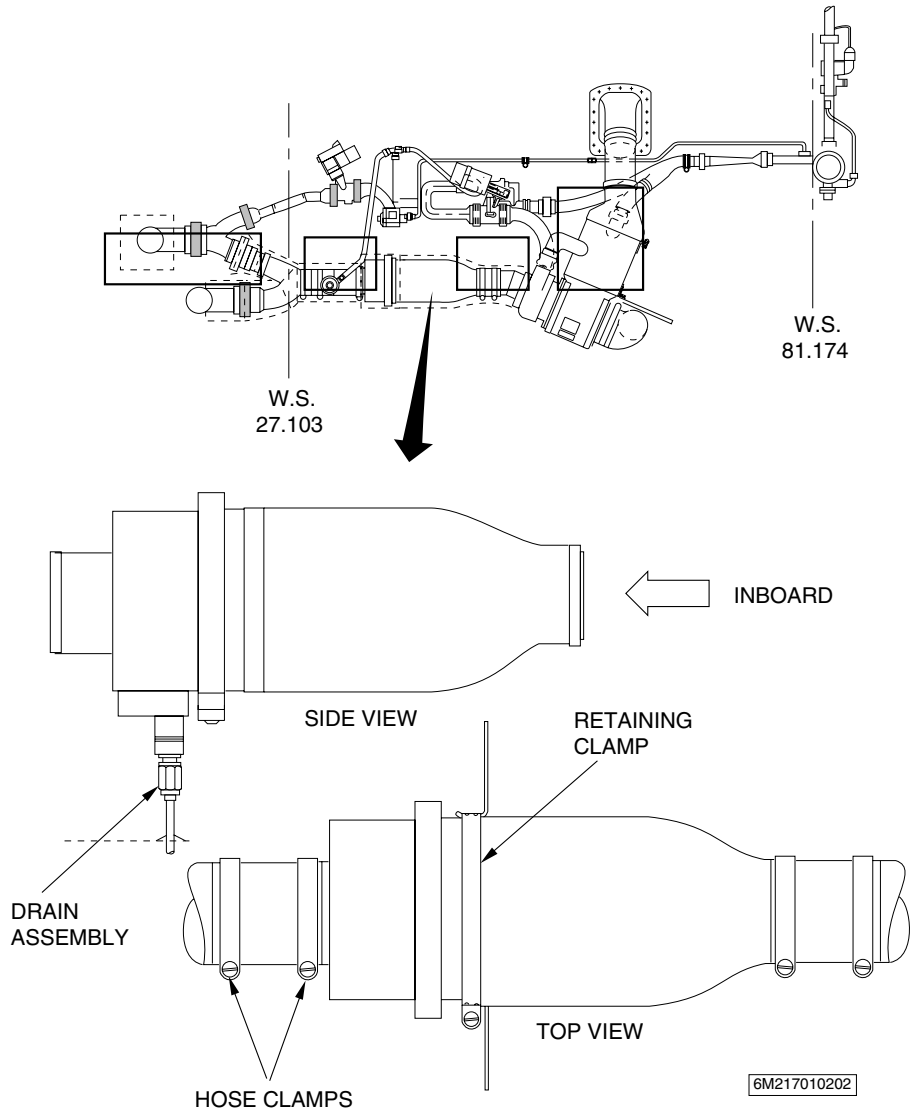
(1) Remove lower wing access panels. Note that the overboard drain extends through one of the access panels.

(2) Remove insulation blanket from separator (on applicable aircraft).

(3) Remove clamps securing hoses to inlet and outlet ducts of separator.

NOTE: It may be necessary to loosen both clamps at each end to obtain enough clearance for removal of separator.

(4) Remove retaining clamp securing separator to wing spar. Remove separator from wing.



Water Separator Installation
Figure 201

B. Installation – Water Separator

(1) Place separator into position in wing. Observe that arrows on housing point inboard and down.

(2) Secure separator to wing spar with retaining clamp.

(3) Install hoses at inlet and outlet ends and secure with clamps.

(4) Tighten all clamps. Install insulation blanket (on applicable aircraft).

(5) Install all lower wing access panels.

NOTE: Use care to avoid damaging the overboard drain where it extends through the access panel.

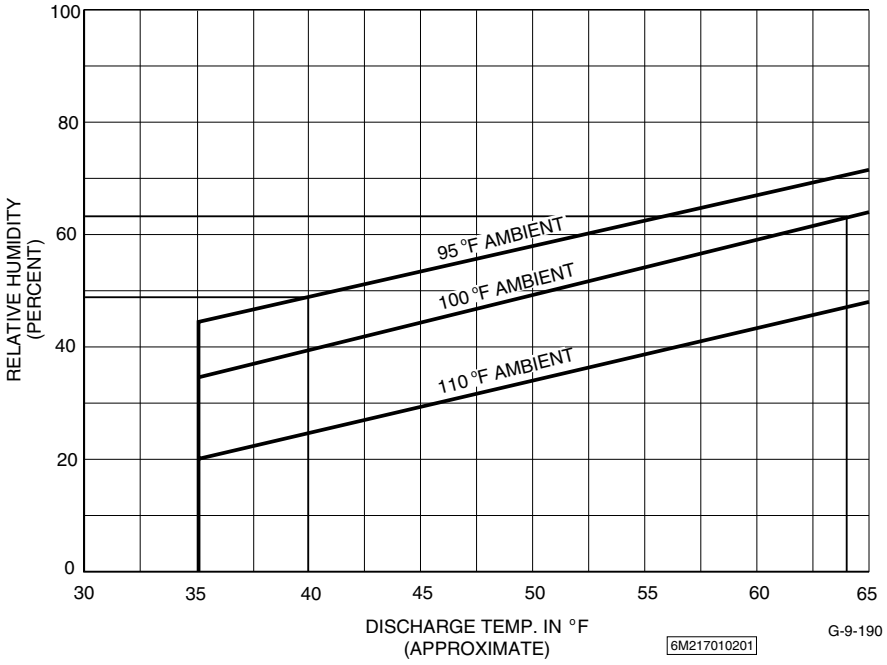
2. Inspection/Check-Water Separator

A. Separator Temperature Vs Relative Humidity

(1) The low limit system is designed to prevent water discharge temperatures below freezing. Water separator temperatures may not always be near the freezing point. Under certain conditions of temperature and humidity the temperature may be considerably higher.

(2) Figure 202, Separator Temperature Vs Relative Humidity, should be used as a trouble-shooting guide to determine the expected discharge temperature with engines at approximately 90% and water separator in maximum cooling.

(3) Temperature readings can be taken with a standard glass thermometer or an all metal type used in automobile air conditioning checks. The thermometer can be inserted into the water separator drain. Depth of insertion should be varied to obtain an average reading.



Separator Temperature vs Relative Humidity Figure 202

(4) To determine approximate discharge temperature, proceed as follows:

(a) Determine relative humidity.

(b) Determine ambient temperature.

(c) At the point of intersection of temperature and humidity on the chart, read down. The result is the approximate discharge temperature.

(d) For example (see Figure 202):

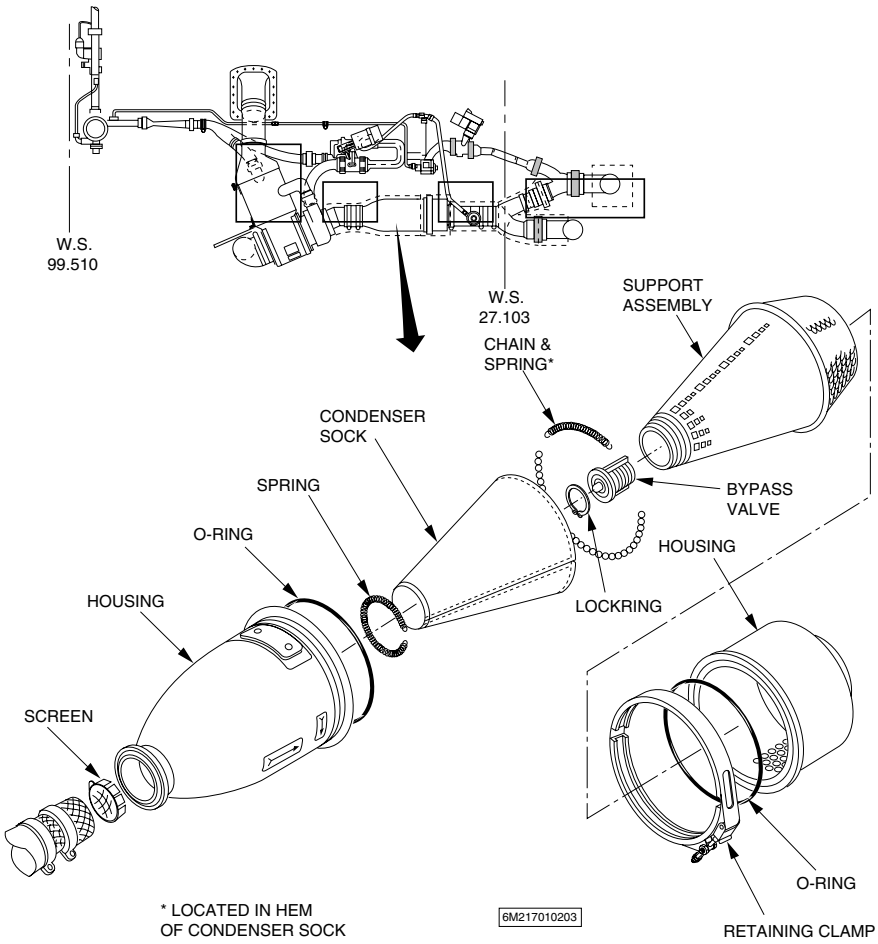
With an ambient temperature of 100 degrees Fahrenheit and a relative humidity of 62%, expect a discharge temperature of approximately 64 degrees Fahrenheit.

With an ambient temperature of 95 degrees Fahrenheit and a relative humidity of 48%, expect a discharge temperature of approximately 40 degrees Fahrenheit.

3. Adjustment/Test – Water Separator

A. Disassembly.

(1) Remove the clamp securing the two halves of the water separator housing (see Figure 203).



Water Separator Disassembly
Figure 203

CAUTION: WEAR GLOVES TO PROTECT HANDS WHILE HANDLING FIBERGLASS CONDENSER.

(2) Remove bypass valve lockring.

(3) Remove bypass valve and inspect for spring action and freedom of operation.

(4) Inspect condenser sock for condition and cleanliness.

NOTE: Use care to avoid stretching condenser sock while sliding off or on support assembly.

B. Cleaning.

(1) Clean the separator sock using a mild detergent. If necessary, sock may be removed from support assembly.

C. Assembly.

(1) Install condenser sock on support assembly (if removed).

(2) Install bypass valve in condenser assembly and secure with lockring.

(3) Inspect O-rings for condition. Replace if necessary.

(4) Join the two halves of the separator and secure with clamp.

ECS DEICING – DESCRIPTION AND OPERATION

1. General

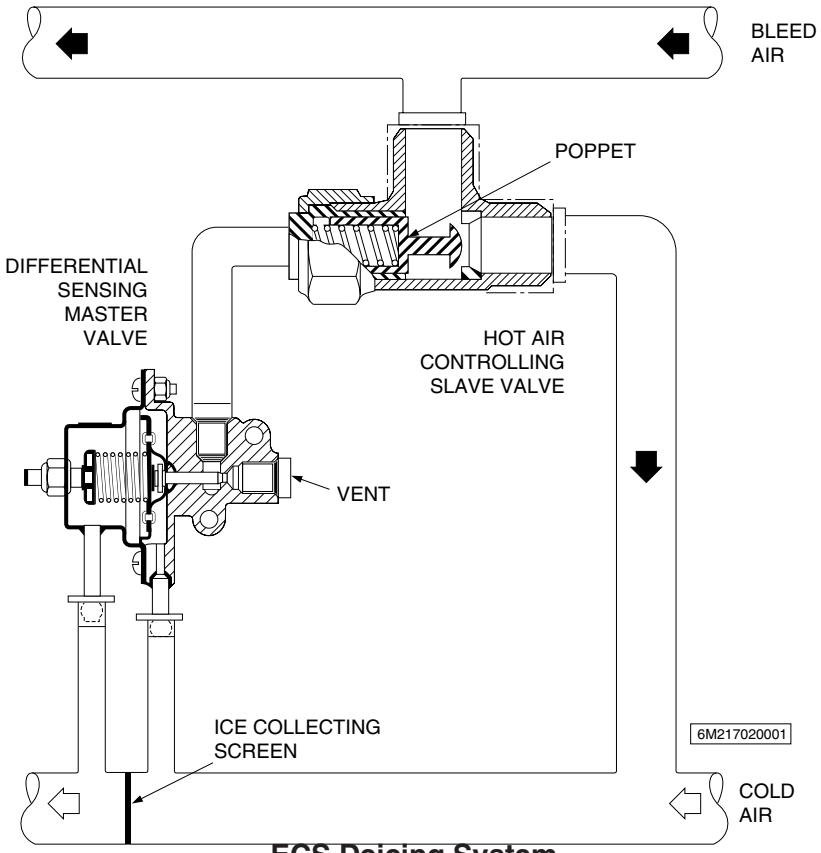
A. The ECS deicing system consists of a differential sensing master valve, an ice collecting screen, and a hot air controlling slave valve. Hot air is introduced into the turbine discharge air to ensure that the water separator does not ice over and to eliminate freezing of the ECS ducts.

B. Engine bleed is routed to the hot air controlling slave valve (see Figure 1). The slave valve is a balanced pressure poppet type valve. Pressure is introduced to both sides of the poppet and hold the valve closed. One of the balanced sides is connected to the master valve.

C. The master valve is a differential valve constructed of two chambers separated by a rubber diaphragm. One chamber is connected to the pressure on one side of the screen. The other chamber is connected to the pressure on the other side of the screen. If ice begins to form on the screen, the pressure on the supply side of the screen will increase. The pressure on the down side of the screen will decrease. When a sufficient differential exists between the two pressures, the master valve will open.

D. Opening of the master valve vents the control side of the slave valve to ambient pressure. The slave valve has been held in the closed position by the balance of pressure within the valve. The difference in pressure allows the poppet to move toward the vented side and allows hot air to be introduced into the cold air duct.

E. When sufficient bleed air has been introduced to deice the screen, a pressure differential will no long exist across the screen. The master valve senses the pressure equalization and closes. The pressure within the slave will balance and the poppet will move to the closed position. The system remains at static rest until a sufficient amount of ice accumulates on the screen and a pressure differential again exists within the system.



ECS Deicing System
Figure 1

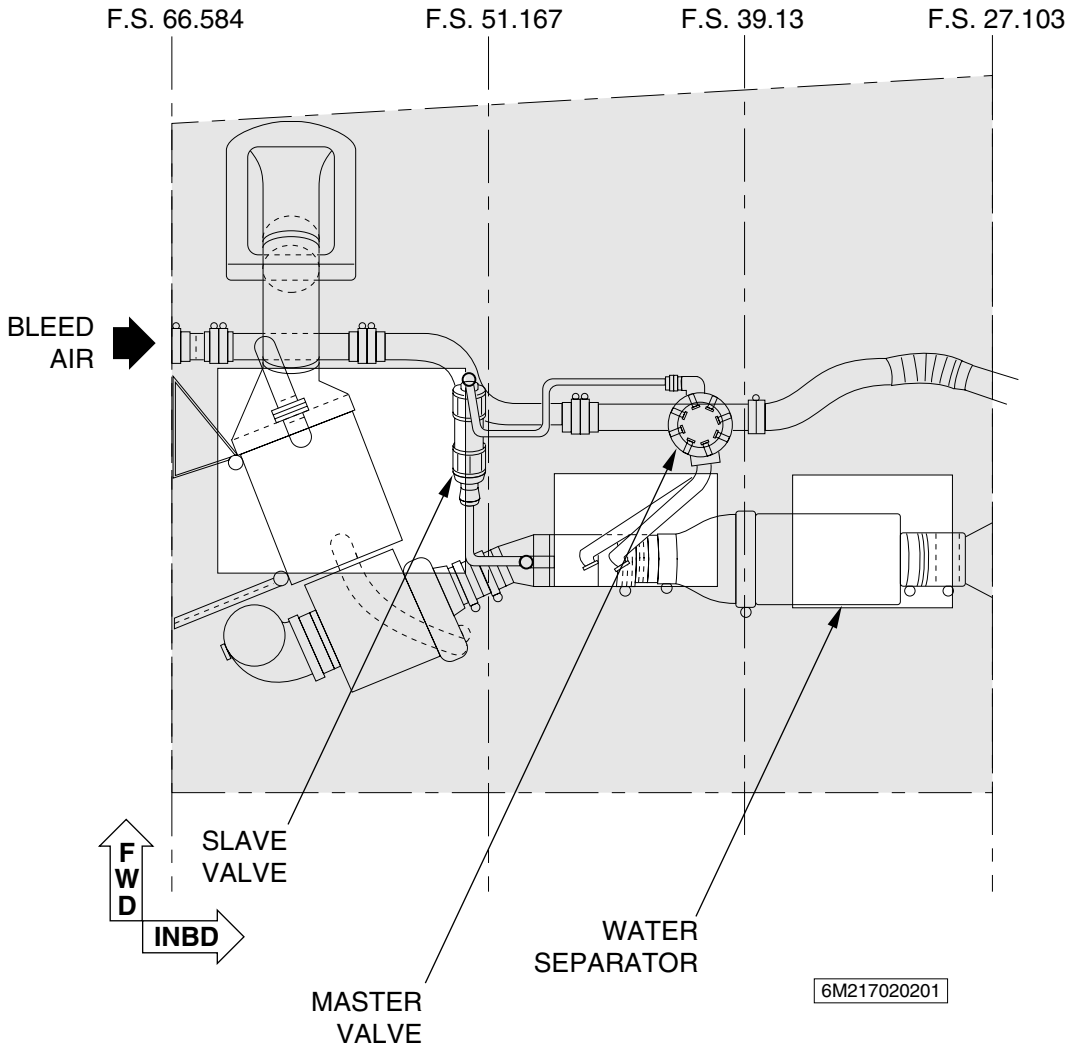
2. Removal/Installation – ECS Deicing

A. Removal – Slave Valve

(1) Remove panel Inbd of F.S. 66.584 to gain access to valve (see Figures 201 and 202).

(2) Disconnect lines attached to valve. Unscrew supply line B-nut and withdraw valve.

(3) Remove end fitting from valve. Do not remove the end fitting that is safetied to valve.



ECS Deicing
Components – Left Side
Figure 201

B. Installation – Slave Valve

(1) Coat threads of new valve with anti-seize (Fel Pro C5-A or equivalent) and install end fitting.

(2) Position valve in place and connect lines.

(3) Install access panels.

C. Removal – Master Valve

(1) Remove panel outbd of F.S. 39.13 to gain access to valve (see Figures 201 and 202).

(2) Tag rubber lines attached to valve and disconnect, remove hard line attached to valve.

(3) Remove two (2) bolts and nuts securing valve to rib and remove valve.

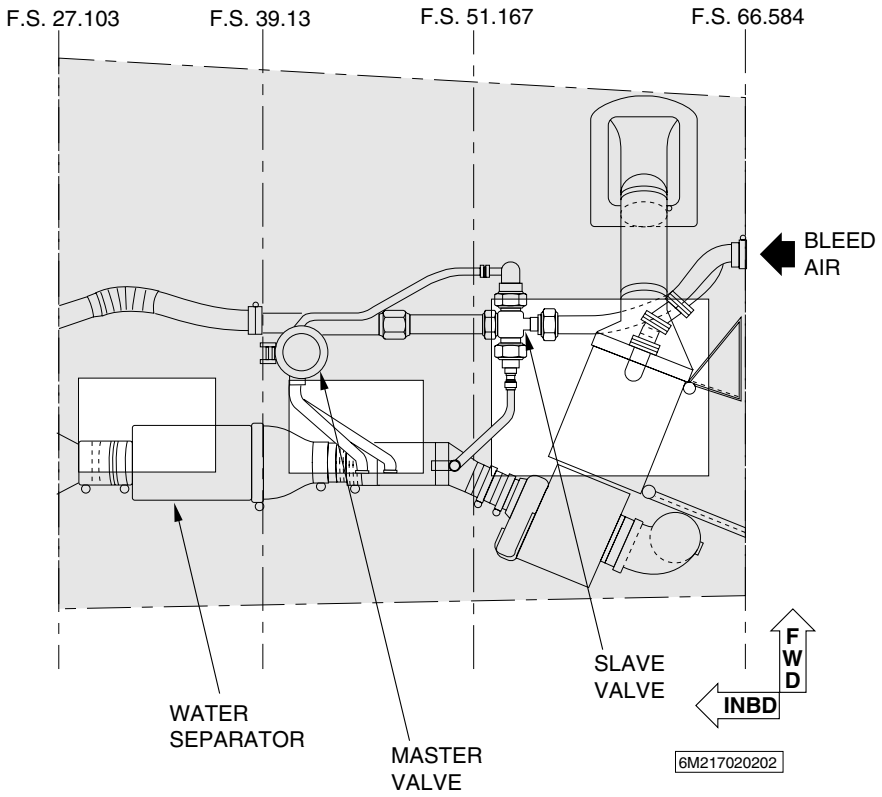
D. Installation – Master Valve

(1) Install valve in wing using original bolts, spacers, and nuts.

(2) Connect lines to valve.

NOTE: The rubber line attached most inboard on the cold air duct system should be connected to the top of the valve. The line connected nearest the cooling turbine should be connected to the bottom of the valve.

(3) Install access panel.



ECS Deicing System Components – Right Side
Figure 202

ECS DEICING – DESCRIPTION AND OPERATION

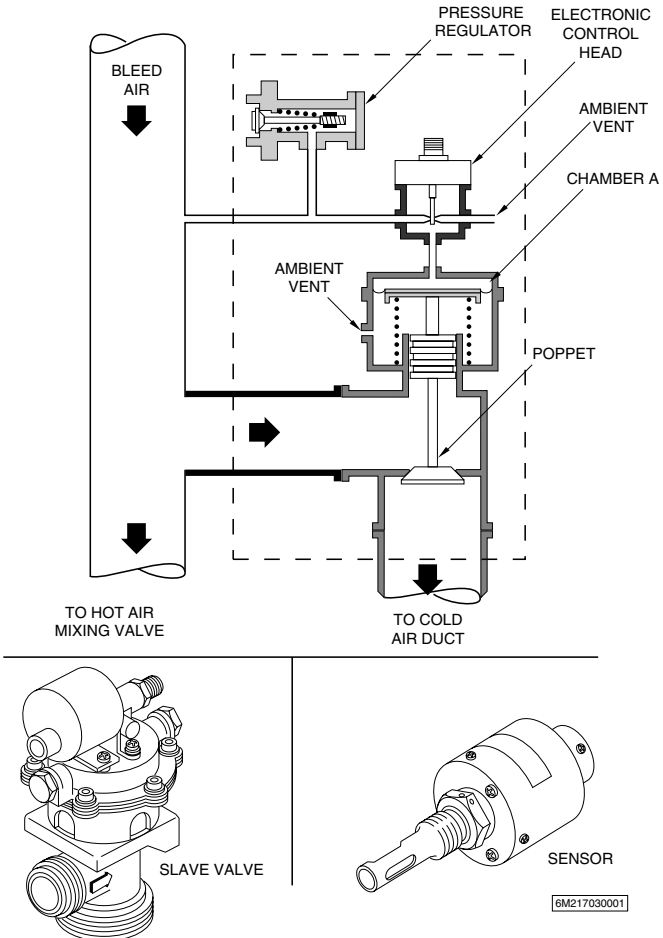
1. General

A. The ECS deicing system consists of a temperature sensor and a hot air controlling slave valve. Hot air is introduced into the turbine discharge air to ensure that the water separator does not ice over and to eliminate freezing of the air conditioning system ducts.

B. Engine bleed air is routed to the hot air controlling slave valve (see Figure 1). The slave valve is a balanced poppet valve, one half of the balance being ambient air and the other half being the pressure present in chamber A. The pressure in chamber A will be dependent upon the position of the electrically operated control head. The control head may allow either regulated pressure from the hot air duct to enter chamber A, ambient pressure to enter chamber A, or any combination of the two. The poppet valve is spring loaded to the closed position.

C. The temperature sensor measures the temperature of the air in the cold air duct and supplies an electrical signal to the slave valve. The master valve attempts to maintain the temperature in the duct at 35°F.

D. As the temperature in the duct varies, the electrical output from the sensor will vary. The slave valve reacts to this signal by either increasing or decreasing the pressure in chamber A. If more bleed air is allowed in, the pressure increases. If more air is allowed to vent to ambient, the pressure decreases. When the pressure in chamber A increases, the poppet will open, introducing bleed air into the cold air duct. The sensor will position the slave valve so that the temperature in the duct remains at 35°F.



ECS Deicing
Figure 1

1. Removal/Installation – ECS Deicing System

A. Removal – Slave Valve

(1) Gain access to slave valve by removing panels as shown in Figures 201 and 202.

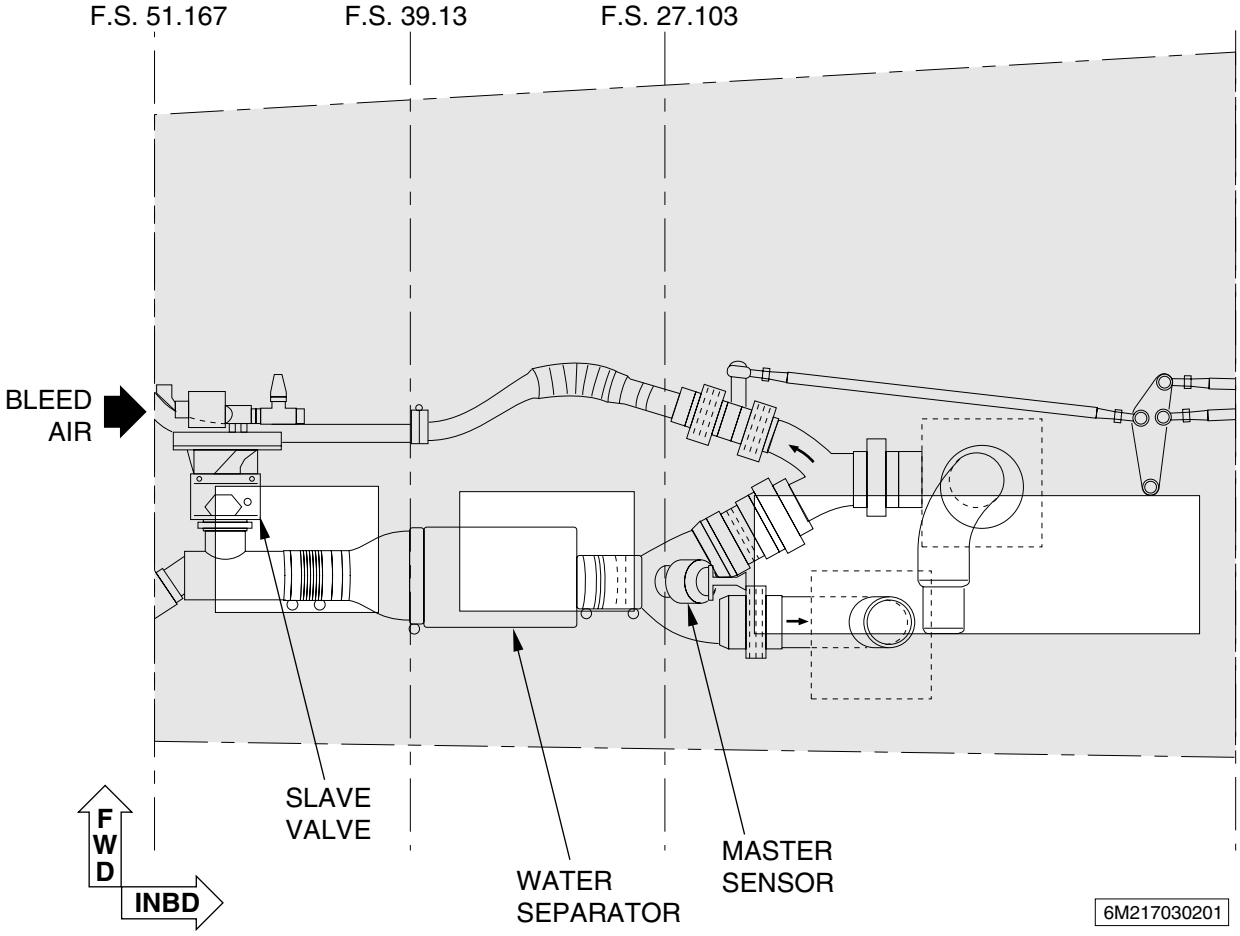
(2) Disconnect electrical connector from valve.

(3) Disconnect small pressure line connected to valve pressure regulator.

(4) Disconnect large supply line from valve inlet.

(5) Cut safety wire and remove three (3) screws securing duct to outlet of valve.

(6) Remove two (2) nuts and bolts securing valve to attaching bracket and withdraw valve from wing.



ECS Deicing Components – Left Side
Figure 201

B. Installation – Slave Valve

(1) Install O-ring in groove on outlet of valve.

(2) Coat threads of inlet of valve with anti-seize (Fel-Pro C5-A or equivalent).

(3) Position valve in place and secure to bracket using original bolts and nuts.

(4) Connect inlet line to valve.

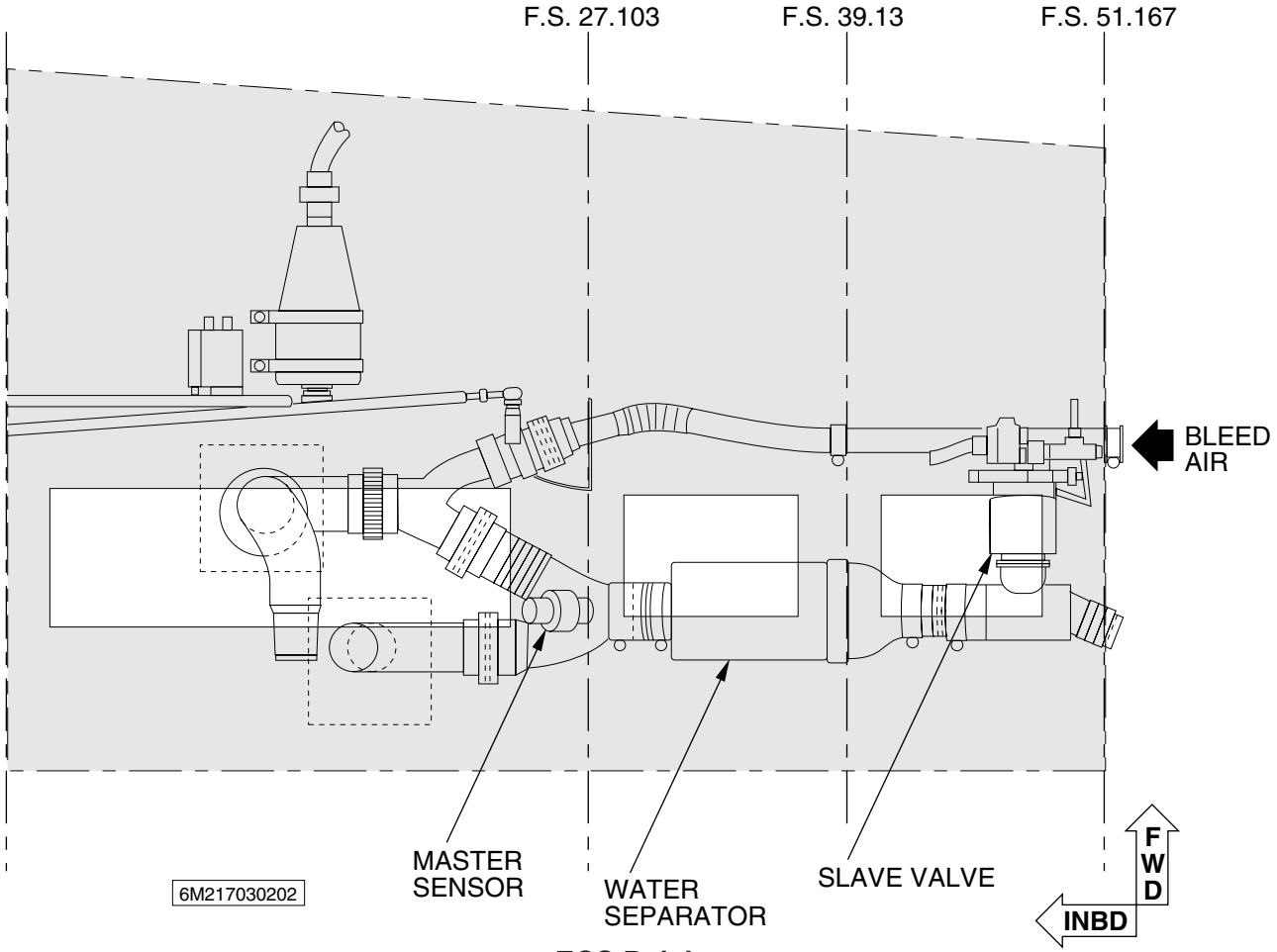
(5) Install screws securing duct to outlet of valve and tighten finger tight plus 1/8 to 1/4 turn.

(6) Safety wire screws in place.

(7) Connect small line to valve pressure regulator.

(8) Install electrical connector to valve.

(9) Install access panels.



ECS Deicing Components – Right Side Figure 202

C. Removal - Master Sensor

(1) Gain access to sensor by removing panels as shown in Figures 201 and 202.

(2) Disconnect electrical connector from sensor.

(3) Unscrew sensor and remove.

NOTE: Do not use pliers on body of sensor.

D. Installation - Master Sensor

(1) Install O-ring on sensor.

(2) Position sensor in place and screw into duct.

(3) Attach electrical connector to sensor.

(4) Install access panels.

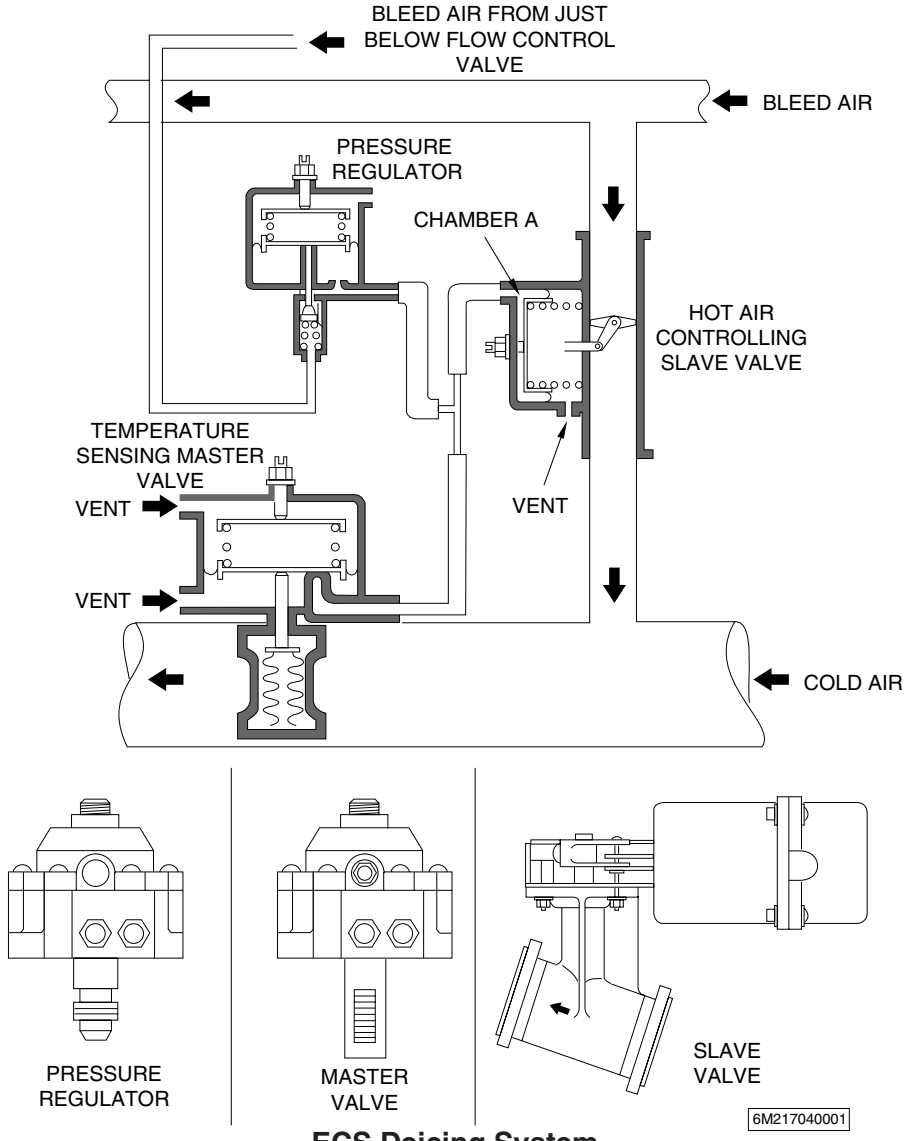
ECS DEICING – DESCRIPTION AND OPERATION

1. General (Figure 1)

A. To ensure the water separator does not ice over and to eliminate freezing of the air conditioning ducts, hot air is introduced into turbine discharge air before air leaving the water separator freezes. The deicing system consists of a temperature sensing master valve, a hot air controlling slave valve, and a supply pressure regulating valve.

2. Hot Air Controlling Slave Valve

A. Engine bleed air is routed to the hot air controlling slave valve. The slave valve is a spring-balanced, pressure operated, butterfly plate control valve. When the pressure in chamber A becomes sufficient to overcome the force of the spring, the valve diaphragm assembly begins to open the butterfly plate, allowing hot air to be introduced into the cold air duct. The spring normally holds the valve closed, and if the pressure in chamber A is reduced to less than the spring value, the valve closes.



ECS Deicing System
Figure 1

3. Temperature Sensing Master Valve

A. The temperature sensing master valve receives chamber A pressure. When the master valve senses a duct temperature of 37°F. or higher, it opens allowing the pressure to be vented to ambient, allowing the spring inside the slave valve to hold the valve closed. If the temperature drops to below 70°F, the master valve begins to close allowing pressure to build in chamber A and open the slave valve. The master valve, using pressure regulation, will open the slave valve and admit the correct amount of hot air to maintain the duct temperature at 37°F.

4. Pressure Regulator

A. The positioning of the slave valve is dependent upon the amount of air the master valve allows to escape and the supply pressure provided. To maintain an accurate supply pressure, a pressure regulator is used. Bleed air system just after the flow control valve and routed to the wing center section. A regulator maintains is extracted pressure at 16 - 18 psi. The pressure regulator output passes to the master and slave valves through a Tee junction equipped with a 0.039 orifice to control the pressure flow to the master valve.

1. Removal/Installation – ECS Deicing

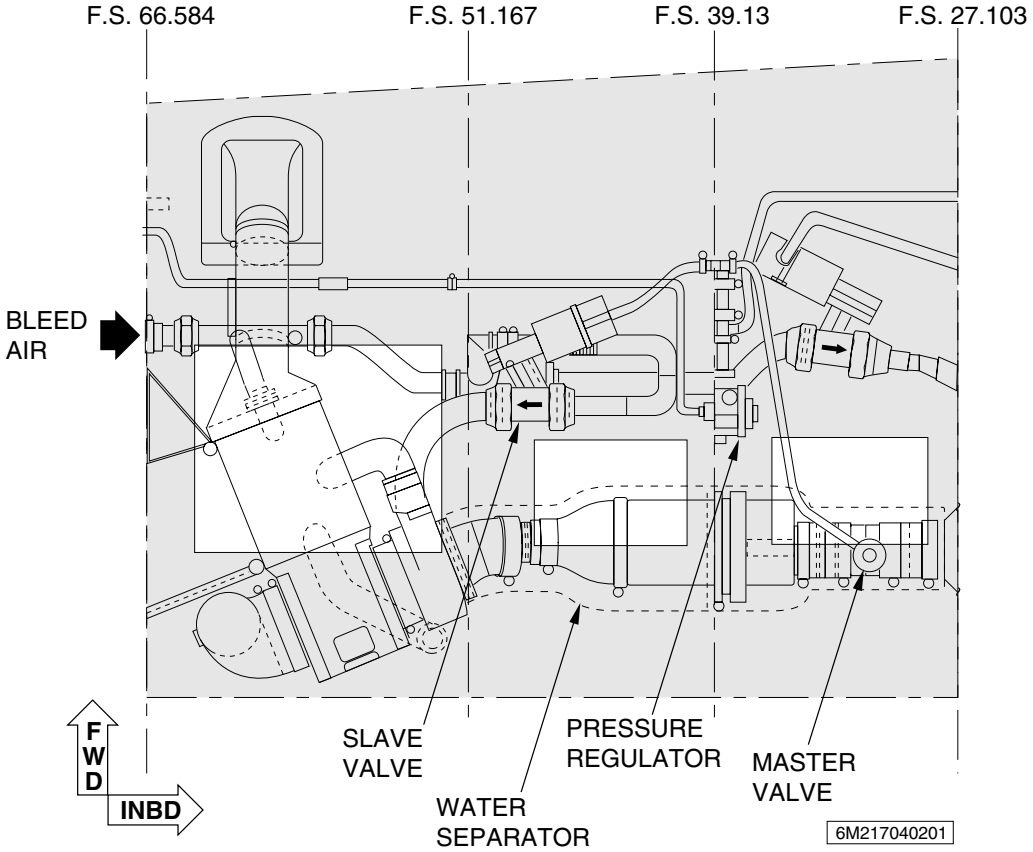
A. Removal – Slave Valve

(1) Gain access to valve by removing panel Inbd of F.S. 51.167 as shown in Figures 201 and 202.

(2) Remove clamp securing rubber hose to valve and remove hose.

CAUTION: REMOVE HOSE CAREFULLY OR DAMAGE TO FITTING MOUNTING BOSS ON VALVE MAY OCCUR. IF NECESSARY, SPLIT HOSE WITH RAZOR BLADE TO REMOVE.

(3) Loosen clamps attaching valve to inlet and outlet tubing. Remove clamps and valve.



ECS Deicing System Components – Left Side
Figure 201

B. Installation – Slave Valve

(1) Inspect gaskets. Replace if necessary.

(2) Position valve in place and install clamps. Torque clamps 35 - 40 inch-pounds.

(3) Install rubber hose and secure with clamp.

(4) Install access panel.

C. Removal – Master Valve

(1) Gain access to valve by removing panel outbd of F.S. 26.103 as shown in Figures 201 and 202.

(2) Remove clamp securing rubber hose to valve and remove hose.

CAUTION: REMOVE HOSE CAREFULLY OR DAMAGE TO FITTING MOUNTING BOSS ON VALVE MAY OCCUR. IF NECESSARY, SPLIT HOSE WITH RAZOR BLADE TO REMOVE.

(3) Remove three screws and nuts securing valve to duct and remove valve. Retain gasket.

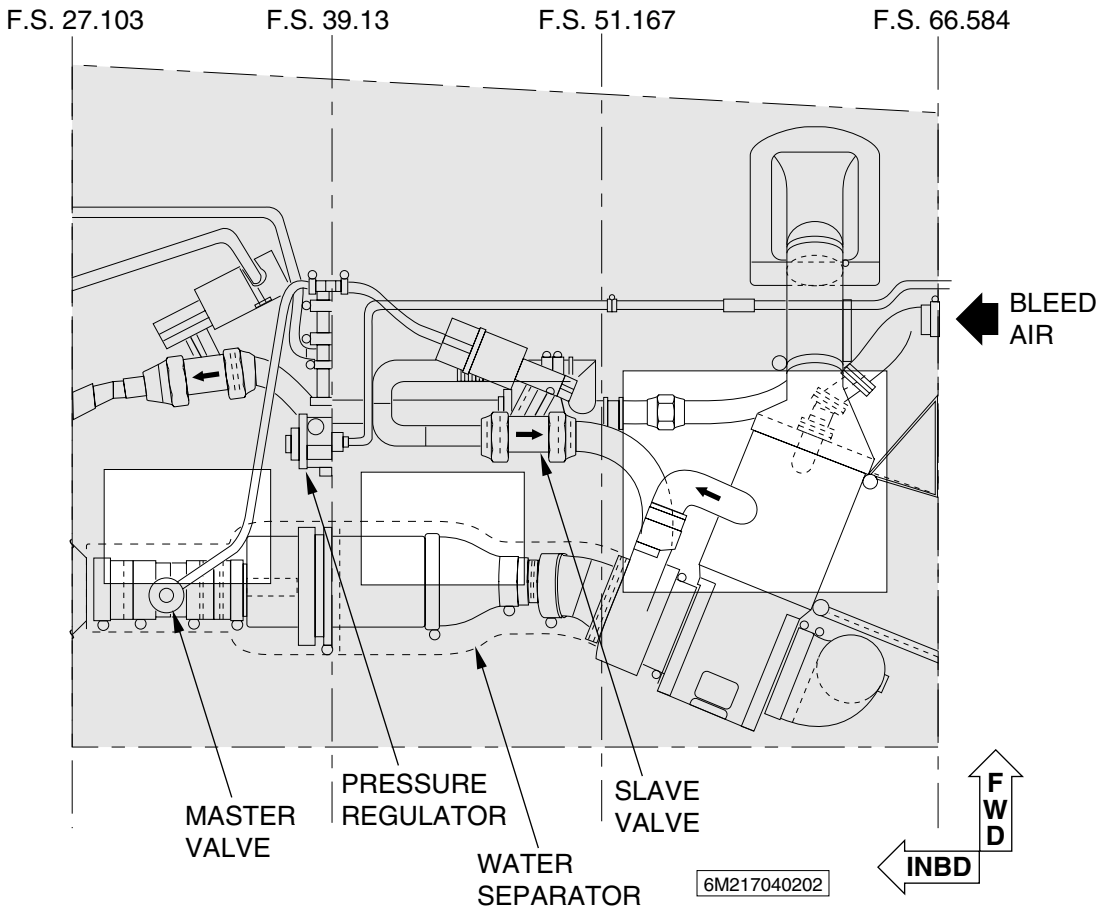
D. Installation – Master Valve

(1) Inspect gaskets. Replace if necessary.

(2) Position valve and gasket in place and secure with screws and nuts.

(3) Install rubber hose and secure with clamp.

(4) Install access panel.



ECS Deicing Components – Right Side
Figure 202

E. Removal – Pressure Regulator

(1) Gain access to regulator by removing panel outbd of F.S. 27.103 and F.S. 39.13 as shown in Figures 201 and 202.

(2) Disconnect supply line B-nut. Remove clamp securing rubber hose to regulator and remove hose.

CAUTION: REMOVE HOSE CAREFULLY OR DAMAGE TO FITTING MOUNTING BOSS ON REGULATOR MAY OCCUR. IF NECESSARY, SPLIT HOSE WITH RAZOR BLADE TO REMOVE.

(3) Remove screws and nuts securing regulator to rib and remove regulator.

F. Installation – Pressure Regulator

(1) Coat inlet threads of regulator with anti-seize (Fel-Pro C5-A or equivalent) and position in place. Secure regulator using original hardware.

(2) Connect supply line B-nut to inlet of regulator. Install rubber hose and secure with clamp.

(3) Install access panels.

2. Inspection/Check – ECS Deicing

NOTE: Perform this check during regularly scheduled inspections, or if a cooling turbine fails. If the Low Limit Duct Sensor is not operating properly, ice may form on turbine wheel, causing imbalance and failure.

A. Remove Master Valve (Low Limit Duct Sensor) (see Removal - Master Valve, this section).

B. Visually inspect bellows assembly for damage.

C. Discard valve if there is evidence of impact damage to the assembly.

D. Install Master Valve.

E. Run cooling turbine with engine advanced to above 90 percent.

F. Select Manual Full Cold on cabin temperature selector.

G. Insert thermometer in water separator drain port.

H. If temperature falls below 35° F, replace Master Valve.

MZ AEROSPACE
An Elbit Systems of America, LLC, Company

MAINTENANCE MANUAL

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AN ELBIT SYSTEMS OF AMERICA, LLC, COMPANY

MAINTENANCE MANUAL

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COMMUNICATIONS – DESCRIPTION AND OPERATION

1. General

A. This chapter describes the provisions for a typical installation regarding standard items of communications equipment. Detailed description is dependent upon avionics installed in individual aircraft.

B. Communications equipment provisions are provided for the installation of two identical but separate VHF transceiver systems. Provisions for a ramp hailer system, providing flight crew to ground communication, is also provided.

C. Provisions for installation of an audio selector panel are included. The audio selector panel permits cockpit headset or speaker monitoring of selected communication and/or navigation equipment signals.

D. Provisions for a Cockpit Voice Recorder (CVR) for T201-215, T(B)276, 292-417, AT001-074, and TC201-999 airplanes, on an option-basis, are presented in Service Bulletin, 226-23-001, COCKPIT VOICE RECORDER INSTALLATION, dated January 9, 1992. These provisions are in compliance with FAR 91.609 and 135.151 requirements.

1. Removal/Installation – Communications

NOTE: The following procedures apply to typical avionics equipment mounted on nose compartment radio rack.

A. Removal – Typical Rack Mounted Component

(1) Open FWD baggage compartment access doors.

(2) Remove radio rack access cover.

(3) Loosen knurled knob(s) securing component to shelf on radio rack.

(4) Disconnect antenna coaxial connectors.

(5) Slide component aft and remove.

B. Installation – Typical Rack Mounted Component

CAUTION: ENSURE PINS OF COMPONENT CONNECTORS ARE STRAIGHT. DO NOT APPLY EXCESSIVE FORCE WHEN INSTALLING UNIT.

(1) Position component on rack and slide in FWD direction until electrical connectors are mated.

(2) Tighten knurled knob(s) securing component to rack.

(3) Connect coaxial antenna cables.

(4) Install radio rack access cover.

(5) Close and secure FWD baggage compartment access doors.

C. Removal – Typical Instrument Panel Mounted Component

(1) Unlock Dzus fasteners at corners of the control head.

(2) Pull control head out of instrument panel.

D. Installation – Typical Instrument Panel Mounted Component

CAUTION: ENSURE PINS OF COMPONENT CONNECTORS ARE STRAIGHT. DO NOT APPLY EXCESSIVE FORCE WHEN INSTALLING UNIT.

(1) Slide control head into its dust cover.

(2) Secure control head with Dzus fasteners.

VERY HIGH FREQUENCY (VHF) TRANSCEIVER – DESCRIPTION AND OPERATION

1. General

A. Provisions for installation of two identical, but separate VHF transceiver communications systems are incorporated in the aircraft. The transceivers are mounted on the radio rack in the nose baggage compartment and are designated VHF COMM No. 1 and No. 2.

B. The VHF transceivers are remotely controlled by control panels mounted across the top of the instrument panel. The VHF COMM No. 1 transceiver antenna is located on the belly of the aircraft, normally at F.S. 196 for T, and F.S. 298 for AT and TC series. The VHF COMM No. 2 transceiver antenna is also located on the belly of the aircraft normally at F.S. 180 for T, and F.S. 278 for AT and TC aircraft. Power to operate the VHF COMM No. 1 and No. 2 transceivers is furnished by the LH and RH Essential buses.

C. Locations of the VHF antennas vary somewhat, depending on the particular avionics package.

VERY HIGH FREQUENCY (VHF) TRANSCEIVER – MAINTENANCE PRACTICES

1. Removal/Installation – Very High Frequency (VHF) Transceiver

A. Removal – VHF Comm Antenna – No. 1 or No. 2 (see Figure 201)

(1) Remove screws (item 3) attaching antenna (item 1) to belly of fuselage.

(2) Lower antenna far enough to remove antenna lead.

NOTE: If VHF Comm Antenna No. 1 or No. 2 coaxial cable is too short to enable removal of antenna lead from exterior of fuselage, loosen coaxial cable from within fuselage.

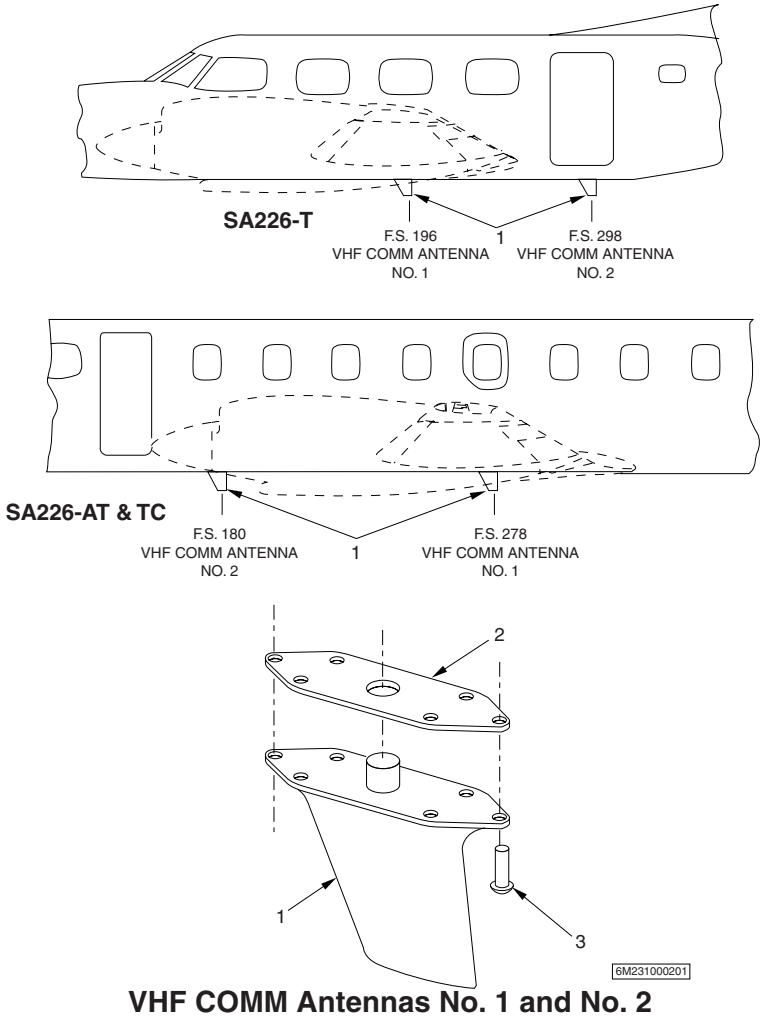
(3) Remove antenna and gasket (item 2).

B. Installation – VHF Comm Antenna – No. 1 and No 2 (see Figure 201)

(1) Position antenna (item 1) with gasket (item 2) over mounting holes on aircraft.

(2) Connect coaxial cable to antenna.

(3) Secure antenna with screws (item 3).



VHF COMM Antennas No. 1 and No. 2
Figure 201

PASSENGER ADDRESS – DESCRIPTION AND OPERATION

1. General

A. A passenger address (PA) system enables the pilot or copilot to speak to the passengers through speakers installed throughout the cabin. The PA amplifier is an integral part of the audio amplifier. Speaker volume is controlled by a rheostat located on the audio amplifier. Speakers are installed throughout the cabin in varying numbers, complementing the two cockpit speakers.

B. Each cockpit speaker, serving pilot/copilot communication needs, is a 5 inch (12.70 cm) diameter speaker mounted within a round cutout in the cockpit headliner. The cabin speakers typically consist of two sizes, 3 inch (7.62 cm) and 5-inch (12.70 cm) in diameter. Each cabin speaker is mounted within a cutout of a side wall, panel assembly. The cabin speakers are variably spaced above the seat locations with the total number varying from one aircraft to another.

C. A cover is provided for each cockpit speaker. Both rectangular and round covers are provided for the cabin speakers.

1. Removal/Installation – Passenger Address

A. Removal – Cockpit Speaker (Figure 201)

(1) Remove clip secured headliner panel (item 2) above pilot/copilot side window to gain access to either cockpit speaker (item 1), using two (2) standard panel removal/installation tools as shown in Figure 202.

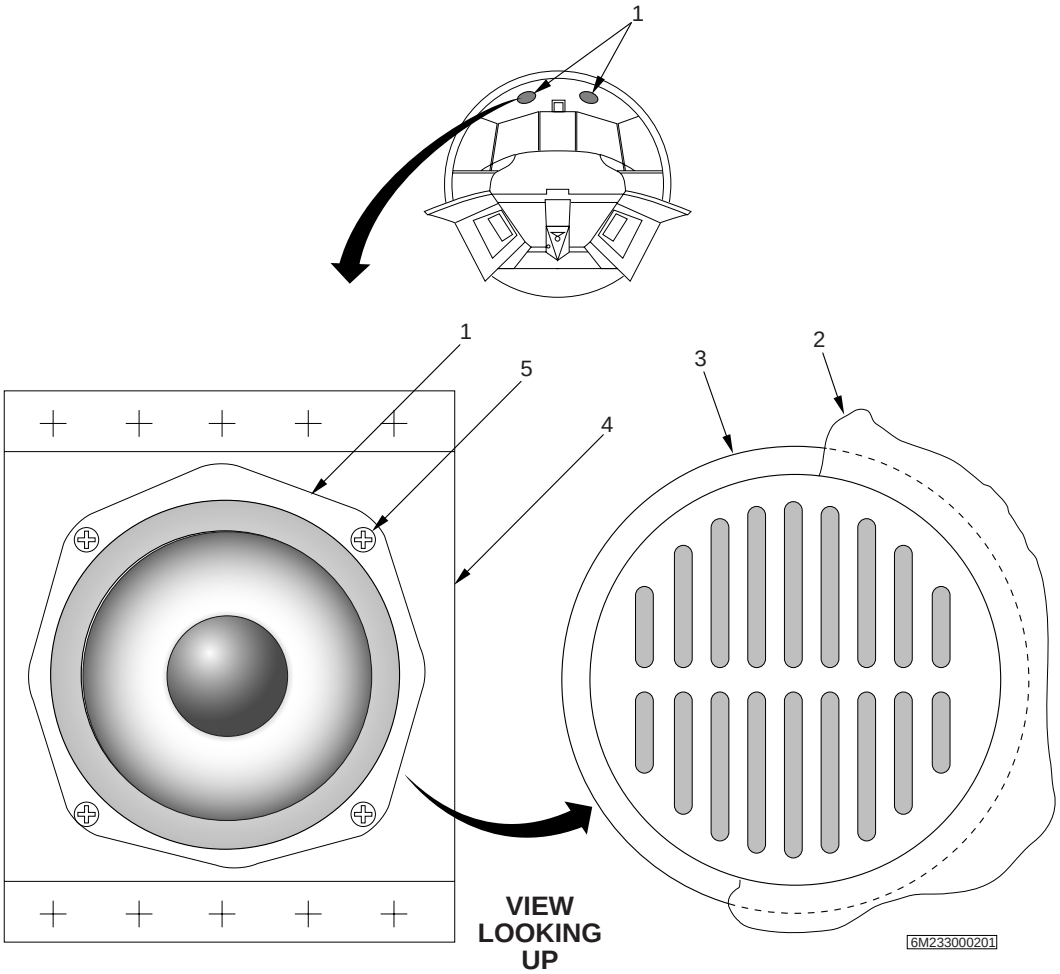
NOTE: Figure 202 depicts a typical removal/installation of either a headliner panel in the cockpit or a side wall panel in the cabin applying simple leverage with two (2) identical tools to remove applicable panel.

(2) Remove cockpit speaker cover (item 3).

(3) Remove screws (item 5) attaching speaker to mounting bracket. (item 4)

(4) Disconnect speaker wiring.

NOTE: Some cockpit speaker wiring requires only a simple terminal disconnect, while other installations are hard-wired and require that the affected wiring be cut.



Cockpit Speakers
Figure 201

B. Installation – Cockpit Speaker (Figure 201)

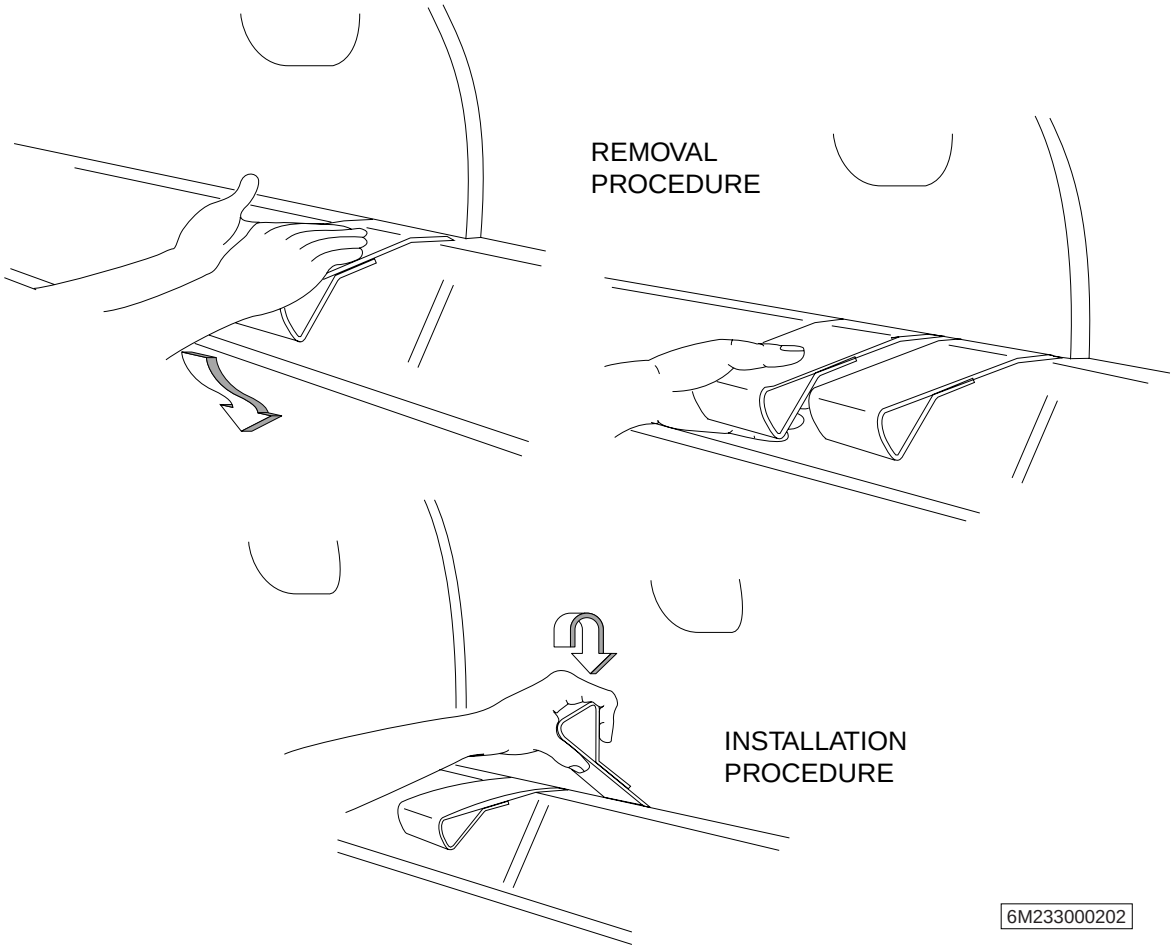
(1) Connect speaker electrical wiring.

NOTE: Some cockpit speaker wiring requires only a simple terminal connection, while other installations require appropriate splicing of affected wires. When connecting wiring, be sure to observe phase-marking on speaker.

(2) Position speaker (item 1) and secure with screws (item 5).

(3) Nest cockpit speaker cover (item 3) carefully within headliner panel cutout.

(4) Install headliner panel (item 2) securing in place behind clips, using two (2) standard panel removal/installation tools as shown in Figure 202.

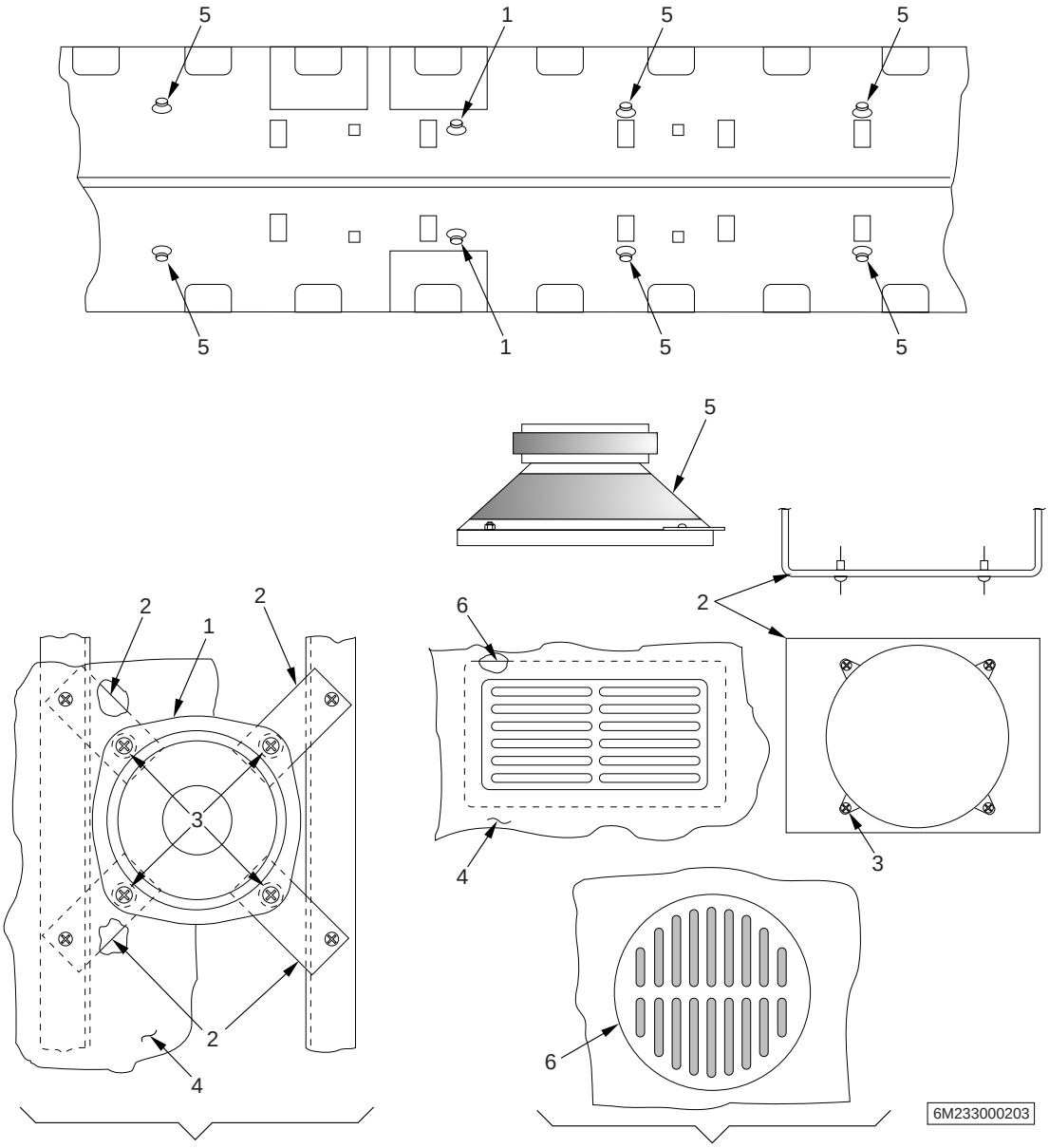


Standard panel Removal/Installation tool
Figure 202.

C. Removal – Cabin Speaker (Figure 203)

(1) Remove side wall panel assembly (item 4) using combination of gentle flexing motion and two (2) standard panel removal/installation tools. Pull panel free of channels located at top, center, and bottom.

NOTE: The use of 3 inch (7.62 cm) and 5 inch (12.70 cm) cabin speakers varies, depending on particular series of aircraft and avionics installation, but all of the cabin speakers represented are removed and installed in the same manner from one aircraft to another.



Cabin Speakers
Figure 203

(2) Lower side panel assembly and remove cabin speaker cover (item 6).

(3) Disconnect cabin speaker electrical wiring.

(4) Remove screws (item 3) securing speaker (item 1 or 5) to either clip assembly or speaker mount (item 2), as applicable.

(5) Remove cabin speaker.

D. Installation – Cabin Speaker (Figure 203)

(1) Connect cabin speaker electrical wiring.

(2) Position speaker (item 1 or 5) on either clip assembly or speaker mount (item 2) and secure with screws (item 3).

(3) For any 5 inch cabin speaker, nest cabin speaker cover (item 6) in place within panel cutout.

(4) Install side panel assembly (item 4) securing panel in place within channels provided at top, center, and bottom. using standard panel removal/installation tools (see Figure 202).

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AUDIO INTEGRATING – DESCRIPTION AND OPERATION

1. General

A. Audio output from the VHF transceivers and receivers DME, ADF, marker beacon, and passenger address (PA) system is integrated to supply the cockpit headsets and speakers. The ADF and VOR inputs have tone/voice filters. All transceiver outputs from the audio amplifier are constant level outputs. Speakers, above and beside the pilot and copilot, permit monitoring of the audio channels without the use of headphones.

B. The pilot's and copilot's audio control panels are located on the upper left and upper right portions of the instrument panels respectively. A group of toggle switches on the audio panels permit phone or speaker activation of audio signals from any of the receivers. A rotary selector switch permits selection of the transceivers, ramp hailer, or PA channels for transmission purposes. A second rotary selector switch permits selection of the ADF/VOR filter.

AUDIO INTEGRATING – MAINTENANCE PRACTICES

1. Removal/Installation – Audio Integrating

A. Removal – Audio Selector Panel

(1) Ensure battery switch is OFF.

(2) Unlock Dzus fasteners at corners of audio panel.

(3) Lift audio panel out of instrument panel.

(4) Disconnect electrical connector to audio panel.

B. Installation – Audio Selector Panel

(1) Connect electrical connector to audio panel.

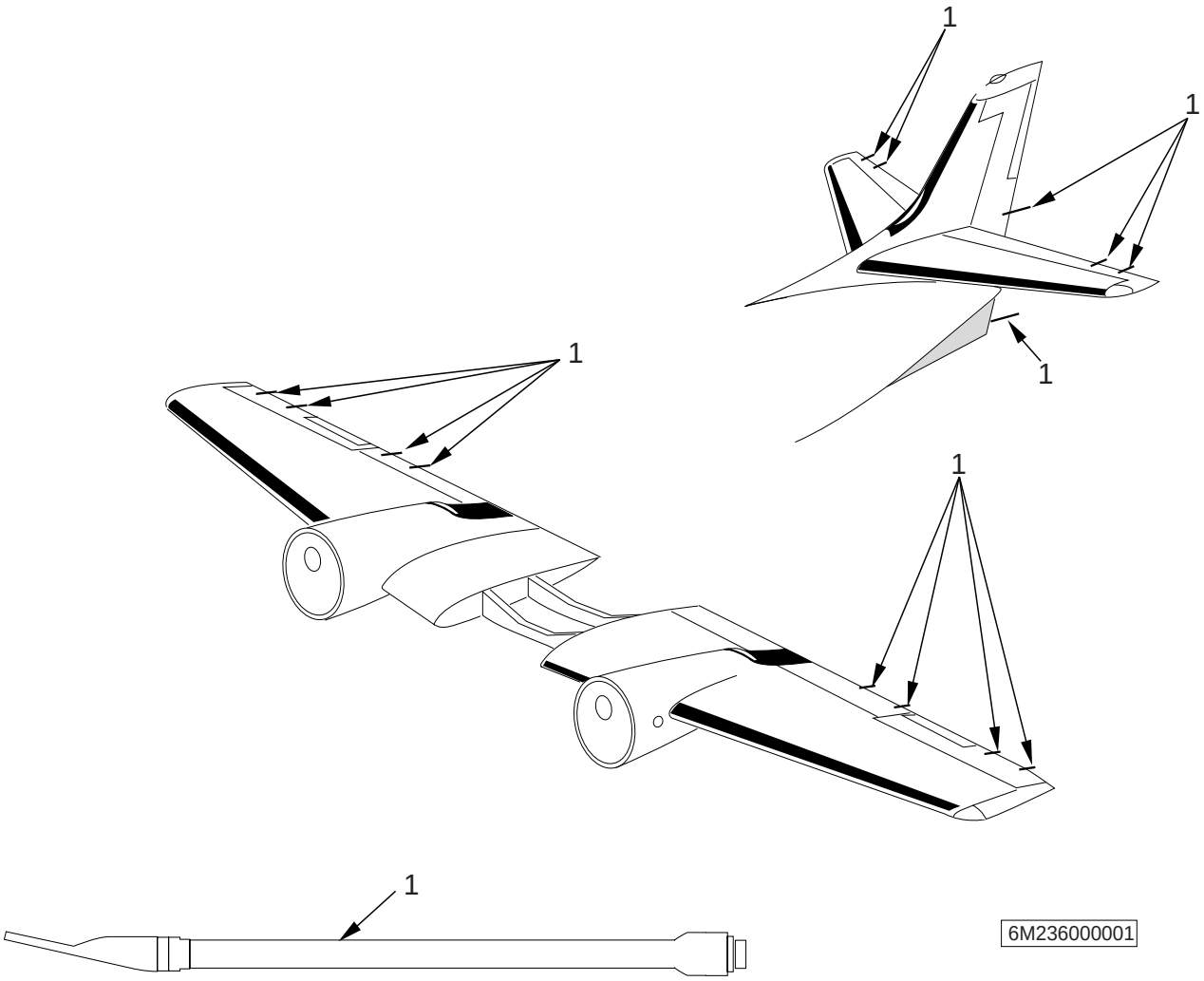
(2) Position audio panel within instrument panel.

(3) Secure audio panel to instrument panel with Dzus fasteners at corners of instrument panel.

STATIC DISCHARGING – DESCRIPTION AND OPERATION

1. General

A. Static discharging wicks (item 1) are installed on flight control surfaces to discharge static electricity. Installation locations include the aileron, wing flap, horizontal stabilizer, rudder, and ventral fin. The static wicks are either screw-mounted or riveted to the applicable control surfaces.



Static Discharging Wick Locations
Figure 1

STATIC DISCHARGING – MAINTENANCE PRACTICES

1. Removal/Installation – Static Discharging

A. Removal – Static Wick Retainer (Figure 201)

(1) Remove Static wick from retainer.

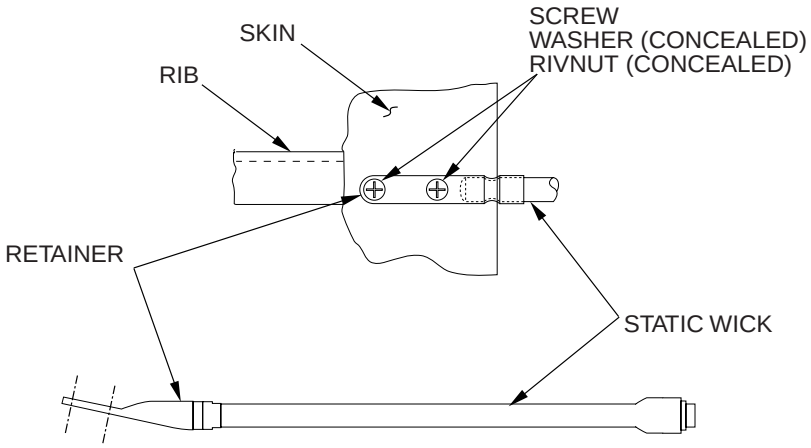
NOTE: T201-318, AT001-073, and TC201-315 use rivet-mounted retainers in place of screw-mounted retainers. Hence some differences apply to removal of retainers.

(2) For T319-999, AT074-999, TC316-999, remove screws securing retainer to airframe skin and rib. For earlier aircraft, carefully drill out rivets through center of each rivet with a #40 drill bit.

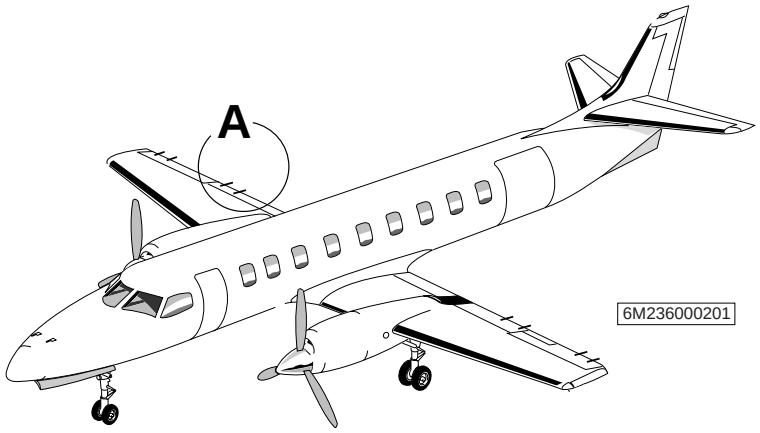
CAUTION: BE CAREFUL NOT TO GOUGE OR TEAR SKIN WHEN USING A PUTTY KNIFE.

(3) If retainers cannot be removed from the control surface, carefully pry retainer from control surface with a putty knife.

(4) Remove static wick retainer.



A TYPICAL STATIC WICK ASSY



Static Discharging
Typical Static Wick Assembly
Figure 201

B. Installation – Static Wick Retainer (Figure 201)

NOTE: Use screws, washers, and rivnuts as replacement attaching hardware for previously riveted mounts.

(1) Install static wick retainers as follows:

(a) Using a felt tip marker, outline retainer base on applicable control surface.

(b) Mask area around of retainer base or apply a locally manufactured template around areas to be worked.

(c) Lightly sand area with 400-600 grit sandpaper to remove any residual epoxy adhesive.

NOTE: On an initial installation, it is not necessary to sand masked area to bare metal. Sand just enough to break glaze on painted surface.

(d) Position one retainer at a time upon sanded control surface and drill required number of holes with a #27 drill bit through airframe skin and rib to match holes in retainer base, as applicable.

NOTE: Number coding each retainer and surface is recommended as it will better ensure proper location of each retainer during installation.

(e) For installations lacking screw-mounted attaching hardware, secure rivnuts to inner face of rib with rivets. Ensure that screw hole of each rivnut is aligned with each screw hole.

(f) Mix a non-conductive Dayton-Granger epoxy adhesive kit or equivalent, then apply a uniform coat to sanded control surface area.

(g) Secure retainer to epoxy-coated control surface with screws, washers, and rivnuts.

(h) Wipe off excess epoxy adhesive around perimeter of retainer. Allow six (6) hours for epoxy to cure.

(i) Perform electrical continuity check IAW paragraph **2.**, B., this section.

WARNING: METHYL PROPYL KETONE (MPK) IS TOXIC, FLAMMABLE, AND CAUSTIC. ISOPROPYL ALCOHOL IS TOXIC AND FLAMMABLE. AVOID OPEN FLAMES AND SPARKS. USE ONLY IN A WELL-VENTILATED AREA. AVOID EYE AND SKIN CONTACT OR BREATHING OF VAPORS. PERSONAL PROTECTIVE EQUIPMENT, CONSISTING OF RESPIRATOR, INDUSTRIAL GOGGLES, AND RUBBER GLOVES IS REQUIRED.

(j) Clean perimeter of retainer base with MPK or isopropyl alcohol.

(k) Fillet seal perimeter of retainer base with bead of Dow Corning RTV 738 sealant to provide a moisture barrier.

(l) Shank seal voids surrounding screw heads with Dow Corning RTV 738 sealant.

2. Inspection/Check – Static Discharging

A. Inspection – Static Wicks (Figure 201)

(1) Inspect static wicks for security and condition.

(2) Ensure outboard wick on elevator is installed.

(3) A maximum of one (1) wick per cluster (except in case of outboard wick) may be missing without hindering flight operations.

B. Inspection – Static Wick Retainers

(1) Using a multimeter or wheatstone bridge, perform a DC resistance check to ensure proper bonding of retainer to control surface. Verify resistance is 0.1 ohm or less on a new installation. Resistance should not exceed 0.5 ohm under any circumstance for installations occurring after initial delivery.

(2) If measured resistance exceeds 0.5 ohm, remove retainer and repeat para **1**. B., steps (c) and (e) through (i), as applicable, until measured resistance is below 0.5 ohm.

NOTE: In instances where measured resistance exceeds 0.5 ohm, control surface skin burning that results from a lightning strike is the most probable cause for the excessive resistance.

(3) Inspect perimeter of retainers and screw heads to verify positive sealing. Repair all sealing voids to ensure integrity of required moisture barrier.

AUDIO AND VIDEO MONITORING – DESCRIPTION AND OPERATION

1. General

A. The cockpit voice recorder system consists of a cockpit voice recorder (CVR) and control unit. This system provides multiple channels for recording transmitted or voice signals received that originate at the pilot's station, copilot's station, passenger address system or a third crewmember's station, in the cockpit and cabin areas, as applicable.

B. The cockpit area microphone is located to pick up and record cockpit voice signals while suppressing turbine noises. Turbine noises are electronically suppressed by the recorder system.

C. The CVR is secured in a vibration mount attached to brackets on the left side of the aft electronics compartment.

D. The control unit, containing the area microphone, preamplifier, and erase and test switches, is located away from the recorder unit in the instrument panel with the specific location varying between avionics installations.

E. The CVR is installed in T, AT, and TC aircraft, with TC aircraft being the predominate recipient of this equipment item. For T201-215, T(B)276, 292-417, AT001-074, and TC201-999, the CVR is installed on an optional basis. The CVR is incorporated on all subsequent T, AT, and TC aircraft certified to transport nine or more passengers and otherwise required to comply with FAR 91.609 and 135.151, as specified in Series Service Bulletin, 226-23-001, COCKPIT VOICE RECORDER INSTALLATION, dated January 9,1992.

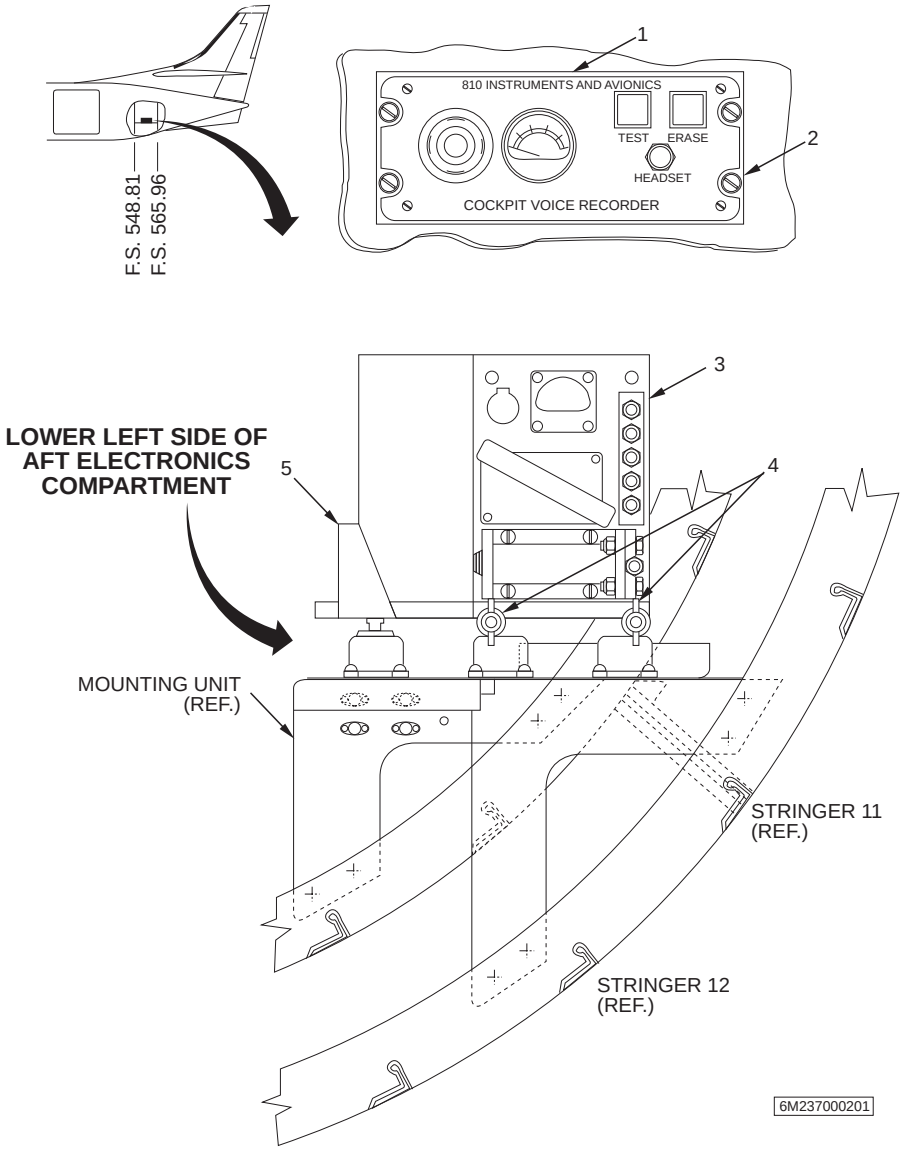
1. Removal/Installation – Audio and Video Monitoring

A. Removal – Cockpit Voice Recorder (CVR) (see Figure 201)

(1) Ensure electrical power is OFF.

(2) Loosen knurled nuts (item 4) on CVR vibration mount (item 5). Swing each knurled nut down.

(3) Slide CVR (item 3) FWD and lift out of vibration mount.



Cockpit Voice Recorder
Figure 201

B. Installation – Cockpit Voice Recorder (CVR) (see Figure 201)

(1) Ensure electrical power is OFF.

(2) Carefully align CVR (item 3) with receptacles in CVR vibration mount (item 5) and slide aft until fully seated.

(3) Swing each knurled nut into position and tighten.

C. Removal – Control unit (see Figure 201)

(1) Ensure electrical power is OFF.

(2) Unlock Dzus fasteners (item 2) located near corners of control unit mounted on instrument panel.

(3) Slide control unit (item 1) aft.

(4) Disconnect electrical connector.

D. Installation – Control unit (see Figure 201)

(1) Ensure electrical power is OFF.

(2) Connect electrical connector.

(3) Slide control unit (item 1) into instrument panel opening until face of control unit is flush with face of instrument panel.

(4) Secure control unit with Dzus fasteners (item 2) to panel at corner locations on face of control unit.

E. Removal – "G" Switch (CVR)

(1) Ensure electrical power is OFF.

(2) Remove center aisle floorboard in cabin over center wing area as necessary to gain access to G-Switch at approximately F.S. 254-272 (TC and AT), or F.S. 151-169 (T and T(B)).

(3) Remove screws with washers securing G-Switch to mount bracket. Retain hardware for reinstallation.

(4) Remove protective boot or sealant, as applicable, from G-Switch electrical terminal posts.

(5) Remove nuts and washers securing wiring to terminal posts on G-Switch and remove switch from aircraft.

F. Installation – "G" Switch (CVR)

(1) Install wiring terminals onto G-Switch terminal posts IAW Wiring Manual, Section 23-70-01. Secure with washers and nuts.

(2) Secure G-Switch to mount bracket with screws and washers.

(3) Perform Functional Check IAW manufacturer's instructions.

(4) After successful verification of CVR operation, apply sealant (Dow Corning 3145 RTV Adhesive, Clear or MIL-A-46146 equivalent) or install protective boot onto G-Switch electrical terminal posts.

(5) Install center aisle floorboard.

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24-35-10	MP	204	DEC 01/01
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ELECTRICAL POWER COMPONENTS – DESCRIPTION AND OPERATION

1. General

A. SA226 series aircraft electrical power system provides 24VDC, 28.5VDC, 26VAC and 115VAC in different output combinations for the airplane electrical requirements. Monitoring and warning devices are provided to inform the crew of each AC or DC operating status.

2. AC Generation

A. AC generation is supported by the AC voltmeter, bus fail warning lights, and overload sensors to ensure essential AC monitoring, measurement, and warning control capabilities during flight operations. The selection and use of this AC generation equipment varies from one series aircraft to another.

B. AC generation installations are provided in three basic forms, consisting of dual 3-phase alternators, dual static inverters, and dual rotary inverters. Each type of AC generation equipment supports a particular group of aircraft.

C. Each AC installation provides 26VAC and 115VAC to avionics and flight instruments through the respective AC circuitry supported by the dual 3-phase alternators, dual static inverters, or dual rotary inverters.

3. DC Generation

A. Basic electrical power for the airplane is provided by 24VDC or 28.5VDC. The DC system consists of two starter-generators, associated control, and protective components, two nickel-cadmium batteries, provisions for a DC external power circuit, and selected protective and indicating components. Electrical power for engine starting is provided by the LH and RH starter-generators and nickel-cadmium batteries. These batteries are also a source of emergency electrical power should generator power not be available.

4. External Power

A. The external power circuit consists of an external power receptacle, ground power unit (GPU) control relay, a portion of the series/parallel relay, and several other supporting components. Power supplied by the external power circuit is routed directly from the series/parallel relay to the battery bus relay and associated system circuitry.

5. AC Electrical Load Distribution

A. With dual 3-phase alternators, static, or rotary inverters, the AC buses are linked together by relays, bus-tie cables and bus-tie circuit breakers (CB's). Power is supplied to the buses by either the LH or RH 3-phase alternator working in conjunction with a 115VA inverter or the No. 1 or No. 2 inverter of each static or rotary inverter system, as applicable.

6. DC Electrical Load Distribution

A. The battery bus forms the central distribution point for DC power. The DC load distribution is linked together by bus-tie switches, a 325VA current limiter, CB's, and smaller bus bars.

ELECTRICAL POWER COMPONENTS – MAINTENANCE PRACTICES

1. Removal/Installation – Electrical Power Components

A. Removal – Electrical Wiring and Connectors

(1) Remove and replace defective wiring and connectors using standard electrical maintenance practices.

B. Installation – Electrical Wiring and Components

(1) Materials.

(a) Unshielded wire conforming to MIL-W-22759/34 or equivalent.

(b) Shielded wire conforming to MIL-C-27500 or equivalent.

(c) High temperature wire conforming to MIL-W-25038 or equivalent. Use wire conforming to MIL-W-25038 when replacing fire detector wires forward of the firewall.

(d) Coaxial cable of Military RG Type or equivalent.

(e) Nylon monofilament cord IAW MIL-C-572E, Type P; waxed fiber twines IAW MIL-T-713C, Type N; waxed nylon twine IAW MIL-T-713C, Type P or equivalent.

(f) Wire clamps conforming to MS 21919G.

NOTE: Vinyl, nylon or other cable tie wrap devices designed specifically for cable supports may be used as substitutes.

(g) Rubber grommets conforming to MS 35489.

NOTE: Nylon grommets, fabricated from nylon strips, Part Numbers G51H-A and G51H-B, may be used as substitutes.

(h) Terminal strips conforming to and fabricated from MS 27212 material or equivalent.

(i) Insulating enamel conforming to Glyptal No. 1201.

NOTE: When thinning is required, use Glyptal No. 1500 or equivalent.

(j) Insulating varnish conforming to Glyptal No. 1286, clear, or equivalent.

NOTE: When thinning is required, use Glyptal No. 1500 or equivalent.

(k) Pressure vessel sealant for brush, fillet, and faying applications conforming to MIL-S-81733.

(l) Insulation sleeving conforming to MIL-I-7444C or equivalent.

NOTE: Fire sleeve tubing shall conform to Deutsch 148-14510S or equivalent.

(m) Terminal covers conforming to MS 25171 nipples or equivalent.

(n) Dow Corning RTV 3145 adhesive sealant or RTV 738 non-corrosive sealant.

(o) Methyl propyl ketone (MPK).

(p) Aliphatic naphtha.

(q) Electrical connector sealant (potting compound) conforming to MIL-S-8516.

(r) Bonding strap conforming to MS 25083.

(s) Silver solder conforming to QQ-B-654 or equivalent.

(2) Procedures.

(a) Refer to FAA Advisory Circular AC 43.13 -1B Chg. 1 or latest FAA approved revision for installation procedures.

2. Inspection/Check – Electrical Power Components

A. Inspection – Wiring (Figures 201 and 202)

(1) Inspect insulation for fraying, burns, chaffed areas, and general condition.

NOTE: Wire insulation may be attacked by spilled fluids or the environment. Damage to insulation may not necessitate wire replacement. The severity of the damage, should be evaluated on a case-by-case basis whereby a sensible decision is made and applied to the situation at hand.

(2) If wires are damaged, check for chafing between wires and moving parts of aircraft.

(3) Check for damage to conductor portion of wire. To determine if wire requires replacement, refer to Table 201.

(4) Inspect wires for overheating. Overheated wires will have a dull brown appearance and less flexibility than normal wire.

(5) If overheated wires are found, determine cause. Overheating may either be caused by an electrical short or by close proximity to an external heat source (bleed air, etc.).

(6) Check wires for security and proper mounting.

(a) Check clamps for security and condition.

(b) Check wire ties and plastic tie wraps for condition and security.

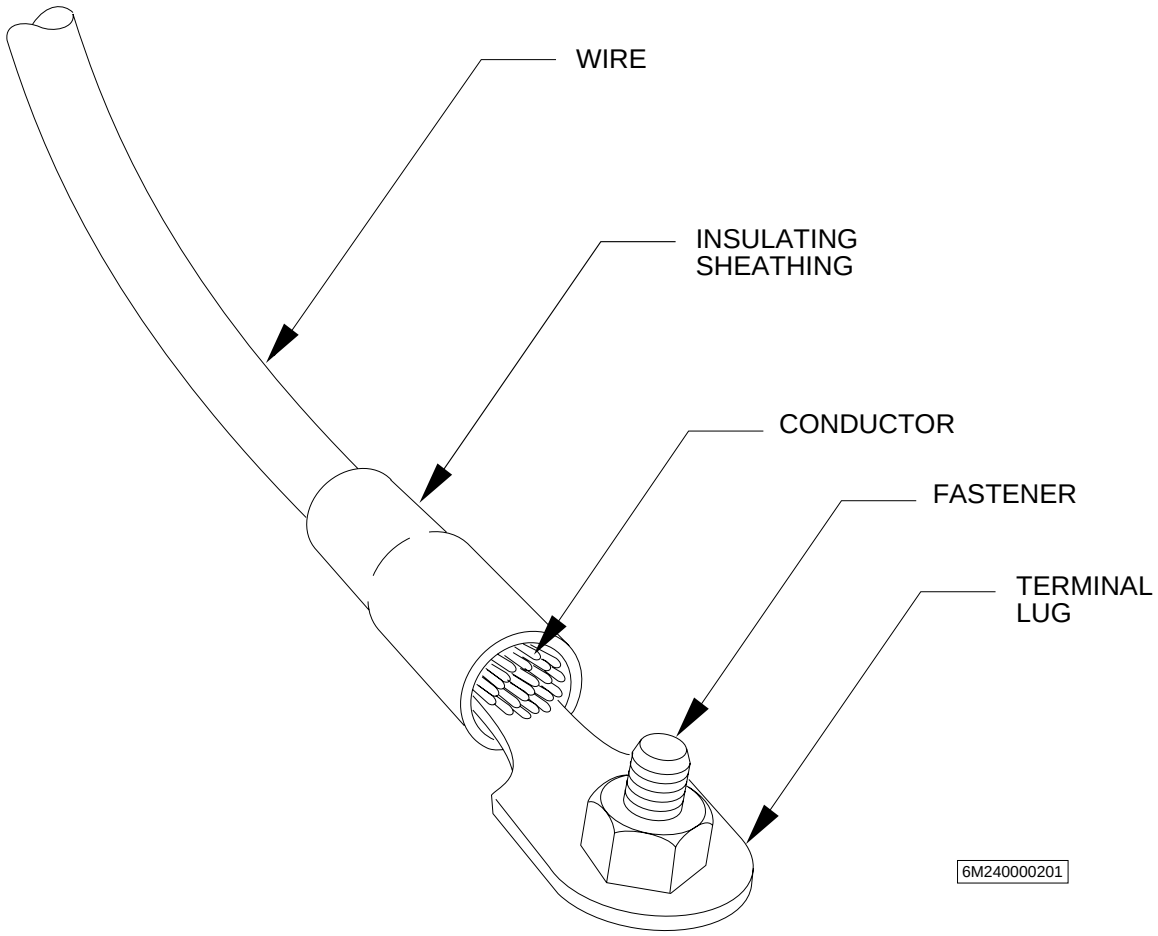
B. Inspection – Swaged Terminal (Figure 201)

(1) Inspect swaged terminal for security and condition.

NOTE: A typical swaged terminal installation is shown in Figure 201.

(2) Ensure wire conductor is just visible at the end of the terminal.

(3) Ensure the insulating sheath is intact.



Swaged Terminal Lug
Figure 201

(4) Check terminal attaching hardware for security and condition.

(5) Inspect terminal-to-wire union for security, corrosion and condition.

(6) Check upper and lower mating surfaces of terminal for corrosion.

NOTE: Corrosion on surfaces of terminal may be removed with wire brush. Corrosion in wire-attaching crimp requires terminal replacement.

C. Inspection – Removable Connectors (Figure 202)

(1) Check end bell for security, condition and corrosion.

(2) Check coupling nut for security and condition. Check for positive safety.

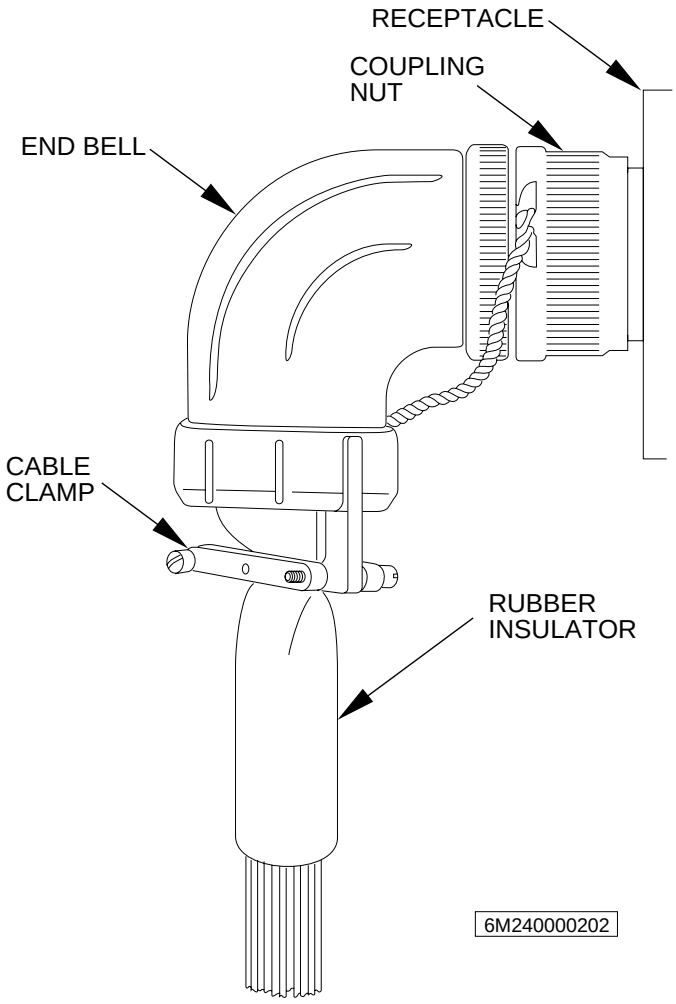
NOTE: Coupling nut should be tight enough to prevent strain on wires.

(3) Check cable clamp for security and condition, clamp screws for tightness.

(4) Check rubber insulator (if equipped) for rips, tears and general condition. Ensure insulator is secured to wire bundle with tie wrap or tie cord.

(5) Inspect coupling pins and receptacle sockets for corrosion and discoloration.

NOTE: A dark powder-appearing substance is an indicator of loose connector pins.



Removable Connector
Figure 202

3. Cleaning/Painting – Electrical Power Components

A. Cleaning – Electrical Power Components

WARNING: METHYL PROPYL KETONE (MPK) AND TRICHLORETHYLENE ARE TOXIC, FLAMMABLE AND CAUSTIC. AVOID OPEN FLAMES AND SPARKS. USE ONLY IN A WELL-VENTILATED AREA. AVOID EYE AND SKIN CONTACT OR BREATHING OF VAPORS. PERSONAL PROTECTIVE EQUIPMENT CONSISTING OF RESPIRATOR, INDUSTRIAL GOGGLES AND RUBBER GLOVES IS REQUIRED IN THE USE OF EITHER CLEANING AGENT.

(1) Use MPK to clean areas to be sealed, wires to be primed or potted.

WARNING: ALIPHATIC NAPTHA AND ISOPROPYL ALCOHOL ARE TOXIC AND FLAMMABLE. AVOID OPEN FLAMES AND SPARKS. USE ONLY IN A WELL-VENTILATED AREA. AVOID EYE AND SKIN CONTACT OR BREATHING OF VAPORS. PERSONAL PROTECTIVE EQUIPMENT CONSISTING OF RESPIRATOR, INDUSTRIAL GOGGLES AND RUBBER GLOVES IS REQUIRED IN THE USE OF EITHER CLEANING AGENT.

(2) Use aliphatic naphtha or isopropyl alcohol to clean surfaces to which insulating enamel is to be applied and to remove flux after soldering operations.

4. Approved Repairs – Electrical Power Components

A. Repair – Electrical Circuits

(1) When repairing damaged circuits, effect repairs in the following order of preference:

(a) Replace damaged wires.

(b) Repair damaged wires with permanent splices.

(c) Repair damaged wires using disconnect splices.

(2) Repair shielded wires with approved shielded wire.

(3) All replacement wire shall be of same type and same or larger gauge as wire being replaced.

(4) Frayed insulation coatings may be repaired by applying one (1) coat of varnish conforming to Glyptal No. 1286 or equivalent.

NOTE: When thinning is required, use Glyptal No. 1500 thinner or equivalent.

(5) Damage to wires not requiring replacement by cable size, considering minimum number of strands and maximum number of damaged strands is shown in Table 201.

CABLE SIZE GAUGE	MINIMUM NUMBER OF STRANDS	MAXIMUM NICKED AND/OR BROKEN STRANDS
24-22	19	0
20-12	19	1
10	41	2
8-4	133	3
26	33	12

Limits For Wires Not Requiring Replacement
Table 201

AC GENERATION – DESCRIPTION AND OPERATION

1. General

A. AC power generation is accomplished by any one of several means, each consisting of a unique mix of AC generation equipment. The different AC generation equipment includes the following:

(1) General AC generation equipment.

(2) Dual 3-phase alternators used in conjunction with single inverter.

(3) Dual static inverters.

(4) Dual rotary inverters.

B. The various AC generation systems provide 26VAC and 115VAC to avionics and flight instruments through the respective AC circuitry of the 3-phase alternators, static inverters or rotary inverters.

C. AC generation is supported by the AC voltmeter, bus fail warning lights and overload sensors to ensure essential AC monitoring, measurement and warning control capabilities during flight operations. The selection and use of this AC generation equipment varies from one group of aircraft to another.

AC VOLTMETER AND BUS FAIL WARNING – DESCRIPTION AND OPERATION

1. General

A. AC Voltmeter and Bus Fail Warning is comprised of a bus selectable voltmeter located on left forward console panel and amber bus failure warning lights on either the left forward console or annunciator panel.

B. Measurements of AC current will appear on the AC voltmeter anytime the 115VAC system is energized. A selector switch permits monitoring either AC bus.

C. Each 115VAC bus also powers a bus failure relay. When voltage is present at the bus, the relay is energized and disconnects power to the AC bus warning light. If power is lost to the bus, the relay relaxes and the warning light illuminates.

D. AC Voltmeter and Bus Fail Warning represents one of three electrical sub-systems included in the tri-unit meter assembly and Central Warning Systems presented in Indicating/Recording Systems, this manual.

2. Warning Lights

A. AC Voltmeter and Bus Fail Warning lights are used for all SA226 aircraft. Some physical design and placement differences from one group of aircraft to another do apply to the warning lights. The functional operation of all, however, is essentially the same. The typical operation of the warning lights is as follows:

(1) L AC BUS light illuminated indicates Left AC Bus is not powered. The selected inverter has failed and the other inverter (as applicable) is not powering the bus, usually indicating a defect in the bus-tie circuit.

NOTE: AC generation achieved through the use of dual 3-phase alternators requires only a single inverter in contrast to the use of two inverters used for all other AC generation systems.

(2) R AC BUS light illuminated indicates Right AC Bus is not powered. The selected inverter has failed and the other inverter is not powering the bus, usually indicating a defect in the bus-tie circuit.

(3) Both L AC BUS light and R AC BUS light illuminated indicates neither AC bus is powered, usually indicating a complete AC power source failure.

B. Installations are represented in three distinct AC generation methods:

(1) Dual 3-phase alternators (in conjunction with a single 600VA inverter).

(2) Dual static inverters consisting of the following:

(a) 250VA.

(b) 350VA.

(c) 600VA.

(3) Dual rotary inverters.

AC OVERLOAD SENSORS – DESCRIPTION AND OPERATION

1. General

A. Overload sensors in the AC system circuitry protect wires in the applicable alternator or inverter circuitry against excessive current when the AC system is energized.

B. Any given wire being protected has the overload sensor installed in series near the wire's source of power. Within the sensor, current flows through a temperature-sensitive conductor mechanically connected to a set of contacts. These contacts provide the control capability of the sensor.

C. With the sensor relaxed, the control contacts are closed and complete the circuit being controlled and complete the circuit being controlled. If the current flowing through the protected wire exceeds the current rating of the overload sensor, the temperature-sensitive conductor changes position and causes the control contacts to open. The control path is now shorted to ground, causing the control circuit breaker to trip. Simultaneously, the relay opens and the protected wire is disconnected from it's power source.

D. The overload sensor will automatically reset itself after the excessive current load has been removed and the temperature-sensitive conductor cools. The control circuit breaker, by design, must be manually reset.

E. To prevent the overload sensor from being "fooled" by momentary changes in current, a specific resistance is built into the control path. If the overload sensor trips momentarily, the current flow will be sufficient to trip the control circuit breaker.

F. Sensors are also used in the dual 3-phase alternator circuits applicable to particular SA226-T and AT aircraft. The control contacts are connected to form a series path through all three overload sensors. If any one sensor is overloaded, the series path in this AC circuitry is broken and all three phases, A, B, and C, are removed from operation.

2. Installations

A. Several different overload sensor designs, with minor differences, are incorporated into the AC circuitry serving T201-256, AT001-041, and TC201-292, utilizing both static and rotary inverters. An improved overload sensor circuitry design, built around dual 600VA static inverters, was developed, and now serves T257-999 and AT042-999.

B. Alternator overload sensors are installed in the AC circuitry serving the dual 3-phase alternators and auxiliary inverter for T201-248 and AT001-002. The 115VAC series path established in this particular circuitry provides essential safeguards for the AC boost pumps and contributes to required interactions with the AC bus relay and speed switch.

AC OVERLOAD SENSORS – FAULT ISOLATION

1. Troubleshooting – AC Overload Sensors

A. Conditions and Procedures Applicable to Overload Sensors

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
CB trips.	Control circuit is shorted to ground.	Locate and eliminate short in circuit.
	Faulty circuit breaker	Replace defective CB.
Control circuit fails to operate.	Control circuit is overloaded.	Troubleshoot wiring for shorts to ground.
	Sensor is defective.	Measure current flowing through protected wire with aMmeter of sufficient capability to make determination.

AC 3-PHASE ALTERNATORS – DESCRIPTION AND OPERATION

1. General

A. Dual 3-phase engine-driven alternators provide AC power for T201-248 and AT001-002 aircraft. A single 600VA static inverter is used during starting and ground operations and as standby power should either or both alternators fail. Automatic change-over from static inverter to the alternators is accomplished through the use of relays and the engine speed sensing switch (speed switch).

B. Each alternator is installed in tandem with a DC system starter-generator.

C. Since the alternators are engine-driven, the output frequency (Hertz, Hz) varies in direct proportion to the engine speed. The equipment used in the aircraft requires a 400 Hz output. From an operating speed of 96 to 100%, the alternator output varies in the frequency range, 390 to 410 Hz.

D. Subsequent aircraft incorporate starter-generator assemblies without alternators.

E. Engine speed less than 90%

(1) The cockpit-installed inverter switch provides a ground to close the inverter relay. If either AC bus relay is relaxed, the inverter switch provides a ground and the inverter supplies 115VAC power to the respective LH and RH buses.

(2) The 115VAC power is routed through the applicable bus tie circuit breakers and relaxed auxiliary inverter relay to each respective 115VAC bus.

(3) Some of the power supplied to each 115VAC bus is fed to an autotransformer to provide 26VAC power to particular indicating and avionics systems.

F. Conditions necessary for the static inverter to operate are:

(1) The inverter relay must have 115VAC power available.

(2) The inverter switch must be able to provide a ground for the inverter relay. This is dependent upon one or both AC bus relays being in the relaxed position.

G. Engine speed greater than 90%

(1) Alternator operation is controlled by a switch installed on the copilot's switch panel. Each switch is part of a loop comprised of the following:

(a) AC bus relay.

(b) Speed switch.

(c) Three overload sensors (see Figure 201 in AC Overload Sensors – Maintenance Practices).

(2) Beginning at the alternator, 115VAC power is routed from the C-phase terminal to the AC bus relay coil and then to the speed switch. Inside the speed switch, a set of contacts allows current flow through the switch only when the engine speed is above 90%.

(3) The three overload sensors form a series path for 115VAC current flow to the alternator switch. If any of the three legs of the alternator are overloaded, the series path will be broken and the AC bus relay will open, disconnecting the alternator. This is a safety feature since the AC boost pumps would be damaged if allowed to operate on less than three phases.

(4) To close the AC bus relay and allow the alternators to carry the AC load, the following conditions are necessary:

(a) The alternator must produce its required output.

(b) The speed switch contacts must be closed.

(c) None of the three overload sensors can be tripped.

(d) The cockpit alternator switch must be in the ON position.

H. Automatic Switching

(1) After the alternators assume the AC load, the inverter relay opens and the inverter output stops. If the engine speed slows below 90%, the speed switch contacts open, breaking the loop necessary to close the AC bus relay. When the AC bus relay relaxes, a ground is again provided for the inverter control relay and, if the inverter switch is ON, the inverter will operate and assume the AC load relinquished by the alternators.

(2) The lack of power output at the C-phase terminal will also cause the auxiliary inverter relay to relax, thus providing a path for inverter power to supply the 115VAC bus. This action is fully independent from LH to RH, and vice versa.

(3) Failure of one alternator will cause the inverter to operate and supply power to its bus. The operable alternator will also supply power to its bus. If the engine speed increases above 90%, the speed switch contacts close and the automatic switching process reverses itself with the alternators again assuming the AC load.

AC 3-PHASE ALTERNATORS – FAULT ISOLATION

1. Troubleshooting

A. Conditions and Procedures Applicable to Dual 3-Phase Alternators.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Alternator will not come on line.	Alternator has no output.	Check for output at AC bus relay, and with engine operating, check for output at all three (3) phases.
	Loop required to close bus relay is broken.	With engine operating at over 90%, position alternator switch OFF. This should indicate 115 VAC power being present at alternator switch.
No power at alternator switch with engine speed above 90% and alternator switch OFF.	Loop between AC bus relay and inverter switch open.	With engines not operating, remove speed switch connector. Check for ground through overload sensors with alternator switch ON.
	One (1) or more overload sensors tripped.	Troubleshoot wiring for shorts, or if overload sensor is found to be defective, replace overload sensor.
	Speed switch not closing.	Swap speed switches, left and right.
	Alternator switch or alternator control CB defective.	Troubleshoot switch and circuit, and repair as necessary.
Inveter and alternators supply only one (1) of two (2) required voltages.	Auto transformer defective (if 26 VAC is not being supplied).	Troubleshoot auto transformer and wiring IAW applicable wiring diagram in Chapter 24 of Wiring Manual.
	AC switching relay is defective.	Replace switching relay.

AC 3-PHASE ALTERNATORS – MAINTENANCE PRACTICES

1. Servicing – AC 3-Phase Alternators

A. All information presented here is applicable to Lear Siegler dual 3-phase alternators approved for T201-248 and AT001-002 aircraft.

2. Removal/Installation – AC-3 Phase Alternator

A. Removal – AC 3-Phase Alternator (See Figure 201)

(1) Remove inboard, outboard, and lower cowling, as applicable, and position out of way.

(2) Verify 115VAC power to starter-generator alternator assembly is OFF.

(3) Place identification tags on any unmarked wires connected to starter-generator terminal block, then disconnect all electrical wires at the terminal block.

(4) Disconnect plug (item 8) from connector (item 7) located atop alternator (item 6).

CAUTION: THE STARTER-GENERATOR ALTERNATOR ASSEMBLY MUST BE SUPPORTED AT ALL TIMES DURING REMOVAL/INSTALLATION. ANY MISALIGNMENT AND/OR BINDING OF THE DRIVE SHAFT CAN REDUCE THE EFFECTIVE SERVICE LIFE OF EQUIPMENT. DO NOT LET ASSEMBLY HANG WITHOUT THE V-BAND CLAMPS AND ATTACHING HARDWARE FULLY TIGHTENED. UNDUE BENDING LOADS EXERTED ON DRIVE SHAFT SHEAR SECTION CAN DAMAGE THE ASSEMBLY.

(5) If installed, disconnect fire extinguisher tube assembly and position out of the way.

(6) Disconnect and remove starter-generator and alternator cooling duct assemblies (items 13 and 17), associated collars (items 10 and 11), and hoses (items 15 and 18). Leave intake duct assembly (item 3) attached to aft end of alternator until starter-generator alternator assembly has been safely removed.

(7) Remove lockwire (item 22) from base of T-bolt nut (item 21) on T-bolt (item 20).

(8) Loosen T-bolt nut.

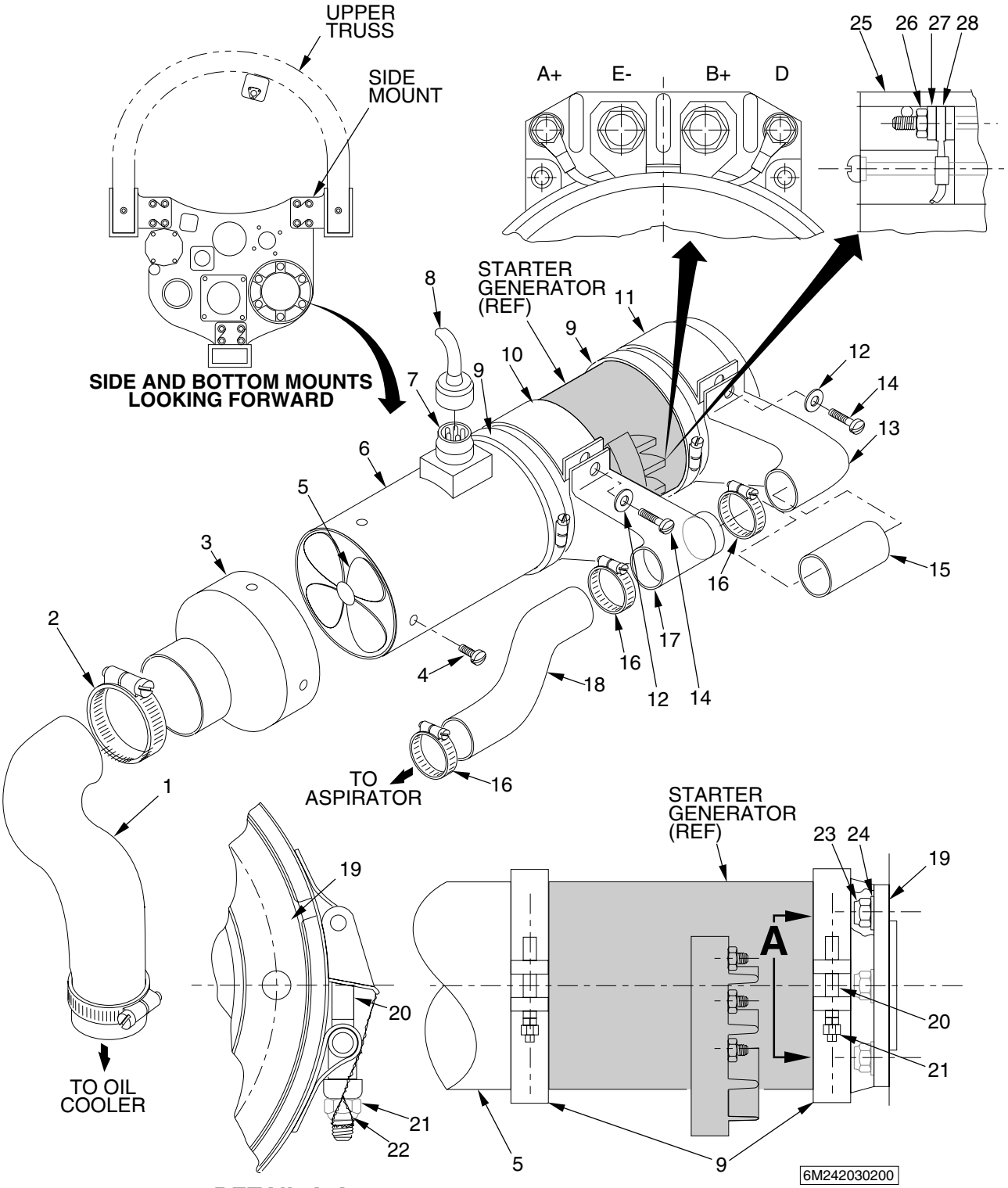
(9) Pivot T-bolt to release V-band clamp (item 9). Unlatch quick attach-detach adapter (QAD pad, item 19) from V-band clamp.

NOTE: To maintain alignment of the alternator and starter-generator, have one mechanic hold the alternator in position by keeping mounting surfaces flush with QAD pad while another mechanic loosens and removes the V-band clamp.

(10) Carefully remove starter-generator alternator assembly from QAD pad, maintaining alignment of spline to prevent binding of the drive shaft.

(11) Remove nuts (item 23) and washers (item 24) securing QAD pad to engine. Remove QAD pad.

(12) Remove intake duct assembly from alternator and attaching hose (item 1) and hose clamp (item 2).



DETAIL A-A

AC 3-Phase Alternator
Figure 201

B. Installation – AC 3-Phase Alternator (see Figure 201).

(1) Prior to installing starter-generator alternator assembly, verify/perform the following:

(a) Assigned mechanics have clear access to installation area.

(b) Entire length of drive shaft is clean and smooth.

(c) Starter-generator guide pins are both clean and visibly free of defects.

(d) Air screen provided over alternator fan (item 5) is removed (if new replacement alternator has been supplied).

(e) Mounting surfaces of starter-generator alternator (item 6) and QAD pad (item 19) forward of starter-generator are clean and free of burrs and any other degrading surface defects.

(f) QAD pin guide pin holes are clean, free of burrs, foreign objects/substances, and otherwise in a condition favorable for achieving a satisfactory installation.

(g) Position QAD pad, whereby, any one (1) of three (3) QAD pad guide pins is positively engaged at 12 o'clock position and 120° apart from each of the other two (2) guide pins. Verify that positioning is correct.

(h) Install nuts (item 23) and washers (item 24) and torque to 120 inch-pounds (13.56 N·m).

(i) Verify 115VAC power to starter-generator alternator assembly is OFF.

CAUTION: THE STARTER-GENERATOR ALTERNATOR ASSEMBLY MUST BE SUPPORTED AT ALL TIMES DURING REMOVAL/INSTALLATION. ANY MISALIGNMENT AND/OR BINDING OF THE DRIVE SHAFT CAN REDUCE THE EFFECTIVE SERVICE LIFE OF EQUIPMENT. DO NOT LET ASSEMBLY HANG WITHOUT THE V-BAND CLAMPS AND ATTACHING HARDWARE FULLY TIGHTENED. UNDUE BENDING LOADS EXERTED ON DRIVE SHAFT SHEAR SECTION CAN DAMAGE THE ASSEMBLY.

(2) Install intake duct assembly (item 3) onto aft end of alternator with screws (item 4), first removing screws from alternator, then slipping ring of intake duct assembly over end of alternator, and installing screws back into alternator.

(3) Clean and lubricate drive shaft spline (not illustrated) thoroughly before installation. Lubricate with Molykote M77, MIL-G-21164 (molybdenum disulfide) or equivalent.

(4) Engage drive shaft spline to accessory drive pad on aircraft, being certain to use two (2) mechanics to accomplish the following:

(a) Align starter-generator alternator assembly (item 6) to QAD pad, avoiding any binding of the drive shaft against the drive shaft spline.

(b) Hold starter-generator alternator assembly in position by keeping mounting surfaces flush with QAD pad, using services of first mechanic to hold the assembly in position, while the second mechanic installs and secures forward V-band clamp (item 9).

(5) Align dowel pins with applicable guide holes and press starter-generator alternator assembly onto mounting adapter.

(6) Position V-band clamp over mating flange of alternator, latch the T-bolt (item 20) and partially tighten T-bolt nut (item 21).

(7) Tap around entire circumference of V-band using a rubber mallet of suitable size to better ensure proper alignment of QAD pad with respect to the alternator starter-generator assembly.

(8) Tighten T-bolt nut to two-thirds (2/3) the recommended torque value stamped on the V-band clamp, approximately 46 inch-pounds (5.20 N·m).

(9) Tap around entire circumference of V-band clamp while tightening T-bolt nut to its recommended torque value of 70 inch-pounds (7.91 N·m).

(10) Safety wire T-bolt by securing lockwire (item 22) at base of T-bolt nut.

(11) Install starter-generator cooling duct assembly (item 13) and supporting hose (item 15), and alternator cooling duct assembly (item 17) and supporting hose (item 18) as follows:

(a) Align, mate, and secure collar (item 11) to starter-generator cooling duct assembly with screw (item 14) and washer (item 12).

(b) Engage forward end of hose (item 15) into inlet of starter-generator cooling duct assembly and surround aft end of hose with loose hose clamp (item 16).

(c) Align, mate, and partially secure collar (item 10) to alternator cooling duct assembly with screw (item 14) and washer (item 12), leaving sufficient slack to engage forward end of hose (item 15) over transition hose flange of alternator cooling duct assembly.

(d) Engage aft end of hose (item 15) over hose flange of alternator cooling duct assembly and secure with hose clamp (item 16).

(e) Engage forward end of hose (item 18) surrounded with hose clamp (item 16) over alternator cooling duct assembly hose flange, then tighten hose clamp securely.

(f) Engage aft end of hose (item 18) surrounded with hose clamp over hose flange of oil cooler aspirator (not illustrated), then tighten hose clamp securely.

CAUTION: ENSURE THAT ALL PINS OF CONNECTOR ARE STRAIGHT AND NOT DAMAGED IN ANY WAY PRIOR TO ENGAGING PLUG WITH CONNECTOR. BINDING OF CONNECTOR PINS OR BENT CONNECTOR PINS WITH POWER APPLIED CAN RESULT IN SERIOUS DAMAGE TO THE ALTERNATOR AND OTHER ELECTRICAL EQUIPMENT.

(12) Engage hose (item 1) onto hose flange of intake duct assembly and secure with hose clamp (item 2).

(13) Position and install fire extinguisher tube assembly, torque tube assembly end fitting 550 to 750 inch-pounds (62.14 to 84.74 N·m).

(14) Connect and secure starter-generator electrical wires (item 28), washers (item 27), and nuts (item 26) on terminal block (item 25).

(15) Engage plug (item 8) into connector (item 7) located atop alternator.

(16) Install inboard, outboard, and lower cowlings.

3. Inspection/Check – 3-Phase Alternators

A. Each time 3-phase alternator is removed and/or installed (in tandem with starter-generator) perform the following operational test:

(1) Place aircraft in a suitable area to operate engines, clear of personnel and equipment not contributing directly to subject inspection/check.

(2) Start engines IAW AFM.

(3) Position alternator switches, LH and RH, on copilot's switch panel ON. Check the following:

(a) Tri-unit meter (with AC voltmeter) on LH console indicates approximately 115VAC.

(b) Adjust speed lever to achieve engine speed less than 90% and make AC system checks, LH and RH, for each of the following:

1 The cockpit-installed inverter switch provides a ground to close the inverter relay.

2 115VAC power is routed through applicable bus tie circuit breaker and relaxed auxiliary inverter relay to 115VAC bus.

3 With static inverter supplying AC power, 26VAC power is available at auto-transformer, providing acceptable oil, fuel, and torque pressure indications.

(c) Adjust speed lever to achieve engine speed greater than 90% and make AC system checks, LH and RH, for each of the following:

1 115VAC power travels from C-phase to AC bus relay coil to speed switch, closing the contacts and no overload sensor tripped.

2 AC power is switched to static inverter, AC boost pump stops operating, and through automatic switching, DC boost pump begins to operate.

3 With alternator supplying AC power, 26VAC power is available at autotransformer, providing acceptable oil, fuel, and torque pressure indications.

(4) Position alternator switches, LH and RH, on copilot's switch panel OFF.

(5) Shut-down engines IAW AFM.

B. Each time 3-phase alternator is removed (in tandem with starter-generator) perform the following (see Figure 201):

(1) Inspect starter-generator alternator assembly mount splines for wear IAW Lear Siegler Maintenance Manual.

(2) Inspect plug (item 8) and connector (item 7) for condition.

(3) Inspect starter-generator and alternator cooling duct assemblies (items 13 and 17) and adjoining collars (items 10 and 11) for serviceability, security, and condition.

CAUTION: PROGRESSIVE WEAR-THROUGH OF OUTER LAYER OF SKEET HOSE TO POINT WHERE SPIRAL WIRE OF HOSE IS EXPOSED AND SUBJECT TO DIRECT CONTACT WITH "HOT" CURRENT OF EXPOSED WIRE OR TERMINAL CAN CAUSE A SHORT-CIRCUIT AND DISABLE STARTER-GENERATOR ALTERNATOR ON EITHER AN INTERMITTENT OR EXTENDED BASIS, AND OTHERWISE DEGRADE THE PERFORMANCE OF THE AC AND DC ELECTRICAL SYSTEMS.

(4) Inspect hoses (items 1, 15, and 18) and hose clamps (items 2 and 16) for condition. Inspect hose (item 15) particularly close as it is subject to abrasive wear on side where it comes in contact with terminal A+ of starter-generator terminal block.

(5) Inspect V-band clamps (item 9) for serviceability, security, and condition.

(6) Inspect electrical wires at terminals (item 28) for security and condition.

(7) Inspect intake duct assembly (item 3) for condition.

(8) Inspect alternator fan (item 5) for serviceability, security, and condition.

AC STATIC INVERTERS – DESCRIPTION AND OPERATION

1. General

A. A single 600VA static inverter is used as standard original equipment for T201-248 and AT001-002, in conjunction with the use of the dual 3-phase starter-generator alternators. The control and operation of this 600VA static inverter closely resembles that for subsequent dual 600VA inverters.

B. Three main classifications of dual static inverters evolved and are used for most T, AT and TC aircraft in the form of 250VA, 350VA, and 600VA dual static inverters. Each of these main static inverter classifications are addressed separately, identifying alignment of each classification with respect to use on particular T, AT, and TC models.

C. In any dual static inverter application, either inverter, No. 1 (LH) or No. 2 (RH), provides AC power for the aircraft, but only one is used at a time.

D. Each static inverter, for any of the volt-ampere classifications, is powered by 28VDC.

E. 600VA static inverters are installed as standard equipment for T and AT models beginning with T249 and AT003, respectively, and as optional equipment for all TC models, TC201-999.

F. 250VA static inverters are installed as standard original equipment for TC213-292 aircraft except for certain aircraft in this series, which on the basis of SB 24-014, are retrofitted with larger capacity 350VA static inverters.

G. 350VA static inverters are installed as standard retrofit equipment for TC213-292 aircraft except those aircraft equipped with optional 600VA static inverters. SB 24-014 constitutes the basis for the installation of the 350VA inverters.

2. Dual Static Inverter Control and Operation

A. Only one inverter is used at any given time. The crew selects the inverter to be used by switching the inverter control switch (INVT NO. 1/NO. 2) located on the copilot's switch panel. In the event of inverter failure (see AC Voltmeter and Bus Fail Warning, this chapter), the crew must select the other inverter.

B. Power for the No. 1 inverter comes from the 25A (NO. 1 INVERTER ESS) CB on the LH Essential Bus. The No. 1 inverter supplies power to the Left 115VAC Bus and the Left 26VAC Bus.

C. Power for the No. 2 inverter comes from the 25A (NO. 2 INVERTER ESS) CB on the RH Essential Bus. The No. 2 inverter supplies power to the Right 115VAC Bus and the Right 26VAC Bus.

D. The application of DC power to each static inverter is controlled by the inverter control relay located on the RH wheel well electrical panel on T201-248 and AT001-002 aircraft, behind the inverters in avionics rack for AT003-024, and behind the inverters on the forward pressure bulkhead for T249-999, AT025-999, and TC213-999. Each inverter control relay is powered by a 1A (INV CONT) CB on the respective essential bus.

E. When an inverter is selected, power from the inverter control CB energizes the inverter control relay, which powers each inverter and respective AC switching relay.

F. Inverter output, both 115VAC and 26VAC, is supplied to the respective bus.

G. The left and right 115VAC buses are tied together through a 2A CB (115VAC BUS TIE). The left and right 26VAC buses are tied together through a 2A CB (26VAC BUS TIE). Both Circuit breakers are located on the RH Non-Essential Bus. When either inverter is operating, power is connected to all four AC buses.

3. Inverter Location

A. During inspections, all components should be checked for security of mounting, obvious physical damage and evidence of overheating. Pertinent descriptive particulars for the static inverters are presented in Table 1, below, considering system components, effectivities, and locations.

COMPONENT	EFFECTIVITY		LOCATION
Auxiliary Inverter, 600VA	T AT	201-248 001-002	Nose compartment, LH aft floorboard.
No. 1 and No. 2 Inverters, 600VA	T	249-999	Nose compartment, upper aft area, on FWD pressure bulkhead.
	AT	003-024	Inside avionics rack, lower shelf, behind copilot's seat.
	AT	025-999	Nose compartment, upper aft area, on FWD pressure bulkhead.
No. 1 and No. 2 Inverters, 250VA	TC	213-292	Nose compartment, upper aft area, on FWD pressure bulkhead.
No. 1 and No. 2 Inverters, 350VA	TC	213-292 293-999*	Nose compartment, upper aft area, on FWD pressure bulkhead.
Inverter control relays and AC switching relays	T	249-999	Behind inverters, on FWD pressure bulkhead.
	AT	003-024	Behind inverters, inside avionics rack.
	AT	025-999	Behind inverters, on FWD pressure bulkhead.
Inverter control relays and AC switching relays	TC	213-292	Behind inverters, on FWD pressure bulkhead.
Inverter control relays and AC switching relays	TC	213-292 293-999*	Behind inverters, on FWD pressure bulkhead.

* Denotes post-SB 24-014 aircraft

Static Inverters Leading Particulars
Table 1

AC STATIC INVERTERS – FAULT ISOLATION

1. Troubleshooting – AC Static Inverters

CAUTION: DURING TROUBLESHOOTING OPERATIONS, DO NOT REMOVE THE ELECTRICAL CONNECTOR TO THE INVERTER WITH POWER APPLIED. ELECTRICAL ARCING CAN RESULT, DAMAGING THE PINS OF THE INVERTER CONNECTOR.

A. Conditions and Procedures Common to 250VA, 350VA, and 600VA Inverters.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Inverter will not operate	Inverter is not reveiving power	Check CB panel to determine if inverter CB has popped. If so, reset CB. If defect persists, disconnect inverter connector and check CB panel once again to see if CB has popped. If so, check for short circuit in inverter wiring. Refer to applicable wiring diagram in Chapter 24 of Wiring Manual.
Inverter is receiving power, but will not operate.	Inverter is not grounded.	Supply auxiliary ground to inverter.
	Inveter is defective.	Switch inverter (No. 1 with No. 2 or vice versa) and apply power. If switched inverter operates, replace defective inverter, as applicable. If switched inverter fails to operate, continue troubleshooting other areas.
Inverter is not receiving power.	Inverter relay is not closing.	Inverter switch to ON – listen for inverter relay closing.
Inverter relay is not closing.	No ground from inverter switch.	Pull AUX CONT inverter circuit breaker and check for ground at inverter relay with inverter switch ON.
	No power to inverter relay.	Reset AUX CONT circuit breaker and check for power at relay. No power is an indication of the over load sensor being tripped or defective wiring.
Inverter is operating, no power to one (1) bus, 115VAC or 26VAC, as applicable.	Inverter relay is defective. Defective bus tie circuit breaker or wiring.	Replace relay. Replace circuit breaker(s) or wiring.
	Auxiliary inverter relay open.	Troubleshoot relays and wiring.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Inverter is operating, but inverter is overheating.	Edge lighted panels in ON state excess load on 115VAC circuitry for aircraft equipped with 250VA inverters.	Incorporate provisions of SB 24-014, whereby, 250VA static inverters are replaced with higher capacity 350VA static inverters abd associated improvements.
	Short in 115VAC circuitry causes overheating from resistance build-up.	Check to ensure all wire splice and terminal connections are secure and where cracked or broken, wires are found and attaching hardware missing, replace with new.
Inverter is operating, and 26VAC circuitry is intermittent with occasional engine instrumentation measurements being noticably distorted, inconsistent, or inoperative.	Overload in 26VAC circuitry.	Troubleshoot transformer, switches, bus ties, and wiring.
	Inverter is not operating, and is experiencing cyclic fault.	Short circuit in 26VAC circuitry. Troubleshoot wiring and assocated 26VAC circuitry.
Power is present only to one bus.	Defective bus tie circuit breaker.	Replace circuit breaker(s).
	Defective wiring betweeen buses.	Troubleshoot circuitry between buses IAW Wiring Manual and replace defective wiring, as necessary.

C. Conditions and Procedures Unique to 600VA Inverter.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Inverter relay is not closing.	Overload sensor is tripped.	Check for power at relay; troubleshoot sensor and wiring IAW Wiring Manual.
Inverter No. 2 is not receiving. power.	25A CB or associated wiring is defective.	Troubleshoot IAW Wiring Manual.
	The loop formed by the inverter switch is not closing.	Jumper the "Remote Loop", externally IAW Wiring Manual.

D. Conditions and Procedures Unique to 600VA Inverter.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Inverter supplies only one (1) voltage, 115VAC or 26VAC, but lacks the other.	Ac switching relay defective.	Check for presence of voltage at inverter. If presence of only one (1) voltage is indicated, replace AC switching relay.
35A inverter PWR/CONT circuit breaker trips after long periods of operation or fuse inside inverter blows after similarly long periods of operation.	Inverter is overloaded.	Examine avionics electrical load in relation to AC capability of the aircraft.

AC STATIC INVERTERS – MAINTENANCE PRACTICES

1. Inspection/Check – AC Static Inverters

CAUTION: DURING INSPECTION/CHECK OPERATIONS, DO NOT REMOVE THE ELECTRICAL CONNECTOR TO THE INVERTER WITH POWER APPLIED. ELECTRICAL ARCING COULD RESULT, DAMAGING THE PINS OF THE INVERTER CONNECTOR.

A. Inspection/Check – AC Static Inverters

(1) Inspect/check the following AC static inverters for security of mounting, obvious physical damage, and evidence of overheating:

(a) 250VA inverters.

(b) 350VA inverters.

(c) 600VA inverters.

(d) 600VA auxiliary inverter.

(2) Inspect/check the following components directly supporting each static inverter for security of mounting, obvious physical damage, and evidence of overheating:

(a) Auxiliary inverter relays.

(b) Inverter switches.

(c) Inverter relays.

(d) Inverter overload sensors.

(e) Inverter power relays.

(f) Inverter control relays.

AC ROTARY INVERTERS – DESCRIPTION AND OPERATION

1. General

A. Two DC powered 100VA rotary inverters, No. 1 (LH) and No. 2 (RH), provide AC power for TC201-212 aircraft. Either inverter's operation is pilot selectable. In the event of an inverter failure, the pilot must select the other inverter.

B. The inverters are controlled by a switch installed on the copilot's switch panel. The No. 1 inverter (LH), is powered from the series parallel relay through a 15A circuit breaker (CB) on the LH essential bus. The No. 2 inverter (RH), is powered through a 15A CB on the RH essential bus.

C. Operation of the 100VA rotary inverters is characterized by the following:

(1) When the inverter switch is positioned to be operated by either No. 1 inverter or No. 2 inverter, power is routed to the selected inverter through the system circuitry, causing the inverter on line to operate.

(2) Power also closes the AC switching relay, enabling the switching relay to connect the 115VAC output to the AC bus. A portion of the 115VAC output is routed to an auto-transformer to provide 26VAC for the aircraft.

AC ROTARY INVERTERS – FAULT ISOLATION

1. Troubleshooting – AC Rotary Inverters

CAUTION: DURING TROUBLESHOOTING OPERATIONS, DO NOT REMOVE THE ELECTRICAL CONNECTOR TO THE INVERTER WITH POWER APPLIED. ELECTRICAL ARCING CAN RESULT, DAMAGING THE PINS OF THE INVERTER CONNECTOR.

A. Conditions and Procedures Common to 100VA Rotary Inverter

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Only one inverter will not operate.	No power supply is available.	Refer to Wiring Manual and check power to the inverter with the battery and inverter switch ON.
Power is present, but inverter does not operate.	Inverter is not grounded. Inverter is defective.	Supply AUX ground to inverter. Replace inverter.
Power is not present for No. 1 inverter.	Overload relay is not closed.	Check for power at relay.
	Overload sensor tripped.	Check wiring to sensor.
	Overload sensor defective.	Replace sensor.
	Defective inverter switch.	Replace switch.
Power is not present for No. 2 inverter.	Defective inverter switch. Defective CB.	Replace switch. Replace CB.
Inverter operates, but has no power to AC bus.	No 115VAC output from inverter.	Measure AC output at switching relay or autotransformer, and if found to be defecient, replace inverter.
	AC switching relay not closing.	Inspect wiring to relay IAW Wiring Manual.
	AC switching relay defective.	Replace relay.
Inverter supplies only one voltage, 115VAC or 26VAC, but not both.	Autotransformer defective (if 26VAC is not being supplied).	Troubleshoot autotransformer and wiring IAW Wiring Manual.
	AC switching relay defective.	Replace relay.
Power is present only to one bus.	Defective bus tie circuit	Replace breaker.
	Defective wiring between buses.	Troubleshoot IAW Wiring Manual.
Inverter circuit breaker trips after long periods of operation.	Inverter is overloaded.	Checkout and establish electrical load in relation to aircraft's AC capability.

AC ROTARY INVERTERS – MAINTENANCE PRACTICES

1. Inspection/Check – AC Rotary Inverters

CAUTION: DURING INSPECTION/CHECK OPERATIONS, DO NOT REMOVE THE ELECTRICAL CONNECTOR TO THE INVERTER WITH POWER APPLIED. ELECTRICAL ARCING CAN RESULT, DAMAGING THE PINS OF THE INVERTER CONNECTOR.

A. Inspection – AC Rotary Inverters.

(1) Inspect the 100VA rotary inverters for security of mounting, obvious physical damage, and evidence of overheating.

(2) Inspect the following components directly supporting each rotary inverter for security of mounting, obvious physical damage, and evidence of overheating:

(a) LH overload sensor.

(b) AC switching relays.

(c) Auto-transformers.

(d) AC control box.

LOCATION OF KEY AC COMPONENTS – DESCRIPTION AND OPERATION

1. General

A. The following tables are provided to aid in the location of electrical components throughout the aircraft. In some cases, customer option installations and avionics installations may have caused certain components to be relocated. Relocated components will, in general, be located near their original location.

B. For illustrated details and pertinent parts relationships of AC generation components not covered in this chapter, refer to applicable figures in Chapter 24 of the Illustrated Parts Catalog.

C. For electrical details and pertinent functional relationships of AC generation components not covered in this chapter, refer to the airplane Wiring Manual.

COMPONENT	EFFECTIVITY	LOCATION
AC warning lights	T201-248 AT001-033 (Exc AT025) TC201-212	LH side console, forward.
AC warning lights	T249-999 AT025, 034-999 TC213-999	Annunciator panel.
AC voltmeter	T201-999 AT001-999 TC201-999	LH side console, forward.
AC voltmeter selector switch	T201-999 AT001-999 TC201-999	LH side console, forward, just aft of voltmeter.
AC bus warning relays	T201-999 AT001-999 TC201-999	LH forward console, below AC voltmeter.

Key AC Components – T201-999, AT001-999, and TC201-999
Table 1

LOCATION OF KEY AC COMPONENTS – DESCRIPTION AND OPERATION

COMPONENT	EFFECTIVITY	LOCATION
Dual 3-phase alternator	T201-248 AT001-002	LH and RH engine, in tandem with starter-generator.
AC bus relay side,	T201-248 AT001-002	LH and RH engine mount, RH near alternator.
Alternator overload sensors els.	T201-248 AT001-002	Behind LH and RH ice light pan-
Speed switch	T201-248 AT001-002	Inside middle door of RH console.
Auxiliary inverter relays	T201-248 AT001-002	RH console, center, near speed switches.
Inverter switch	T201-248 AT001-002	RH switch panel.
Inverter relay	T201-248 AT001-002	RH wheel well, electrical panel.
Inverter overload sensor	T201-248 AT001-002	RH wheel well, electrical panel.
Auxiliary Inverter, 600VA	T201-248 AT001-002	Nose baggage compartment, LH aft floorboard.
Auto transformers	T201-248 AT001-002	LH console, center area.

Key AC Components – T201-248 and AT001-002
Table 2

LOCATION OF KEY AC COMPONENTS – DESCRIPTION AND OPERATION

COMPONENT	EFFECTIVITY	LOCATION
Static Inverter, 600VA	AT003-024 (Exc AT014)	Inside avionics rack, lower shelf.
Static Inverter, 600VA	AT014	LH side of nose baggage compartment, aft of baggage door.
Static Inverter, 600VA	AT025-041 and AT003E	Nose compartment, upp aft area, on FWD pressure bulkhead.
Static Inverter, 600VA	T249-256	Nose compartment, upper aft area, on FWD pressure bulkhead.
AC switching relay	AT003-024	Next to inverter, in avionics rack.
AC switching relay	T249-256 AT025-041	Behind inverters, on forward pressure bulkhead.
Inverter power relay	AT003-024	Next to inverters, in avionics rack.
Inverter power relay	T249-256 AT001-002	Behind inverters, on forward pressure bulkhead.
Inverter power relay	AT003E	Near inverter, in nose baggage compartment.
Inverter overload sensor	T249-256 AT003-041	RH wheel well electrical panel.

Key AC Components – T249-256 and AT003-041
Table 3

LOCATION OF KEY AC COMPONENTS – DESCRIPTION AND OPERATION

COMPONENT	EFFECTIVITY	LOCATION
Static Inverters, 250 VA	TC213-292 TC293-999 (Post SB 24-014)	Nose compartment, upper aft area, on pressure bulkhead.
Static Inverters, 350 VA	TC293-999	Nose compartment, upper aft area, on pressure bulkhead.
Inverter control relay	T257-999 AT042-999 TC213-999	Behind inverters, on forward pressure bulkhead.
AC switching relay	T257-999 AT042-999 TC213-999	Behind inverters, on forward pressure bulkhead.

Key AC Components– T257-999, AT042-999, and TC213-999
Table 4

LOCATION OF KEY AC COMPONENTS – DESCRIPTION AND OPERATION

COMPONENT	EFFECTIVITY	LOCATION
Rotary Inverters, 100VA	TC201-212	Under RH nose compartment floorboard.
LH overload sensor	TC201-212	RH wheel well, electrical panel.
AC switching relays	TC201-212	AC control box.
Auto-transformers	TC201-212	AC control box.
AC control box	TC201-203	Nose baggage compartment, ceiling, aft of baggage door.
AC control box	TC204-212	Nose baggage compartment, RH side ceiling, forward of baggage door.

Key AC Components – TC201-212
Table 5

DC GENERATION – DESCRIPTION AND OPERATION

1. General

A. Two engine driven starter-generators provide the aircraft's primary source of DC power.

(1) Starter-Generator.

(a) Alternator Starter-Generator.

(b) Starter-Generator.

B. Two nickel cadmium batteries are used for standby power in the event of generator failure and engine starting. Warning systems are used to monitor output and condition of various DC power sources.

(1) Batteries.

(a) Battery Control.

(b) Battery Disconnect Warning.

(c) Battery Overheat Light.

(d) Battery Fault Detection.

(e) Auxiliary Battery System.

C. Power is distributed to the various systems through the use of a bus system and associated wiring.

2. Components

A. Airframe-installed components that significantly contribute to DC generation, consist of the following:

(1) Solid-state Regulator.

(a) The solid-state regulator and protection panel provide the following functions:

1 Regulate generator output voltage.

2 Protect against generator wiring failure.

3 Overvoltage protection.

4 Reverse current protection.

5 Generator parallel load sharing.

(2) Generator Output Sense Relay.

(a) This relay is energized when the respective generator switch is ON. It is used to arm the battery fault circuit (TC models only), allow the auxiliary battery to be charged (TC models only), and act as a lockout to prevent a series battery start with a generator operating (all models).

(3) Start Disconnect Relay.

(a) The A-field connection between the voltage regulator and the A-terminal of the alternator starter generator (for early models) and starter generator (for later models) must be broken during engine starts or high voltage spikes sometimes generated during this starting phase, may flow from the generator circuit, protecting the regulator against these damaging intermittent signals from the generator during engine starts.

(4) Generator Limiting Resistor.

(a) The 2.5 ohm, 100W resistor (often referred to as the "armadillo resistor" because of its shape) limits current output of each generator during engine start. This resistor is installed in series between the voltage regulator and the A-terminal of the starter-generator on early models. On later models, with SB 24-031 having been incorporated, the generators are self-exciting from their own output and flashing the field is not required.

(5) Flashing Limiting Resistor.

(a) The 400 ohm, 10W resistor limits current supplied to the generator during reset operations. On later models, with SB 24-031 having been incorporated, the generators are self-exciting from their own output and flashing the field is not required.

(6) Generator Switches.

(a) A generator switch for each engine, installed on the pilot's switch panel, controls the connection of the generator to the DC bus.

(b) After engine start, positioning the generator switch to RESET supplies power to the generator fault system. The generator positioned to RESET also supplies power to the A-terminal of the generator to flash the field. On later models, with SB 24-031 incorporated, the generators are self-exciting from their own output and flashing the field is not required.

3. System Operation

A. During engine start, DC power is supplied to the starter portion of the starter-generator portion of the starter-generator, causing a fault condition that is sensed by the DC Feeder Fault Detection System.

B. After the engine is started, the feeder fault circuit is tripped and the generator switch positioned to RESET.

(1) When the generator switch is moved to RESET, power is also applied through a load resistor to terminal A and associated wire of the generator to flash the field. This circuitry design is effective for T201-248, AT001-024 and TC201-208.

(2) On subsequent aircraft, T249-999, AT025-999, and TC209-999 (post-SB 24-031), the field is no longer flashed when the generator is moved to RESET.

BATTERIES – DESCRIPTION AND OPERATION

1. General

A. Two nickel cadmium batteries are provided for engine starts and standby power in the event of generator failure. The batteries are securely mounted to withstand flight loads and are vented overboard to prevent fumes and liquids from accumulating within the aircraft. Several different shapes, sizes, and capacities of batteries are used. In no case should types or sizes be mixed in one installation. All information presented in this chapter is of a general nature; in all cases, FOLLOW THE BATTERY MANUFACTURER'S RECOMMENDATIONS.

BATTERIES – MAINTENANCE PRACTICES

1. Servicing – Batteries

A. Establish an area specifically for storage, maintenance and servicing of batteries.

B. Where possible, use a GPU for engine starts and during long periods of maintenance requiring electrical power.

C. Begin a program of battery maintenance, as simple or as elaborate as the operation requires. A sample form that might be used to record battery maintenance follows (see Figure 201).

D. Batteries require more attention during summer months and when aircraft are continuously operated in extremely warm environments. Generally, the higher the ambient temperatures, the more frequent batteries should be serviced, inspected and charged.

E. Battery temperature indication does not require a special hourly maintenance check, since it is required on each preflight check by the AFM. The battery overheat sensor should be checked after the battery is deep cycled and must be checked each time the battery overheat warning occurs.

BATTERY SERIAL NUMBER _____ DATE OF MANUFACTURE _____

MANUFACTURERS MODEL NUMBER _____ MANUFACTURERS PART NUMBER _____

DATE REMOVED	DATE INSTALLED	INSPECTED BY	DEEP CYCLED	NO. HOURS SINCE LAST SERVICED	AVERAGE AMBIENT TEMP THIS PERIOD	POSITION IN AIRCRAFT

6M243010201

Sample Battery Maintenance Form
Figure 201

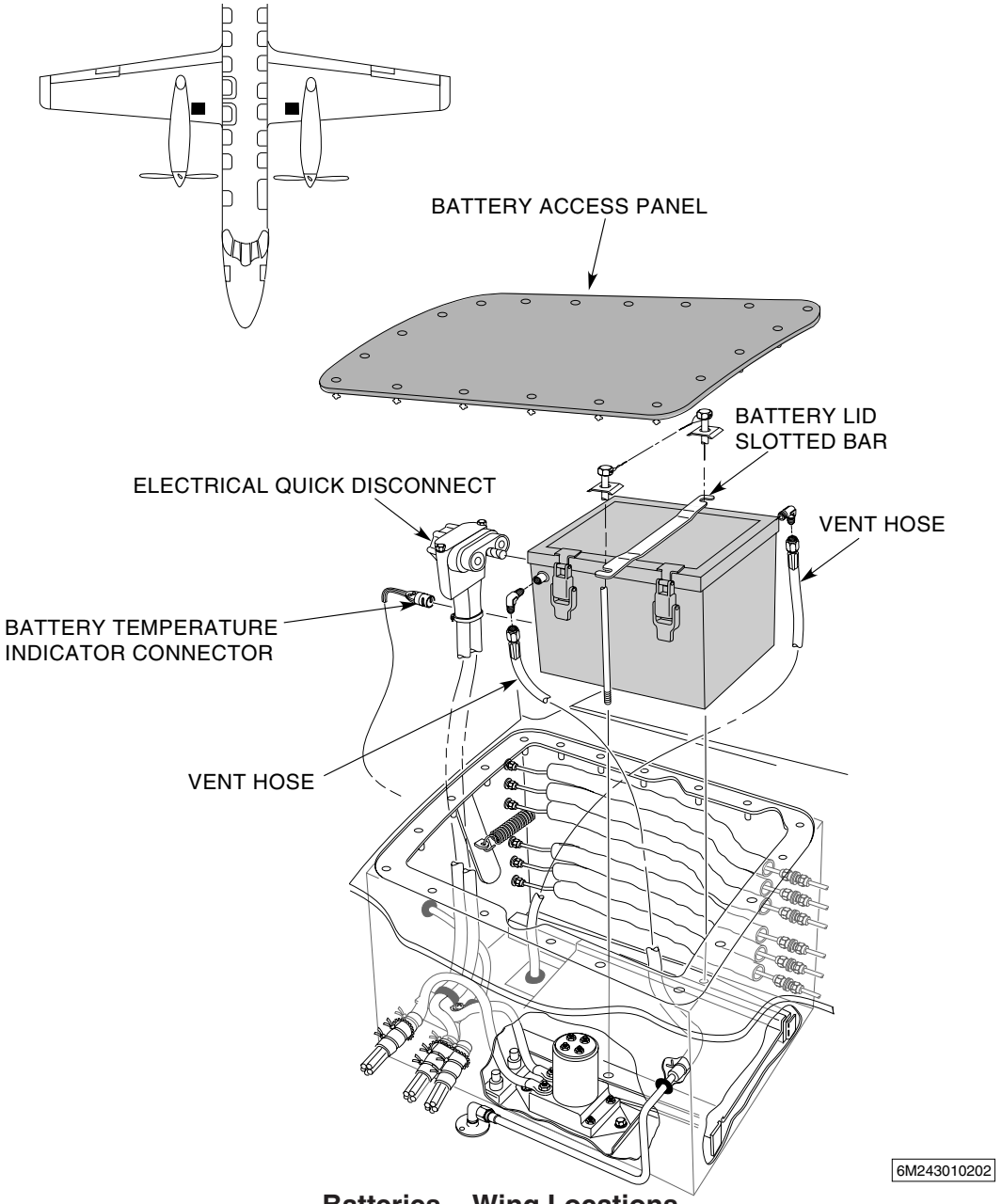
2. Removal/Installation – Batteries

A. Removal – Batteries – T201-248, AT001-074, and TC201-999 (see Figure 202)

(1) Remove battery access panel by unfastening the 1/4-turn Camloc fasteners.

(2) Disconnect battery electrical quick-disconnects.

(3) Disconnect battery temperature indicator plug located directly below battery quick-disconnect.



Batteries – Wing Locations
Figure 202

(4) Remove battery retaining bolts on AT and TC aircraft.

NOTE: The battery lid bar is slotted to allow retaining bolt removal.

(5) Remove battery from aircraft.

B. Installation – Batteries – T201-248, AT001-074, and TC201-999 (see Figure 202)

(1) Inspect battery to ensure no damage has occurred during maintenance, shipment, or handling.

CAUTION: LOOSE ELECTRICAL CONNECTIONS MAY CAUSE DAMAGE TO THE BATTERY.

(2) Check all electrical connections for tightness and security.

(3) Ensure battery well is clean and clear of obstructions.

(4) Set battery into battery well.

(5) Connect vent tube hoses. Torque hose fittings 45 inch-pounds (5.08 N·m).

(6) Connect battery temperature indicator plug.

(7) Connect battery electrical quick-disconnect.

(8) Install battery retaining bolts on AT and TC aircraft.

(9) Torque battery retaining bolts 60 inch-pounds (6.78 N·m) and safety wire.

(10) Install battery access panel and secure Camloc fasteners.

C. Removal – Batteries – T249-999 (see Figure 203)

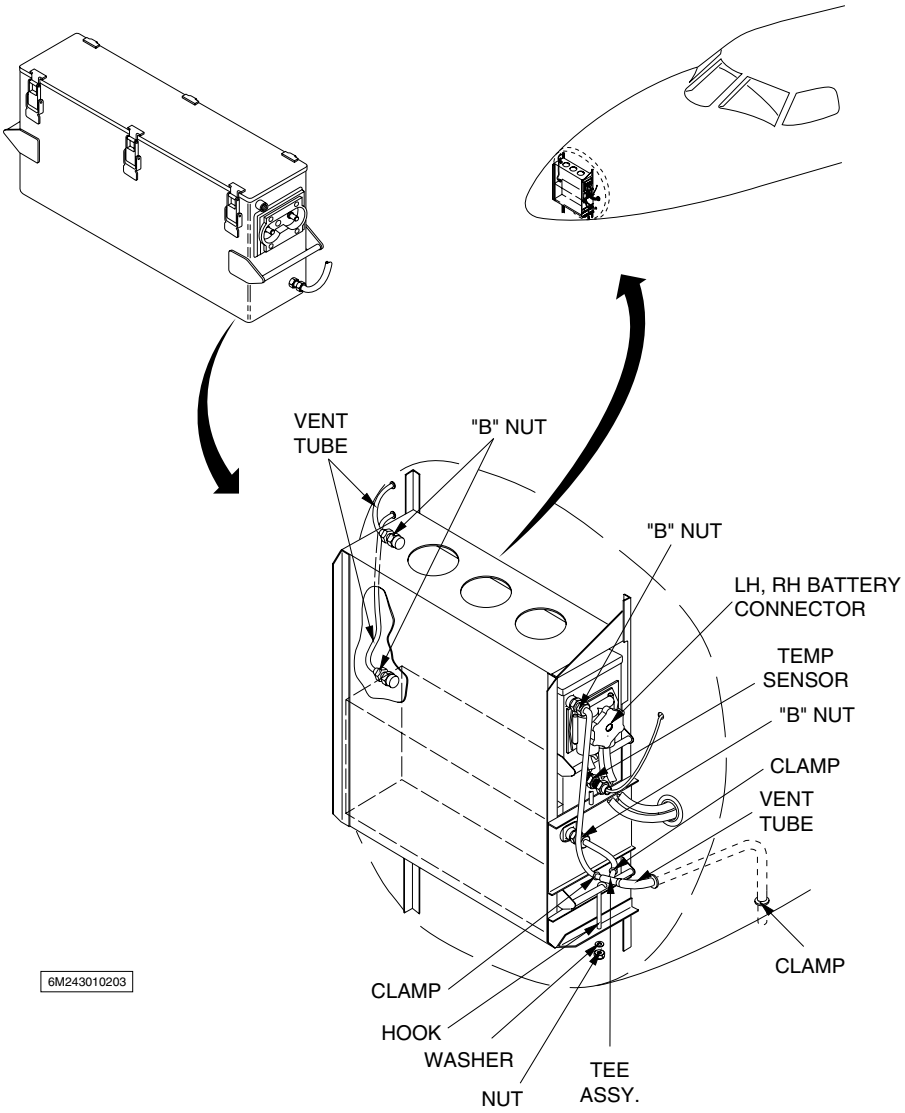
(1) Remove radome.

(2) Disconnect LH, RH battery connector.

(3) Disconnect battery temperature sensor located directly below battery connector.

(4) Remove battery retaining hooks.

(5) Remove battery.



Batteries – Radome Location
Figure 203

D. Installation – Batteries – T249-999 (see Figure 203)

(1) Inspect battery to ensure no damage has occurred during maintenance, shipment, or handling.

CAUTION: LOOSE ELECTRICAL CONNECTIONS MAY CAUSE DAMAGE TO THE BATTERY.

(2) Check all electrical connections for tightness and security.

(3) Ensure battery well is clean and clear of obstructions.

(4) Set battery into battery well.

(5) Connect vent tube hoses. Torque hose fittings 45 inch-pounds (5.08 N·m).

(6) Connect battery temperature sensor.

(7) Connect LH, RH battery connector.

(8) Install battery retaining hooks.

(9) Install radome.

3. Disassembly and Deep Cycle

A. Each time the battery is cycled it should be disassembled. The following procedure is an outline only. Follow battery manufacturer's instructions.

(1) Discharge battery using a variable resistance. Monitor individual cell voltages. When cell reaches 1.0 volt or below, use shorting clip to fully discharge each cell.

(2) Inspect top of battery, looking for electrolyte leakage and corrosion accumulation.

(3) If a cell looks suspicious, mark top center of filler cap using a marking pen. Rinse top of battery with tap water.

(4) Loosen all bolts securing battery links before removing. After all bolts are loose, remove bolts and links and place in a suitable container for cleaning.

(5) Remove cells, spacer plates and battery overheat sensor carefully; observing each cell for possible cracks, swelling or symptoms of overheating (discoloration, localized warping or distorted areas in the plastic.)

(6) Any cell having a mark on top (from step 3) should be more carefully inspected.

(7) Keeping cell caps tightly in place, wash outside of cells using tap water.

(8) Inspect battery case for cracks, condition of latches, and distortion of the case.

(9) Inspect battery lid, with special attention paid to the rubber seal in the lid and how well the lid fits the case.

(10) Install cells, spacer plates, and battery overheat sensor.

(11) Install links and bolts tightening only finger tight. When all bolts have been installed, begin tightening them slowly in a symmetric fashion and torque to manufacturer's specifications.

(12) Ensure that all cells are correctly oriented in polarity. The first cell must have positive terminal connected to positive battery terminal. From this point, cells must be connected positive/negative all through the battery so that the last cell ends with negative side connected to negative battery terminal.

(13) Charge the battery. Because of differences in manufacturer's batteries and battery chargers, no charging instructions are provided. Become familiar with the battery charger to be used and read the battery manufacturer's instructions on charging his batteries. Most manufacturers will list several different methods of charging batteries. Find one compatible with the battery charger in use and follow it.

(14) During the last portion of the charge, electrolyte may be required. Use only distilled water for this purpose and wear suitable eye and clothing protection.

(15) When charging is complete, the battery must be allowed to vent, preferably one (1) hour or more.

(16) When venting is complete, install cell caps and tighten with cap tool.

(17) Rinse top of battery with water and blow dry with compressed air.

(18) Install battery lid.

(19) Wipe battery case with a clean cloth.

(20) Mark date of servicing.

4. Inspection/Check – Batteries

A. Inspection/checks include:

(1) Inspection After Battery Overheat Warning.

(2) Battery Quick-disconnect Connectors.

(3) Battery Fault Functional Check-out Procedure.

(4) Flight Line Battery Fault Check-out Procedure.

B. Inspection/Check – Inspection After Battery Overheat Warning

(1) Determine elapsed time since overheat warning occurred.

NOTE: Since the battery must reach a temperature of 150°F (65.6°C) to activate the battery overheat light, a cool battery (shortly after the illumination of the battery overheat light) is an indication of a malfunctioning system and not a thermal runaway.

CAUTION: IF THERMAL RUNAWAY HAS OCCURRED AND ONLY A SHORT PERIOD OF TIME HAS ELAPSED, THE BATTERY WILL BE EXTREMELY HOT.

(2) If one (1) hour or less has elapsed since overheat warning light illuminated and battery is cool:

(a) Gain access to battery and perform Battery Overheat Sensor Check.

(b) With aircraft electrical power ON and battery overheat sensor disconnected, the needle of the appropriate overheat meter will travel up scale and warning and caution lights will illuminate.

(3) If one (1) hour or less has elapsed since overheat warning light illuminated and battery is hot:

(a) Remove battery from aircraft as soon as practical.

(b) Inspect battery well and exterior of aircraft for any evidence of electrolyte.

(c) Flush battery well and nearby aircraft structure thoroughly with water.

(d) If electrolyte is found, first neutralize the area with a 5% solution of boric acid and water; then flush with water.

(e) Remove battery to a safe area, preferably outside.

(f) Let the battery stand until it has reached ambient temperature.

WARNING: IF BATTERY IS DISASSEMBLED 'HOT', THERE MAY BE PRESSURE INSIDE THE CELLS AND SUITABLE PROTECTIVE CLOTHING (RUBBER APRON, GOGGLES, ETC.) MUST BE WORN.

(g) After the battery has cooled, disassemble IAW para **3**., this section.

NOTE: If badly burned cells are found, the cells nearby probably have suffered internal damage because of heat and should be replaced. Refer to the particular battery manufacturer's instructions.

(h) Deep-cycle serviceable cells before reuse.

(i) Inspect battery case for condition and permanent distortion.

(4) If one (1) hour or more has elapsed since overheat warning light occurred, the battery may have cooled or may never have been hot, proceed as follows:

(a) Perform steps listed in para **4.**, B., (2), this section.

(b) If a malfunction of the battery overheat system cannot be confirmed, assume that the battery suffered a thermal runaway and accomplish steps listed in para **4.**, B., (3), this section.

C. Inspection/Check – Battery Quick-disconnect Connectors

(1) Quick-disconnect connectors are used to connect aircraft wiring to the battery. Each time the battery is removed, the connectors should be carefully inspected for condition and security.

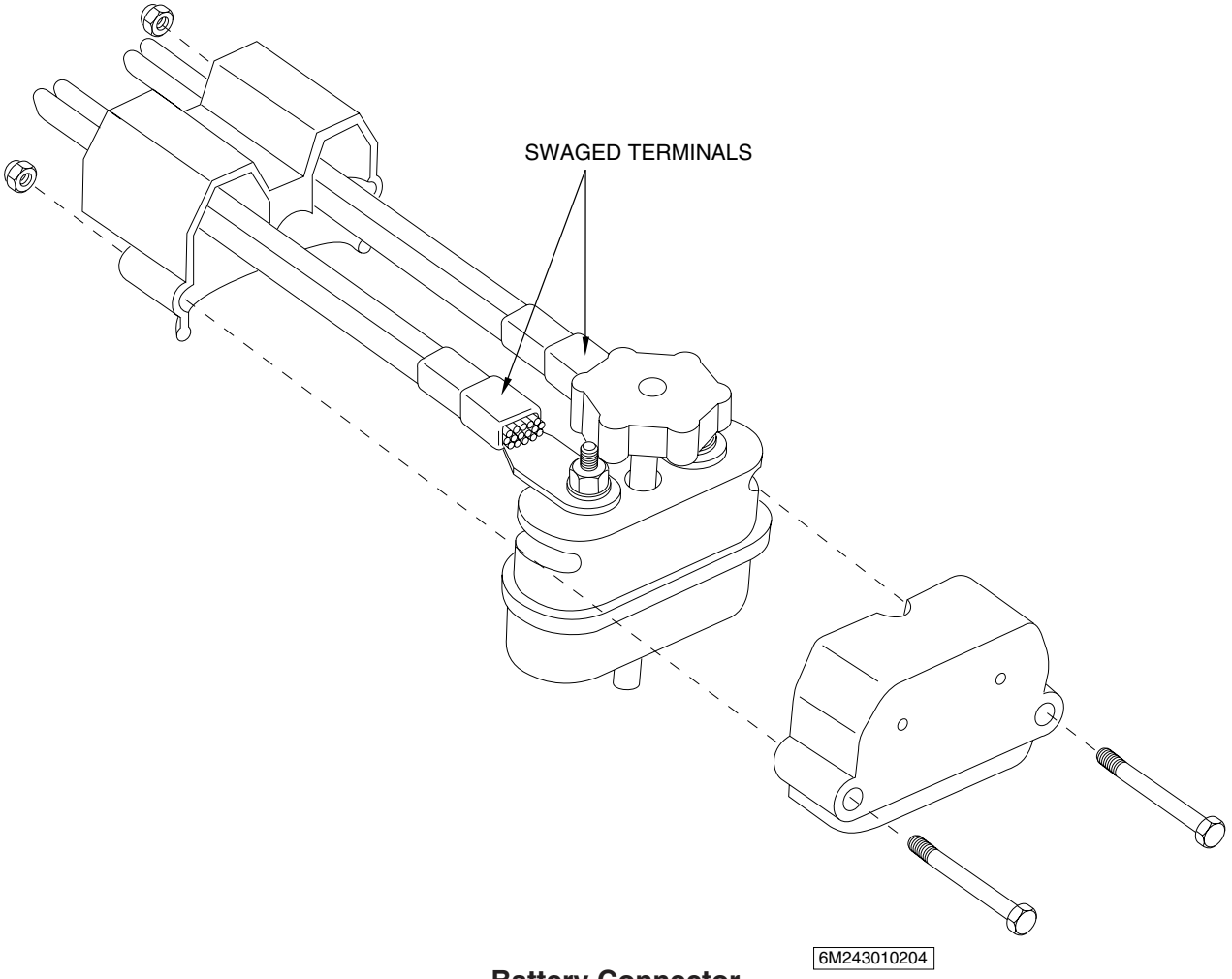
(2) Disassemble connectors.

(3) Inspect each connector half for condition.

(4) Inspect swaged terminals for condition and security.

(5) Inspect swaged terminal connector nuts for security and condition.

(6) Assemble connector halves.



Battery Connector
Figure 204

D. Inspection/Check – Battery Fault Functional Check-out Procedure

(1) Open all CB's on LH ESS circuit breaker panel except LH Start Control No. 2 and annunciator CB.

(2) Open all CB's on RH ESS circuit breaker panel except RH Start Control No. 2.

(3) Open all CB's on NON-ESS circuit breaker panel.

(4) Obtain a test lead of 20 gauge wire with alligator clips on both ends of test lead.

(5) Connect one (1) end of test lead to LH ESS bus bar input side to Start Control No. 2 CB and annunciator CB. Connect the other end of test lead to terminal No. 1 on entrance light switch.

(6) With RH and LH battery connected to aircraft ensure entrance light CB is closed.

(7) Switch both LH and RH generator switches and both LH and RH battery switches ON.

(8) Verify applicable annunciator lamps are ON and ensure battery fault, LH battery disconnect, and RH battery disconnect lamps in annunciator panel are OFF.

(9) Cycle LH battery switch OFF and verify LH battery disconnect lamp in annunciator panel illuminates.

(10) Cycle LH battery switch ON and verify LH battery disconnect lamp in annunciator panel extinguishes.

(11) Repeat steps (9) and (10) for RH battery.

(12) Obtain a D-cell, 1.5 volt DC battery, or equivalent DC voltage source.

(13) Momentarily connect the D-cell battery across terminals X1 and X2 of LH battery fault transformer located in J-Box, aft of pilot's seat.

(14) Verify BATTERY FAULT lamp and L BAT DISC lamp in annunciator panel illuminates and remains illuminated with D-cell battery removed from terminals X1 and X2 of fault transformer.

(15) Turn LH battery switch OFF; then RESET; then back ON. Verify battery fault lamp and LH battery disconnect lamp in annunciator panel extinguishes.

(16) If any one (1) of the conditions are not met in steps (9) through (15), identify the discrepancy and repair. Repeat steps (9) through (11). Steps (1) through (8) may require a repeated check-out.

(17 Repeat steps (12) through (15) for RH battery.

(18) Remove jumper installed in step (4) and return aircraft to service.

E. Flight Line Battery Fault Check-out Procedure

(1) Close all CB's in LH ESS, RH ESS and NON ESS circuit breaker panels.

(2) Start one (1) engine and bring generator on line, with both batteries installed.

(3) Position both battery switches and both generator switches ON.

(4) Repeat steps (9) through (18) of para **4.**, D, this section.

BATTERY CONTROL AND FAULT DETECTION – DESCRIPTION AND OPERATION

1. General

NOTE: Because of an A.D. note requirement, most early aircraft have been retrofitted with some type of dual battery switch arrangement. If the aircraft records indicate compliance with Service Bulletin 24-003 (formerly numbered 24-30-2034, -3038, -4038), refer to Battery Control and Fault Detection – Maintenance Practices, this chapter. If the aircraft has some other means of disconnecting the batteries, refer to the information provided by the disconnect system manufacturer.

A. A battery master switch, located on the pilot's switch panel, controls the connection of the batteries to the bus. Each battery's output is connected to a battery relay. Each battery relay connects to the applicable battery bus relay and the battery bus relay connects to the battery bus. The battery switch provides a ground, through the battery fault box for the battery relays and battery bus relay. Inside the battery fault box, the current path is made through pin N, and the battery switch to ground. To place the batteries on the line, the following conditions must exist:

(1) Each battery must have sufficient voltage to close its battery relay.

(2) The battery switch must be ON.

(3) The control contacts within the fault box must be closed.

2. System Operation

A. After the aircraft engines are operating and each generator has an output, the two generator output sense relays will close and the battery fault system becomes active. If a battery fault is sensed, the control contacts within the fault panel will open, breaking the path necessary to provide a ground for the battery system relays. When this occurs, the following operations will occur:

(1) Each battery relay will open, disconnecting the battery.

(2) The battery bus relay will open disconnecting the battery wiring from the bus. The red battery fault light, installed on the annunciator panel, illuminates when this condition occurs.

(3) If the battery switch is turned OFF, the series path through the generator sense relays and the battery switch is broken. When the battery switch is turned ON, and both generators are operating, the fault system again becomes active.

B. Each battery switch has a spring-loaded third position, RESET. When either battery switch is moved to RESET, power is applied to a ground fault sensing reset relay. The normally-closed contacts of this relay and the two generator sense relays form a series loop which arms the battery fault system. When either battery switch is moved to RESET, the following occur:

(1) The reset relay breaks the loop and disarms the battery fault system.

(2) When the RESET position is released (both generators operating), the battery fault system again becomes active.

BATTERY CONTROL AND FAULT DETECTION – FAULT ISOLATION

1. Troubleshooting – Battery Control and Fault Detection

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
One (1) battery will not come on the line.	Battery has no output.	
	Battery relay is not receiving a ground.	Check for ground with battery switch in the ON position.
	Battery relay is defective.	Jumper power to relay, move battery switch to ON and listen for relay actuation.
Neither battery will-come on the line.	Battery fault box is not receiving a ground for circuit.	Remove fault box connector and jumper pin N to ground with both battery switches ON.
Both batteries come on the line with pin N jumpered to ground.	Battery fault box is not receiving a ground.	Check for grounds at pin K and pin L.
	Battery fault box is defective.	Replace box.
Neither battery will come on line with pin N jumpered to ground.	Defective wiring from relays to fault box.	Refer to applicable wiring diagram and check wiring.
	Defective battery bus relay.	Jumper power to relay with either battery switch ON and listen for relay actuation.
Battery fault occurs when equipment or systems are operating.	Spurious signal being injected into the fault transformer wiring. Check battery.	Inspect wiring to affected system and isolate from battery fault system.

BATTERY CONTROL AND FAULT DETECTION – MAINTENANCE PRACTICES

1. Inspection/Check – Battery Control and Fault Detection

A. Each time a battery is removed:

(1) Inspect the battery relay for security of mounting.

(2) Inspect the electrical connections at the battery relay and the battery quick disconnect connector for security.

(3) Inspect the mating contacts of the battery connector for corrosion.

B. Inspection – Battery Control and Fault Detection:

(1) Inspect the battery system bus wire terminations IAW DC Electrical Load Distribution – Maintenance Practices, this chapter.

BATTERY CONTROL AND FAULT DETECTION – DESCRIPTION AND OPERATION

1. General

NOTE: This section also applies to those aircraft retrofitted by SB 24-003 (formerly 24-30-2034, -3038, and -4038).

A. Two battery switches, located on the pilot's switch panel, control the connection of the batteries to the bus. Each battery's output is connected to a battery relay; each battery relay connects to the battery bus relay; and the battery bus relay connects to the battery bus. Each battery switch provides a ground, through the battery fault box, for the respective battery relay and the battery bus relay. Inside the battery fault box, the current path is in pin N, through the closed control contacts and out pin L to ground. To place a battery on the line, the following conditions must exist:

(1) The battery must have sufficient voltage to close the battery relay.

(2) The battery switch must be ON.

(3) The control contacts within the fault box must be closed.

2. System Operation

A. After the engines are operating and each generator has an output, the two generator output sense relays will close and the battery fault system becomes active. If a battery fault is sensed, the control contacts within the fault panel will open, breaking the path necessary to provide a ground for the battery system relays. When this occurs, each battery relay will open, disconnecting the batteries; the battery bus relay will open, disconnecting the battery wiring from the bus. A battery fault light is installed and illuminates when this condition occurs. Any time the battery relay is in the open position, the battery fault light will be illuminated (see Battery Overheat Light – Maintenance Practices, this chapter).

B. Each battery switch has a spring-loaded third position, RESET. When either battery switch is moved to RESET, power is applied to a ground fault sensing reset relay. The normally closed contacts of this relay, along with the two generator sense relays, form a series loop which arms the battery fault system. When either battery switch is moved to RESET, the reset relay breaks the loop which disarms the battery fault system. When the RESET position is released, if both generators are operating, the fault system again becomes active.

BATTERY CONTROL AND FAULT DETECTION – FAULT ISOLATION

1. Troubleshooting – Battery Control and Fault Detection

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
One (1) battery will not come on line.	Battery has no output.	Check battery.
	Battery relay is not receiving a ground.	Check for ground with battery switch on the ON position.
	Battery relay is defective.	Jumper power to relay and move battery switch to ON and listen for relay actuation.
Neither battery will come on line.	Battery fault box is not providing a ground for circuit.	Remove fault box connector and jumper pin L to ground.
Both batteries come on the line with pin L jumpered to ground.	Battery fault box is not receiving a ground.	Check for grounds at pins K and L with battery switch ON.
	Battery fault box is defective.	Replace box.
Neither battery will come on line with pin L jumpered to ground.	Defective wiring from relays to fault box.	Refer to Airplane Wiring Manual.
Battery fault occurs when equipment or systems are operated.	Defective battery bus relay.	Jumper power to relay and listen for actuation.
	Spurious signal being interjected into fault transformer wiring.	Inspect wiring to affected system and isolate from battery fault system.
Fault light will not press-to-test.	Defective bulb.	Substitute known good bulb.
Battery fault occurs and no fault light illuminates.	Defective wiring from fault box to light.	Troubleshoot and repair wiring*.
	Fault box defective.	Replace box.

* The press-to-test circuitry and light powering circuitry are incorporated into the annunciator panel effective T249-999, AT025, AT034-999, and TC213-999.

A. Replace the annunciator as an assembly when troubleshooting reveals a definite deficiency in the annunciator. The following test will determine if the problem is with the aircraft wiring or within the annunciator panel:

(1) Remove the small (37-pin) and large (50-pin) connectors from the back of the annunciator panel.

(2) With battery switches ON, verify 28VDC at pin 28 of the small connector.

NOTE: If voltage measurement is negative or the light will not press-to-test, the problem is within the annunciator panel.

BATTERY CONTROL AND FAULT DETECTION – MAINTENANCE PRACTICES

1. Inspection/Check – Battery Control and Fault Detection

NOTE: This section also applies to those aircraft retrofitted by SB 24-003 (formerly 24-30-2034, -3038, and -4038).

A. Each time a battery is removed:

(1) Inspect the battery relay for security of mounting.

(2) Inspect the electrical connections at the battery relay and the battery quick-disconnect connector for security.

(3) Inspect the mating contacts of the battery connector for corrosion.

B. Inspection – Battery Control and Fault Detection:

(1) Inspect the battery system bus wire terminations IAW DC Electrical Load Distribution System – Maintenance Practices, this chapter.

BATTERY DISCONNECT WARNING – DESCRIPTION AND OPERATION

1. General

A. A battery disconnect light, when illuminated, indicates the respective battery relay has opened and disconnected the battery from the bus system. The circuit consists of contacts installed inside the battery relay and are shorted to ground when the relay is open. The disconnect light function is turned off during the start cycle. Later aircraft were designed with the disconnect circuitry incorporated into the annunciator panel.

BATTERY DISCONNECT WARNING – FAULT ISOLATION

1. Troubleshooting – Battery Disconnect Warning

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Disconnect light will not press-to-test.	Defective bulb.	Replace bulb
Battery disconnect light inoperative.	Circuit breaker near battery.	Reset or replace circuit
	Start disconnect system keeping system disconnected.	Check other systems for proper operation of engine start sequence 55% function (battery overheat).

Original battery disconnect warning system circuitry, located in the LH console on inside of aft access door, was effective for T201-248, AT001-033 (Exc AT025), and TC201-212. IAW provisions of SB 24-003.

SB 24-008 subsequently provided a panel-mounted voltmeter an selector switch to improve the voltage monitoring capability of DC voltage sources for AT026-999 and TC209-999 as standard equipment, and as optional equipment for T201-248, AT001-024, and TC201-24-08. Later models reflect additional changes in the battery disconnect warning circuitry, including a modified press-to-test capability and relocation of the system circuitry and associated. indicators to the annunciator panel

The battery disconnect warning circuitry and indicators incorporated within the annunciator panel are effective with the following:

T249-999

AT025, AT034-999

TC213-999

A. When the battery disconnect warning system, is known to be defective, replace the annunciator as a complete assembly. The following test will determine if the problem is with the aircraft wiring or within the annunciator panel:

(1) Remove the two (2) connectors, small and large, from the back of the annunciator panel. The small and large connectors contain 37 and 50 pins, respectively.

(2) Using a multimeter at pins 1 and 35 of the small (37 pin) connector, check for 28VDC with the battery switch ON.

(3) At pin 26 (for the LH battery) or pin 27 (for the RH battery) of the larger (50 pin) connector, check for zero ohms to ground with the battery switch OFF.

(4) At pin 26 of the small (37 pin) connector, perform check to ensure 28VDC is not applied.

BATTERY OVERHEAT LIGHT – DESCRIPTION AND OPERATION

1. General

NOTE: This section also applies to all aircraft retrofitted by SB 24-003 (formerly 24-30-2034, -3038, and -4038).

A. A battery overheat light monitors the temperature of the battery as an indication of the battery's charging rate. With the battery operating in a series mode, the following conditions exist:

(1) The overheat light is disconnected during start.

(2) Due to the series start capability, the right battery overheat light illuminates each time the batteries goes into series.

(3) A press-to-test function checks the lamp driver circuitry.

(4) The overheat sensor and wiring are under continuous test during normal operation.

B. The overheat circuitry for T201-248, AT001-033 (Exc AT025), and TC201-212 is installed on a separate driver board assembly. Later aircraft were designed with the circuitry incorporated into the annunciator panel. The same principal is employed in both circuits.

(1) A driver transistor is used as a switch to turn the overheat light ON.

(2) The current enabling the driver transistor to turn ON is routed to ground through the closed overheat sensor.

(3) If the battery exceeds 150°F (65.6°C), the sensor opens and the current, no longer able to flow to ground through the sensor, causes the driver transistor to turn ON and illuminate the overheat light.

(4) During the press-to-test function, power from an independent source turns the driver transistor ON and causes the overheat light to illuminate.

C. During start, the overheat system is deactivated. A relay opens and breaks the path for current necessary to turn on the driver transistor. On later aircraft, a second transistor is used as an auxiliary means of stopping the current from reaching the driver transistor.

BATTERY OVERHEAT LIGHT – FAULT ISOLATION

1. Troubleshooting – Battery Overheat Light

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Overheat light will not press-to-test (single light only).	Defective bulb or transistor T-1.	lamp driver circuitry defective. Replace bulb.
Neither overheat light will press-to-test.	No power supply or relay stuck closed.	Check for power supply.
No overheat light when lead at battery disconnected.	If system will press-to-test, the wire from cockpit to battery is shorted to ground.	Troubleshoot or replace wire.
	If system will not press-to-test,	Troubleshoot or replace.

*The lamp driver circuitry is contained in a board installed behind the last door on the LH console effective these aircraft:

T201-248

AT001-033 (Exc AT025)

TC201-212

A. The circuitry is discrete and may be troubleshot, repaired, and components replaced in the normal manner of electrical servicing. Complete new board assemblies are available and may be exchanged for the existing unit if desired.

BATTERY OVERHEAT LIGHT – MAINTENANCE PRACTICES

1. Inspection/Check – Battery Overheat Light

NOTE: This section also applies to all aircraft retrofitted by SB 24-003 (formerly 24-30-2034, -3038, and -4038).

A. Each time battery is removed:

(1) Inspect electrical connection to overheat sensor inside battery.

(2) Inspect overheat sensor wiring inside battery compartment for evidence of electrolyte attack or physical damage.

(3) With temperature lead disconnected, turn aircraft batteries ON. Verify overheat light illuminates.

2. Adjustment/Test – Battery Overheat Light

A. Perform the following test on overheat sensor:

(1) Remove panels as necessary to gain access to batteries.

(2) Remove “handshake” connector to battery overheat sensor.

(3) Turn battery switch ON. Verify battery overheat light illuminates.

(4) Repeat same procedure for other battery.

(5) Remove both batteries.

(6) Remove battery cover. Remove bolts securing temperature sensor link to battery.

CAUTION: IF THE LINK MUST BE REMOVED WITH THE BATTERY CHARGED, INSERT PIECES OF STIFF CARDBOARD BETWEEN THE TEMPERATURE SENSOR LINK AND OTHER BATTERY LINKS.

(7) Attach a length of string to temperature sensor lead and remove sensor link assembly from battery.

(8) To test switch, the following will be required:

(a) Container of approximately one (1) pint capacity.

(b) One (1) pint of water or oil.

(c) Immersion heater or other means of applying heat to the container.

(d) Mercury-filled liquid thermometer with a range including 180°F (82.2°C), or other device for measuring temperature to an accuracy of 1°F (-17.2°C).

(e) Ohmmeter or continuity tester.

(9) Test procedure:

(a) Fill the container.

(b) Immerse temperature link assembly.

(c) Begin applying heat to container. Sensor should indicate continuity until reaching 150°F (65.6°C), ± 5°F (± [-15.0°C], [-20.6°C], respectively), then sensor should open. Remove heat and sensor closes with 5°F (-15.0°C) applied. If sensor does not open and close within the tolerances specified, replacement of sensor is required.

(10) Apply a thin coating of Dow Corning RTV 3145 over the sensor to help prevent corrosion and install in battery.

(11) Install batteries.

(12) Turn battery switches ON.

(13) Verify that battery overheat light is not illuminated.

(14) Check battery overheat light press-to-test function.

BATTERY OVERHEAT LIGHT – DESCRIPTION AND OPERATION

1. General

A. A battery overheat light monitors the temperature of the battery as an indication of the battery's charging rate. With the battery operating in a series mode, the following conditions exist:
   (1) The overheat light is disconnected during start.

   (2) Due to the series start capability, the right battery overheat light illuminates each time the batteries goes into series.

   (3) A press-to-test function checks the lamp driver circuitry.

   (4) The overheat sensor and wiring are under continuous test during normal operation.

B. The overheat circuitry for T249-999, AT025, and 034-999, and TC213-999 is installed within the annunciator. Previous aircraft were designed BATTERY OVERHEAT LIGHT – FAULT ISOLATION with the circuitry incorporated in a separate driver board. The same principal is employed in both circuits and reflects the following operating conditions:

   (1) A driver transistor is used as a switch to turn ON the overheat light.

   (2) The current enabling the driver transistor to turn ON is routed to ground through the closed overheat sensor.

   (3) If the battery exceeds 150°F (65.6°C), the sensor opens and the current, no longer able to flow to ground through the sensor, causes the driver transistor to turn ON and illuminate the overheat light.

   (4) During the press-to-test function, power from an independent source turns the driver transistor ON and causes the overheat light to illuminate.

C. During start, the overheat system is deactivated. On this and earlier model aircraft, a relay opens and breaks the path for current necessary to turn on the driver transistor. On these particular models, a second transistor is used as an auxiliary means of stopping the current from reaching the driver transistor.

BATTERY OVERHEAT LIGHT – FAULT ISOLATION

1. Troubleshooting – Battery Overheat Light

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Overheat light will not press-to-test (single light only).	Defective bulb or transistor T-1.	If system will not press-to-test lamp driver circuitry defective.
Neither overheat light will press-to-test.	No power supply or relay	Replace bulb.
No overheat light when lead at battery disconnected.	stuck closed.	
	If system will press-to-test the wire from cockpit to battery is shorted to ground.	Check for power supply.

Troubleshoot or replace wire.

Troubleshoot or replace.

*The lamp driver circuitry is contained in a board installed behind the last door on the LH console effective these aircraft:

T249-999

AT025 and AT034-999

TC213-999

A. When the battery overheat circuitry is known to be defective, replace the annunciator as a complete assembly. The following test will determine if the problem is with the aircraft wiring or within the annunciator panel.

(1) Remove the two (2) connectors, small and large, from the back of the annunciator panel. The small connector contains 37 pins, the large connector contains 50 pins. At pin 14 of the small connector, with battery switch ON, measure and verify 28 VDC. At pin 26 of the small connector, measure and verify 0 volts.

WARNING: CLEAR PROPELLERS PRIOR TO ENGINE START.

(2) Depress the start button and check that voltage at pin 26 of small connector indicates 28 VDC. Do not hold the start button down for longer than 1 or 2 seconds; in no case allow the engine to exceed 5 or 6% RPM. Check to see that when the start button is released, the voltage at pin 26 of the small connector disappears.

(3) At pins 12 and 13 of the large connector, verify there is zero (0) resistance to ground.

NOTE: If the results of all voltage measurement checks indicate positive voltages, the problem is within the annunciator panel.

BATTERY OVERHEAT LIGHT – MAINTENANCE PRACTICES

1. Adjustment/Test – Battery Overheat Light

A. Perform the following test on overheat sensor:

(1) Remove panels as necessary to gain access to batteries.

(2) Remove “handshake” connector to battery overheat sensor.

(3) Turn battery switch in the cockpit ON. Verify battery overheat light illuminates.

(4) Repeat same procedure for other battery.

(5) Remove both batteries.

(6) Remove battery cover. Remove bolts securing temperature sensor link to battery.

CAUTION: IF THE LINK MUST BE REMOVED WITH THE BATTERY CHARGED, INSERT PIECES OF STIFF CARDBOARD BETWEEN THE TEMPERATURE SENSOR LINK AND THE OTHER BATTERY LINKS.

(7) Attach a length of string to temperature sensor lead and remove temperature sensor link assembly from battery.

(8) To test switch, the following will be required:

(a) Container of approximately one (1) pint capacity.

(b) One (1) pint of water or oil.

(c) Immersion heater or other means of applying heat to container.

(d) Mercury-filled liquid thermometer with a range including 180°F (82.2°C), or other device for measuring temperature to an accuracy of 1°F (-17.2°C).

(e) Ohmmeter or continuity tester.

(9) Test procedure:

(a) Fill container.

(b) Immerse temperature link assembly.

(c) Begin applying heat to container. Sensor should indicate continuity until reaching 150°F (65.6°C), ± 5°F (± [-15.0°C], [-20.6°C], respectively), then sensor should open. Remove heat and ensure sensor closes with 5°F (-15°C) applied. If sensor fails to open and close within the tolerances specified, replacement of sensor is required.

(10) Apply a thin coating of Dow Corning RTV 3145 over the sensor.

(11) Install batteries.

(12) Turn battery switches ON.

(13) Verify battery overheat light is not illuminated.

(14) Check battery overheat light press-to-test function.

2. Inspection/Check – Battery Overheat Light

A. Each time battery is removed:

(1) Inspect electrical connection to overheat sensor inside battery.

(2) Inspect overheat sensor wiring inside battery compartment for evidence of electrolyte attack or physical damage.

(3) With temperature lead disconnected, turn aircraft batteries ON. Verify overheat light illuminates.

MAINTENANCE MANUAL

AUXILIARY BATTERY SYSTEM – DESCRIPTION AND OPERATION

1. General

A. SA226 TC201-334 are equipped with a nose mounted auxiliary battery to power the pilot's maplight, the nose baggage compartment lights, the cargo compartment lights, and the cargo door warning test circuit.

B. A circuit breaker, in the LH side of the nose wheel well, controls the output of the battery and should be pulled when the aircraft is undergoing maintenance or when the cargo door is to be open for long periods.

2. System Operation

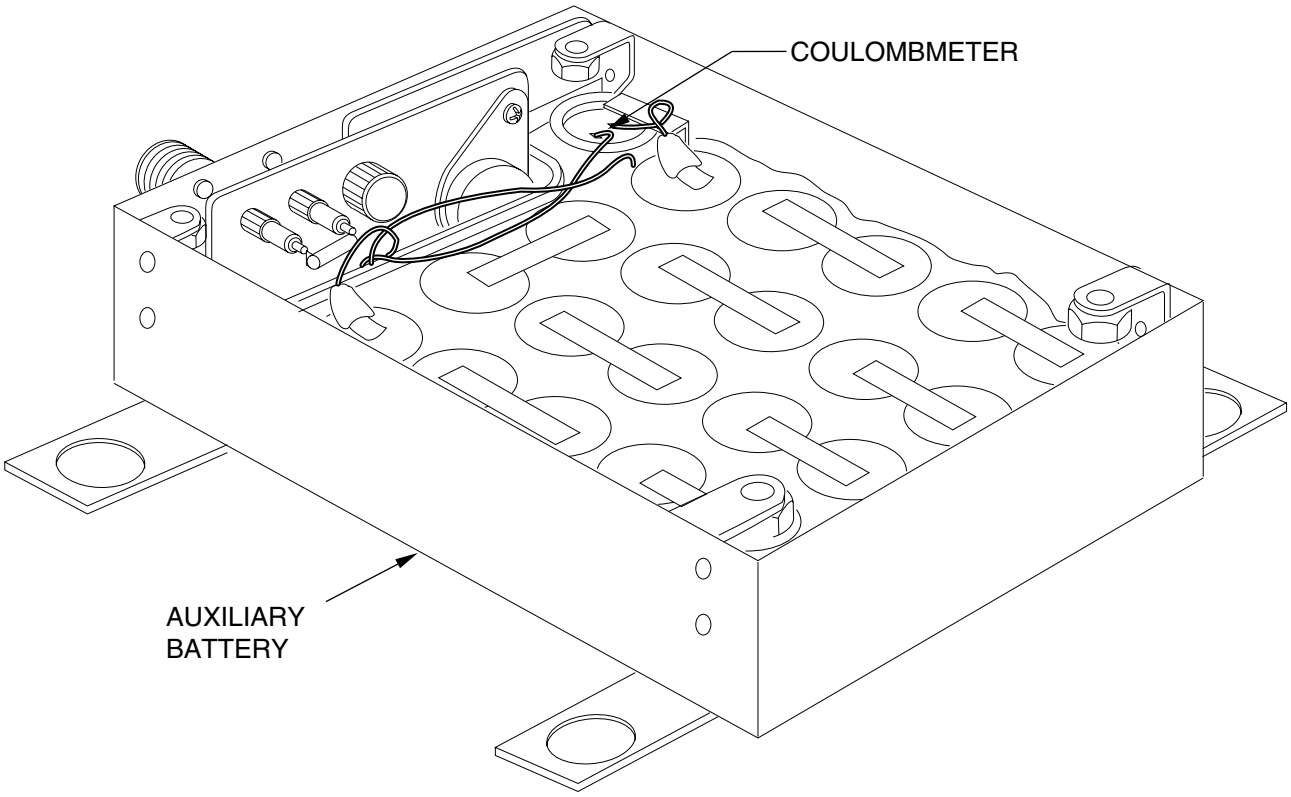
A. With the battery switch ON, a power transfer relay disconnects the auxiliary battery and uses the main batteries to power the circuits. After the engines are operating and a generator has an output, the battery charging relay will close and a circuit breaker on the non-essential bus charges the battery.

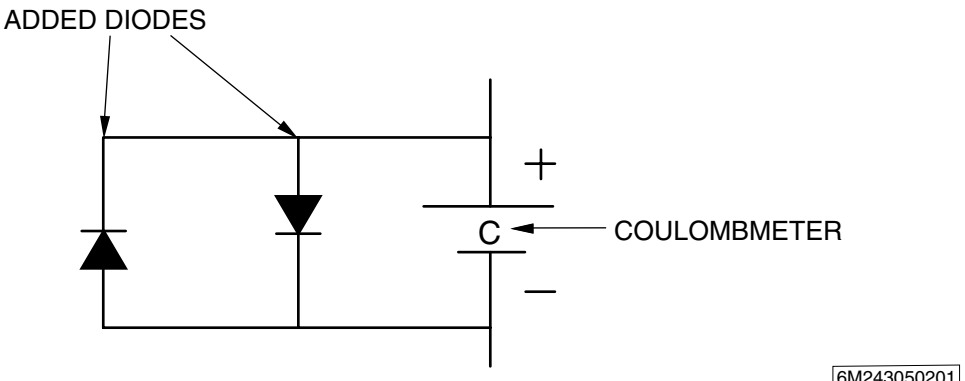
MAINTENANCE MANUAL

AUXILIARY BATTERY SYSTEM – FAULT ISOLATION

1. Troubleshooting – Auxiliary Battery System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No apparent power from auxiliary battery.	Battery is dead.	Turn on main battery and check for auxiliary battery systems power.
Battery is dead.	No charging current to battery.	With battery switch ON, generator operating, check for 28VDC at battery connector.
	Battery is defective.	Replace battery.
Battery frequently low.	No bus power transfer.	Troubleshoot bus transfer relay IAW applicable wiring diagram.





CHARGING CIRCUIT

Deep Cycling Auxiliary Battery
Figure 201

AUXILIARY BATTERY SYSTEM – MAINTENANCE PRACTICES

1. Servicing – Auxiliary Battery System

NOTE: During routine inspections, all components of the system should be inspected for security of installation and obvious physical damage.

A. Auxiliary Battery Deep Cycle (see Figure 201)

(1) Remove battery from aircraft. Remove screws securing battery cover and remove cover.

(2) Coulombmeter must be shorted across using two (2) 1N2069 diodes. The circuit for installing these diodes is shown in Figure 201.

(3) Discharge battery at 330 mA constant current, or with a 73.0 ohm fixed resistance, until total battery voltage is 20VDC.

(4) Charge battery with a constant current power supply at 80 mA for 16 hours.

(5) Two (2) minutes prior to end of charge, read and record cell and coulombmeter voltages. Cells can only be read in pairs and voltages of each pair should be 2.80 to 2.96 volts. Coulombmeter should read 0.80 volt.

(6) Discharge battery at 330 mA constant current, or with a 73.0 ohm fixed resistance, until total battery voltage is 21.0 volts. Minimum time allowed to 21.0 volts is 1.6 hours and each pair of cells must indicate 2.0 volts or above.

(7) If battery fails to meet criteria of step **1.**, A., (6), repeat steps **1.**, A., (1) through (5) at least three (3) times. If battery still fails, replace battery.

(8) If battery meets the criteria of step **1.**, A., (6), remove diodes from coulombmeter. Charge battery with a constant current power supply at 35 mA for 16 hours.

MAINTENANCE MANUAL

STARTER-GENERATOR ALTERNATOR – DESCRIPTION AND OPERATION

1. General

A. Two engine driven starter-generator alternators provide AC and DC power for the aircraft systems. This section addresses the starter-generator (DC generation) portion of this assembly.

B. Switches installed in the copilot's switch panel control each starter-generator connection to the DC bus system.

C. A solid state voltage regulator and protection panel are used to provide the following supporting functions:

(1) Voltage regulation.

(2) Protection against generator wiring failures.

(3) Overvoltage protection.

(4) Reverse current protection.

(5) Parallel sharing of the load by each starter-generator.

D. The starter-generator is installed in tandem with the alternator as the starter-generator alternator assembly. For corresponding descriptions of a stand alone starter-generator configuration reflecting a different circuitry design, providing self-exciting provisions, see Starter-Generator – Description and Operation, this chapter.

E. Starting is accomplished by separate series windings complementing those already included in the armature circuit for generator operation.

F. Cooling is accomplished by an integral fan mounted on the armature shaft. The fan also serves as a portion of the inlet duct mated to the starter-generator alternator.

STARTER-GENERATOR ALTERNATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Starter-Generator Alternator

A. Removal – Starter-Generator Alternator (see Figure 201)

(1) Remove inboard, outboard, and lower cowling, as applicable, and position out of way.

(2) Verify 115VAC power to starter-generator alternator (item 10) assembly is OFF.

(3) Place identification tags on unmarked wires connected to starter-generator terminal block (item 25), then disconnect all electrical wires at terminal block.

(4) Disconnect plug (item 7) from connector (item 6) located on alternator.

CAUTION: STARTER-GENERATOR ALTERNATOR ASSEMBLY MUST BE SUPPORTED AT ALL TIMES DURING REMOVAL/INSTALLATION. ANY MISALIGNMENT AND/OR BINDING OF THE DRIVE SHAFT CAN REDUCE THE EFFECTIVE SERVICE LIFE OF EQUIPMENT. DO NOT LET ASSEMBLY HANG WITHOUT THE V-BAND CLAMPS AND ATTACHING HARDWARE FULLY TIGHTENED. UNDUE BENDING LOADS EXERTED ON DRIVE SHAFT SHEAR SECTION CAN DAMAGE THE ASSEMBLY.

(5) Disconnect and remove starter-generator and alternator cooling duct assemblies (items 13 and 17), associated collars (items 9 and 11) and hoses (items 15 and 18). Leave intake duct assembly (item 3) attached to aft end of alternator until starter-generator alternator assembly has been removed.

(6) Remove lockwire (item 22) from base of T-bolt nut (item 21).

(7) Loosen T-bolt nut.

(8) Pivot T-bolt to release V-band clamp (item 8). Unlatch quick attach-detach adapter (QAD, item 19) from V-band clamp.

NOTE: To maintain alignment of alternator and starter-generator, have one mechanic hold alternator in position by keeping mounting surfaces flush with QAD pad while another mechanic loosens and removes V-band clamp.

(9) Carefully remove starter-generator alternator assembly from QAD pad, maintaining alignment of spline to prevent binding of drive shaft.

(10) Remove nuts (item 23) and washers (item 24) securing QAD pad to engine. Remove QAD pad.

(11) Remove intake duct assembly from alternator and attaching hose.

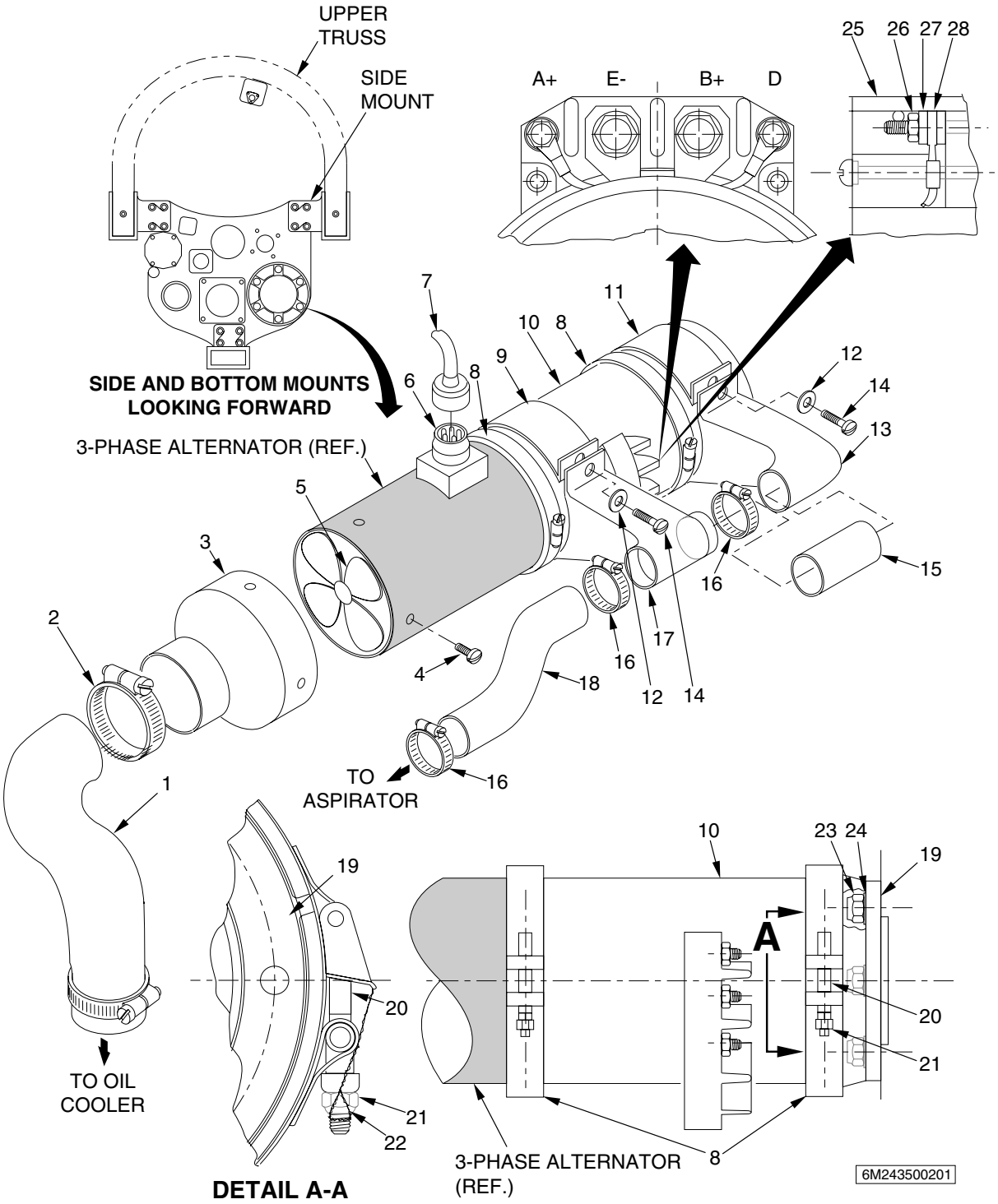
B. Installation – Starter-Generator Alternator (see Figure 201)

(1) Prior to installing starter-generator alternator assembly, verify/perform the following:

(a) Assigned mechanics have clear access to installation area.

(b) Entire length of drive shaft is clean and smooth.

(c) Starter-generator guide pins are both clean and visibly free of defects.



Starter-Generator Alternator
Figure 201

(d) Air screen provided over alternator fan (item 5) is removed (if new replacement alternator has been supplied).

(e) Mounting surfaces of starter-generator alternator (item 10) and QAD pad (item 19) forward of starter-generator are clean and free of burrs and any other surface defects.

(f) QAD pin guide pin holes are clean, free of burrs, foreign objects/substances, and otherwise satisfactory.

(g) Position QAD pad whereby any one (1) of three (3) QAD pad guide pins is positively engaged at 12 o'clock position and 120° apart from each of the other two (2) guide pins. Verify that positioning is correct.

(h) Install nuts (item 23) and washers (item 24). Torque nuts 120 inch-pounds (13.56 N·m).

(i) Verify 115VAC power to starter-generator alternator assembly is OFF.

CAUTION: STARTER-GENERATOR ALTERNATOR ASSEMBLY MUST BE SUPPORTED AT ALL TIMES DURING REMOVAL/INSTALLATION. ANY MISALIGNMENT AND/OR BINDING OF THE DRIVE SHAFT CAN REDUCE THE EFFECTIVE SERVICE LIFE OF EQUIPMENT. DO NOT LET ASSEMBLY HANG WITHOUT THE V-BAND CLAMPS AND ATTACHING HARDWARE FULLY TIGHTENED. UNDUE BENDING LOADS EXERTED ON DRIVE SHAFT SHEAR SECTION CAN DAMAGE THE ASSEMBLY.

(2) Install intake duct assembly (item 3) onto aft end of alternator with screws (item 4), first removing screws from alternator, then slipping ring of intake duct assembly over end of alternator, and installing screws back into alternator.

(3) Clean and lubricate drive shaft spline (if required) thoroughly before installation. Lubricate with Molykote M77, MIL-G-21164 (molybdenum disulfide) or equivalent.

(4) Engage drive shaft spline to accessory drive on aircraft, being certain to use two (2) mechanics to accomplish the following:

(a) Align starter-generator alternator assembly to QAD pad, avoiding any binding of drive shaft against the drive shaft spline.

(b) Hold starter-generator alternator assembly in position by keeping mounting surfaces flush with QAD pad, using services of first mechanic to hold the assembly in position, while the second mechanic installs and secures V-band clamp (item 8).

(5) Align dowel pins with applicable guide holes and press starter-generator alternator assembly onto mounting adapter.

(6) Position V-band clamp over mating flange of alternator, latch T-bolt (item 20) and partially tighten T-bolt nut (item 21).

(7) Tap around entire circumference of V-band using a rubber mallet of suitable size to ensure proper alignment of QAD pad with respect to the starter-generator alternator assembly.

(8) Tighten T-bolt nut to two-thirds (2/3) the recommended torque value stamped on V-band clamp, approximately 46 inch-pounds (5.20 N·m).

(9) Tap around entire circumference of V-band clamp while tightening T-bolt nut to its recommended torque value of 70 inch-pounds (7.91 N·m).

(10) Safety wire T-bolt by securing lockwire (item 22) at base of T-bolt nut.

(11) Install starter-generator cooling duct assembly (item 13) hose (item 15), alternator cooling duct assembly (item 17), and hose (item 18) as follows:

(a) Align, mate and secure collar (item 11) to starter-generator cooling duct assembly with screw (item 14) and washer (item 13).

(b) Engage end of hose (item 15) into inlet of starter-generator cooling duct assembly and surround other end of hose with loose hose clamp (item 16).

(c) Align, mate, and partially secure collar (item 9) to alternator cooling duct assembly with screw (item 14) and washer (item 12), leaving sufficient slack to engage open end of hose (item 15) over transition hose flange of alternator cooling duct assembly.

(d) Engage hose (item 15) over hose flange of alternator cooling duct assembly.

(e) Engage forward end of hose (item 18) surrounded with hose clamp (item 16) over alternator cooling duct assembly hose flange, then tighten hose clamp securely.

(f) Engage aft end of hose (item 18) surrounded with hose clamp over hose flange of oil cooler aspirator, then tighten hose clamp securely.

(g) Engage hose (item 1) over hose flange of intake duct assembly and secure with hose clamp (item 2).

CAUTION: ENSURE THAT ALL PINS OF CONNECTOR ARE STRAIGHT AND NOT DAMAGED IN ANY WAY PRIOR TO ENGAGING PLUG WITH CONNECTOR. BINDING OF CONNECTOR PINS OR BENT CONNECTOR PINS WITH POWER APPLIED CAN RESULT IN SERIOUS DAMAGE TO THE ALTERNATOR AND OTHER ELECTRICAL EQUIPMENT.

(12) Position and install fire extinguisher tube assembly. Torque tube assembly end fitting 550 to 750 inch-pounds (62.14 to 84.74 N·m).

(13) Connect and secure electrical wires (item 28), washers (item 27), and nuts (item 26) onto terminal block (item 25).

(14) Engage plug (item 7) into connector (item 6).

(15) Install inboard, outboard and lower cowling.

2. Adjustment/Test – Starter-Generator Alternator

A. Perform the following operational test each time starter-generator is removed and installed:

(1) Place aircraft in a suitable area to run engines, clear of personnel and equipment.

WARNING: CLEAR PROPELLERS PRIOR TO ENGAGING STARTER.

(2) Start engines IAW AFM.

(3) Place generator switches to RESET, then ON. Check for approximately 28VDC on aircraft voltmeter.

(4) Turn both battery switches OFF.

(5) Turn both generator switches OFF. Verify that both ammeters indicate zero load.

(6) Turn LH generator switch ON. Verify that LH ammeter indicates a load.

(7) Turn LH generator switch OFF.

(8) Turn RH generator switch ON. Verify that RH ammeter indicates a load.

(9) Turn RH generator switch OFF.

(10) Repeat steps (6), (7), (8), and (9).

(11) Turn both battery switches ON.

(12) Perform engine shutdown IAW AFM.

B. Voltage Regulator Paralleling Test (Figure 202)

NOTE: Ensure that the indicating systems have been checked for accuracy prior to initiating this paralleling test.

On later aircraft, a voltage regulator adjustment panel is installed on the J-box behind the pilot's seat. Paralleling procedures for aircraft with and without an adjustment panel are provided in this test.

(1) Move aircraft to run up area.

(2) For aircraft not equipped with a voltage regulator adjustment panel on J-box, T201-291, AT001-074, and TC201-348 (not including provisions of SB 24-018), remove access panel from forward side of J-box behind pilot's seat.

(3) For aircraft equipped with a voltage regulator adjustment panel, T292-999, AT075-999, TC292-999, and earlier aircraft incorporating provisions of SB 24-018, gain access to adjustment panel on inboard side of J-box behind pilot's seat (see Figure 202).

WARNING: CLEAR PROPELLERS PRIOR TO ENGAGING STARTER.

(4) Start engines IAW AFM, with generator switches OFF.

(5) Position generator switches to RESET; then ON.

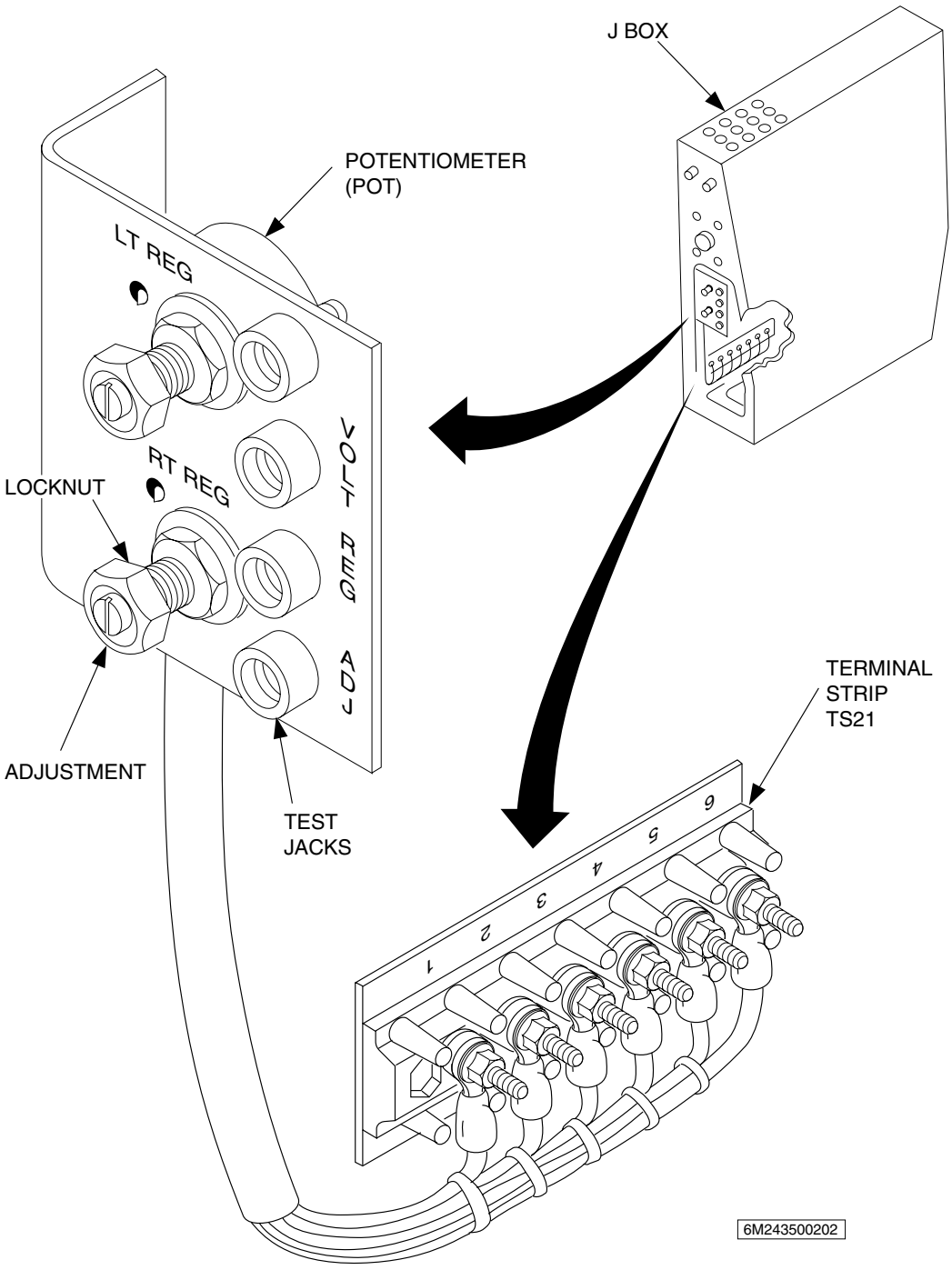
NOTE: If difficulty is experienced getting one (1) or both generators on line, turn one (1) generator on at a time, and adjust voltage to approximately 28 VDC for each. Then turn both ON.

(6) With both generators on line, allow ten (10) minute warm-up for regulators before making paralleling adjustments.

(7) Position both speed levers to LOW.

(8) Position both power levers to GROUND IDLE.

(9) Position generator switches to OFF.



Voltage Regulator for Adjustment Panel
(Incorporating Provisions of SB 24-018)
Figure 202

(10) For aircraft without regulator adjustment panel at J-Box behind pilot:

(a) Connect voltmeter to Black and Red jacks on LH generator control unit (GCU) panel.

(b) Position LH generator switch to RESET; then to OFF.

NOTE: Ensure that approximately 28 VDC is indicated on cockpit voltmeter.

(c) Adjust LH regulator voltage adjustment pot until connected voltmeter indicates 28.5 VDC.

(d) Connect voltmeter to Black and Red jacks on RH generator control unit (GCU) panel.

(e) Position RH generator switch to RESET then to OFF.

NOTE: Ensure that approximately 28 VDC is indicated on cockpit voltmeter.

(f) Adjust RH regulator voltage adjustment pot until connected voltmeter indicates 28.5 VDC.

(g) Remove voltmeter leads from LH and RH generator control unit (GCU) panels.

(h) Turn both generator switches to ON.

(i) Operate electrical equipment that requires high amperage (landing lights, fresh air fan, taxi light, etc.).

(j) Verify that load sharing (on LH and RH Ammeters) is within 10% of applied load.

(k) Shut down engines IAW AFM.

(l) Record LH and RH voltage regulator adjustments in aircraft log.

(11) For aircraft having regulator adjustment panel on J-box behind pilot (Figure 202):

(a) Position LH generator switch to RESET; then to OFF.

(b) Connect voltmeter leads to top two (2) test jacks (LH regulator) of adjustment panel on J-box assembly.

(c) Loosen locknut on top (LH) potentiometer stem.

(d) Using a common screwdriver, adjust top potentiometer (LH REG) until voltmeter indicates precisely 28.5 VDC.

(e) Tighten locknut and check for correct voltage.

(f) Connect voltmeter leads to bottom two test jacks (RH regulator) of adjustment panel on J-box assembly.

(g) Position RH generator switch to RESET; then to OFF.

(h) Loosen locknut on bottom (RH) potentiometer stem.

(i) Using a common screwdriver, adjust bottom potentiometer (RH REG) until voltmeter indicates precisely 28.5 VDC.

(j) Tighten locknut and check for correct voltage.

(k) Remove voltmeter leads from test jacks.

(l) Position both generator switches to ON.

(m) Operate equipment on aircraft that demands high amperage (e.g., pitot heat, prop heat, landing lights, fresh air fan).

(n) Verify load sharing is within 10% of applied load.

(12) Shut down engines IAW AFM.

(13) Record LH and RH voltage regulator adjustments in aircraft log.

3. Inspection/Check – Starter-Generator Alternator

A. Starter-Generator

NOTE: All information presented in this section is applicable to Lear Siegler, Inc. (LSI) starter-generators approved for SA226 aircraft, unless otherwise noted.

(1) Each time the starter-generator is removed:

(a) Inspect the starter-generator accessory drive splines for wear IAW Garrett Maintenance Manual and lubricate before installation with Molykote M77 or MIL-G-21164 (Molybdenum Disulfide). Engines which have had GTEC SB72-0416 complied with, require no lubrication of drive spline.

(b) Inspect starter-generator drive splines IAW Lear Siegler or APC Overhaul Manual instructions.

(c) Inspect wire terminals at starter-generator leads.

(d) Inspect starter-generator blast and cooling tubes for condition and security.

NOTE: If brushes are to be replaced, a brush "run in" procedure must be accomplished IAW manufacturers instructions.

B. Starter-Generator/Alternator

(1) Each 500 hours:

(a) Separate alternator from starter-generator and lubricate interconnecting splines IAW instructions in **3.,** B., (1).

NOTE: If it is necessary to replace either the alternator or the starter-generator, the two units should be separated and only the defective unit returned.

On aircraft equipped with Aircraft Parts Corp. starter-generators, remove the starter-generator for brush inspection. Remove the shortest brush only. Do not remove more than one (1) brush, as brush seating will be affected. Calculate remaining brush life IAW para D, this section.

(2) Each 1000 hours:

(a) Remove and return starter-generator to the applicable Authorized APC or LSI Repair Station for overhaul.

C. Aircraft Parts Corp. Model 300SG Starter-Generator Brush Wear Sample Calculation Procedure (see Figure 203)

(1) Calculate brush life IAW Table 201:

Maximum wear length of a new 300SG1009-5 Brush:			0.700
Brush wear left (as measured from shortest brush):			0.310
Total Brush Wear			0.390
Hours brush has operated:			500
Total brush wear (from above):			0.390
0.390 / Total brush wear	500 = Hours brush operated	0.00078	Rate of brush wear per hour
0.310 / Brush wear left (from above)	0.00078 = Rate of brush wear per hour (from above)	397	Hours on brush wear left before brushes must be replaced

Brush Life
Table 201

(2) If 150 hours or more of brush life remains, install brush of starter-generator removed for inspection/check and operate generator for remainder of brush life, as calculated above, to a maximum of 1000 hours. At that time, return the unit to an Authorized Repair Station for overhaul.

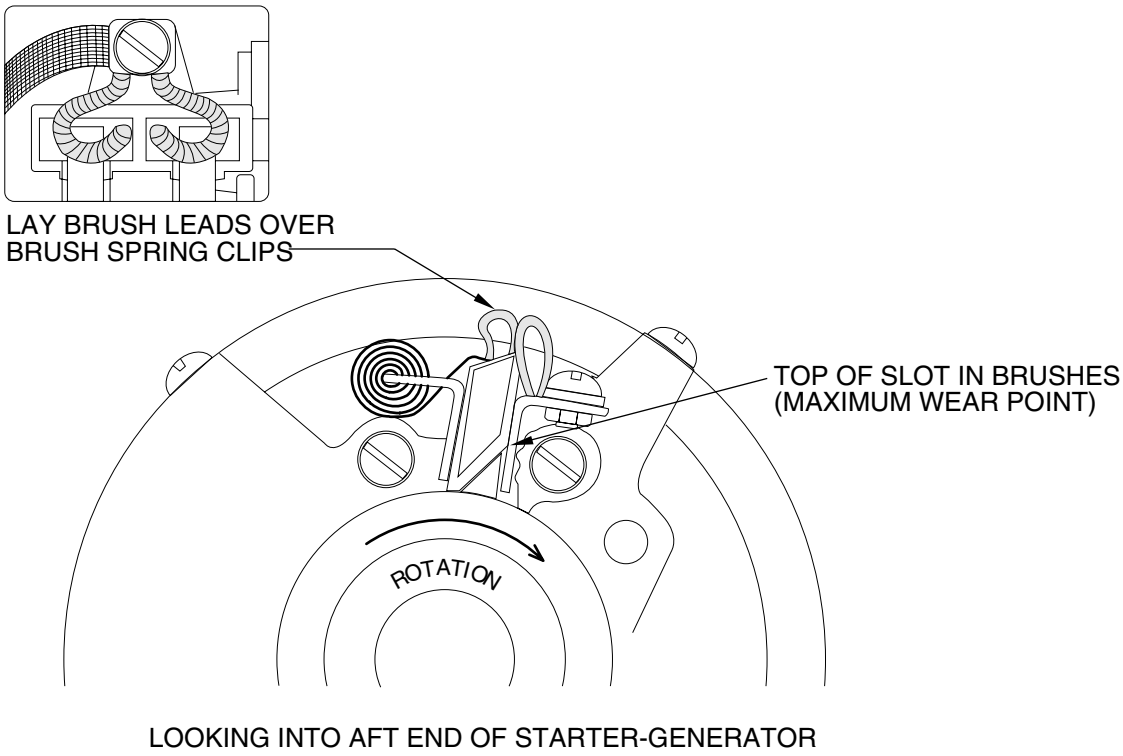
(3) Remove generator after the additional hours established by your calculations have been reached.

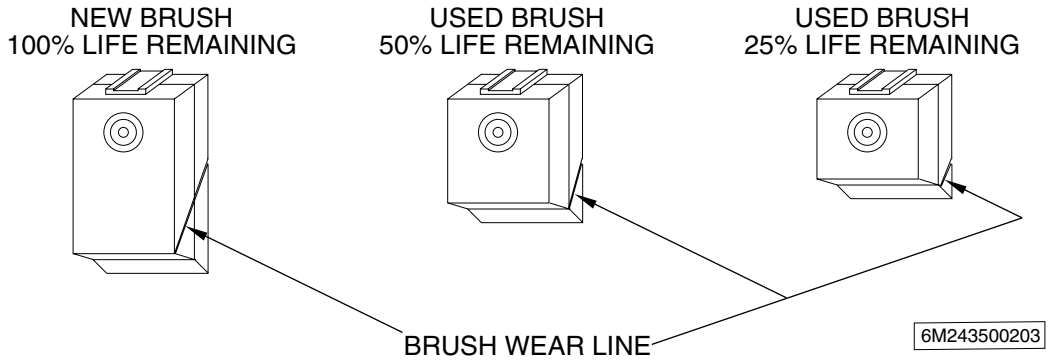
(4) Inspect the brushes.

(5) If all brushes are below maximum brush wear mark, generator MTBO has been established.

(6) Two or three generators can be tested to establish an average brush life.

NOTE: Remove the APC 300A starter-generator for overhaul after it has operated for 800-1000 hours without brush change.





Starter-Generator Brush Wear
Figure 203

D. Generator Paralleling

(1) Check indicating systems for accuracy before adjusting the paralleling of the generators.

NOTE: Generators frequently out of parallel usually indicate that a problem exists within the sensing lines to the regulator (pins A, D, C, F, G, or K). If voltage output of the two (2) generators is close enough to the same (1/2 volt or less difference), the voltage regulator panels can adjust the output of the generators to evenly share the load. How well the generators share the load will be dependent upon the accuracy used in the paralleling operation. If the generators have been paralleled and are operating satisfactorily, a sudden out-of-parallel condition is an indication of system malfunction or an indication problem.

E. Generator Voltmeter and Ammeter

(1) Check voltmeter and ammeter for accuracy, if possible, before beginning paralleling operations. Verify that meter needle indicates zero with power OFF.

F. Generator Voltage Regulator Panel

(1) Ensure that voltage regulator panel is accessible during the paralleling operation.

NOTE: Aircraft with regulators installed in the cockpit may require removal of pilot and copilot seats. Later aircraft require removal of voltage regulator access panels installed in each wheel well. Aircraft with voltage regulator panels installed in wheel wells also require some means of communication between the engines and the person adjusting the regulator.

G. Equipment

(1) Ensure availability of precision voltmeter with a minimum 1/2% accuracy.

STARTER-GENERATOR – DESCRIPTION AND OPERATION

1. General

A. Two engine-driven starter-generators provide DC power for the aircraft systems. DC generation equipment is installed in two circuitry configurations, each fulfilling the same DC power-providing function, but differing somewhat in the manner this is achieved.

B. Conventional starter-generator circuitry incorporating field current and reset resistors, reset and generator relays and associated diodes are installed on SA226 T249-275, 277-291, AT025-074, and TC209-999 aircraft not modified by SB 24-031 and SB 24-033.

(1) These particular SB's enabled the DC starter-generator control system to be self-exciting and operate off its respective generator output.

(2) SB 24-033 is limited to providing an improved generator control relay on an optional basis for pre-SB 24-031 and 033 aircraft.

C. Post SB 24-031 and 24-033 and aircraft serial numbers T292-999, AT075-999, and TC420-999, incorporate the self-exciting control system circuitry and the improved generator control relay.

D. To accommodate the larger starter-generator, minor substitutions in several attaching items are required to provide proper clearance between the fuel shutoff bellcrank bolt and the improved starter-generator.

STARTER-GENERATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Starter-Generator

A. Removal – Starter-Generator (Figure 201)

(1) Remove inboard, outboard, and lower cowling, as applicable, and position out of way.

CAUTION: THE STARTER-GENERATOR MUST BE SUPPORTED AT ALL TIMES DURING REMOVAL/INSTALLATION. ANY MISALIGNMENT AND/OR BINDING OF THE DRIVE SHAFT CAN REDUCE THE EFFECTIVE SERVICE LIFE OF EQUIPMENT. DO NOT LET ASSEMBLY HANG WITHOUT THE V-BAND CLAMPS AND ATTACHING HARDWARE FULLY TIGHTENED. UNDUE BENDING LOADS EXERTED ON DRIVE SHAFT SHEAR SECTION CAN DAMAGE THE ASSEMBLY.

(2) Verify power to starter-generator is OFF.

(3) Place identification tags on unmarked wires connected to starter-generator terminal block, then disconnect all electrical wires at the terminal block, including engine fire-warning temperature sensor wire (red and white diagonal pattern) if included in starter-generator wire bundle.

(4) If installed, disconnect fire extinguisher line and position out of the way.

(5) Disconnect and remove starter-generator cooling duct assembly (item 14), associated collar (item 11) and hoses (items 16, 19, and 21). Leave intake duct assembly (item 3) attached to aft end of starter-generator until starter-generator has been removed.

NOTE: Provide safe storage and accountability for all attaching hardware (hose clamps, screws, washers, and nuts) if replacement starter-generator is not immediately installed following removal of defective assembly.

(6) Remove lockwire (item 24) from base of T-bolt nut (item 23).

(7) Loosen T-bolt nut.

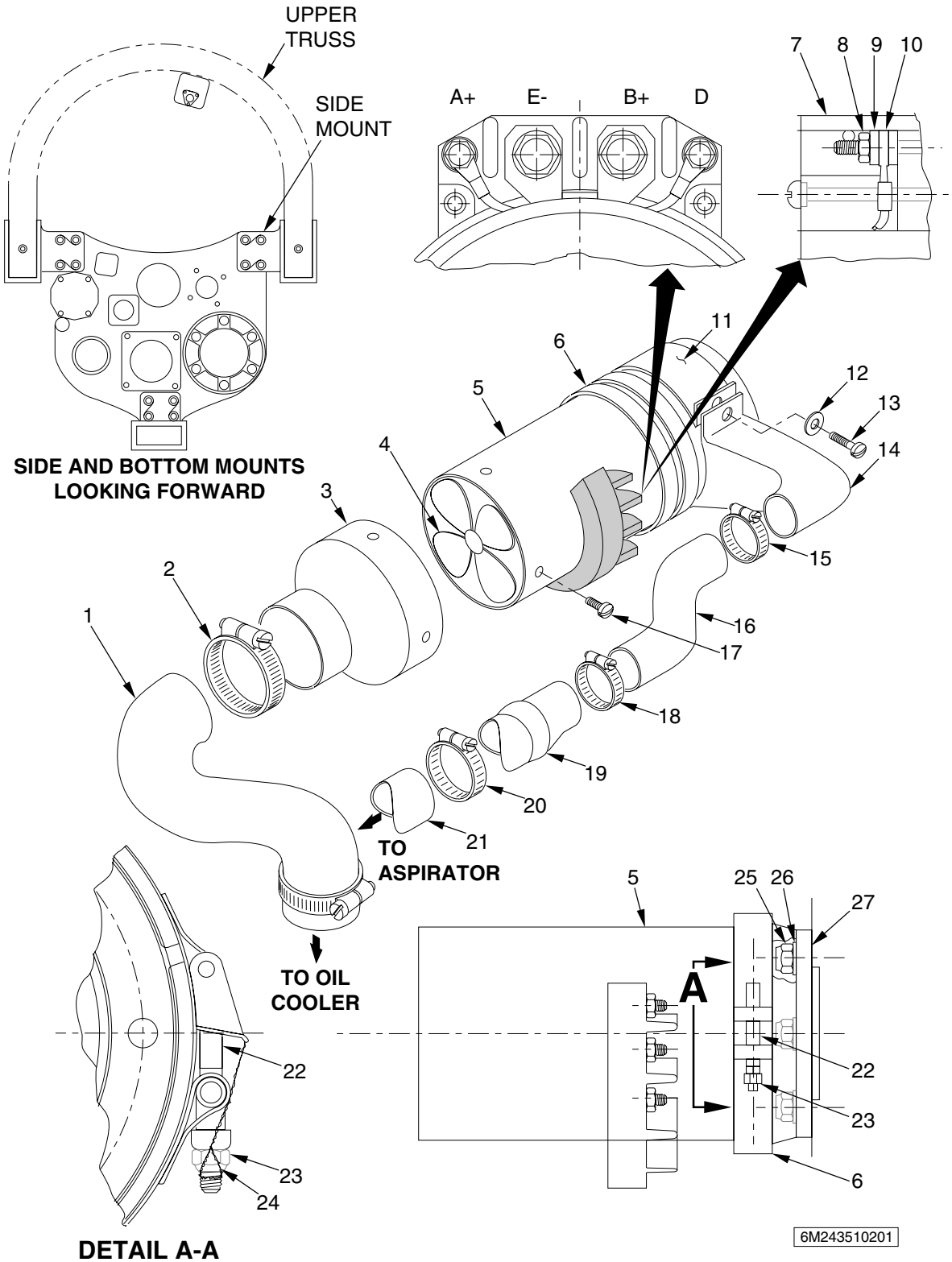
(8) Pivot T-bolt to release V-band clamp (item 6). Unlatch quick attach-detach adapter (QAD, item 27) from V-band clamp.

NOTE: To maintain alignment of the starter-generator, have one mechanic hold the alternator in position by keeping mounting surfaces flush with QAD pad while another mechanic loosens and removes the V-band clamp.

(9) Carefully remove starter-generator from QAD pad, maintaining alignment of spline to prevent binding of the drive shaft.

(10) Remove nuts (item 25) and washers (item 26) securing QAD pad to engine. Remove QAD pad.

(11) Remove intake duct assembly from starter-generator and attaching hose (item 1).



Starter-Generator
Figure 201

B. Installation – Starter-Generator (Figure 201)

(1) Prior to installing starter-generator verify/perform the following:

(a) Assigned mechanics have clear access to installation area.

(b) Entire length of drive shaft is clean and smooth.

(c) Starter-generator guide pins are both clean and free of defects.

(d) Remove air screen provided over starter-generator fan (item 4) (if new replacement starter-generator has been supplied).

(e) Mounting surfaces of starter-generator and QAD pad (item 27) forward of starter-generator (item 5) are clean and free of burrs and surface defects.

(f) QAD pad guide pin holes are clean, free of burrs, foreign objects/substances, and in a satisfactory condition.

(g) Position QAD pad, whereby, any one (1) of three (3) QAD pad guide pins is positively engaged at 12 o'clock position and 120° apart from each of the other two (2) guide pins. Verify correct positioning.

(h) Install nuts (item 25) and washers (item 26) and torque to 95 to 110 inch-pounds (10.73 N·m to 12.43 N·m).

(i) Verify power to starter-generator is OFF.

CAUTION: STARTER-GENERATOR MUST BE SUPPORTED AT ALL TIMES DURING REMOVAL/INSTALLATION. ANY MISALIGNMENT AND/OR BINDING OF THE DRIVE SHAFT CAN REDUCE THE EFFECTIVE SERVICE LIFE OF EQUIPMENT. DO NOT LET ASSEMBLY HANG WITHOUT THE V-BAND CLAMPS AND ATTACHING HARDWARE FULLY TIGHTENED. UNDUE BENDING LOADS EXERTED ON DRIVE SHAFT SHEAR SECTION CAN DAMAGE THE UNIT.

(2) Install intake duct assembly (item 3) onto aft end of starter-generator with screws (item 17), first removing screws from starter-generator, then slipping ring of intake duct assembly over end of starter-generator, and installing screws back into starter-generator.

(3) Clean and lubricate drive shaft spline (if required) thoroughly before installation. Lubricate with Molykote M77, MIL-G-21164 (molybdenum disulfide) or equivalent.

NOTE: Engines which have had GTEC SB72-0416 complied with, require no lubrication of drive spline.

(4) Engage drive shaft spline with engine accessory drive pad, being certain to use two (2) mechanics to accomplish the following:

(a) Align starter-generator (item 5) with QAD pad, avoiding any binding of drive shaft against drive shaft spline.

(b) Hold starter-generator in position by keeping mounting surfaces flush with QAD pad, using services of first mechanic to hold the assembly in position, while the second mechanic installs and secures V-band clamp (item 6).

(5) Align dowel pins with applicable guide holes and press starter-generator onto mounting adapter.

(6) Position V-band clamp over mating flange of alternator, latch T-bolt (item 22) and partially tighten T-bolt nut (item 23).

(7) Tap around entire circumference of V-band using a rubber mallet to ensure proper alignment of QAD pad with respect to starter-generator alternator assembly.

(8) Tighten T-bolt nut to two-thirds (2/3) the recommended torque value stamped on the V-band clamp, approximately 46 inch-pounds (5.20 N·m).

(9) Tap around entire circumference of V-band clamp while tightening T-bolt nut to its recommended torque value of 70 inch-pounds (7.91 N·m).

(10) Safety wire T-bolt by securing lockwire (item 24) at base of T-bolt nut.

(11) Install starter-generator cooling duct assembly (item 14) and adjoining hose (item 16), as follows:

(a) Align, mate, and partially secure collar (item 9) to starter-generator cooling duct assembly with screw (item 13) and washer (item 12), leaving sufficient slack to engage forward end of hose (item 16) over transition hose flange of starter-generator cooling duct assembly.

(b) Secure forward end of hose (item 16) with hose clamp (item 15) and surround aft end of hose with loose hose clamp (item 18).

(c) Engage aft end of hose (item 16) over forward hose flange of reducer assembly (item 19) and secure with hose clamp (item 18).

(d) Engage forward end of hose (item 21) surrounded with hose clamp (item 20) over aft end of reducer assembly, then tighten hose clamp securely.

(e) Engage aft end of hose (item 21) surrounded with hose clamp over hose flange of oil cooler aspirator, then tighten hose clamp securely.

CAUTION: ENSURE THAT ALL CONNECTOR PINS ARE STRAIGHT AND NOT DAMAGED IN ANY WAY PRIOR TO ENGAGING PLUG WITH CONNECTOR. BINDING CONNECTOR PINS OR BENT CONNECTOR PINS CAN RESULT IN SERIOUS DAMAGE TO THE ALTERNATOR AND OTHER ELECTRICAL EQUIPMENT WHEN POWER IS APPLIED.

(12) Position and install fire extinguisher tube assembly. Torque tube assembly end fitting 550 to 750 inch-pounds (62.14 to 84.74 N·m).

(13) Connect and secure electrical wires (item 10), washers (item 9) and nuts (item 8) of starter-generator onto terminal block (item 7).

(14) Install inboard, outboard, and lower cowling.

2. Inspection/Check – Starter-Generator

A. Each time starter-generator (item 5) is removed and/or installed, perform the following operational test:

(1) Place aircraft in a suitable area to operate engines, clear of personnel and equipment not contributing directly to subject inspection/check.

(2) Prior to starting the engine, and depending on the type starter-generator installed, perform the following, as applicable:

(a) For self-exciting type starter-generators, turn battery switches OFF, IAW SB 24-031.

(b) For non-self exciting type starter-generators (pre-SB 24-031), turn both battery switches ON.

WARNING: CLEAR PROPELLERS PRIOR TO ENGINE START.

(3) Start engines IAW AFM.

(4) Position generator switches, LH and RH, on copilot's switch panel to RESET, then ON. Check the following:

(a) Check for approximately 28VDC indicated on aircraft voltmeter.

(b) Verify annunciator GEN FAIL light extinguishes.

(5) Position starter-generator switches, LH and RH, OFF or ON, IAW type starter-generator used.

(6) Position LH starter-generator switch OFF. Verify that LH ammeter indicates zero (0) load.

(7) Position LH generator switch ON. Verify that LH ammeter indicates a load that is within 10% of opposite load.

(8) Repeat steps (6) and (7) for RH side.

(9) Position both starter-generator switches OFF.

(10) Position both battery switches ON.

(11) Shut-down engines IAW AFM.

B. Each time starter-generator is removed (see Figure 201):

(1) Inspect starter-generator splines for wear IAW Lear Siegler or Aircraft Parts Corporation Maintenance Manual, as applicable.

(2) Inspect starter-generator IAW manufacturer's Component Maintenance Manual.

(3) Inspect wire terminals at starter-generator for condition.

(4) Inspect starter-generator blast and cooling tubes for condition and security.

(5) Lubricate before installation, using Molykote M77 or MIL-G-21164 (Molybdenum Disulfide). Engines in compliance with GTEC SB72-0416, require no lubrication of drive spline.

(6) Inspect starter-generator cooling duct assembly (item 14) and adjoining collar (item 11) for serviceability, security, and condition.

CAUTION: PROGRESSIVE WEAR THROUGH OUTER LAYER OF SKEET HOSE TO POINT WHERE SPIRAL WIRE IS EXPOSED AND SUBJECT TO DIRECT CONTACT WITH EXPOSED WIRE OR TERMINAL CAN CAUSE A SHORT-CIRCUIT AND DISABLE STARTER-GENERATOR ALTERNATOR ON EITHER AN INTERMITTENT OR EXTENDED BASIS AND OTHERWISE DEGRADE THE PERFORMANCE OF THE AC AND DC ELECTRICAL SYSTEMS.

(7) Inspect hoses (items 1, 16, and 21), reducer assembly (item 19), and hose clamps (items 2, 15, 18, and 20) for condition. Inspect hose (item 16) particularly close, as it is subject to abrasive wear on side where it frequently comes in contact with terminal A+ of starter-generator terminal block.

(8) Inspect V-band clamp (item 6) for serviceability, security, and condition.

(9) Inspect electrical wires and terminals (item 11) for security and condition.

(10) Inspect intake duct assembly (item 3) for condition.

(11) Inspect starter-generator fan (item 4) for serviceability, security, and condition.

DC VOLTMETER AND SELECTOR SWITCH – DESCRIPTION AND OPERATION

1. General

NOTE: This section also applies to aircraft retrofitted by SB 24-008 (formerly SB 24-30-2036, -3066, and -4066).

A. A voltmeter and selector switch are used to monitor any one of five power sources. Each source, providing a signal to the voltmeter, is provided with a circuit breaker for protection. The functions of each switch position are as follows:

NOTE: As a customer option, a position is provided to measure the voltage output of a GPU connected to the aircraft. Voltage upstream of the battery bus relay is monitored. After the battery switch is moved to ON, this voltage will be approximately equal to bus voltage.

(1) LH or RH battery position:

(a) Monitors the battery voltage of the battery side of the battery relay. In order to obtain an accurate reading, the battery switch should be OFF to isolate the battery from the bus.

(b) If a measurement is taken with the battery switch ON, the battery voltage will be approximately equal to bus voltage.

(2) LH or RH generator position:

(a) Monitors the generator voltage on the generator side of the generator relay.

(b) In order to obtain an accurate reading, the generator switch should be OFF, isolating the generator from the bus.

(c) If a measurement is taken with the generator switch ON, the generator voltage will be approximately equal to bus voltage.

(3) Bus position:

(a) Monitors the voltage at the battery bus in the electrical panel behind the pilot's seat. This voltage will be the average of the voltages applied to the bus. For example, with one battery switch ON and no generators operating, the bus and battery voltage will be approximately equal.

GENERATOR CONTROL AND REGULATING – DESCRIPTION AND OPERATION

1. General

A. Two generator switches, installed on the pilot's switch panel, control the connection of the generators to the bus. Each generator switch also has a spring-loaded RESET position used to reset the generator fault systems and flash the field after engine start.

2. System Operation

A. During the engine start cycle, power is applied to the No. 1 start control relay as soon as the start button is depressed. The actuation of this relay breaks the A-lead of the generator circuit, protecting the regulator against spurious voltage signals from the generator during engine start.

B. After engine start, the generator switch is moved to the RESET position. Power from the No. 2 start control circuit breaker is applied to the generator A-lead to flash the field. A loop is completed between two pins and power is applied to a third pin. This resets the fault system inside the protection panel.

C. When the generator switch is moved to the ON position, a loop between the applicable pins is completed, and if the proper conditions exist (see Starter-Generator Alternator – Description and Operation), the generator relay closes and the generator begins to supply the load.

D. When the generator switch is moved to the OFF position, the loop between the two applicable pins is broken and the regulator panel will open the generator relay. With the generator relay open, the generator output will not necessarily drop to zero and the generator sense relay may remain closed.

GENERATOR CONTROL AND REGULATING – MAINTENANCE PRACTICES

1. Inspection/Check – Generator Control and Regulating

CAUTION: NEVER CONNECT OR DISCONNECT THE VOLTAGE REGULATOR ELECTRICAL CONNECTOR WITH THE GENERATOR TURNING. EVEN IF ALL CIRCUIT BREAKERS ARE PULLED AND THE GENERATOR SWITCH IS IN "OFF" POSITION, DAMAGE TO THE REGULATOR MAY STILL OCCUR.

A. Each time a starter-generator is removed:

(1) Inspect starter-generator drive splines for wear and lubricate before installation.

NOTE: This inspection of spline only applies to engines not in compliance with GTEC SB72-0416.

(2) Inspect wire terminals at starter-generator leads.

(3) Inspect starter-generator blast and cooling tubes for condition and security.

B. Each time pilot and/or copilot seat(s) are removed, before installation:

(1) Inspect each voltage regulator, mounted below crew seats, for security of mounting.

(2) Inspect electrical connector and wiring to each voltage regulator for damage.

NOTE: Exception to the above applies to those aircraft in this effectivity series which have been modified under the provisions of SB 24-018.

C. Each 500 hours:

(1) Inspect LH and RH essential bus wire terminations and grounds for damage and security.

2. Removal/Installation - Generator Control Unit

A. Removal – Generator Control Unit

NOTE: Generator control panel assembly for each regulator (left or right) is located on the left side of the respective wheel well.

(1) Ensure aircraft electrical power is OFF.

(2) Remove screws (unlock Dzus fasteners if applicable) securing generator control panel to wheel well.

(3) Pull panel out just far enough to gain access to voltage regulator. Be careful not to pull or chafe wires.

(4) Remove safety wire securing electrical connector to generator control unit.

(5) Disconnect electrical connector from generator control unit.

(6) Remove screws, washers, and nuts securing voltage regulator to panel.

(7) Remove generator control unit.

B. Installation – Generator Control Unit

(1) Ensure aircraft electrical power is OFF.

(2) Position generator control unit in generator control panel assembly.

(3) Secure with screws, washers, and nuts.

(4) Connect electrical connector to generator control unit.

(5) Safety wire electrical connector.

(6) Position generator control panel in wheel well and secure with screws (Dzus fasteners if applicable).

GENERATOR CONTROL AND REGULATING – DESCRIPTION AND OPERATION

1. General

A. Two generators switches, installed on the pilot's switch panel, control the connection of the generators to the bus. Each generator switch also has a spring loaded RESET position used to reset the generator fault system and flash the field after engine start.

2. System Operation

A. During the engine start cycle, power is applied to the regulator output control relay from the No. 1 start control relay. The actuation of the output control relay breaks the A-lead of the generator circuit, protecting the regulator against spurious signals from the generator during start.

B. After engine start, moving the generator switch to the RESET position supplies power to close the generator reset relay. The closing of the relay resets the fault systems inside the voltage regulator by completing a loop between two pins and suppling power to a third pin. The RESET relay also supplies power to the A-terminal of the generator to flash the field.

C. If the voltmeter selector switch is moved to the respective generator position, the flashing voltage will be indicated when the generator switch is moved to RESET.

D. Moving the generator switch to the ON position supplies power to close the generator ON relay. The closing of the generator ON relay completes a loop between two pins, and if the proper conditions exist (see Starter-Generator – Description and Operation, this chapter), the generator control relay closes allowing the generator to begin supplying power to the bus system.

E. When the generator switch is placed in the OFF position, the power necessary to the generator ON relay is removed and the relay opens, breaking the loop between two specific pins. The regulator senses the breaking of this connection and opens the generator control relay, disconnecting the generator from the bus system. The generator output will not necessarily drop to zero nor will the generator sense relay open.

GENERATOR CONTROL AND REGULATING – MAINTENANCE PRACTICES

1. Inspection/Check – Generator Control and Regulating

CAUTION: NEVER CONNECT OR DISCONNECT THE VOLTAGE REGULATOR ELECTRICAL CONNECTOR WITH THE GENERATOR TURNING. EVEN IF ALL CIRCUIT BREAKERS ARE PULLED AND THE GENERATOR SWITCH IS OFF, DAMAGE TO THE REGULATOR MAY STILL OCCUR.

A. Each time a starter-generator is removed:

(1) Inspect starter-generator drive splines for wear and lubricate before reinstallation.

NOTE: This inspection of spline only applies to engines not in compliance with GTEC SB72-0416.

(2) Inspect wire terminals at starter-generator leads.

(3) Inspect starter-generator blast and cooling tubes for condition and security.

B. Each time components within either wheel well are removed, before installation:

(1) Inspect voltage regulator mounted to its panel for security of mounting.

(2) Inspect electrical connector and wiring to voltage regulator for damage.

C. Each 500 hours:

(1) Inspect the LH and RH essential bus wire terminations and grounds for damage and security.

2. Removal/Installation - Generator Control Unit

A. Removal – Generator Control Unit

NOTE: Generator control panel assembly for each regulator (left or right) is located on the left side of the respective wheel well.

(1) Ensure aircraft electrical power is OFF.

(2) Remove screws (unlock Dzus fasteners if applicable) securing generator control panel to wheel well.

(3) Pull panel out just far enough to gain access to voltage regulator. Be careful not to pull or chafe wires.

(4) Remove safety wire securing electrical connector to generator control unit.

(5) Disconnect electrical connector from generator control unit.

(6) Remove screws, washers, and nuts securing voltage regulator to panel.

(7) Remove generator control unit.

B. Installation – Generator Control Unit

(1) Ensure aircraft electrical power is OFF.

(2) Position generator control unit in generator control panel assembly.

(3) Secure with screws, washers, and nuts.

(4) Connect electrical connector to generator control unit.

(5) Safety wire electrical connector.

(6) Position generator control panel in wheel well and secure with screws (Dzus fasteners if applicable).

LOCATION OF KEY DC COMPONENTS – DESCRIPTION AND OPERATION

1. General

A. The following tables are provided to aid in the location of electrical components throughout the aircraft. In some cases, customer option installations and avionics installations may have caused certain components to be relocated. Relocated components will, in general, be located near their original location.

B. For illustrated details and pertinent parts relationships of DC components not covered in this chapter, refer to applicable figures in Chapter 24 of the Illustrated Parts Catalog.

C. For electrical details and pertinent functional relationships of DC components not covered in this chapter, refer to applicable wiring diagrams in Chapter 24 of the Airplane Wiring Manual.

COMPONENT	EFFECTIVITY	LOCATION
Starter-generator	T201-999 AT001-999 TC201-999	LH and RH engines.
Batteries	T201-248 (except T245) AT001-999 TC201-999	LH and RH wing center section.
Batteries	T245, T249-999	Under the nose radome.
Auxiliary battery	TC201-334	Nose baggage compartment, LH side, under aft floorboard.
Voltage regulator and protection panel	T201-248 AT001-024 TC201-208	Under the pilot and copilot seats.
Voltage regulator and protection panel	T249-999 AT025-999 TC209-999	In LH and RH wheel wells, LH side, large rectangular panel.

Key DC Components
Table 1

LOCATION OF KEY DC COMPONENTS – DESCRIPTION AND OPERATION

COMPONENT	EFFECTIVITY	LOCATION
Battery relay	T201-248 (Exc AT025) AT001-999 TC201-999	Next to each battery in the respective battery well.
Battery relay	T245, T249-999	On electrical panel in forward nose baggage compartment.
Battery bus relay	T201-999 AT001-999 TC201-999	In J-box, behind pilot's seat.
Battery fault box	T201-210 AT001-024 TC201-212	Under pilot's seat, near voltage regulator
Battery fault box	T249-999 AT025-999 TC213-999	Installed on forward LH console door.
Battery fault box	T211-248	Under center cockpit floor at F.S. 120.
Ground fault sensing reset relay	T235-248 AT025-999 TC213-999	Under copilot's seat, attached to seat rail supporting structure.
Battery overheat and disconnect circuit board	T201-248 AT001-033 (Exc AT025) TC201-212	Inside LH console, on inside of aft access door.
Battery overheat and disconnect circuit board	T249-999 AT025, 034-999 TC213-999	Annunciator.

Key DC Components
Table 1 (Continued)

LOCATION OF KEY DC COMPONENTS – DESCRIPTION AND OPERATION

COMPONENT	EFFECTIVITY	LOCATION
Generator relay	T201-248 AT001-024 TC201-208	LH and RH side console at F.S. 100.
Generator relay	T249-999 AT025-999 TC209-999	LH and RH nacelle, RH side of each nacelle.
LH generator shunt	T201-999 AT001-999 TC201-999	LH wheel well, inside RH keelson.
RH generator shunt	T201-248 AT001-024 TC201-208	In RH wheel well electrical panel. lower center area of panel.
RH generator shunt	T249-999 AT026-999 TC209-999	In RH wheel well electrical panel.
Battery relay disconnect circuit breaker	T239-248 AT013-999 TC206-999	Next to battery relay in each battery compartment.
Battery relay disconnect circuit breaker	T249-999	On electrical panel inside nose baggage compartment forward area.
Battery voltmeter circuit breaker	AT025-999 TC209-999	Next to battery relay in each battery compartment.

Key DC Components
Table 1 (Continued)

LOCATION OF KEY DC COMPONENTS – DESCRIPTION AND OPERATION

COMPONENT	EFFECTIVITY	LOCATION
Battery voltmeter circuit breaker	T249-999	On electrical panel inside nose baggage compartment, forward area.
Generator voltmeter circuit breaker	T249-999 AT025-999 TC209-999	Next to each generator relay.
Auxiliary battery charging relay	TC201-334	LH side of nose compartment under floorboard.
Auxiliary battery power transfer relay	T201-334	LH side of nose compartment under floorboard.
Generator RESET relay	T249-999 AT025-999 TC209-999	Installed on the regulator panel in each wheel well.
Generator ON relay	T249-999 AT025-999 TC209-999	Installed on the regulator panel.
Ground fault sensing reset relay	T256-999 AT042-999 TC217-999	RH side console at F.S. 110.

Key DC Components
Table 1 (Continued)

LOCATION OF KEY DC COMPONENTS – DESCRIPTION AND OPERATION

COMPONENT	EFFECTIVITY	LOCATION
Generator sense relays	T201-210 AT001-003 TC201-202	Under crew seats.
Generator sense relays	T211-225	Under center cockpit floorboard at F.S. 120.
Generator sense relays	AT004-041 TC203-216	Under LH and RH floorboards at F.S. 107.
Generator sense relays	T256-999 AT042-999 TC217-999	Inside RH console at F.S. 110.

Key DC Components
Table 1 (End)

EXTERNAL POWER – DESCRIPTION AND OPERATION

1. General

A. External Power Circuit

(1) The external power circuit consists of:

(a) An external power receptacle located in any one of three locations on the aircraft.

1 On SA226 T201-248 (including provisions of SB 24-007, battery relocation to nose) and T249-999, the GPU receptacle is mounted on the RH nose section just aft of the radome.

2 On SA226 AT001-074, and TC201-419, including provisions of SB 24-024, but not including provisions of SB 24-028, the GPU receptacle is mounted on the outboard side of RH nacelle.

3 On SA226 AT001-074 and TC201-419, including the provisions of SB 24-024 and SB 24-028, the GPU receptacle is mounted on the aft right or left lower wing fairing.

(b) Ground power unit (GPU) plug in switch.

(c) A portion of series/parallel relay.

(d) A start mode control relay.

(e) A GPU control relay.

(f) Battery bus relay.

(g) Interconnecting wires.

B. External Power Source

(1) The external power source should be capable of delivering:

(a) Continuous load of 60 amperes at 28.5 volts.

(b) Up to 1,000 amperes for 0.1 of a second during engine starts

2. System Operation

A. Power supplied by the GPU is applied directly from the series/parallel relay to the battery bus relay. Power from the small pin of the GPU plug closes the APU relay, breaking the ground for the series/parallel relay to prevent the batteries from switching to series during engine start.

B. On SA226 T201-245, AT001-074, except AT070, and TC201-419, the following options may be incorporated into the GPU circuitry, in addition to the standard GPU items:

(1) A remounting of diode in GPU circuitry to improve engine start – added by SB 24-004.

(2) A GPU plug-in warning light circuit– added by SB 24-024.

C. On SA226 AT001-074 and TC201-419, the following additional options may be incorporated into the GPU circuitry:

(1) A GPU receptacle located either in the aft right or left lower wing fairing – added by SB 24-028.

1. Connect/Disconnect – External Power

A. Observe the following precautions prior to connecting and disconnecting external power:

(1) Always apply power from ground power unit (GPU) before switching battery voltage on. If batteries are removed from aircraft, connect an external battery parallel to the external power unit prior to energizing ground power unit (GPU.) Batteries may be damaged if exposed to voltages higher than 30 volts for extended periods of time.

(2) Use only an external power source that is negative grounded. If polarity of power source is unknown, determine polarity with a voltmeter before connecting unit to aircraft.

(3) Before connecting or disconnecting an external power source, turn all radio equipment, generator switches and battery switches OFF.

(4) Do not operate aircraft generators with a GPU connected to the bus. Internal damage to aircraft wiring system may result.

(5) After connecting external power plug and turning on GPU, switch batteries ON before turning on any other equipment.

(6) Leave battery ON during external power operation to protect transistorized equipment against transient voltage spikes.

(7) Do not operate avionics with aircraft batteries OFF. Certain types of ground power units, especially the "inverter", produce voltage variations or spikes in their output. These spikes may cause solid state avionics to behave erratically or damage the avionics. Batteries serve as large "capacitors" and smooth out these voltage spikes. Battery switches must be ON to allow GPU use.

B. Connection – External Power Receptacle

(1) Gain access to external power receptacle by opening receptacle door.

(2) Push GPU plug completely into receptacle.

(3) Start GPU and apply power to aircraft by turning GPU power switch ON.

(4) Check GPU voltage by rotating DC voltmeter switch to GPU position. This voltage should be approximately equal to bus voltage.

(5) After completion of operations turn OFF all electrical and battery switches. Turn OFF GPU power switch, turn OFF GPU and disconnect GPU plug from aircraft.

AC ELECTRICAL LOAD DISTRIBUTION – DESCRIPTION AND OPERATION

1. General

A. SA226 T, AT, and TC aircraft use a bus system to distribute electrical power throughout the aircraft. Circuit breakers (CB's) and current limiters protect the buses and their sources against excessive current. The pilot can control the application or removal of power to any bus.

B. Power from each bus is distributed to the individual load systems through the use of small bus bars and CB's. CB's connected to the bus are ganged together by smaller bus bars. The bus bars are connected together and connected to the bus lead.

C. The AC bus system is linked by the two bus tie CB's only when the inverter is supplying the AC power. When the 3-phase alternators are powering the system, each bus is independent. Since the 26VAC power is obtained from each 115VAC bus through the use of an auto-transformer, the two 26VAC buses are not linked together.

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DC ELECTRICAL LOAD DISTRIBUTION – DESCRIPTION AND OPERATION

1. General

A. The DC electrical load distribution point is located in the cockpit J-box behind the pilot's seat on all T, AT, and TC aircraft. DC power is distributed to any one or all three distribution buses, as selected by the bus tie switches.

B. The LH and RH essential buses (located on the LH and RH consoles, respectively) and the non-essential bus on the RH console are selected either ON or OFF by a bus tie switch mounted on the respective console.

C, Individual circuits for control and/or actuation of nine distinct aircraft system circuits are connected through circuit breakers (CB's) to distribution boxes. These circuits may be switched individually to either the LH essential bus or RH essential bus by means of transfer switches located on the LH console. The nine circuits provide power to the following systems:

(1) Fuel Crossflow.

(2) Pilot's Turn and Bank (provides signal for yaw damper on SA226–T).

(3) Surface Deicer Boots.

(4) Landing Gear Control.

(5) Landing Gear and Flap Position Indicator.

(6) Cabin Pressure Dump.

(7) LH Engine Intake Heat.

(8) RH Engine Intake Heat.

(9) LH Windshield Heat.

2. Battery Bus

A. The battery bus must close for battery power to be connected to the battery bus and essential buses or for the generator to charge the batteries. The generators are protected by two (2) 325A current limiters.

3. Essential/Non-Essential Buses

A. Essential buses are protected by two 225A current limiters and a 150A CB.

(1) Bus tie switches are used to disconnect the essential buses should it become necessary.

(2) DC power for inverters comes from LH and RH buses.

B. Each essential bus is connected to its respective battery bus through the 150A non-essential CB and the non-essential bus tie switch. The pilot can control the application or removal of power to the bus.

BATTERY BUS SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The battery bus forms the central distribution point for power. Each battery is connected through a battery relay to the battery bus relay, and then to the battery bus. A 150A circuit breaker (CB) supplies power to the non-essential bus. Power to the bus is controlled by a bus tie switch. Each generator supplies power to the battery bus through a 325A current limiter. The LH and RH essential buses are connected to the respective generator paths by 225A current limiters. Power is distributed to the various CB's by smaller bus bars.

BATTERY BUS SYSTEM – MAINTENANCE PRACTICES

1. Inspection/Check – Battery Bus System

A. Inspect 4 gauge and 0 gauge wire (bus wire) termination points. Each bus wire termination must be inspected for security, condition, corrosion, and evidence of arcing. The termination points of the DC bus wires and their locations are as follows:

(1) Starter-Generator System – see Table 201.

(2) Battery System – see Table 202.

(3) Distribution Wiring – see Table 203.

ITEM/CONNECTION	EFFECTIVITY	LOCATION
Starter-generator connections	All models	LH and RH starter-generators.
Firewall connections	T201-260 TC201-222	LH and RH firewall, RH side of each firewall.
LH generator shunt connections	All models	LH wheel well, inside RH keelson.
RH generator shunt connections	T201-248 AT001-999 TC201-999	RH wheel well electrical panel. Lower center area of panel.
RH generator shunt connections	T249-999	Inside RH wing ice light panel.
Generator relay connections	T201-248 AT001-024 TC201-208	Inside LH and RH side consoles at F.S. 103.
Generator relay connections RH	T249-999 AT025-999 TC209-999	Inboard side of LH wheel well at N.S. 100 and outboard side of wheel well at N.S. 100.
LH starter-generator ground	All Models	LH wheel well, stud protruding through the inboard keelson.
RH starter-generator ground	T201-248 AT001-999 TC201-999	RH wheel well electrical panel, just below center of panel.
RH starter-generator ground	T249-999	RH wheel well, stud protruding through the outboard keelson.
Starter relays	T201-248 (Exc T245) AT001-999 TC201-999	RH wheel well electrical panel, upper LH corner.
Starter relays	T249-999	Nose compartment electrical panel, lower center.

Starter-Generator System
Table 201

ITEM/CONNECTION	EFFECTIVITY	LOCATION
Battery quick disconnect	T201-248 (Exc T245) AT201-999 TC201-999	LH and RH wing battery well panels.
Battery quick disconnect	T245, 249-999	Under nose radome.
Battery quick disconnect ment,	T249-999	Inside nose baggage compart-
(Customer Option)	AT001-999 TC201-999.	LH and RH sides.
Battery relay connections	T201-248 AT001-999 TC201-999	LH and RH wings, inside battery well.
Battery relay connections	T249-999	Nose baggage compartment electrical panel.
LH battery ground connection	T201-248 AT001-999 TC201-999	LH wheel well, stud protruding through the inboard keelson, near N.S. 100.
LH battery ground connection nose	T249-999	Stud protruding through the RH
		wheel well keelson, near F.S. 14.
APU receptacle connections	T201-248 AT001-999 TC201-999	RH nacelle, just below and FWD of wing ice light panel.
APU receptacle connections	T249-999	RH side of nose, just aft of the radome.
APU ground connection ing	T201-248	RH wheel well, on a stud protrud-
	AT001-999 TC201-999	through the outboard keelson.
APU ground connection	T249-999	Stud protruding through RH nose wheel well keelson, near F.S. 14.

Battery System
Table 202

ITEM/CONNECTION	EFFECTIVITY	LOCATION
Series parallel relay connections	T201-248 AT001-999 TC201-999	RH wheel well electrical panel, top center area.
Series parallel relay ground	T201-248 AT001-999 TC201-999	RH wheel well, on stud protruding through outboard keelson, at approximately N.S. 105.
Series parallel relay ground nose	T249-999	Stud protruding through the RH wheel well keelson, near F.S. 14.

Battery System
Table 202 (End)

ITEM/CONNECTION	EFFECTIVITY	LOCATION
J-Box behind pilot's seat tions.	All Models	All four (4) gauge wire termina- All bus bar connections. All current limiter connections.
Bus tie switch connections	All Models	LH essential bus tie switch connections. RH essential bus tie switch connections. Non-essential bus tie switch connections.
Bus connections	All Models	LH essential bus connections. RH essential bus connections. Non-essential bus connections.

Distribution Wiring
Table 203

ESSENTIAL/NON-ESSENTIAL BUS – DESCRIPTION AND OPERATION

1. General

A. A three-bus system distributes electrical power to the aircraft systems.

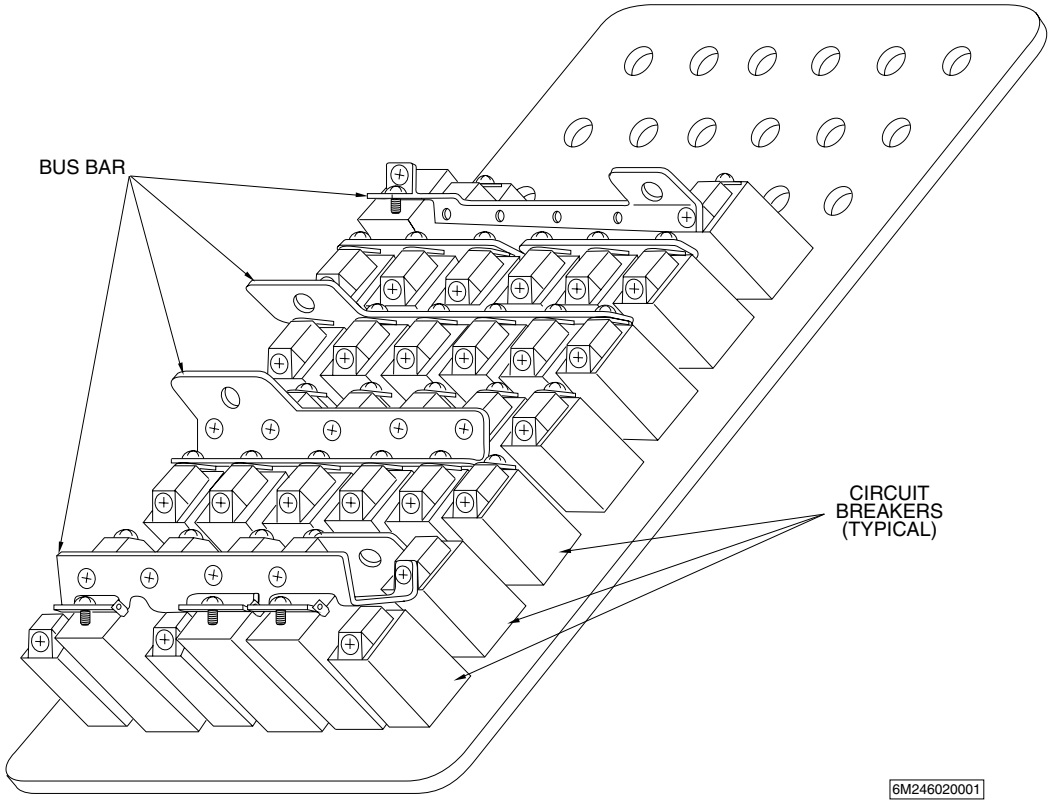
(1) Circuit breakers (CB's) and current limiters protect the buses and their sources against excessive current.

(2) The pilot can control the application or removal of power to any bus.

(3) Space is provided on the aft portion of the LH and RH consoles for avionics installation.

B. Power from each bus is distributed to the individual load systems through the use of small bus bars and CB's. CB's connected to the bus are ganged together by smaller bus bars and are finally connected to the bus lead.

C. Non-essential bus is connected to the battery bus through the 150A non-essential CB and the non- essential bus tie switch. This system design enables the pilot to control the application or removal of power to the non-essential bus.



Essential/Non-Essential Bus Circuit Breaker Installation (Typical)
Figure 1

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MAINTENANCE MANUAL

CHAPTER 25 – EQUIPMENT/FURNISHINGS

LIST OF EFFECTIVE PAGES

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25-EFF PGS	-	2	JUN 01/01	25-20-00	MP	201	JUN 01/01
25-CONTENTS	-	1	JUN 01/01	25-20-00	MP	202	JUN 01/01
25-CONTENTS	-	2	JUN 01/01	25-20-01	MP	201	JUN 01/01
25-00-00	DO	1	JUN 01/01	25-20-02	MP	201	JUN 01/01
25-10-00	DO	1	JUN 01/01	25-20-03	MP	201	OCT 01/06
25-10-00	MP	201	JUN 01/01	25-20-04	MP	201	JUN 01/01
25-10-00	MP	202	JUN 01/01	25-20-04	MP	202	JUN 01/01
25-10-01	MP	201	JUN 01/01	25-20-04	MP	203	JUN 01/01
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25-20-00	DO	2	JUN 01/01	25-30-00	DO	1	JUN 01/01
25-20-00	DO	3	JUN 01/01	25-30-00	DO	2	JUN 01/01
25-20-00	DO	4	JUN 01/01	25-40-00	DO	1	JUN 01/01
25-20-00	DO	5	JUN 01/01	25-40-00	DO	2	JUN 01/01

MAINTENANCE MANUAL

CHAPTER 25 – EQUIPMENT/FURNISHINGS

LIST OF EFFECTIVE PAGES

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MAINTENANCE MANUAL

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MAINTENANCE MANUAL

CHAPTER 25 – EQUIPMENT/FURNISHINGS

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DESCRIPTION AND OPERATION			AT	001-999
			TC	201-999

EQUIPMENT/FURNISHINGS – DESCRIPTION AND OPERATION

1. General

A. Equipment and furnishings described in this chapter are those items related to comfort, convenience, and safety of the passengers and crew. Basic equipment and furnishings of all SA226 series aircraft consists of crew seats mounted on seat tracks, passenger seats, movable bulkheads, snap-on carpeting, individual passenger reading lights, ash trays, heating outlets, and cold air (eye-ball) outlets.

B. Interior furnishings and fittings of the Merlin III, model SA226 – T, Merlin IV, model SA226 – AT, and Metro, model SA226 – TC presented in this chapter cover standard aircraft configuration, not counting for customized manufacturing possibility.

FLIGHT COMPARTMENT – DESCRIPTION AND OPERATION

1. General

A. The flight compartment has accommodation for dual flight controls, instruments, and electrical and electronic control panels conveniently located for crew accessibility. Upholstered seats are provided for the pilot and copilot. Seats are adjustable vertically and horizontally, and are removable. Each seat is provided with a lap type safety belt equipped with quick-release buckles. Each seat is mounted on two parallel tracks, bolted to the compartment floor panel which allows the seat to move forward or aft. The flight compartment is sound dampened and insulated with flame-resistant material. A curtain is provided to separate the flight compartment from the passenger compartment.

B. The Merlin III SA226 – T model aircraft contains a two-section sliding bulkhead door which separates the flight compartment from the passenger cabin.

1. Inspection/Check – Flight Compartment

A. Inspection – Flight Compartment

(1) Inspect pilot and copilot seats for damage, freedom of movement, proper operation of control levers, security of seat track installation, and cleanliness.

(2) Inspect fire extinguisher for security of mounting.

(3) Perform fire extinguisher weight check to ensure serviceability IAW chapter 26.

(4) Inspect safety belts for security of installation and wear.

(5) Inspect cockpit for cleanliness.

(6) Inspect optional emergency equipment for security of mounting and serviceability.

(7) Inspect curtain and attachments for security, wear, and cleanliness (SA226 – AT/TC).

(8) Inspect sliding bulkhead doors for cracks, warpage, and general condition (SA226 – T).

B. Inspection – Restraining Systems

NOTE: Periodic inspection of seat belts are required to ensure structural integrity of the crew restraining system. Perform the following procedures to determine if restraining systems are within acceptable wear limits.

(1) Inspect individual components of the restraining system for cracks, fractures, and functioning of movable parts. Remove defective systems from service.

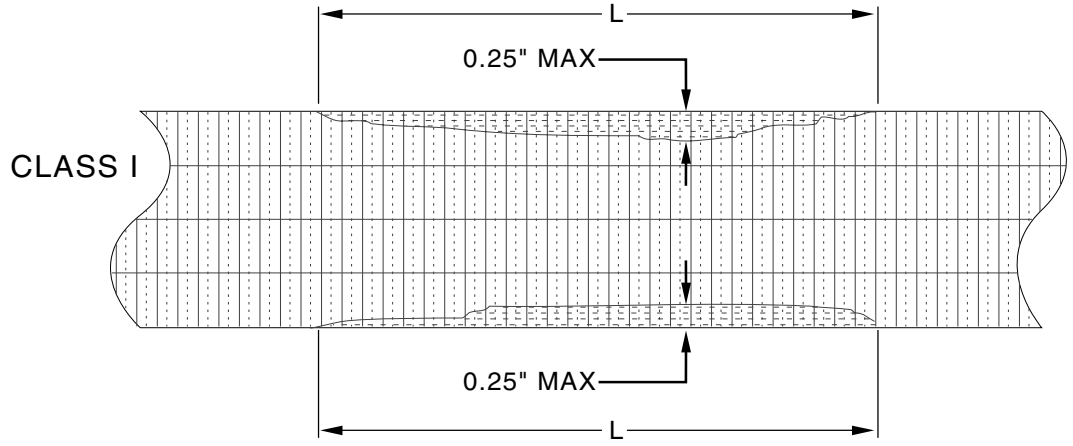
C. Inspection – Restraining Systems (Webbing Only)

(1) Visually inspect all seat belt webbing to determine what class of wear exists (see Figure 201 for explanation of wear classes).

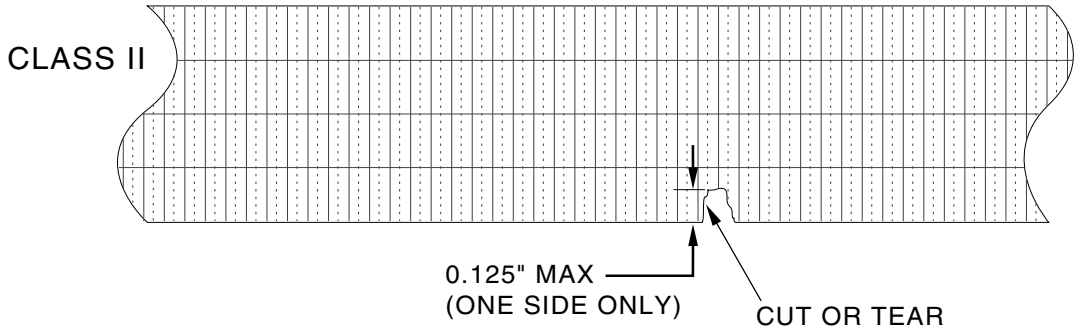
(2) If the webbing has fraying that is less than or equal to either class I or class II, the webbing is within acceptable wear limits.

(3) If the fraying is a combination of class I and class II or is greater than either class, replace or refurbish the webbing to original appearance and performance.

NOTE: Repair/Overhaul of webs must be performed by the manufacturer or by FAA authorized repair stations IAW FAR part 43. Repair and overhaul must conform to manufacturers quality assurance provisions. Each assembly must be 100% proof loaded.



L = LENGTH OF WORN AREA
(PERMISSIBLE TO BE ENTIRE LENGTH OF WEB)



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Webbing Wear
Figure 201

PILOT / COPILOT SEATS – MAINTENANCE PRACTICES

1. Removal/Installation – Pilot / Copilot Seats

A. Removal – Pilot / Copilot Seats

(1) Remove bolts from forward end of seat track.

(2) Remove clamp and bungee assembly from rear of seat.

(3) Slide seat forward approximately 10 inches.

(4) Lift seat upward and remove.

B. Installation – Pilot / Copilot Seats

(1) Position seat on track so that seat engages with notch in seat track.

(2) Slide seat aft and install clamp and bungee assembly.

(3) Insert and secure bolts on track forward of the seat.

COCKPIT ENCLOSURE – MAINTENANCE PRACTICES

1. Removal/Installation – Cockpit Enclosure

A. Removal – Curtain (SA226-AT / TC)

(1) Remove upper and lower stop pins from curtain track.

(2) Slide curtain from top to bottom and out of the track.

B. Installation – Curtain (SA226-AT / TC)

(1) Install curtain edges in track.

(2) Install stop pins.

C. Removal – Bulkhead Sliding Door (SA226-T)

(1) Enter flight compartment, and position pilot or copilot's seat all the way forward for improved working accessibility to bulkhead.

(2) From flight compartment side, remove six wood screws from face of outer bulkhead panel (two screws from center of panel and four screws from bottom edge of panel.

(3) Separate outer bulkhead panel from inner panel.

(4) Remove attaching screws from metal brackets at top and bottom of inner bulkhead panel.

(5) Lift out inner bulkhead panel and sliding door assembly.

D. Installation – Bulkhead Sliding Door (SA226-T)

(1) If bulkhead and sliding door are fully assembled, separate bulkhead halves by removing six wood screws securing outer panel to inner panel. (This leaves sliding door exposed which is attached to inner bulkhead panel).

(2) Position inner bulkhead panel and sliding door assembly into place from flight compartment side behind pilot's or copilot's seat.

(3) Install and secure screws in bulkhead attaching brackets at top and bottom of panel.

(4) Position outer bulkhead panel against inner panel, install and secure six wood screws attaching outer panel to inner panel.

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PASSENGER COMPARTMENT – DESCRIPTION AND OPERATION

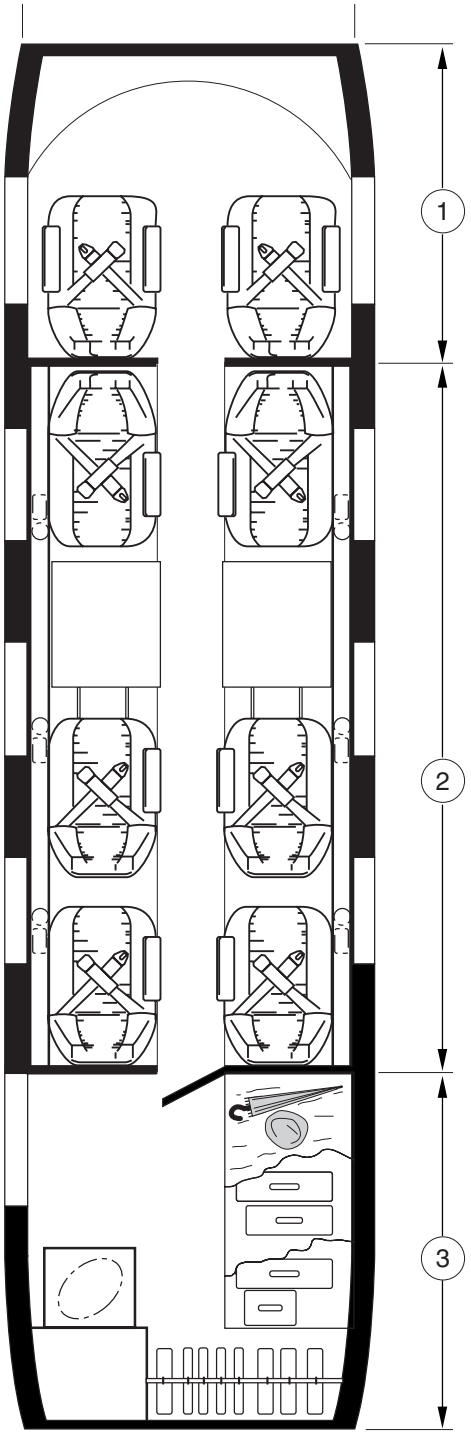
1. General

A. The passenger compartment extends from the sliding bulkhead door at the flight compartment to the bulkhead aft of the entrance door.

B. The Merlin III model SA226-T aircraft has six passenger chairs with reclining backs and folding armrests; three chairs on each side of the cabin aisle (see Figure 1 for floor plan). The front chair on each side of the aisle is installed facing rearward to accommodate a built in card table between the first and second chairs. The card table is hinged at the cabin wall, and may be lifted upward and stowed into a stowage area along the wall as desired. The second chair on each side of the aisle is mounted on tracks to provide fore and aft adjustment as desired by the passenger.

C. Curtains are provided on each cabin window for passenger comfort. A hinged door separates the passengers compartment from the entrance vestibule, aft baggage compartment, galley cabinet, and pull-out lavatory.

PASSENGER COMPARTMENT – DESCRIPTION AND OPERATION



1. COCKPIT AREA
   a. Two adjustable chairs with folding armrests.
   b. Ash trays.
   c. Cockpit bulkhead with sliding doors.
   d. Cold air outlets - one for each crew seat.
   e. Hot air outlets and foot warmers at each crew seat.
   f. Oxygen outlets at each crew seat.
2. CABIN AREA
   a. Four fixed-mounted chairs with reclining backs, folding armrests.
   b. Two track-mounted chairs with reclining backs. (When chairs are pulled to forward position and backs are reclined, they form excellent beds).
      NOTE: All chairs are fabric covered trimmed in English leather (Connally Bros.) for the standard configuration. Optionally all leather covered chairs are available. Two center and two forward chairs are equipped with removable head rests.
   c. Two card tables.
   d. Reading lights at each seat location.
   e. Drink holder at each seat location.
   f. Ash tray at each seat location.
   g. Cold air outlet at each seat location.
   h. Oxygen outlet at each seat location.
   i. Window blinds.
   j. Floor lights.
   k. Hot air outlets at floor.
   l. Rear cabin bulkhead and door separating cabin from vestibule.
3. VESTIBULE AREA
   a. Overhead lights (entrance and vestibule lighting).
   b. Three-shelf luggage rack with web retaining straps.
   c. Coat closet also suitable for golf clubs.
   d. Bar galley cabinet provisioned for two half-gallon jugs (hot/cold), four liquor decanters, ice bucket, and soft drinks, and liquid drain (equipment not included). General stowage for gum, napkins, ect.
   e. Pull out lavatory which hides away while not in use.

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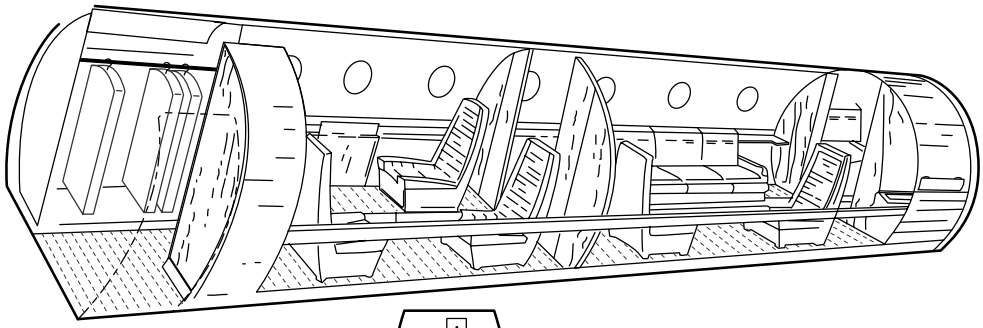
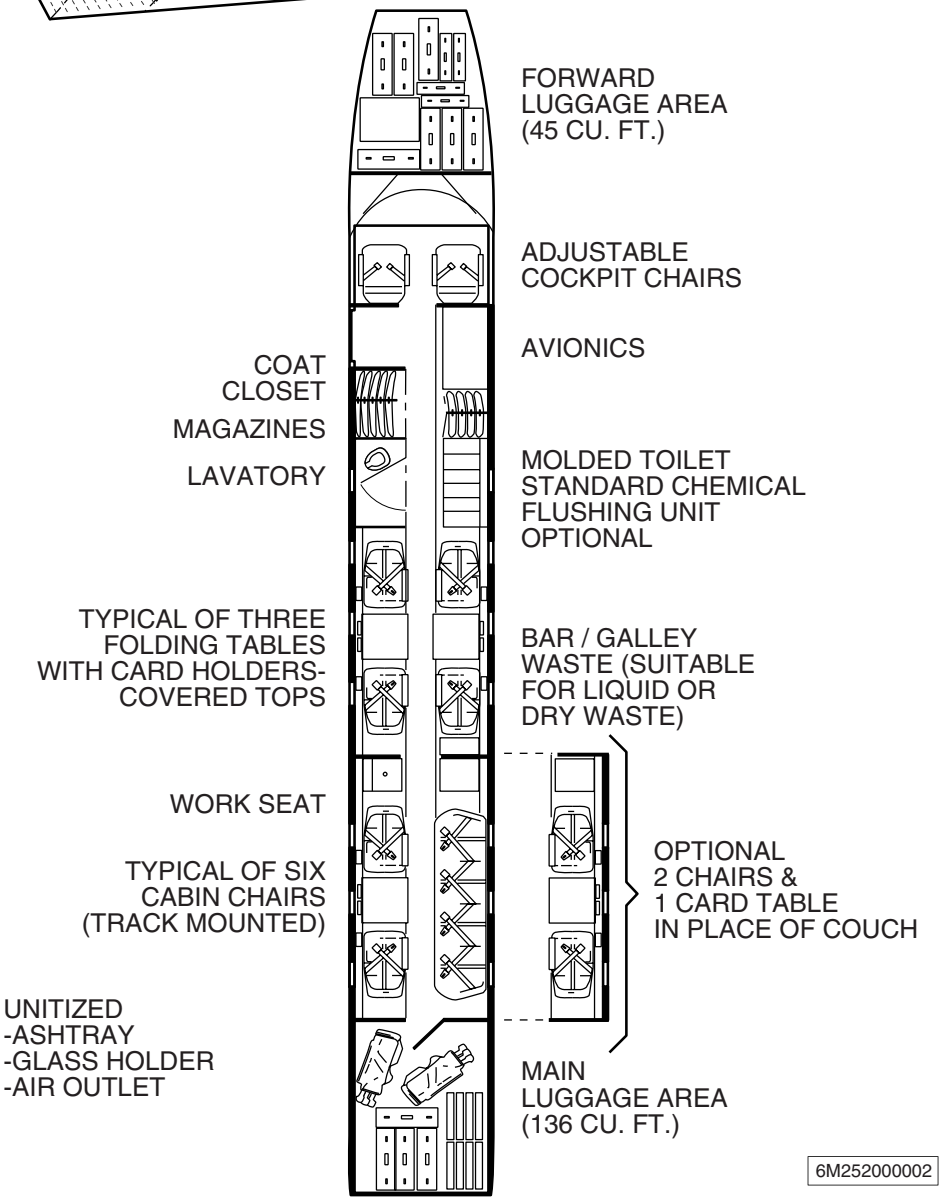
Floor Plan (SA226-T) Figure 1

PASSENGER COMPARTMENT – DESCRIPTION AND OPERATION

D. The Merlin IVC model SA226-AT aircraft is fitted for a 10 passenger executive transport configuration. The passenger compartment extends from the front entrance door through the aft baggage compartment (see Figure 2). The passenger cabin contains six chairs with reclining backs and folding armrests. Three chairs face rearward and three face forward with a folding card table between the forward and rearward facing chairs. A four-place divan is installed along the right rear wall of the cabin.

E. The passenger area is divided into two compartments separated by a bulkhead and optional hinged door. The forward compartment seats four passengers, the aft compartment seats six, including the four-place divan. A toilet compartment is provided and is separated from the forward passenger compartment by a bulkhead with hinged doors. A baggage compartment is provided at the aft end of the cabin separated from the passenger compartment by a bulkhead and hinged door.

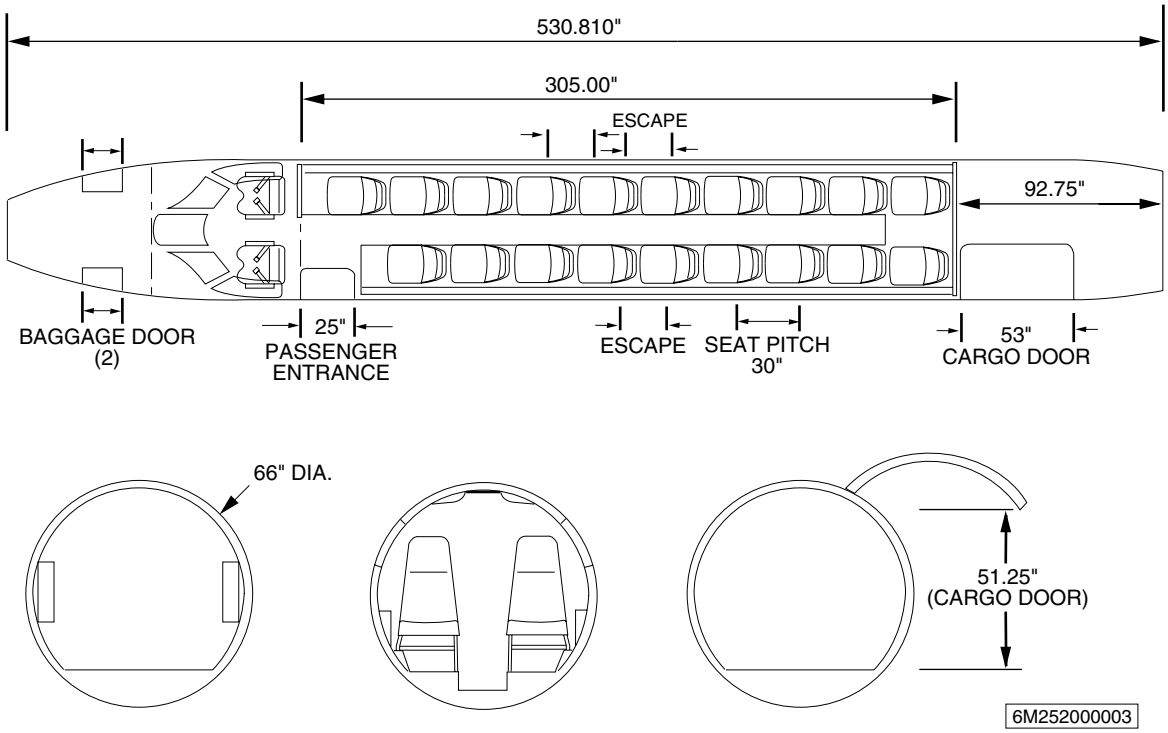




Floor Plan
(SA226-AT)
Figure 2

PASSENGER COMPARTMENT – DESCRIPTION AND OPERATION

F. The Metro model SA226-TC aircraft contains 19 seats in the passenger compartment. The seat backs are contoured and fabricated from durable material formed in place with an energy-absorbent foam material. The seats are folded outboard against the bulkhead when conversion of the passenger compartment into a cargo compartment is desirable. Each seat contains folding armrests and reclining back, and a pocket in the back for literature and passenger emergency instructions (see Figure 3 for a passenger compartment floor plan). The passenger compartment is convertible, in 30-inch increments, from an all passenger configuration to all cargo, or any mixed configuration desired without removing the seats. A movable bulkhead separates the passengers from the cargo.



Floor Plan
(SA226-TC)
Figure 3

G. The interior trim in the passenger compartment is divided into several sections such as the carpeting, bulkhead panels, window trim panels, and sidewall panels. All panels are made primarily of synthetic materials that are light, durable, and easy to maintain

H. Carpeting

(1) The passenger compartment floors are covered either with wool or vinyl, snap-on carpeting, installed over one-quarter inch polyurethane (spongy, synthetic material) padding.

I. Window Trim Panels

(1) The window trim panels are preformed, tedlar plastic material of doublewall construction with insulation material between the panel and the wall. The window trim panels are screw mounted.

J. Sidewall Panels

(1) The trim panels above the windows extend up to the ceiling. These panels are finished in tedlar plastic and butt joined at the ceiling and the side.

K. Insulation

(1) The insulation between the fuselage skin and the panels is of bagged fiberglass batting. A 3M cement is used to hold the insulation in place while the panels are being installed.

2. Passenger Warning Lights

A. The passenger warning lights consist of the FASTEN SEAT BELTS – NO SMOKING sign located above the cockpit bulkhead. The sign lighting is controlled by a switch on the copilot's switch panel.

PASSENGER COMPARTMENT – MAINTENANCE PRACTICES

1. Inspection/Check – Passenger Compartment

A. Inspection – Passenger Compartment

(1) Inspect passenger seats for serviceability, smooth operation, security, structural soundness, cleanliness, tears, and corrosion.

(2) Inspect safety belts for security, wear, and cleanliness.

(3) Inspect panels, flooring, and carpeting for security, cleanliness, tears, and any irregularities.

(4) Inspect optional emergency equipment for security, serviceability, and cleanliness.

(5) Inspect windows for cleanliness, defects, and scratches.

(6) Inspect movable bulkhead and components for security, operation, and cleanliness.

(7) Inspect lights, air outlets, and ashtrays for cleanliness, serviceability, and security.

(8) Inspect (if installed) refreshment galley, equipment/storage cabinet, or refreshment cabinet for security of accessories, doors, and latches for proper operation.

(9) Inspect all passenger and service doors to insure jamb areas are clean and free of debris, and that doors will lock and unlock securely.

B. Inspection – Restraining Systems

NOTE: Periodic inspection of seat belts are required to ensure structural integrity of the passenger restraining system. Perform the following procedures to determine if restraining systems are within acceptable wear limits.

(1) Inspect individual components of the restraining system for cracks, fractures, and functioning of movable parts. Remove defective systems from service.

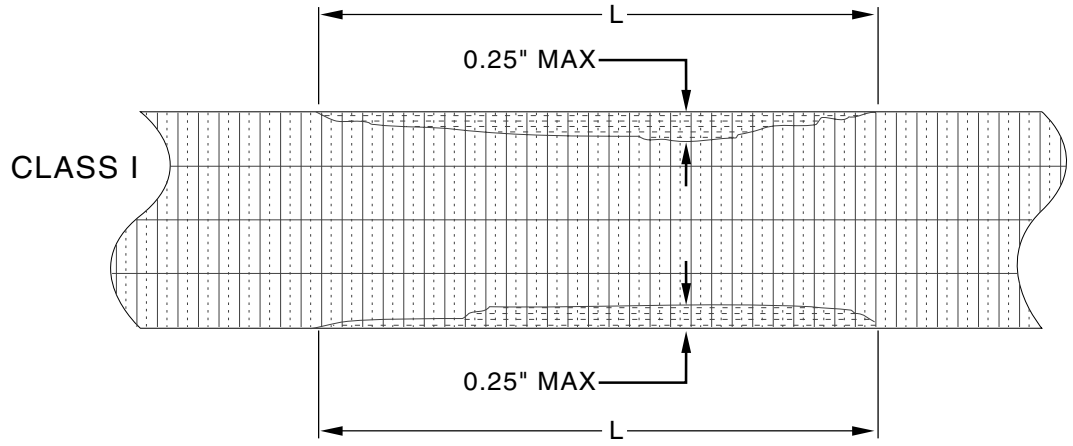
C. Inspection – Restraining Systems (Webbing Only)

(1) Visually inspect all seat belt webbing to determine what class of wear exists (see Figure 201 for explanation of wear classes).

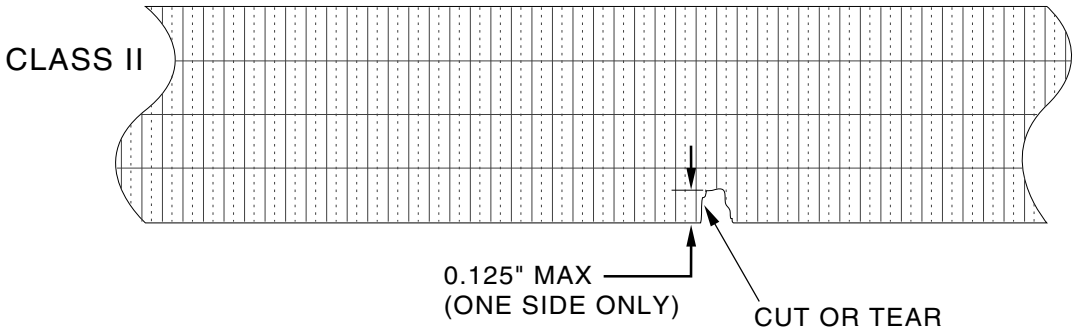
(2) If the webbing has fraying that is less than or equal to either class I or class II, the webbing is within acceptable wear limits.

(3) If the fraying is a combination of class I and class II or is greater than either class, replace or refurbish the webbing to original appearance and performance.

NOTE: Repair/Overhaul of webs must be performed by the manufacturer or by FAA authorized repair stations IAW FAR part 43. Repair and overhaul must conform to manufacturers quality assurance provisions. Each assembly must be 100% proof loaded.



L = LENGTH OF WORN AREA
(PERMISSIBLE TO BE ENTIRE LENGTH OF WEB)



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Webbing Wear
Figure 201

PASSENGER SEATS – MAINTENANCE PRACTICES

1. Removal/Installation – Passenger Seats

A. Removal – Passenger Seats

(1) Push in on lock pins and raise stud fittings from chair anchors.

(2) Remove bolt from chair hinge at fuselage side.

(3) Remove seat.

B. Installation – Passenger Seats

(1) Align chair hinges in bracket, install bolt and secure.

(2) Align chair legs over anchor plates.

(3) Press stud fitting into place. Insert collar with spring-loaded lock pin.

CARPETING – MAINTENANCE PRACTICES

1. Removal/Installation – Carpeting

A. Removal – Carpeting

(1) Stow seats against sides of aircraft (remove from SA226-T aircraft).

(2) Unsnap the fasteners securing the carpeting to the floorboards.

(3) Remove carpeting.

B. Installation – Carpeting

(1) Align carpeting to match the fasteners.

(2) Snap in place.

(3) Check to assure security and uniformity.

(4) Position and attach seats (install in SA226-T aircraft).

2. Cleaning / Painting – Carpeting

A. Materials

(1) Vacuum cleaner.

(2) Mild detergent and water.

(3) Approved grade commercial carpet cleaner.

(4) Fiber bristle brush.

B. Cleaning

NOTE: Depending on type of material and soilage, use approved materials to clean carpeting (SA226-T contains vinyl runners).

(1) Vacuum carpet thoroughly.

(2) Use appropriate agent and clean carpet thoroughly.

(3) Vacuum again as necessary.

INSULATION – MAINTENANCE PRACTICES

1. Removal / Installation – Insulation

A. Removal – Insulation

(1) Remove panels.

(2) Remove insulation by sections.

(3) Use a cloth dampened in solvent or methylpropylketone to remove old sealant and to prepare area for installation.

WARNING: SOLVENT AND METHYLPROPYLKETONE ARE HAZARDOUS WHEN INHALED OR LEFT IN CONTACT WITH SKIN. USE PROTECTIVE PERSONAL EQUIPMENT AND WASH BODY AREA WITH SOAP AND WATER IF SPILLAGE ON THE SKIN OCCURS.

B. Installation – Insulation

(1) Trim insulation material to fit.

(2) Apply thin coat or small spots of 3M Sealant to insulation backing.

(3) Press insulation into place.

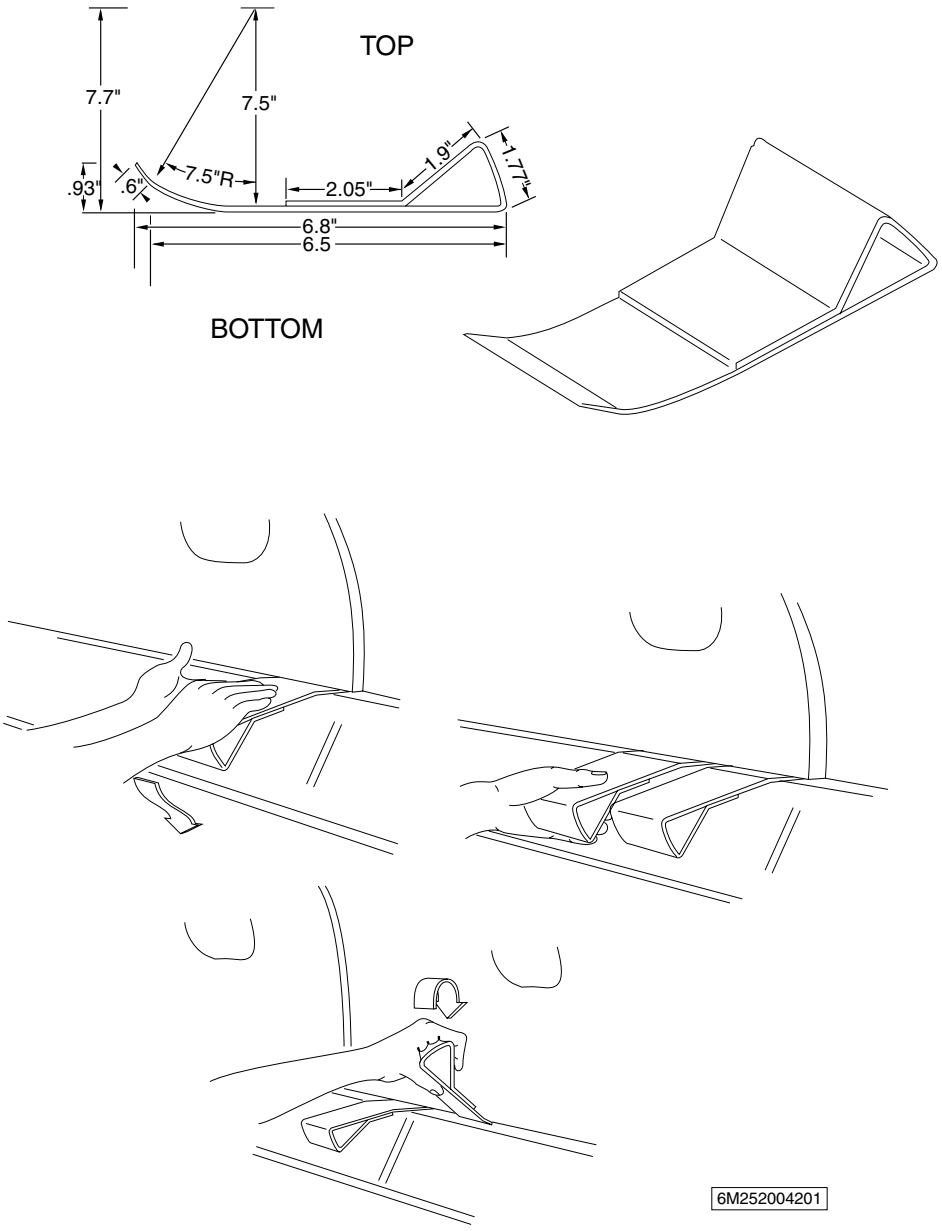
PASSENGER COMPARTMENT HEADLINER – MAINTENANCE PRACTICES

1. Removal / Installation – Passenger Compartment Headliner

A. Removal – Passenger Compartment Headliner (Figure 201)

NOTE: For convenience and to protect the headliner, use a headliner installation tool (two required) manufactured to dimensions in Figure 201.

Blade of tool is 3 inches wide, and fabricated from 0.060 stainless steel or 2024-T6 aluminum sheet.



Headliner Installation Tool
Figure 201

(1) Slide one of the headliner installation tools, under a bottom corner edge of the headliner.

(2) Slide the other tool under the headliner edge next to the first one.

(3) With a forward movement of the outside tool, pop the headliner out.

(4) Remove the outside tool and place it under the panel edge forward of the second tool.

(5) Pop the headliner out with second tool.

(6) Continue the process by switching the tools along the bottom edge of the panel until the panel is free.

(7) Remove panel.

(8) Unhook sceet hoses.

B. Installation – Passenger Compartment Headliner (Figure 201)

(1) Hook up sceet hoses.

(2) Position the headliner in place.

(3) Use a headliner tool, to pop the headliner into position at one corner.

(4) Position the second headliner tool next to the first and pop the headliner into place.

(5) Continue the process using both tools until the headliner in entirely seated along its lower edge.

2. Application of Water Borne Topcoats

A. Materials Required

(1) Coatings

(a) Interplan 750 Primer (aluminum only) (HSH) or equivalent.

(b) Interplan 1065B Topcoat (HSH) or equivalent.

(2) Spray Gun

(a) GFG-616-43 EX (DeVilbiss) or equivalent.

(b) JGHV-5338 FF (DeVilbiss) or equivalent.

(3) Cleaners

(a) Interplan 150AL (aluminum only) (HSH) or equivalent.

(4) Fillers/Additives

(a) Interplan 100SQ (HSH) or equivalent.

(b) Interplan 100SP (HSH) or equivalent.

(c) Interplan 100AL (aluminum filler) (HSH) or equivalent.

(7) Cleaning Cloths

(a) No. 42 Cotton Wipers (Kamen) or equivalent.

(b) Golden Tan Tack Cloth (MWB Ind.) or equivalent.

(8) Gloves

(a) Chem-Pro #224 (Briggs-Weaver) or equivalent.

B. Personnel Requirements

(1) All personnel performing coating operations shall be trained IAW the following:

(a) Ability to properly measure, dispense and mix the components, and obtain a sprayable admix.

(b) Ability to properly set spray gun controls and prepare spray gun for production use for both conventional spraying, and textured application.

(c) Ability to apply any one of the approved coatings IAW acceptance criteria.

(d) Ability to mix and apply fillers to obtain an acceptable edge or surface finish.

C. Facilities

(1) Perform all coating operations in a well ventilated area. If ventilation level is uncertain, use breathing apparatus as specified by the Safety department.

(2) Suspend all coating operations whenever the environment exceeds the limits listed in Table 201.

TEMPERATURE	HUMIDITY
55°F to 95°F (12.78°C to 35°C)	20% to 70% Recommended, 80% Max.

Environmental Spraying Limitations
Table 201

(3) Ensure the compressed air supply system is capable of supplying air that is clean, dry, and free from visible oil and water. Check filters and traps daily and maintain on a regular basis.

(4) Ensure areas where coatings are being applied are free from sources of ignition such as pilot lights, sparking electrical equipment, etc.

D. Storage

(1) Store all coating components indoors at temperatures between 40°F and 100°F (4.44°C and 37.78°C).

(2) Shelf life of coatings and primers is a maximum of twelve (12) months from date of manufacture. Discard components older than twelve (12) months.

E. Color

(1) Standard interior color shall be FED-STD-595 No. 27722.

(2) Custom colors require a FED-STD-595 number, an internal customer color number that has already been color matched, or a sample for color matching.

F. Cleaning

NOTE: Ensure all surfaces to be coated are clean and free from dirt, grease, silicones, fingerprints, or any other contaminates that can affect adhesion of the coatings.

(1) For new surfaces, lightly abrade the surface with a very fine nylon abrasive pad to promote adhesion, and wipe surface with Interplan 150AL cleaner.

(2) For previously coated surfaces, or if contamination is present, use Interplan 150AL cleaner with the abrasive pad to loosen and remove the contamination.

(3) Remove cleaner residue with #42 cotton cloth wet with tap water; immediately wipe dry with a clean cloth.

G. Priming

NOTE: Primers are required only on metal surfaces, but may be used on composites and plastics as necessary to promote adhesion.

Solvents may be used on metals as necessary to remove contamination, but shall be removed by subsequent cleaning with Interplan 150AL cleaner and shall never be used as final cleaner.

(1) Thin primer with water as necessary to produce a spraying viscosity as listed in Table 202.

(2) Apply primer in a thin first coat (about 0.75 mils), with a cross coat following in fifteen to thirty (15 to 30) minutes, as necessary to achieve full hide.

(3) Allow primer to dry at least one (1) hour before applying topcoat.

H. Topcoat

(1) Thin topcoat with tap, distilled, or deionized water to meet viscosity listed in Table 202. Any additives or special preparation steps shall be IAW manufacturers instructions.

Coating	Viscosity	Dry To Touch	Full Cure		Cup Pressure
			Ambient	Accelerated	
Interplan 1065B	30-40 Sec #2 Zahn	5-10 Minutes	72 Hours	30 Min. at 160 - 180°F (71.11 - 82.22°C)	6-10 psi (0.42-0.70 kg/cm^2)

Coating Parameters
Table 202

CAUTION: WATER BORNE COATINGS MAY APPEAR THICKER THAN EQUIVALENT SOLVENT BASED COATINGS, THEREFORE, FOLLOW THE VISCOSITY REQUIREMENTS AND DO NOT THIN BY APPEARANCE AS OVER THINNING OF THE TOPCOAT WILL REDUCE PRODUCT PERFORMANCE.

(2) If no texture is required, apply topcoat in a thin first coat (about 0.75 mils), with a cross coat following in fifteen to thirty (15 to 30) minutes, as necessary to achieve full color and hide.

(3) When a "leather" effect texture is required, apply the first coat using a spray gun set for a spatter effect IAW instructions in **2.**, H. Apply the second cross coat, if necessary, with another gun, or with the same gun reset for smooth surface application.

(4) Ensure parts are "dry to touch" prior to handling.

(5) Ensure parts are fully cured (see Table 202) prior to being exposed to weather, or damaging fluids, and prior to being put into service.

I. Application of Texture

NOTE: Application of a textured "leather look" finish is achieved by spattering on the coating to get the rough surface. Since application is predominately "technique", the following instructions are only a starting point for operators, who must use trial and error to find the proper adjustments for them.

(1) Prepare a workmanship sample for comparison with the applied finish. Always compare the finished panel to the workmanship sample and try to maintain the identical texture.

(2) Apply texture using a conventional spray gun. An HVLP gun and a pressure pot is recommended. The same nozzles and metering may be used as with conventional spraying.

(3) Adjust the pot pressure a little higher than normal to supply more paint than normal.

(4) Adjust the atomization pressure lower than normal, until the paint begins to spatter.

(5) Mix the coating on the thicker side of the recommended viscosity and avoid any undue thinning of the coating.

(6) A little greater distance is recommended than with conventional spraying, but find a distance that works and keep it very constant. This will keep the finished surface consistent in texture.

CAUTION: LIMIT MULTIPLE LAYER THICKNESS TO ABOUT ONE EIGHTH (1/8) INCH TO AVOID CRACKING.

(7) Ensure the size and density of the raised texture is consistent. If too rough, follow with a light overcoat to smooth the texture. If too shallow, a second application of spatter may be necessary.

J. Damaged Surface Repair

NOTE: Minor dents and dings may only require a single application of filler, while larger damaged areas will require multiple applications.

(1) Apply a spreadable type filler using a spatula or trowel. Ensure thickness per layer does not exceed 0.010 inch (0.254 mm) for IP 100SQ and 100BR.

(2) Build up damaged area with several thin applications, dried between layers, rather than a single heavy layer. This will provide faster dry time and superior performance.

NOTE: Fillers and Primer/Fillers tend to settle due to their high solids content. Always mix Interplan products thoroughly prior to use.

Interplan fillers may be mixed together to achieve virtually any viscosity. For example if you need IP 100SP with increased spreadability, add some IP 100BR or IP 100SQ. Be sure to mix thoroughly to insure uniform consistency.

(3) Allow to dry at least sixty (60) minutes.

NOTE: Warm rooms, heaters, or heat lamps may be used to accelerate filler or coating drying times. Do not allow temperatures to exceed 140°F (60°C). Best results are obtained by allowing moisture to "flash off" at room temperature for fifteen to thirty (15 to 30) minutes before force drying at elevated temperatures.

(4) Sand surface to desired contour using 120 to 320 grit aluminum oxide sandpaper.

(5) Remove sanding residue with a Golden Tan tack cloth.

(6) Repeat **2.**, I. (1) through (5) as necessary.

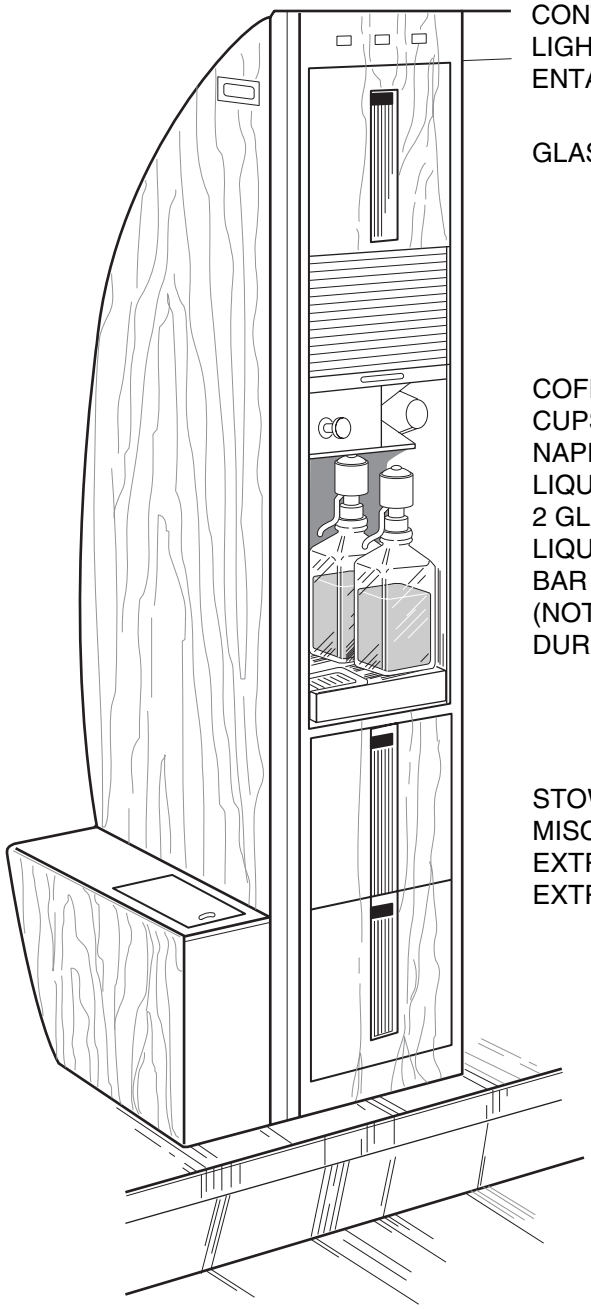
BUFFET / GALLEY – DESCRIPTION AND OPERATION

1. General

A. The Merlin IV model SA226-AT and Merlin III model SA226-T aircraft contain a food and beverage refreshment bar and stowage area suitable for glassware, heated and cold beverages. Provisions are included for ice and miscellaneous supplies. A waste container suitable for liquid or dry waste is attached to the refreshment bar (see Figure 1).

B. The refreshment bar on the Merlin IV model SA226-AT aircraft is located just aft of the entrance door in the forward passenger compartment. The heated beverage and glass cabinet slide out during use.

C. The Merlin III model SA226-T aircraft incorporates the refreshment bar in the aft end of the passenger cabin adjacent to the entrance door. The galley cabinet is in the entrance vestibule along with the aft baggage compartment.



Refreshment Bar
(SA226-T / AT)
Figure 1

LAVATORIES – DESCRIPTION AND OPERATION

1. General

A. The Merlin III model SA226-T and Merlin IV model SA226-AT aircraft incorporates a slide out toilet and lavatory. These provisions are stowed away from sight into a cabinet when not in use. A standard chemical flushing unit is optionally included as desired.

B. The Merlin III model SA226-T aircraft incorporates the slide out toilet and lavatory in the aft cabin vestibule. The vestibule area is separated from the main passenger cabin by a rear cabin bulkhead and door.

C. The Merlin IV model SA226-AT aircraft incorporates the lavatory and slide out toilet just aft of the forward main entrance door. These provisions are in the same general area with the coat closet and separated from the forward passenger compartment by the forward bulkhead and hinged door.

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MAINTENANCE MANUAL

CHAPTER 26 – FIRE PROTECTION

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26-20-20	MP	202	DEC 01/01
26-20-20	MP	203	DEC 01/01
26-20-30	DO	1	DEC 01/01
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MAINTENANCE MANUAL

CHAPTER 26 – FIRE PROTECTION

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CHAPTER 26 – FIRE PROTECTION

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FIRE PROTECTION – DESCRIPTION AND OPERATION

1. General

A. The fire protection systems consist of:

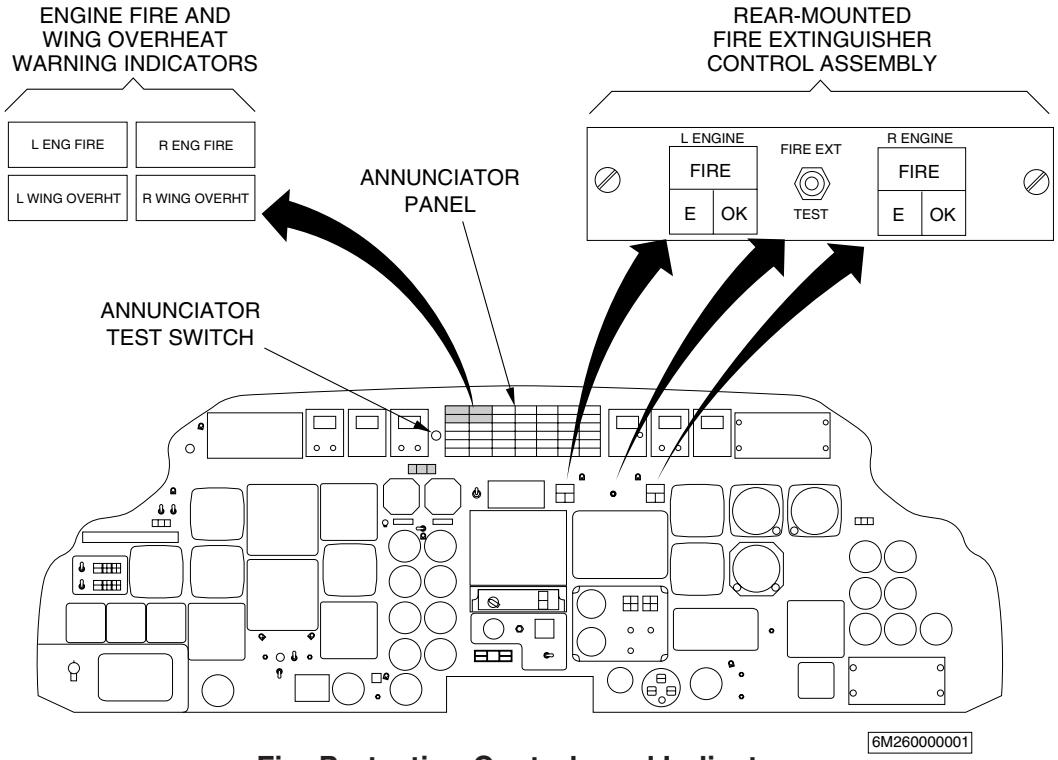
(1) Engine fire detection.

(2) Wing overheat detection.

(3) Engine fire extinguisher.

(4) Portable fire extinguishers.

B. Fire protection controls and indicators (Figure 1) are incorporated in the annunciator panel and instrument panel rear-mounted fire extinguisher control assembly.



Fire Protection Controls and Indicators
Figure 1

2. Engine Fire and Wing Overheat Detection

A. The engine fire detectors and wing overheat thermal switches are strategically located in the engine and wing areas, respectively.

(1) In the event rated temperatures of the various fire detectors and thermal switches are exceeded, fire detectors and thermal switches act as electrical switches to activate and control engine fire and wing overheat warning lights located on the annunciator panel.

(2) Engine fire warning lights on the annunciator panel are identified as follows:

(a) L ENG FIRE.

(b) R ENG FIRE.

(3) Wing overheat warning lights on the annunciator panel are identified as follows:

(a) L WING OVHT.

(b) R WING OVHT.

(4) An annunciator test switch on the annunciator panel confirms continuity of the engine fire and wing overheat detectors, and illuminates the lights on the annunciator panel and the fire lights on the instrument panel side of the fire extinguisher control assembly.

3. Engine Fire Extinguisher System

A. An optional engine fire extinguisher system consists of one 86 cubic inch (218.44 cc) bottle containing bromotrifluoromethane, CBrF3, located on FWD right side of each nacelle.

(1) Fire extinguishing agent is routed from the fire extinguisher bottle through the engine firewall to affected engine area through stainless steel tubes.

(2) Stainless steel tubing is routed from a discharge port to direct fire extinguishing agent onto the following:

(a) Fuel control.

(b) Starter-generator.

(c) Oil cooler.

B. A pressure gauge is installed on each fire extinguisher bottle to indicate internal pressure.

C. Each fire extinguisher bottle is equipped with a thermal relief valve to enable excess internal pressure to vent overboard.

4. Engine Fire and Wing Overheat Controls and Indicators

A. An instrument panel rear-mounted engine fire extinguisher control assembly is installed to provide:

(1) Fire extinguisher discharge control.

(2) Fire extinguisher charge status on SA226 T398, AT075 , and TC398 and subsequent.

(3) System continuity is confirmed when L and R ENG FIRE lights on annunciator panel illuminate.

B. The wing overheat indicating system consists of 250°F, 350°F, and 450°F thermal switches, in conjunction with L and R wing flashers, resistors, and wing overheat relay. L and R WING OVHT lights on the annunciator panel function as indicators for wing overheat.

5. Portable Fire Extinguishers

A. Two hand-held portable fire extinguishers are provided in the event of cabin and/or cockpit fire emergencies.

(1) One fire extinguisher is located in the cockpit area, aft and to the right of the copilot's seat.

(2) A second fire extinguisher is mounted just aft of the passenger door.

(3) Specific locations of both fire extinguishers vary somewhat, depending on particular aircraft. Both are liquid bromodifluoromethane, CBrCLF2, nitrogen-charged bottles.

ENGINE FIRE DETECTION – DESCRIPTION AND OPERATION

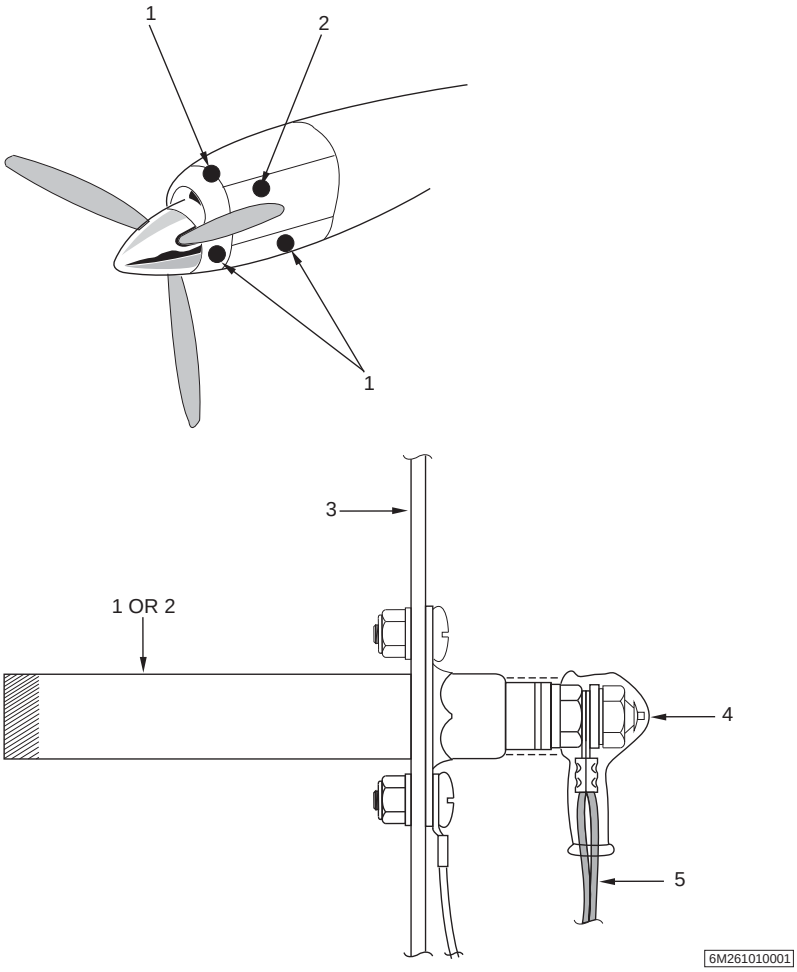
1. General

A. Engine Fire Detection consists of four fire detectors (Figure 1) located in each engine compartment.

(1) Three fire detectors (item 1) are rated at 450°F. One fire detector (item 2) is rated at 600°F.

(2) If any of the detectors detect a temperature equal to or greater than its rated value, a signal will be transmitted through system wiring (item 5) and associated circuitry to the fire extinguisher control panel. This signal illuminates the L and /or R red ENGINE FIRE light on the face side of the fire extinguisher control assembly on the instrument panel, and the corresponding L and/or R ENG FIRE lights on the annunciator panel.

(3) Each fire detector is secured to a mount bracket (item 3) and the electrical connection hardware of each is protected with a removable rubber boot (item 4). The mount brackets are secured to the engine, engine truss, or fire wall.



Engine fire Warning Sensor Location
Figure 1

1. Removal/Installation - Engine Fire Detection

A. Removal – Fire Detectors (Figure 201)

(1) Ensure that left and right battery switches (item 2) are OFF.

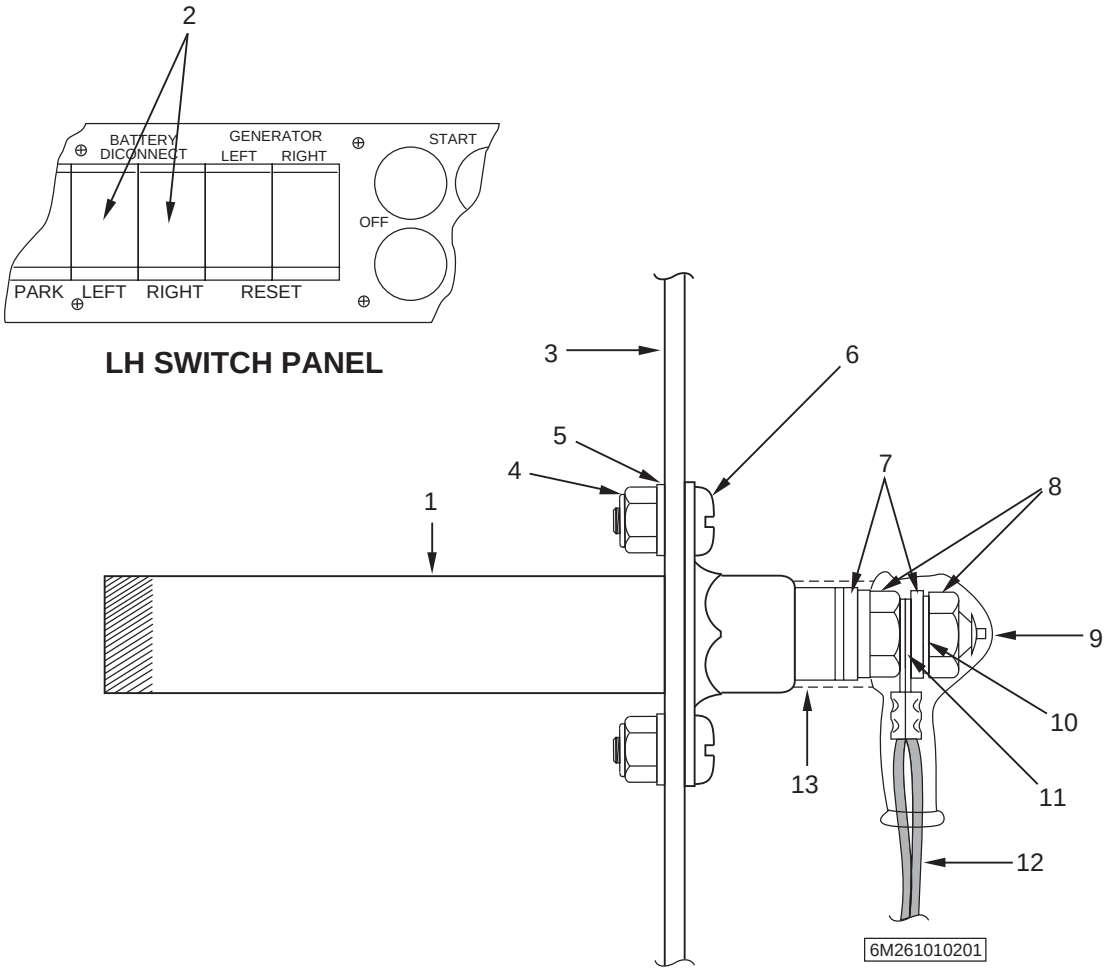
(2) Open engine cowling and locate engine fire detector (item 1) to be removed. Locations of the four (4) fire detectors are as follows:

(a) Fire detector No. 1 (450°F), left forward side of engine, directly below starter-generator.

(b) Fire detector No. 2 (450°F), lower left side of engine, attached to oil cooler inlet duct.

(c) Fire detector No. 3 (600°F), left forward-center of engine near unfeathering pump.

(d) Fire detector No. 4 (450°F), forward upper-center of engine, forward of aft intake heat valve.



Typical Engine Fire Detector
Figure 201

(3) Remove rubber boot (item 9) from terminal end of fire detector and slide downward over fire detector wires (item 12) approximately one (1) inch (2.54 cm), exposing attaching hardware on terminal end of fire detector.

CAUTION: EXERCISE CARE NOT TO DAMAGE CERAMIC PORTION OF FIRE DETECTOR WHEN REMOVING SEALER.

(4) Remove Dow Corning RTV 734 sealer (item 13).

CAUTION: EXERCISE CARE DURING REMOVAL OF ENGINE FIRE DETECTOR. DROPPING OR BUMPING FIRE DETECTOR COULD RESULT IN CRACKING OF CERAMIC PORTION.

(5) Remove nuts (item 8), flat washers (item 7), lock washer (item 10), and wire terminals (item 11) from each engine fire detector to be removed. See Figure 1 for locations of the fire detectors.

(6) Remove screws (item 6), nuts (item 4), and washers (item 5) from mount bracket (item 3) and remove fire detector.

(7) Check condition of fire detector wires (item 12) and wire terminals (item 11) and replace if necessary.

B. Installation – Fire Detector (Figure 201)

(1) Ensure that left and right battery switches are OFF.

WARNING: METHYL PROPYL KETONE (MPK) IS TOXIC, FLAMMABLE, AND CAUSTIC. ISOPROPYL ALCOHOL IS TOXIC AND FLAMMABLE. AVOID OPEN FLAMES AND SPARKS. USE ONLY IN A WELL-VENTILATED AREA. AVOID EYE AND SKIN CONTACT OR BREATHING OF VAPORS. PERSONAL PROTECTIVE EQUIPMENT CONSISTING OF RESPIRATOR, INDUSTRIAL GOGGLES, AND RUBBER GLOVES IS REQUIRED.

(2) Clean electrical connection and ceramic area of fire detector (item 1) with MPK or isopropyl alcohol, using a small brush and/or clean, moistened lint-free cloth.

(3) Apply an even coat of RTV 734 sealant to completely cover ceramic portion of detector.

CAUTION: EXERCISE CARE DURING INSTALLATION OF ENGINE FIRE DETECTOR IN MOUNT BRACKET. DROPPING OR BUMPING FIRE DETECTOR COULD RESULT IN CRACKING OF CERAMIC PORTION.

(4) Position fire detector into mount bracket (item 3) and secure with screws (item 6), nuts (item 4), and washers (item 5).

(5) Install flat washers (item 7), lockwasher (item 10), and nuts (item 8) onto threaded terminal post of each fire detector.

(6) Slide each rubber boot (item 9) to end of fire detector wires (item 12) and position in place over electrical connection.

(7) Wipe electrical bond area of each fire detector with clean lint-free cloth as follow-up measure to cleaning of electrical bond area.

(8) Apply Dow Corning RTV 734 sealer (item 13) to entire electrical bond area between boot and body of each fire detector.

(9) Close and secure engine cowling.

2. Adjustment/Test – Engine Fire Detection

A. Functional Test – Fire Detectors

(1) Perform functional check after removal/installation of fire detector(s) as follows:

(a) Position left and right battery switches ON.

(b) Depress PRESS-TO-TEST annunciator panel switch. Verify L ENG FIRE and R ENG FIRE lights are illuminated.

(c) Depress PRESS-TO-TEST fire extinguisher panel switch. Verify L ENG FIRE and R ENG FIRE lights are illuminated.

(d) Using jumper wire, connect fire detector on affected engine-to-aircraft ground wire and verify L and/or R ENG FIRE light is illuminated.

(e) Position left and right battery switches OFF.

3. Inspection/Check – Engine Fire Protection

A. Inspection – Fire Detectors

WARNING: METHYL PROPYL KETONE (MPK) IS TOXIC, FLAMMABLE, AND CAUSTIC. ISOPROPYL ALCOHOL IS TOXIC AND FLAMMABLE. AVOID OPEN FLAMES AND SPARKS. USE ONLY IN A WELL-VENTILATED AREA. AVOID EYE AND SKIN CONTACT OR BREATHING OF VAPORS. PERSONAL PROTECTIVE EQUIPMENT CONSISTING OF RESPIRATOR, INDUSTRIAL GOGGLES, AND RUBBER GLOVES IS REQUIRED.

(1) Clean ceramic area of each detector with MPK or isopropyl alcohol.

(2) Seal ceramic area of engine fire detector with Dow Corning RTV 734 sealer.

(3) Check rubber boots for rips, hardness, security, and general condition.

(4) Check fire detector tie mounts for proper sealing of electrical bond areas and condition of Dow Corning RTV 734 sealer.

(5) Check fire detectors for proper mounting and security.

(6) Using good light (min 100fc) and 1.5X magnifying power, Detail Visual Inspect (DVI) each installed detector for impact or other physical damage IAW Service Bulletin 226-26-005, steps 2. A. and B.

(7) If damage is found on a detector:

(a) Remove engine detectors IAW instructions in this section and test the detector IAW Service Bulletin 226-26-005 step 2. C. Suggested Test Method.

(b) Remove wheel well detectors IAW section 26-10-20 in this manual and test the detector IAW Service Bulletin 226-26-005 step 2. C. Suggested Test Method.

(c) Replace any detector that fails test.

(d) Install detector IAW instructions in this section.

MAINTENANCE MANUAL

WING OVERHEAT DETECTION – DESCRIPTION AND OPERATION

1. General

A. Wing overheat detection consists of one 350°F thermal switch (item 1), one 450°F thermal switch (item 2), and one 250°F thermal switch (item 3) for each wing. Thermal switch locations (Figure 1) for each type represented vary accordingly.

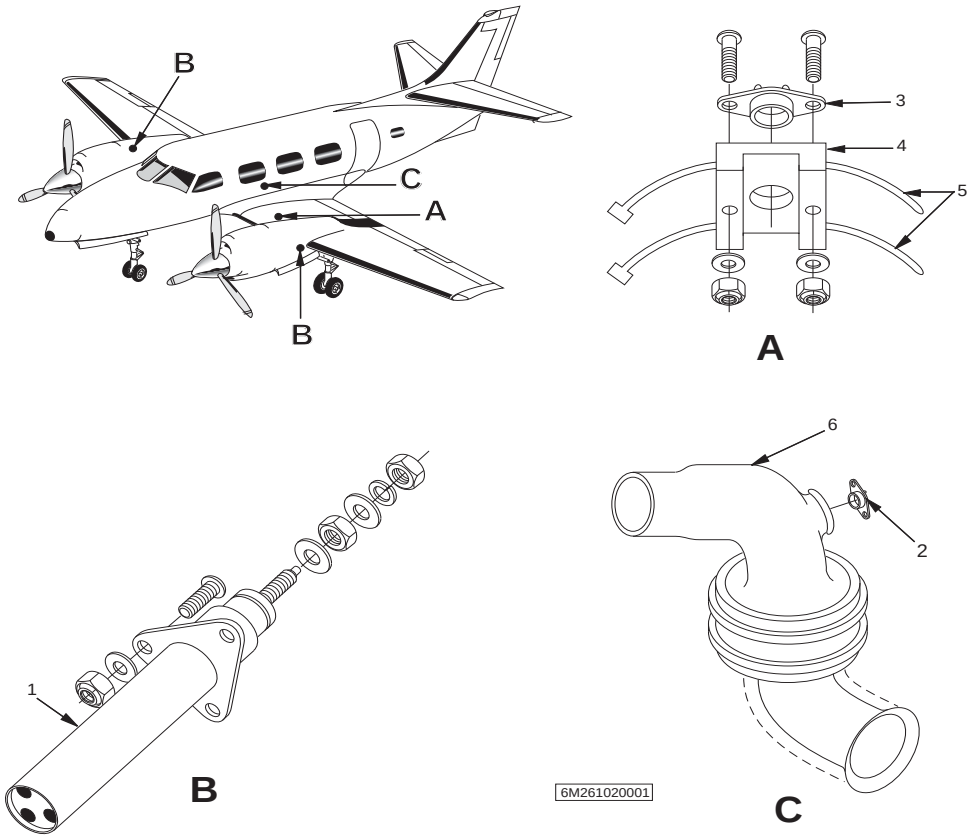
B. The 250°F thermal switch (Figure 1, Detail A), located in the leading edge of each wing, will detect any overheat condition in that area.

(1) Each switch is installed within a tie mount (item 4). The tie mount is secured to a four gauge feeder wire bundle with cable ties (item 5).

(2) The thermal switch monitors the temperature of engine bleed-air ducts and four gauge feeder wires routed inside the leading edge of each wing.

(3) The thermal switch is located in each wing leading edge cavity, inboard of the engine, at W.S. 51.00. This thermal switch is in contact with generator wires and secured with cable ties.

(4) If the temperature in this area exceeds 250°F, the switch will close and energize L and/or R flasher causing the red L WING OVHT and/or R WING OVHT annunciator panel light to flash.



Wing Overheat Detection Thermal Switch Locations
Figure 1

C. The 350°F thermal switch (Figure 1, Detail B), located in each wheel well just below and forward of the mass-flow control valve, is used to detect a rupture of the bleed air line. If the temperature in this area exceeds 350°F, the switch will close, causing the red L WING OVHT and/or R WING OVHT annunciator panel light to illuminate steadily.

D. The 450°F thermal switch (Figure 1, Detail C), located within each inlet air duct (item 6) just inside the cabin under the floorboard, is used to detect a failure of the air conditioning turbine. If the turbine should fail, the temperature in this duct will rise above 450°F and actuate the switch, causing the red L WING OVHT and/or R WING OVHT light in the annunciator panel to illuminate steadily.

E. All three types of thermal switches control L WING OVHT and R WING OVHT annunciator panel lights.

F. Flasher

(1) The left and right wing flashers are installed as part of wing overheat cockpit electrical assembly located behind LH aft console access panel, EP9, for T201-248, AT001-033 (Exc 025), and TC201-212.

(2) The left and right wing flashers are an integral part of solid state annunciator panel circuitry on T249-291, AT025, 034-074, and TC213-419.

(3) Each flasher operates in conjunction with the 250°F thermal switch to cause the red L WING OVHT and/or R WING OVHT annunciator panel light to flash.

G. Test Relay

(1) A test relay is installed as part of the wing overheat cockpit electrical assembly located behind LH aft console access panel, EP9, on T201-248, AT001-024, AT026-033 (Exc 025), and TC201-212.

(2) The test relay is an integral part of solid state annunciator panel circuitry on T249-291, AT025 and 034-074, and TC213-419.

(3) The test relay is used to test wing overheat detection circuitry for its integrity and operability. Closing the annunciator panel test switch activates circuitry to check the integrity of the wiring, flasher and annunciator panel L WING OVHT and/or R WING OVHT lights.

1. Removal/Installation – Wing Overheat (Figures 201 and 202)

A. Removal – 250°F Thermal Switch (Figure 201, Detail A)

NOTE: Removal/installation of the 250°F thermal switch is accomplished between W.S. 27.10 and 81.74 within the cavity of lower wing panel, directly aft leading edge scoop.

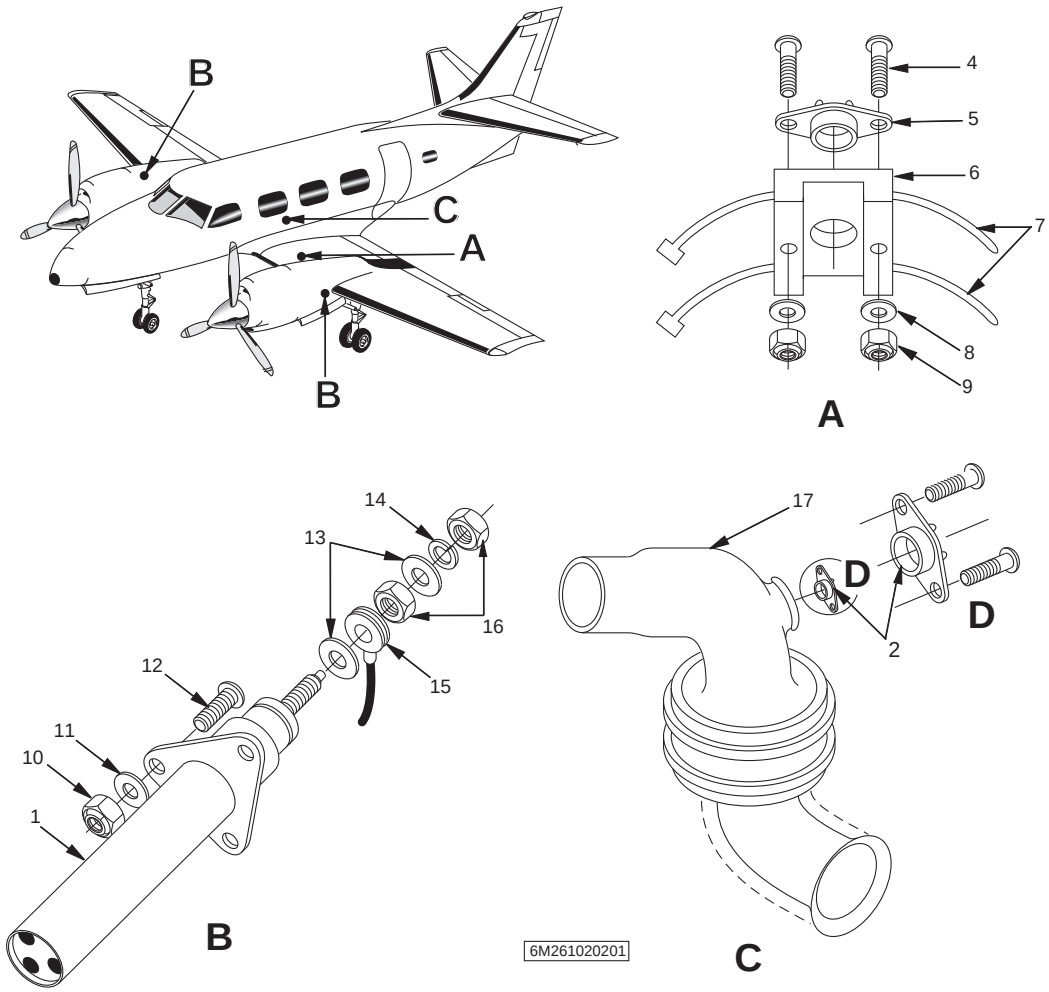
(1) Ensure left and right battery switches OFF.

(2) Remove lower wing panel directly aft of leading edge air scoop.

(3) Reach up toward leading edge of wing. Grasp four gauge feeder wire bundle and locate thermal switch (item 5) by feel.

(4) Cut plastic cable ties (item 7) securing thermal switch to four gauge feeder wire bundle.

(5) Remove thermal switch with tie mount (item 6) still secured.



Wing Overheat Detection Thermal Switch Details
Figure 201

(6) Tag wire connections and cut 22-gauge sensor wires to free thermal switch at most practical point within six (6) inches (15.24 cm) of connecting end.

(7) Remove thermal switch from tie mount.

B. Installation – 250°F Thermal Switch (Figure 201, Detail A)

(1) Ensure that left and right battery switches are OFF.

(2) Mount thermal switch (item 5) onto tie mount ((item 6). Secure with screws (item 4), nuts (item 9), and washers (item 8).

(3) Connect electrical wires to thermal switch. Apply Dow Corning RTV 734 sealer to wire connections.

(4) Thread cable ties (item 7) into tie mount.

(5) Place tie mount into lower wing opening and secure to four gauge feeder wire bundle with cable ties.

(6) Perform functional test of thermal switch IAW instructions in Adjustment/Test – Wing Overheat, this section.

(7) Install lower wing panel.

C. Removal – 350°F Fire Detector (Figure 201, Detail B)

NOTE: Removal/installation of 350°F fire detector is accomplished within each wheel well between W.S. 81.74 and 99.51.

(1) Ensure that battery switch is OFF.

(2) Remove generator control panel on forward left side of main gear wheel well, enabling access to fire detector mounted on inner side of panel.

(3) Disconnect electrical wires from fire detector.

(4) Remove fire detector.

D. Installation – 350°F Fire Detector (Figure 201, Detail B)

(1) Ensure that battery switch is OFF.

(2) Mount fire detector (item 1) to forward left side of generator control panel in wheel well. Secure with screws (item 12), washers (item 11), and nuts (item 10).

(3) Connect electrical wires (item 15) to fire detector.

WARNING: METHYL PROPYL KETONE (MPK) IS TOXIC, FLAMMABLE, AND CAUSTIC. ISOPROPYL ALCOHOL IS TOXIC AND FLAMMABLE. AVOID OPEN FLAMES AND SPARKS. USE ONLY IN A WELL-VENTILATED AREA. AVOID EYE AND SKIN CONTRACT OR BREATHING OF VAPORS. PERSONAL PROTECTIVE EQUIPMENT CONSISTING OF RESPIRATOR, INDUSTRIAL GOGGLES, AND RUBBER GLOVES IS REQUIRED.

(4) Clean wire connections and ceramic area of fire detector with MPK or isopropyl alcohol, using a small brush.

(5) Apply Dow Corning RTV 734 sealer to wire connections and ceramic area of fire detector.

(6) Perform functional test of fire detector IAW instructions in Adjustment/Test – Wing Overheat, this section.

(7) Install generator control panel on forward left side of main gear wheel well.

E. Removal – 450°F Thermal Switch (Figure 201, Details C and D)

NOTE: Removal/installation of 450°F thermal switch is accomplished at F.S. 159.00 on right and left sides for T and T(B), and at F.S. 261.70 for AT and TC within inlet air duct mounting pad opening, just inside cabin under floorboard.

(1) Ensure that left and right battery switches are OFF.

(2) Remove seats, carpet, and floorboards, as necessary, to gain access to thermal switch.

(3) Locate thermal switch (item 2) on mounting pad of inlet air duct (item 17).

(4) Tag wire connections and cut wires to free thermal switch at most practical point near connecting end of switch.

(5) Remove thermal switch from mounting pad of inlet air duct.

F. Installation – 450°F Thermal Switch (Figure 201, Details C and D)

(1) Ensure that left and right battery switches are OFF.

(2) Mount thermal switch (item 2) on mounting pad of inlet air duct (item 17) with screws (item 18).

(3) Solder wire connections to thermal switch.

WARNING: METHYL PROPYL KETONE (MPK) IS TOXIC, FLAMMABLE, AND CAUSTIC. ISOPROPYL ALCOHOL IS TOXIC AND FLAMMABLE. AVOID OPEN FLAMES AND SPARKS. USE ONLY IN A WELL-VENTILATED AREA. AVOID EYE AND SKIN CONTACT OR BREATHING OF VAPORS. PERSONAL PROTECTIVE EQUIPMENT CONSISTING OF RESPIRATOR, INDUSTRIAL GOGGLES, AND RUBBER GLOVES IS REQUIRED.

(4) Clean wire connections of thermal switch with MPK or isopropyl alcohol, using a small brush.

(5) Apply Dow Corning RTV 734 sealer to wire connections.

(6) Perform functional test of thermal switch IAW instructions in Adjustment/Test – Wing Overheat, this section.

(7) Install floorboards, carpet, and seats, as required.

G. Removal/Installation – Flasher/Relay (See Figure 202)

(1) Ensure that left and right battery switches (item 2) are OFF.

(2) Remove annunciator panel (item 1).

(3) Repair or replace annunciator panel.

H. Removal – Flasher, Adjustable Resistor, or Test Relay

(1) Ensure that left and right battery switches are OFF.

(2) Remove LH aft console access panel, EP9, located between F.S. 107.06 and 116.56, to gain access to flashers (item 4), adjustable resistors (item 8), and wing overheat test relay (item 9).

(3) Remove flasher, adjustable resistor and/or test relay from wing overheat cockpit electrical assembly.

I. Installation – Flasher, Adjustable Resistor, or Test Relay

(1) Ensure that left and right battery switches are OFF.

CAUTION: EXERCISE CARE TO PREVENT DAMAGE TO CONNECTORS WHEN INSTALLING FLASHER.

(2) Install flasher, adjustable resistor and/or test relay, as applicable.

(3) Perform functional test of flasher, adjustable resistor, and/or test relay IAW instructions in Adjustment/Test – Wing Overheat, this section.

(4) Install LH aft console access panel, EP9.

2. Adjustment/Test – Wing Overheat (See Figure 202)

A. Adjustment/Test – Adjustable Resistor

(1) Remove access panel, EP9, in LH aft console area to enable access to wing overheat cockpit electrical assembly.

(2) Loosen adjustable resistor wiper set screw (item 6).

NOTE: Wing overheat detection incorporates two (2) adjustable resistors, one providing adjustment control for left wing overheat detection, and one for right wing overheat detection. Adjustment/Test can be made to either or both resistors.

(3) Position left/right adjustable resistor wiper (item 7) to obtain 42 ± one (1) ohm.

(4) Tighten left/right adjustable resistor wiper set screw.

B. Functional Test – Wing Overheat Detection

(1) Position left and right battery switches (item 2) ON.

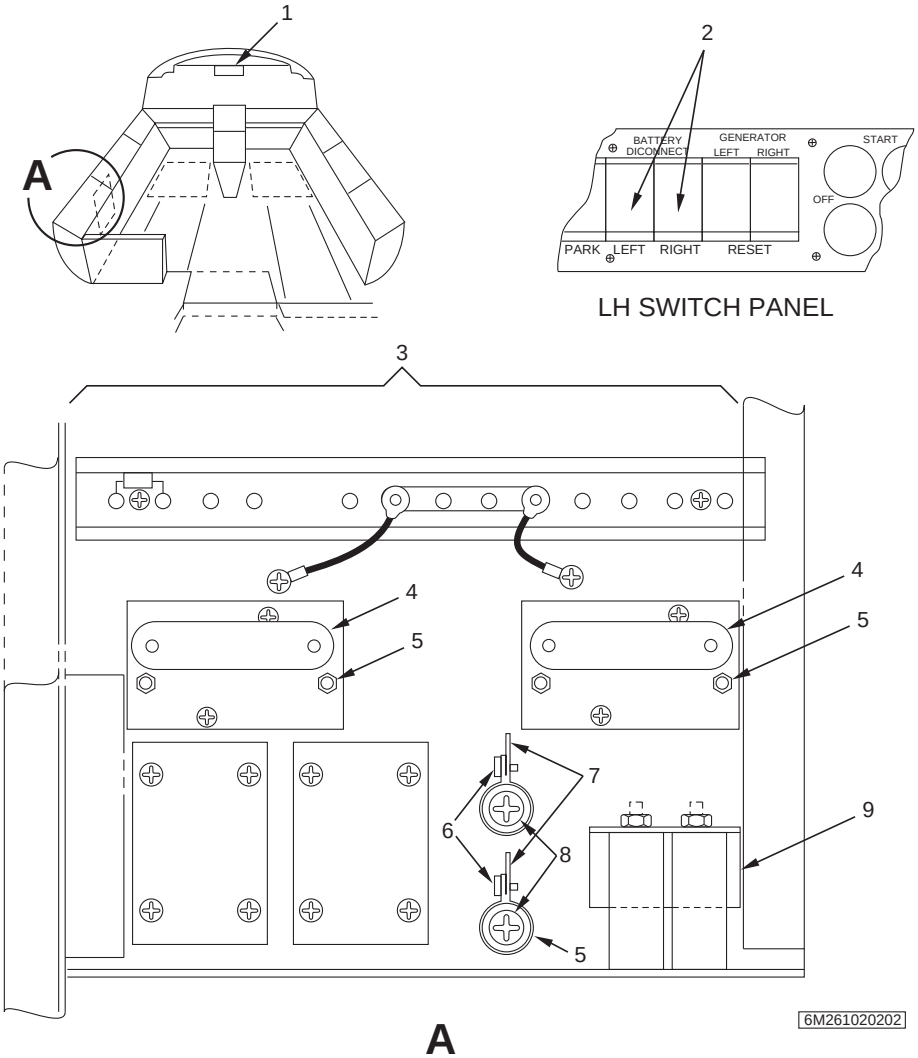
(2) Press annunciator test switch, located adjacent to annunciator panel.

(3) Verify that L and R WING OVHT lights on annunciator panel illuminate:

(a) On aircraft T201-248, AT001-032 (except AT025 and AT026), press-to-test annunciator switch may cause the L/R WING OVERHEAT lamps to come on steady momentarily then flash until the press-to-test switch is released (mechanical flashers).

(b) On aircraft T249-291, T(B) 276, 292-417, AT025, AT026, AT033-074, TC213-396, TC398-413, TC418, and TC419, press-to-test annunciator switch will cause the L/R WING OVERHEAT lamps to immediately begin to flash until the press-to-test switch is released (electronic flashers).

(4) Position left and right battery switches OFF.



Wing Overheat Detection
Cockpit Electrical Assembly
Figure 202

FIRE EXTINGUISHING – DESCRIPTION AND OPERATION

1. General

A. The optional fire extinguishing system uses a one-shot fire extinguishing bottle for each engine and a control assembly in the cockpit. The system will receive an engine fire warning signal from any of the four fire detectors located on each engine.

B. The fire extinguishing system is an optional installation on SA226 T201-999, AT001-999, and TC201-397, and standard equipment for TC398-999.

C. The fire extinguisher bottle is installed just aft of the engine firewall on the lower right side of each nacelle. A stainless steel tube from the outlet of the bottle is routed through the firewall and branches out to direct the fire extinguishing agent to strategic points on the engine.

D. A pressure gauge on each bottle is visible through a window in the access door on the right side of each nacelle and indicates normal pressure range of the agent.

E. The control assembly can indicate a normal or thermal discharge through sensors located in the fire bottle.

FIRE EXTINGUISHER CONTROL – DESCRIPTION AND OPERATION

1. General

A. Fire Extinguisher Control Assembly

(1) The optional control assembly contains:

(a) Two logic modules.

(b) Two control indicators, each containing three indicator lights and a firing switch.

(c) A test toggle switch.

(d) Two test relays.

(e) Two bus ties.

(2) The control assembly is rear-mounted and secured with two Dzus fasteners on the backside of the instrument panel below the annunciator panel.

(3) Pushing the lens of the L/R ENGINE FIRE indicator lights fires the extinguishers. Since the system is powered from both essential buses through isolation diodes, pushing the L/R FIRE switch will fire the respective bottle if power is on the buses. Each lens is covered by a clear plastic spring-loaded guard to avoid accidental firing.

(4) A red FIRE light, an amber E (empty) light, and a green OK light comprise the three indicator lights. The lights are operated by the logic module. The fire extinguisher test switch is located between the two control indicators on the instrument panel.

(a) If the bottle is thermally discharged, the amber E light will illuminate.

(b) If the bottle is fired from the cockpit, the E light will illuminate and the FIRE light will remain illuminated until the fire is extinguished.

(c) When the fire extinguisher test toggle switch is selected Left or Right, all three lights on the respective control indicator will normally illuminate.

(d) If the bottle has been discharged or either the FIRE or the E lights are on, the green OK light will not illuminate when the test switch is pressed.

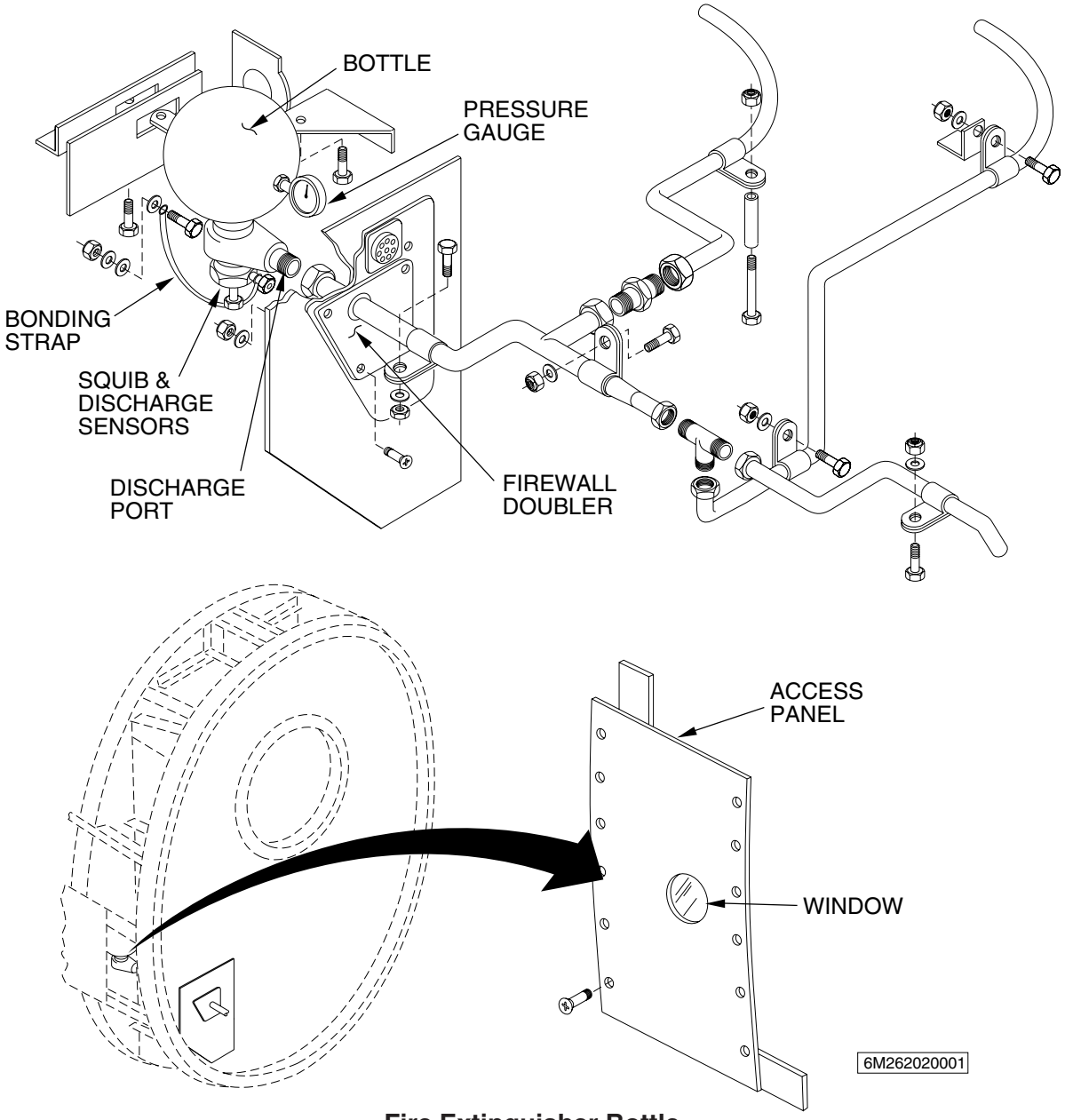
B. Pressing the annunciator panel test switch will test the engine fire warning system for continuity as well as illuminate the red FIRE light on the control indicator. Should the fire detectors on the engines sense an overheat condition, the FIRE light will illuminate.

FIRE EXTINGUISHER BOTTLE – DESCRIPTION AND OPERATION

1. General

A. An optional 86 cubic inch (218.44 cc) fire extinguisher bottle (Figure 1), containing bromotrifluoromethane, CBrF3, is installed on the right side of each nacelle behind the firewall and is secured to the airframe structure with two bolts. Each bottle incorporates a pressure gauge, thermal relief valve, discharge squib and normal discharge port. Each bottle weighs 3.35 pounds (1.52 kg) when fully charged.

B. The fire extinguisher bottle is optional for SA226 T201-999, AT001-999, and TC201-397, and standard equipment for TC398-999.



Fire Extinguisher Bottle
Figure 1

2. Controls

A. The engine fire extinguisher system uses 28.5 VDC from the essential buses to energize the discharge squib on the bottles. A push-type switch for each engine extinguisher is located on the face side of the fire extinguisher control assembly on the instrument panel, below the annunciator panel. Pressing the FIRE switch energizes the discharge squib, allowing the extinguisher agent to be released.

3. **Plumbing**

A. Stainless steel tubing, attached to each fire extinguisher bottle, routes the fire extinguishing agent through the firewall into smaller stainless steel tubes to direct the agent into strategic engine compartment areas. These areas are:

(1) The fuel control (left side of engine).

(2) Starter-generator (FWD right side of engine).

(3) The engine oil cooler (rear of engine).

B. The engine fire extinguisher plumbing is secured to the engine truss assembly and bracket with clamps. The primary tube is secured to the firewall with screws, washers, and nuts.

FIRE EXTINGUISHER BOTTLE – MAINTENANCE PRACTICES

1. Removal/Installation – Fire Extinguisher Bottle

WARNING: WHEN ANY MAINTENANCE IS PERFORMED ON OR NEAR FIRE EXTINGUISHER BOTTLES, ENSURE ELECTRICAL POWER IS OFF. FAILURE TO OBSERVE THIS WARNING COULD RESULT IN ACCIDENTAL DISCHARGE OF FIRE EXTINGUISHING AGENT, CAUSING A POTENTIALLY SERIOUS HEALTH RISK TO PERSONNEL IN WORK AREA.

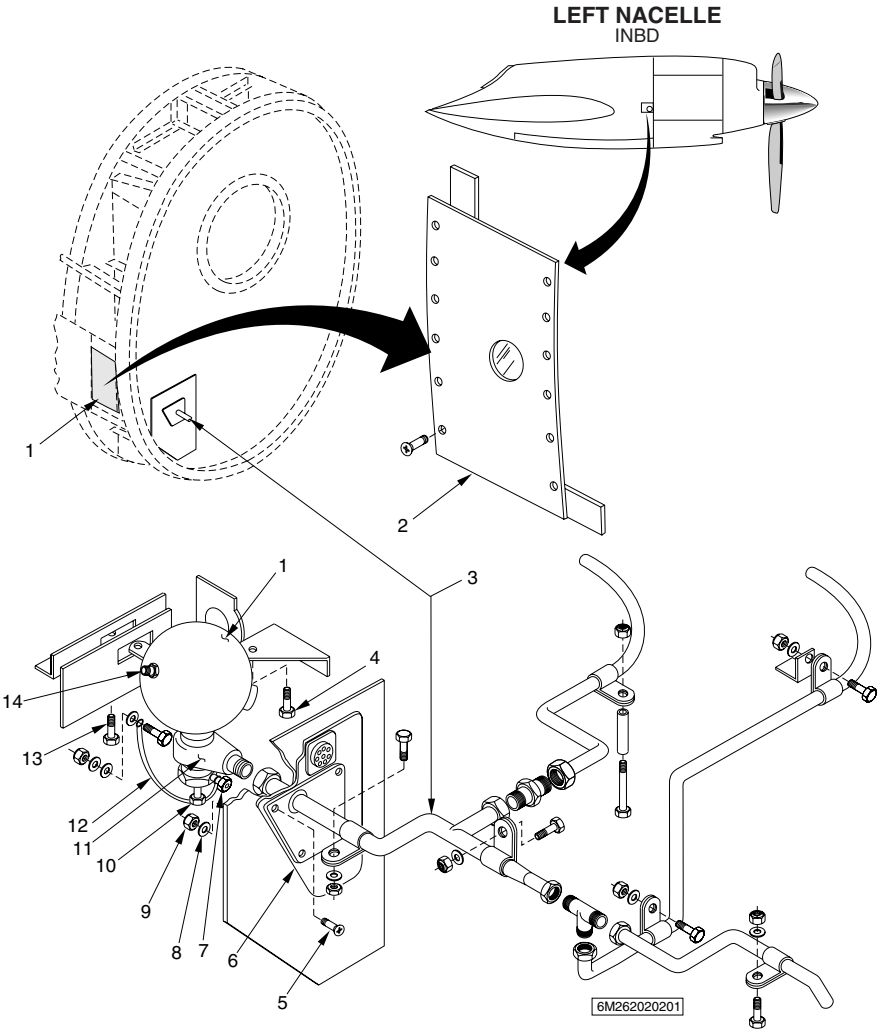
A. Removal – Fire Extinguisher Bottle (see Figure 201)

(1) Ensure that battery switch is OFF.

(2) Remove fire extinguisher access panel (item 2) located on right side of each nacelle.

(3) Disconnect power supply and sensor wires from squib terminal (item 7).

(4) Disconnect ground wire (item 12) from squib (item 10).



Fire Extinguisher Bottle
Major System Components
Figure 201

(5) Remove upper forward access panel on outboard side of right wheel well (RH bottle), or remove upper forward access panel on inboard side of left wheel well (LH bottle) to gain access to mounting hardware.

(6) Disconnect tubing (item 3) from squib nozzle (item 11).

(7) Disconnect thermal relief plumbing and fittings from thermal relief valve (item 4).

(8) Remove screws (item 5), washers (item 8), and nuts (item 9) attaching firewall doubler (item 6) to forward side of firewall.

(9) Working through access panel removed in step (5), remove aft bolt (item 4) attaching bottle to bracket.

(10) Working through access panel removed in step (2), remove FWD bolt (item 13) attaching bottle to bracket.

(11) Remove fire extinguisher bottle (item 1).

B. Installation – Fire Extinguisher Bottle (see Figure 201)

(1) Position bottle (item 1) with squib (item 10), down, and squib nozzle (item 11) outlet, FWD.

(2) Secure fire extinguisher bottle to bracket with FWD bolt (item 13), working through fire extinguisher access panel.

(3) Secure fire extinguisher bottle to bracket with aft bolt (item 4), working through forward access panel.

(4) Connect primary discharge tube (item 3) to squib nozzle (item 11). Torque retaining nut 550 to 750 inch-pounds (60.14 to 84.74 N·m).

NOTE: Ensure that primary discharge tube drain hole is at lowest point possible (6 o'clock position).

(5) On aircraft equipped with thermal relief plumbing, connect thermal relief tube and fittings to thermal relief valve (item 14).

(6) Secure firewall doubler to forward side of firewall, and secure with screws (item 5), washers (item 8), and nuts (item 9).

NOTE: Firewall doubler consists of two (2) irregular-shaped doublers joined with screws, washers, and nuts, and is characterized by a notched cutout in upper corner.

(7) Connect power and sensor wires to squib terminals (item 7).

(8) Connect ground wire (item 12) to squib (item 13).

(9) Install forward access panel of RH wheel well (RH bottle) or install upper forward access panel on inboard side of LH wheel well (LH bottle), as applicable.

(10) Install fire extinguisher access panel (item 2) on right side of nacelle.

2. Adjustment/Test – Fire Extinguisher Bottle

A. Leak Test

(1) Material Required: Electronic-Type Leak Detector.

(2) Remove fire extinguisher bottle (item 1).

(3) Using clean, dry compressed air or gaseous nitrogen, blow any dirt or contaminants from the bottle outlet.

WARNING: DO NOT USE FLAME-TYPE HALOGEN LEAK DETECTOR DUE TO CLOSE PROXIMITY OF AIRCRAFT FUEL LINES IN AIRCRAFT.

(4) Using electronic-type leak detector, check all pipe threads, fittings, valves, welded bosses, and seams for indication of leakage.

B. Squib Test (Detonator Test)

(1) Material Required: Squib Tester, Model 115, American Standard, Monrovia, California, or equivalent.

WARNING: DO NOT USE A VOLT-OHMMETER, BATTERY POWERED CONTINUITY LIGHT OR SIMILAR TEST DEVICE TO TEST THE SQUIB ASSEMBLY. USE OF SUCH TEST DEVICES THAT GENERATE MORE THAN 30mA OF ELECTRIC CURRENT COULD DETONATE THE SQUIB ASSEMBLY, CAUSING AN ACCIDENTAL DISCHARGE OF FIRE EXTINGUISHING AGENT.

(2) Using squib tester, check squib assembly for nominal resistance of 1.15 + 0.25 ohms.

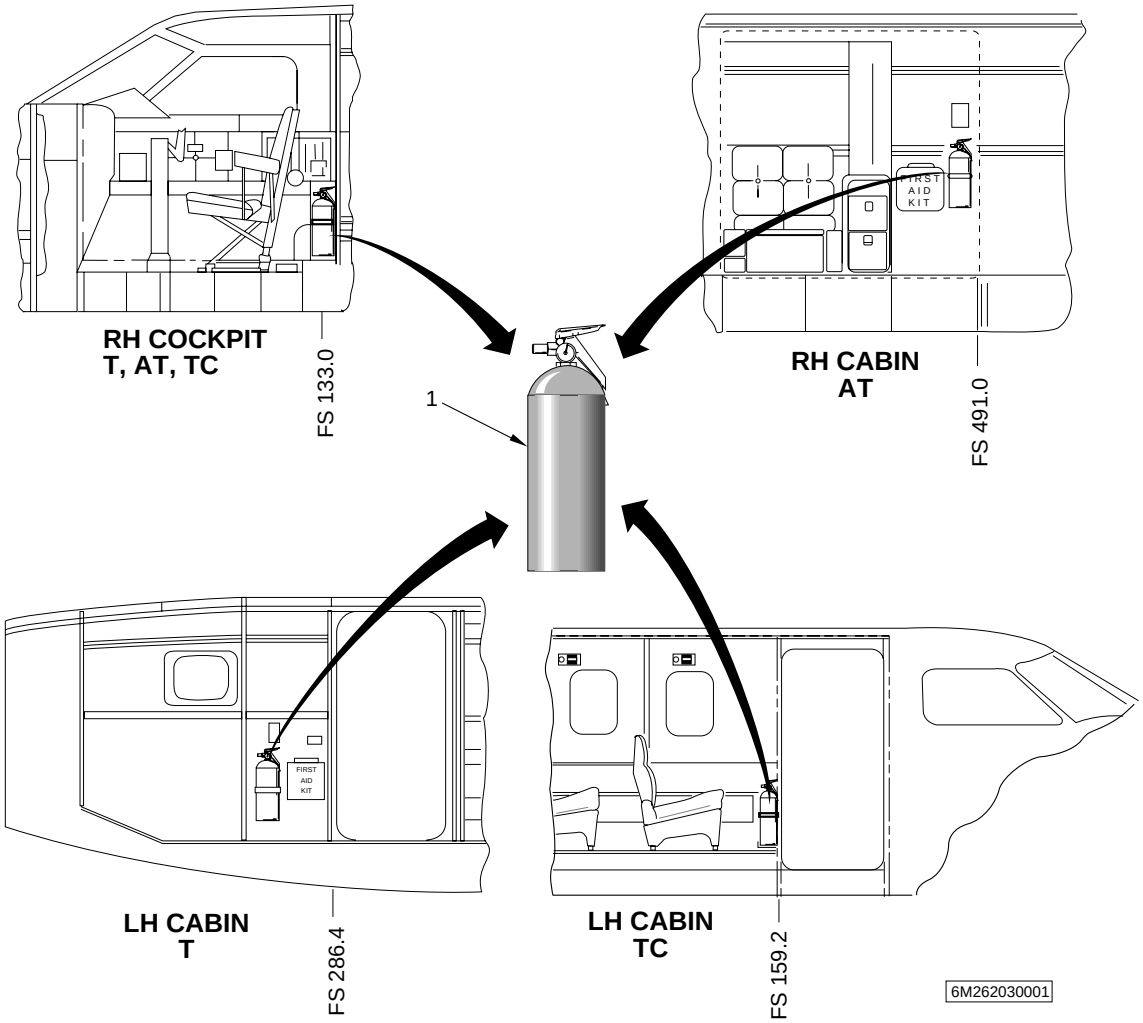
PORTABLE FIRE EXTINGUISHER – DESCRIPTION AND OPERATION

1. General

A. One hand-held portable fire extinguisher (item 1, Figure 1) is located in the cockpit on the copilot's RH console, aft of copilot's seat. A second hand-held portable fire extinguisher is mounted just aft of the passenger door, on the LH side of cabin for T and TC aircraft, and on the RH side of cabin for AT aircraft. Both units are bromochlorodifluoromethane, CBrF2, liquid chemical gaseous nitrogen-charged fire extinguishers.

B. When actuated, the vapor pressure causes the agent to expand so that the discharge stream is a mixture of liquid droplets and vapor. Under normal pressurization, a typical bromodifluoromethane portable fire extinguisher has a horizontal stream range of 9 to 15 feet (2.7 to 4.6 m) and, because of its chemical makeup, is not affected by wind as much as CO2 extinguishers.

C. Deep-seated Class A fires may need to be broken apart to effect complete extinguishment. On Class B fires, the discharge of bromodifluoromethane agent is applied in a side-to-side motion, gradually progressing toward the back of the fire. The extinguisher should be initially discharged from a distance not closer than 8 feet (2.4 m) to prevent splashing when applied.



Portable Fire Extinguisher Locations
Figure 1

PORTABLE FIRE EXTINGUISHER – MAINTENANCE PRACTICES

1. Inspection/Check – Portable Fire Extinguisher

A. Inspection – Bromodifluoromethane, CBrF2, Fire Extinguisher (Figure 201)

(1) Inspect the nozzle for obstructions and obvious damage.

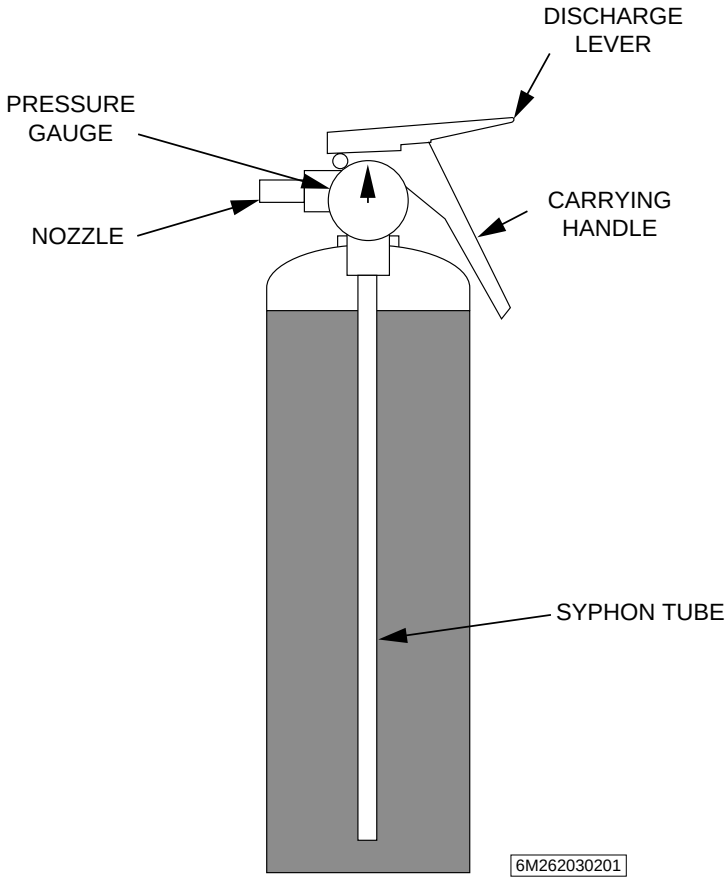
(2) Ensure that fire extinguisher is properly charged, verifying that needle of pressure gauge falls within operable range.

(3) Verify that locking seal is intact and free of damage.

(4) Weight check fire extinguishers every six (6) months. Recharge if gross weight is below 4 pounds 3 ounces (1.90 kg), or if pressure is below operable range at normal temperature. (Not applicable to fire extinguishers with gauges installed).

(5) Assuming no full or major depletion-use of fire extinguisher has been made and pressurization indicates fire extinguisher is still within operable range, recharge portable fire extinguishers every six (6) years, except as noted below.

(6) Perform a hydrostatic test every twelve (12) years IAW National Fire Protection Association Publication 10. Perform initial test twelve (12) years from date of manufacture.



Portable Fire Extinguisher
Figure 201

MZ AEROSPACE
An Elbit Systems of America, LLC, Company

MAINTENANCE MANUAL

CHAPTER 27 – FLIGHT CONTROLS

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27-30-01	MP	201	OCT 07/17
27-30-01	MP	202	OCT 07/17
27-30-01	MP	203	OCT 07/17
27-30-01	MP	204	OCT 07/17
27-30-02	DO	1	DEC 31/14
27-30-02	MP	201	OCT 07/17
27-30-02	MP	202	OCT 07/17
27-30-02	MP	203	OCT 07/17
27-30-02	MP	204	OCT 07/17
27-30-02	MP	205	OCT 07/17
27-30-02	MP	206	OCT 07/17
27-30-03	DO	1	OCT 07/17
27-30-03	MP	201	OCT 07/17
27-30-03	MP	202	OCT 07/17
27-30-03	MP	203	OCT 07/17
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27-30-03	MP	205	OCT 07/17
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27-30-03	MP	209	OCT 07/17
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27-30-11	DO	2	JUN 01/01	27-31-06	MP	219	APR 30/08
27-30-11	MP	201	JUN 01/01	27-31-06	MP	220	DEC 01/01
27-30-11	MP	202	APR 01/05	27-31-06	MP	221	APR 30/06
27-30-11	MP	203	APR 01/05	27-31-06	MP	222	APR 01/05
27-30-11	MP	204	JUN 01/01	27-31-07	DO	1	DEC 01/01
27-30-11	MP	205	JUN 01/01	27-31-07	DO	2	DEC 01/01
27-30-11	MP	206	JUN 01/01	27-31-07	MP	201	DEC 01/01
27-30-12	DO	1	DEC 01/01	27-31-07	MP	202	DEC 01/01
27-30-12	MP	201	DEC 01/97	27-31-08	DO	1	DEC 01/01
27-31-06	DO	1	DEC 01/01	27-31-08	MP	201	DEC 01/01
27-31-06	DO	2	DEC 01/01	27-31-08	MP	202	DEC 01/01
27-31-06	DO	3	DEC 01/01	27-31-08	MP	203	DEC 01/01
27-31-06	DO	4	DEC 01/01	27-31-08	MP	204	DEC 01/01
27-31-06	DO	5	APR 30/08	27-31-08	MP	205	DEC 01/01
27-31-06	FI	101	DEC 01/01	27-31-08	MP	206	DEC 01/01
27-31-06	FI	102	DEC 01/01	27-31-08	MP	207	APR 01/05
27-31-06	FI	103	DEC 01/01	27-31-09	DO	1	DEC 01/01
27-31-06	FI	104	DEC 01/01	27-31-09	DO	2	DEC 01/01
27-31-06	FI	105	DEC 01/01	27-31-09	MP	201	DEC 01/01
27-31-06	FI	106	DEC 01/01	27-31-09	MP	202	DEC 01/01
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27-31-06	FI	112	DEC 01/01	27-32-06	DO	6	DEC 31/14
27-31-06	FI	113	DEC 01/01	27-32-06	FI	101	DEC 31/14
27-31-06	MP	201	DEC 01/01	27-32-06	FI	102	DEC 31/14
27-31-06	MP	202	DEC 01/01	27-32-06	FI	103	DEC 31/14
27-31-06	MP	203	DEC 01/01	27-32-06	FI	104	DEC 31/14
27-31-06	MP	204	DEC 01/01	27-32-06	FI	105	DEC 31/14
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27-32-06	MP	211	DEC 31/14	27-40-00	DO	1	DEC 01/98
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27-32-06	MP	232	DEC 31/14	27-40-01	MP	207	DEC 31/14
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27-32-06	MP	234	DEC 31/14	27-40-01	MP	209	DEC 31/14
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27-32-07	MP	201	AUG 01/98	▌27-40-02	MP	207	OCT 07/17
27-32-07	MP	202	JUN 01/85	27-41-03	DO	1	DEC 31/14
27-32-08	DO	1	AUG 01/98	27-41-03	DO	2	DEC 31/14
27-32-08	MP	201	AUG 01/98	27-41-03	FI	101	DEC 31/14
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27-32-09	DO	2	OCT 01/05	27-41-03	MP	204	DEC 31/14
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27-42-03	MP	208	DEC 31/14	27-70-03	MP	201	SEP 28/12
27-50-00	DO	1	DEC 01/98	27-70-04	DO	1	APR 01/04
27-50-01	DO	1	DEC 01/98	27-70-04	DO	2	APR 01/04
27-50-01	MP	201	MAY 01/03	27-70-04	MP	201	DEC 01/97
27-50-01	MP	202	JUN 01/93	27-70-04	MP	202	DEC 01/97
27-50-01	MP	203	JUN 01/93	27-70-04	MP	203	APR 01/04
27-50-02	DO	1	MAY 01/03	27-SERVC	SI	1	OCT 01/06
27-50-02	DO	2	MAY 01/03	27-SERVC	SI	2	APR 01/04
27-50-02	MP	201	MAY 01/03	27-SERVC	SI	3	APR 01/04
27-50-03	DO	1	MAY 01/03	27-SERVC	SI	4	APR 01/04
27-50-03	DO	2	MAY 01/03	27-SERVC	SI	5	APR 01/04
27-50-03	MP	201	MAY 01/03	27-SERVC	SI	6	APR 01/04
27-50-03	MP	202	MAY 01/03	27-SERVC	SI	7	APR 01/04
27-50-04	DO	1	MAY 01/03	27-SERVC	SI	8	APR 01/04
27-50-04	MP	201	MAY 01/03	27-SERVC	SI	9	APR 01/04
27-50-04	MP	202	MAY 01/86	27-SERVC	SI	10	APR 01/04
27-50-05	DO	1	MAY 01/03	27-SERVC	SI	11	APR 01/04
27-50-05	MP	201	JUN 01/85	27-SERVC	SI	12	APR 01/04
27-50-06	DO	1	JUN 01/85	27-SERVC	SI	13	APR 01/04
27-50-06	MP	201	MAY 01/03	27-SERVC	SI	14	APR 01/04
27-51-06	DO	1	MAY 01/03	27-SERVC	SI	15	APR 01/04
27-51-06	MP	201	MAY 01/03	27-SERVC	SI	16	APR 01/04
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27-52-06	DO	1	MAY 01/03	27-SERVC	SI	19	APR 01/04
27-52-06	MP	201	MAY 01/03	27-SERVC	SI	20	APR 01/04
27-52-06	MP	202	MAY 01/03	27-SERVC	SI	21	APR 01/04
27-52-06	MP	203	MAY 01/03	27-SERVC	SI	22	APR 01/04
27-52-06	MP	204	MAY 01/03	27-SERVC	SI	23	APR 01/04
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27-70-00	DO	2	APR 01/04	27-SERVC	SI	26	APR 01/04
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SAS SERVO IDLE CONTROL MAINTENANCE PRACTICES	27-32-13	201	T AT TC	276, 292-999 – – – – – –
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PITCH TRIM SYSTEM AND CONTROL FAULT ISOLATION	27-40-01	101	T AT TC	201-999 001-999 201-999
PITCH TRIM SYSTEM AND CONTROL MAINTENANCE PRACTICES	27-40-01	201	T AT TC	201-999 001-999 201-999
PITCH TRIM ACTUATOR DESCRIPTION AND OPERATION	27-40-02	1	T AT TC	201-999 001-999 201-999
PITCH TRIM ACTUATOR MAINTENANCE PRACTICES	27-40-02	201	T AT TC	201-999 001-999 201-999
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MAINTENANCE MANUAL

CHAPTER 27– FLIGHT CONTROLS

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FLIGHT CONTROLS – SERVICE INFORMATION

1. Introduction

A. The following information is supplied as a supplement to this chapter. Subject matter is included which is not applicable to an ATA 100 format, but is still of interest to Merlin and Metro owner/operators. The information in this section is not part of the basic manual, and may be removed or retained at the discretion of the owner/operator.

B. Revisions and additions to this section will be made as additional information is available, Input from individual owner/operators is encouraged, and may be submitted to:

M7 Aerospace, LP
Post Office Box 790490
San Antonio, Texas 78279-0490
ATTN: Technical Services Department

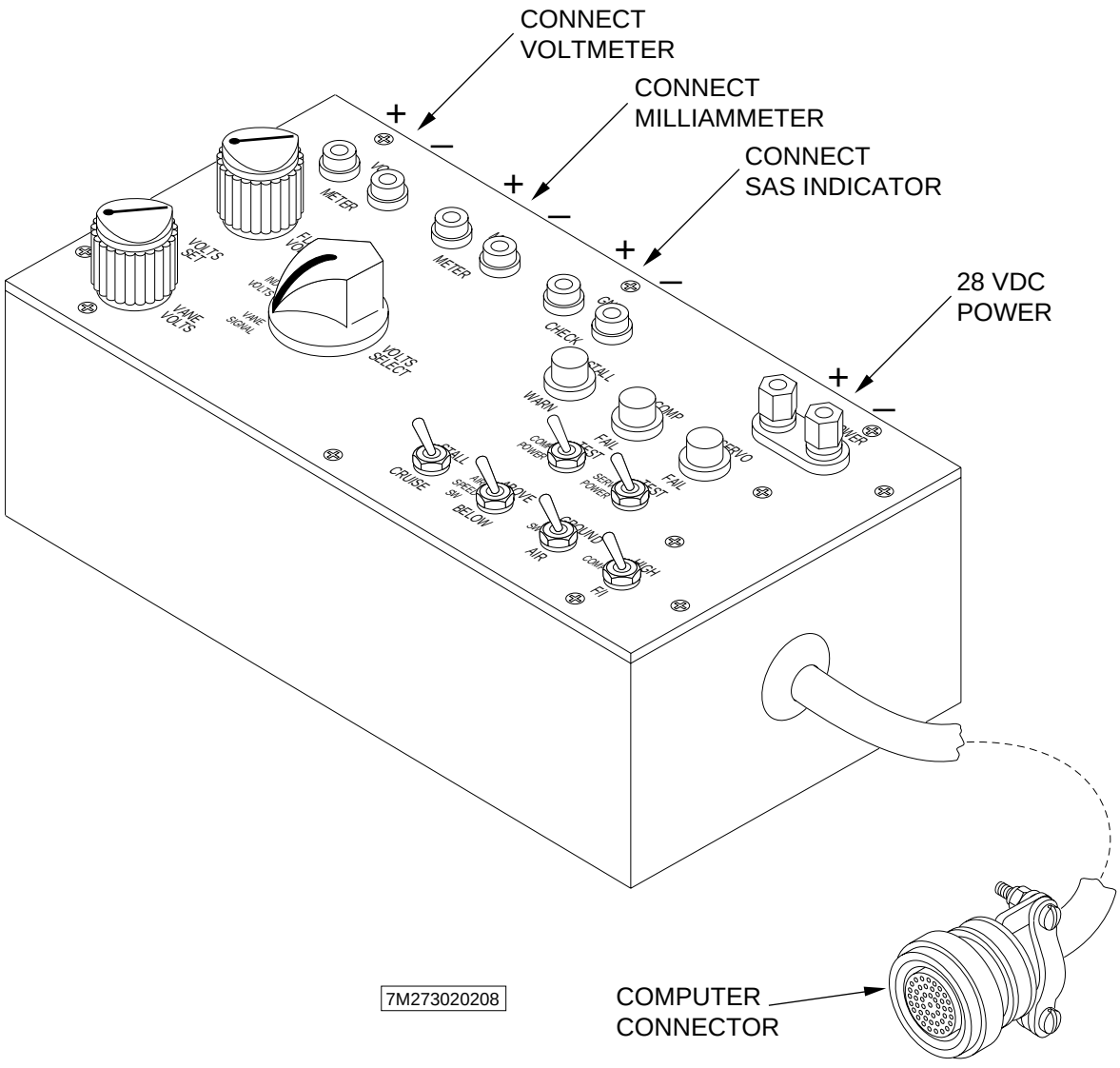
C. Flight Controls Test Equipment

(1) The following pages contain equipment necessary for the set-up, rigging, and balancing of flight control components and surfaces. Included are test sets and corresponding data forms used for the ground and flight set-up of the SAS system. Also included are several test fixtures for SAS set-up, and for rigging of the flight controls, and balancing of the control surfaces. This equipment is also useful in troubleshooting of flight control systems, and a substantial saving of manhours can be realized by using it.

(2) Materials for this equipment were chosen primarily on the basis of their availability. Specific brand names were, where ever possible, deliberately deleted. Deviations from the listed materials is completely acceptable provided the end result performs the intended function.

D. Conrac SAS Computer Calibration (TS27-01)

(1) This test set is used for bench check, ground set-up, and extraction of values set in the computer after in-flight test. These values can be found in the aircraft log on the SAS DATA SHEET and SAS COMPUTER CALIBRATION graph. Procedures for using the tester are found in Section 27-30-06. This method of SAS calibration is highly accurate, and substantial savings in manhours and in-flight set-up time can be realized.



Conrac SAS Computer Calibration
TS27-01
Figure 1

E. SAS Calibrator Construction (TS27-01)

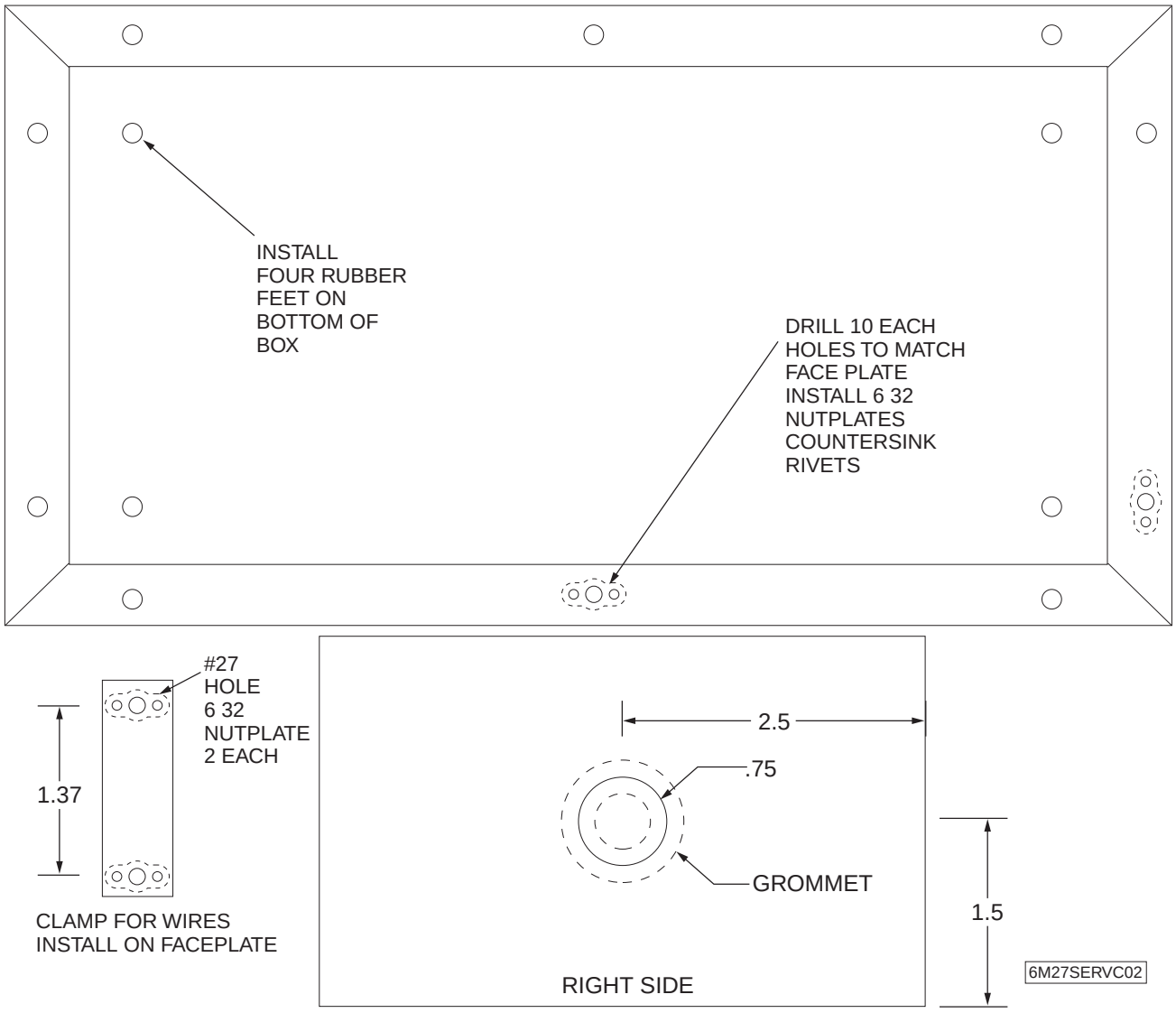
(1) The tester can be built into a box approximate size and construction shown in Figures 2 and 3. It is recommended that the components be situated on the faceplate of the tester and be marked clearly and indelibly as shown in Figure 4. Banana or pin type jacks can be used; red for positive, and black for negative connections to meters and power. A digital voltmeter and milliammeter can be built directly into the tester if desired. Operation and set-up procedures of the computer dictate that a separate voltmeter and milliammeter be used. The VOLTS SELECT switch (S7) should be a dual stack rotary type with five or more positions, or a single stack double sectioned type. For the accuracy needed in calibration of the SAS computer the two potentiometers, FLAP VOLTS SET (R3) and VANE VOLTS SET (R2), must be ten turn potentiometers. For the flap pot any ohm value from 500 to 1,500 may be used; for the vane pot any ohm value from 1,000 to 2,000 may be used. Both flap and vane pot may be the same ohm value. If different ohm values are used the flap pot must be the lower. The three lights (LT1, LT2, LT3) should be 28 VDC rated and have replaceable bulbs. Sockets must not be case grounded, and lenses should be red. The test switch (S1) must be spring loaded to both sides. Switches (S2, S2, S4, and S5) must be spring loaded to one position. On schematic Figure 5 the dot in the symbol for these switches denotes the normally closed position for the switch, and the triangle denotes the spring loaded position. All wires should be 22 AWG. Test leads should be clamped to the back of faceplate using the clamp bar in Figure 2. Wires should be insulated at this point. The test lead should extend from the box approximately one foot. Connect wires to all pins in connector (P1). Run wires into the box, and connect it to the components. On the schematic Figure 5, wires marked (NC) are not connected. Cap and stow these wires.

MATERIAL LISTING TS27-01

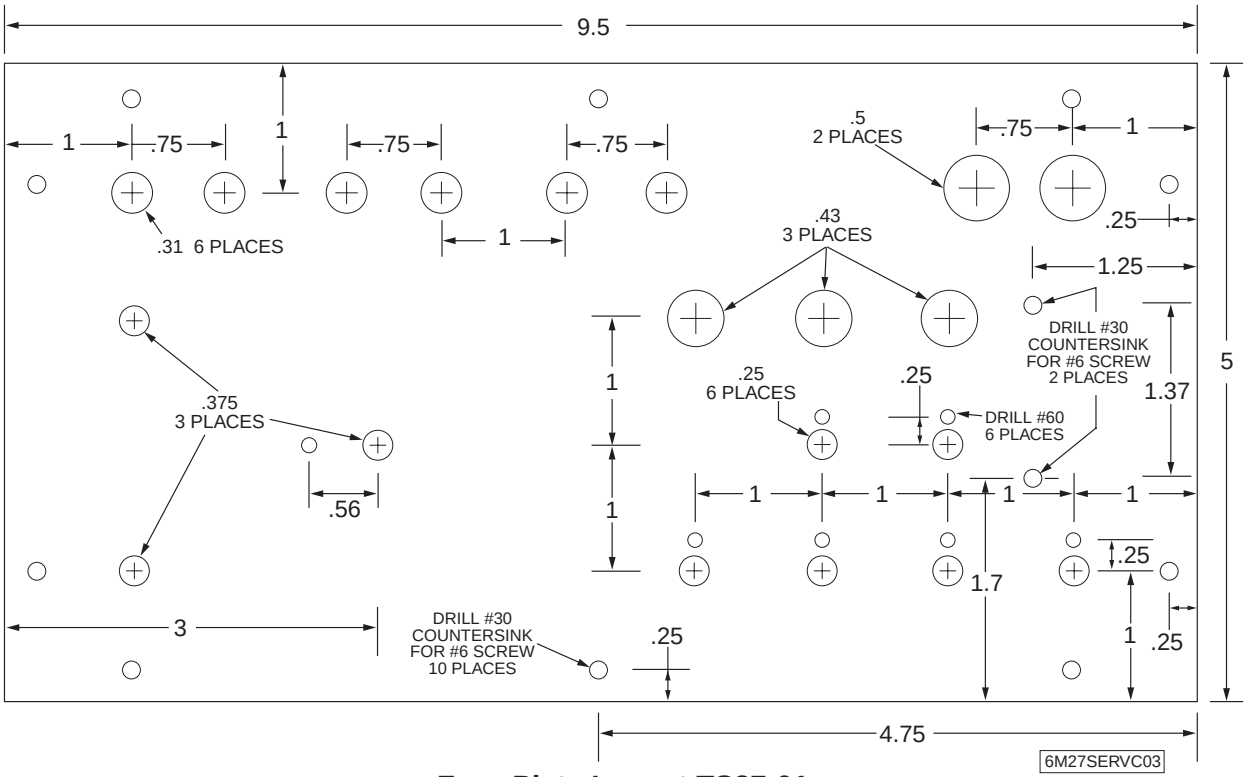
ITEM NO.	NAME	PART NUMBER/ SPECIFICATION	DESCRIPTION	QTY.
1.	BOX,BASE	ANY	9 in. X 5 in. X 3 in.	1
2.	FACE PLATE	ANY	9 in. X 5 in. X 1/16 in.	1
3.	GROMMET	ANY	1/2 in. I.D.	1
4.	FEET, RUBBER	ANY	ANY	4
5.	SCREW	ANY	6-32 (Countersunk)	12
6.	NUTPLATE/ RIVNUT	ANY	6-32	12
7.	CLAMP, WIRE	ANY	1 5/8 in. X 1/2 in. X 1/16 in.	1
8.	(P1) CONNECTOR	MS3126F8-32S	– – – –	1
9.	JACK, BANANA (J1, J3, J5)	ANY	BLACK	3
10.	JACK, BANANA (J2, J4, J6)	ANY	RED	3
11.	JACK, BANANA POWER (J7) DOUBLE	ANY	BLACK AND RED	1
12.	LIGHT (LT1, LT2) LT3)	B355A or MS 25041-5 or EQUIVALENT	TWO TERMINAL TYPE (RED LENS)	3
13.	BULB	327 or MATCHING ITEM 12	28 VDC	3
14.	SWITCH TEST (S1)	MS27716-27 (8A1041)	SP-DT SPRING LOADED CENTER OFF (MINIATURE)	1
15.	SWITCH (S2, S3, S4, S5, S6)	MS17716-29 (8A1061)	SP-DP SPRING LOADED ONE SIDE (MINIATURE)	5

MATERIAL LISTING TS27-01 (continued)

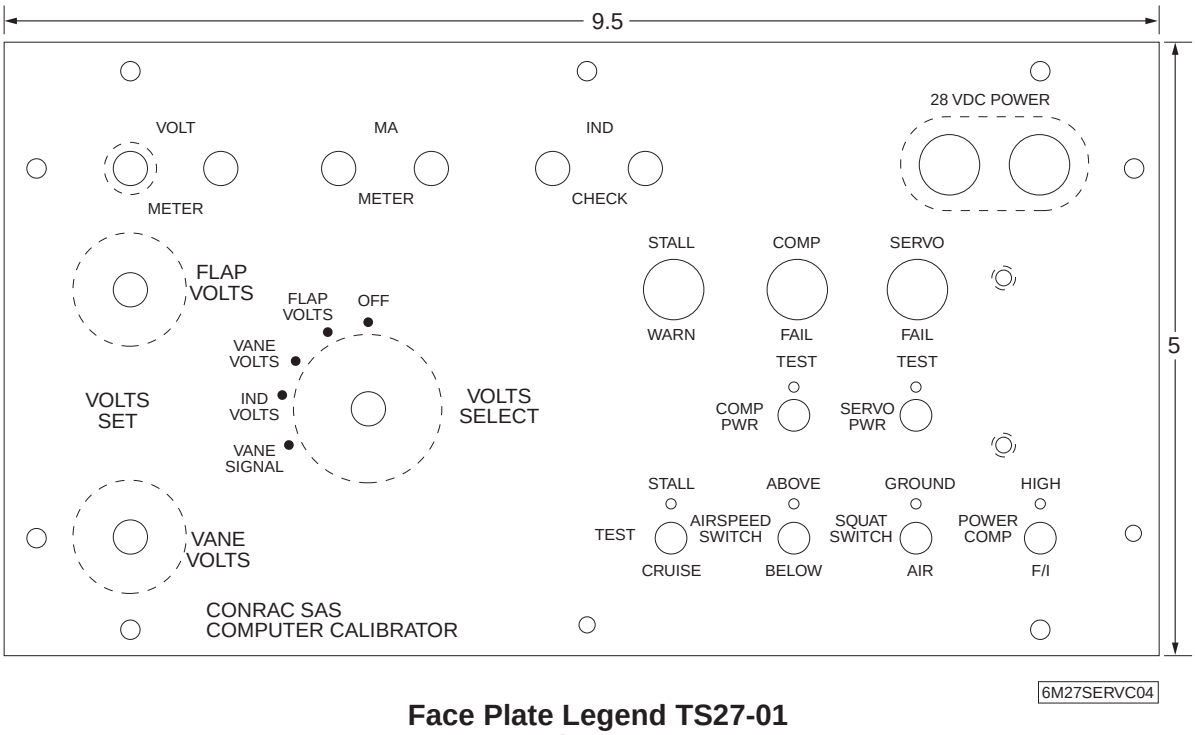
ITEM NO.	NAME	PART NUMBER/ SPECIFICATION	DESCRIPTION	QTY.
16.	SWITCH (S7)	ANY	DUAL STACK ROTARY OR SINGLE STACK DOUBLE WIPER ROTARY	1
17.	RESISTOR (R1)	ANY	200 OHM 10 WATT	1
18.	POTENTIO-METER (R2)	522-5129 MFG TYPE 8400 ALLIED STK# 822-96RC OR EQUIVALENT	10 TURN MICROPOT 1000 OHMS CAN BE 1000 TO 2000 OHMS	1
19.	POTENTIO-METER (R3)	522-5126 MFG TYPE 8400 ALLIED STK# 822-96RC OR EQUIVALENT	10 TURN MICROPOT 5000 OHMS CAN BE 500 TO 1000 OHMS	1
20.	WIRE 22 AWG	ANY	STRANDED	AR



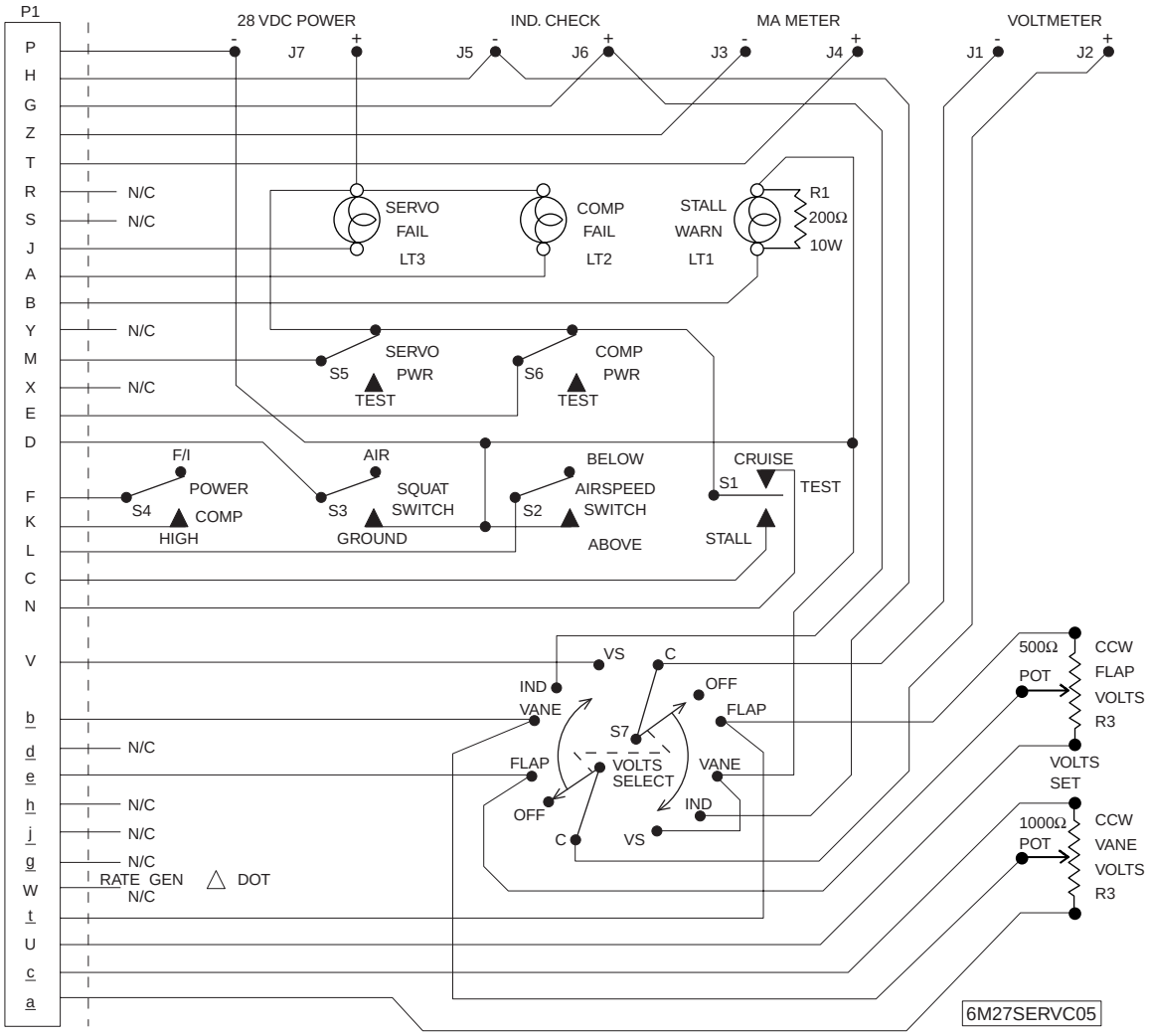
Base Layout TS27-01
Figure 2



Face Plate Layout TS27-01
Figure 3



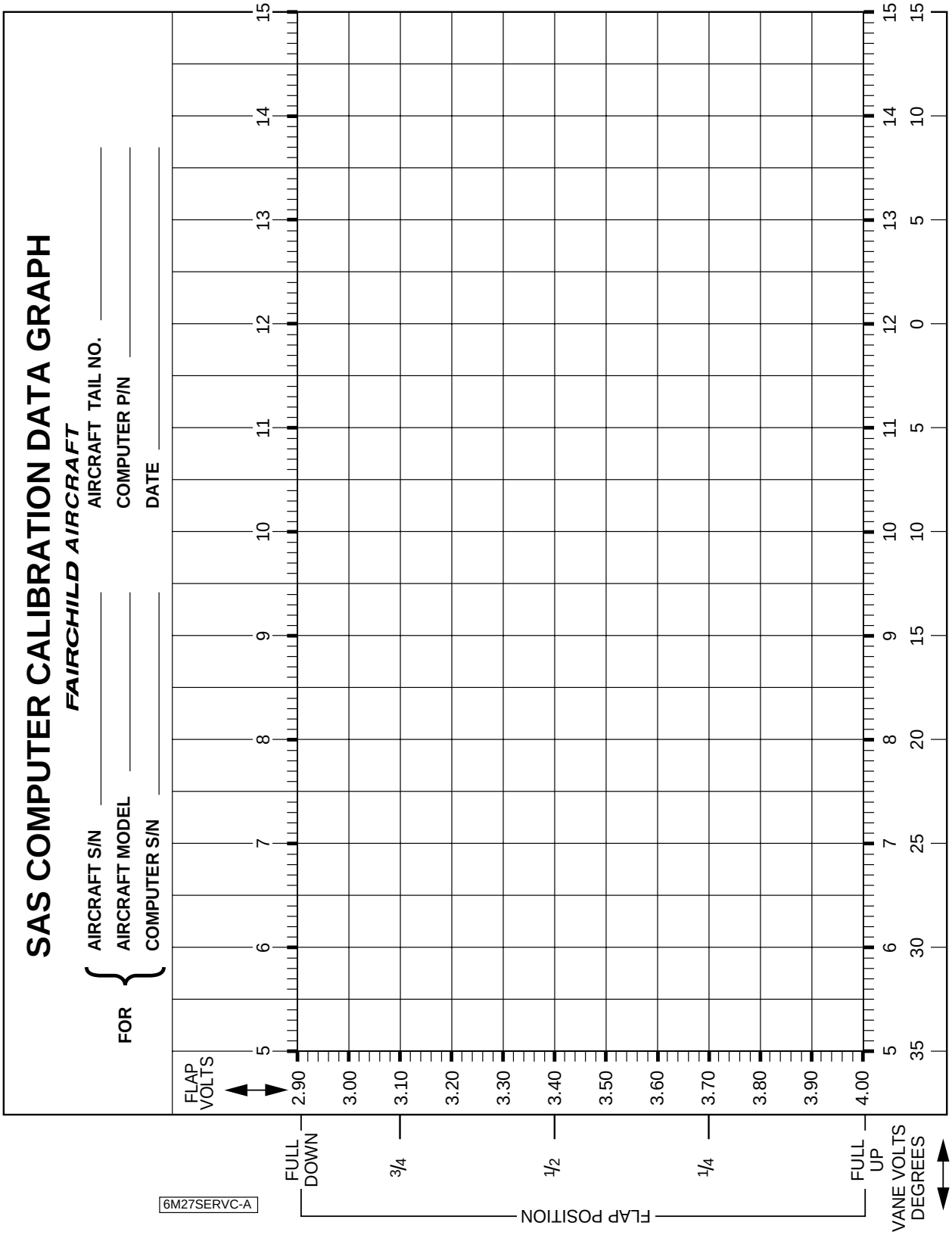
Face Plate Legend TS27-01
Figure 4

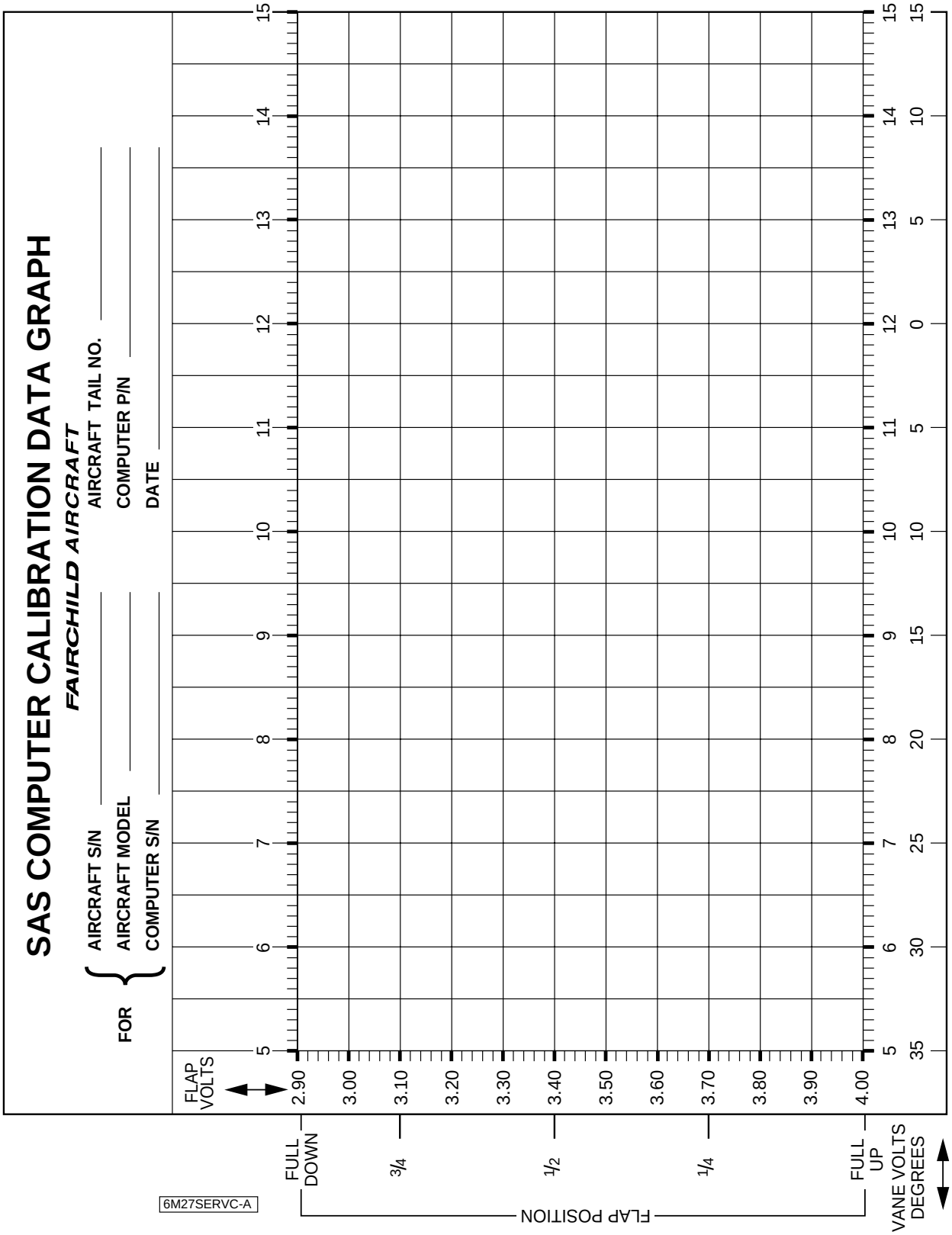


Schematic TS27-01
Figure 5

F. SAS Data Sheet and SAS Computer Calibration Graph

(1) The next four pages are blank data sheets and graphs for CONRAC SAS GROUND set-up.


SAS COMPUTER CALIBRATION DATA GRAPH
FAIRCHILD AIRCRAFT
FOR
AIRCRAFT S/N
AIRCRAFT MODEL
COMPUTER S/N
AIRCRAFT TAIL NO.
COMPUTER P/N
DATE
FLAP VOLTS
2.90
3.00
3.10
3.20
3.30
3.40
3.50
3.60
3.70
3.80
3.90
4.00
FULL DOWN
¾
½
¼
FULL UP
FLAP POSITION
VANE VOLTS DEGREES
6M27SERVC-A


SAS COMPUTER CALIBRATION DATA GRAPH
FAIRCHILD AIRCRAFT
FOR
AIRCRAFT S/N
AIRCRAFT MODEL
COMPUTER S/N
AIRCRAFT TAIL NO.
COMPUTER P/N
DATE
FLAP VOLTS
2.90
3.00
3.10
3.20
3.30
3.40
3.50
3.60
3.70
3.80
3.90
4.00
FULL DOWN
3/4
1/2
1/4
FULL UP
FLAP POSITION
VANE VOLTS
DEGREES
6M27SERVC-A

SAS DATA SHEET

AIRCRAFT S/N_____

COMP S/N _____ CUM _____

DATE _____

	COLUMN					
	A	B	C	D	E	F
STEP	SET FLAP VOLTS TO	SET VANE VOLTS TO	ADJUST COMPUTER POT	TO OBTAIN IND VOLTS	TO OBTAIN MILLAMPS	SAS INDICATOR V/Vs
1	4.00		0 STALL	0.00	N/A	1.0
2	4.00		0 CRUISE	4.59	N/A	1.3
3	3.70		3/8 STALL	0.00	N/A	1.0
4	3.70		3/8 CRUISE	4.59	N/A	1.3
5	3.40		5/8 STALL	0.00	N/A	1.0
6	3.40		5/8 CRUISE	4.59	N/A	1.3
7	2.90		FULL STALL	0.00	N/A	1.0
8	2.90		FULL CRUISE	4.59	N/A	1.3
9	TURN SLOWLY CW	DO NOT TOUCH	N/A CHECK ONLY	4.59	N/A	1.3
IF IND VOLTS NOT PLUS OR MINUS .10 REPEAT STEPS 1 THROUGH 8						
10	2.90	TURN SLOWLY CCW	WARN HORN	STALL WARN LIGHT ON AT 1.75	N/A	1.12
11	2.90	TURN SLOWLY CCW	PUSHER ONSET / PUSHER CURRENT	MA INCREASE AT 0.20	INCREASE / 51 MA	1.02
*12	2.90	N/A	RAMP CURRENT	AT 4.59	25 MA	1.3
*13	2.90	N/A	RAMP ONSET	11.30	8 MA	1.8
14	4.00	N/A	CRUISE TEST	4.59	N/A	1.3
15	4.00		N/A CHECK ONLY	1.75	N/A	1.12
16	4.00		N/A CHECK ONLY	0.20	51 MA	1.02
17	4.00	12.00	N/A CHECK ONLY		M/A	

* Merlin III - T S/N Only

6M27SERVC-B

SAS DATA SHEET

AIRCRAFT S/N_____

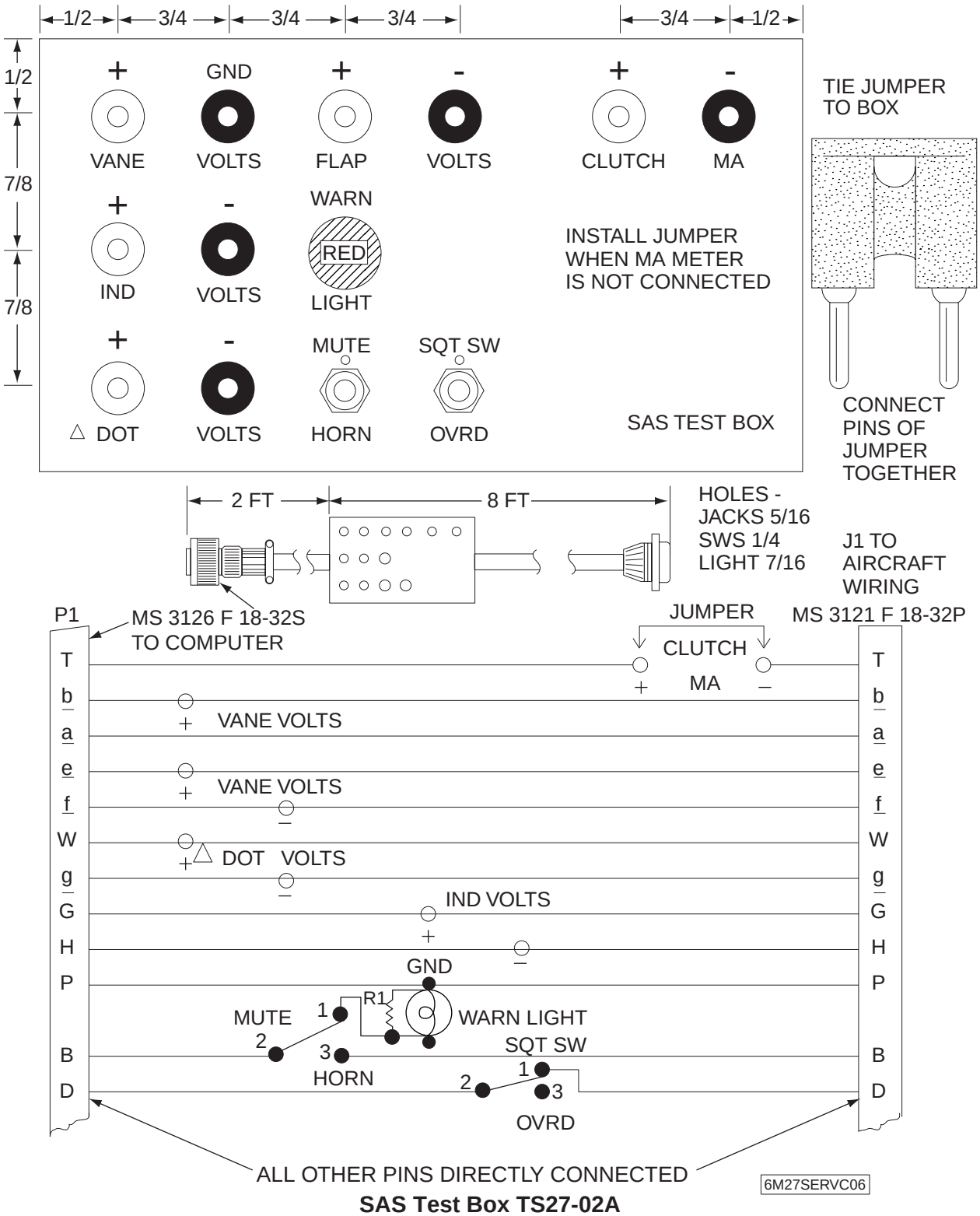
COMP S/N _____ CUM _____

DATE _____

	COLUMN					
	A	B	C	D	E	F
STEP	SET FLAP VOLTS TO	SET VANE VOLTS TO	ADJUST COMPUTER POT	TO OBTAIN IND VOLTS	TO OBTAIN MILLAMPS	SAS INDICATOR V/Vs
1	4.00		0 STALL	0.00	N/A	1.0
2	4.00		0 CRUISE	4.59	N/A	1.3
3	3.70		3/8 STALL	0.00	N/A	1.0
4	3.70		3/8 CRUISE	4.59	N/A	1.3
5	3.40		5/8 STALL	0.00	N/A	1.0
6	3.40		5/8 CRUISE	4.59	N/A	1.3
7	2.90		FULL STALL	0.00	N/A	1.0
8	2.90		FULL CRUISE	4.59	N/A	1.3
9	TURN SLOWLY CW	DO NOT TOUCH	N/A CHECK ONLY	4.59	N/A	1.3
IF IND VOLTS NOT PLUS OR MINUS .10 REPEAT STEPS 1 THROUGH 8						
10	2.90	TURN SLOWLY CCW	WARN HORN	STALL WARN LIGHT ON AT 1.75	N/A	1.12
11	2.90	TURN SLOWLY CCW	PUSHER ONSET / PUSHER CURRENT	MA INCREASE AT 0.20	INCREASE / 51 MA	1.02
*12	2.90	N/A	RAMP CURRENT	AT 4.59	25 MA	1.3
*13	2.90	N/A	RAMP ONSET	11.30	8 MA	1.8
14	4.00	N/A	CRUISE TEST	4.59	N/A	1.3
15	4.00		N/A CHECK ONLY	1.75	N/A	1.12
16	4.00		N/A CHECK ONLY	0.20	51 MA	1.02
17	4.00	12.00	N/A CHECK ONLY		M/A	

* Merlin III - T S/N Only

6M27SERVC-B



SAS Test Box TS27-02A
Figure 6

G. Conrac SAS Test Box (TS27-02)

(1) The test box is used for ground set-up, voltage checks, in-flight set-up, and troubleshooting the CONRAC SAS system. Because of the location of the SAS computer on the Merlin IIIB, the TS27-02 must be used for in-flight set-up of SAS.

H. Construction of TS27-02

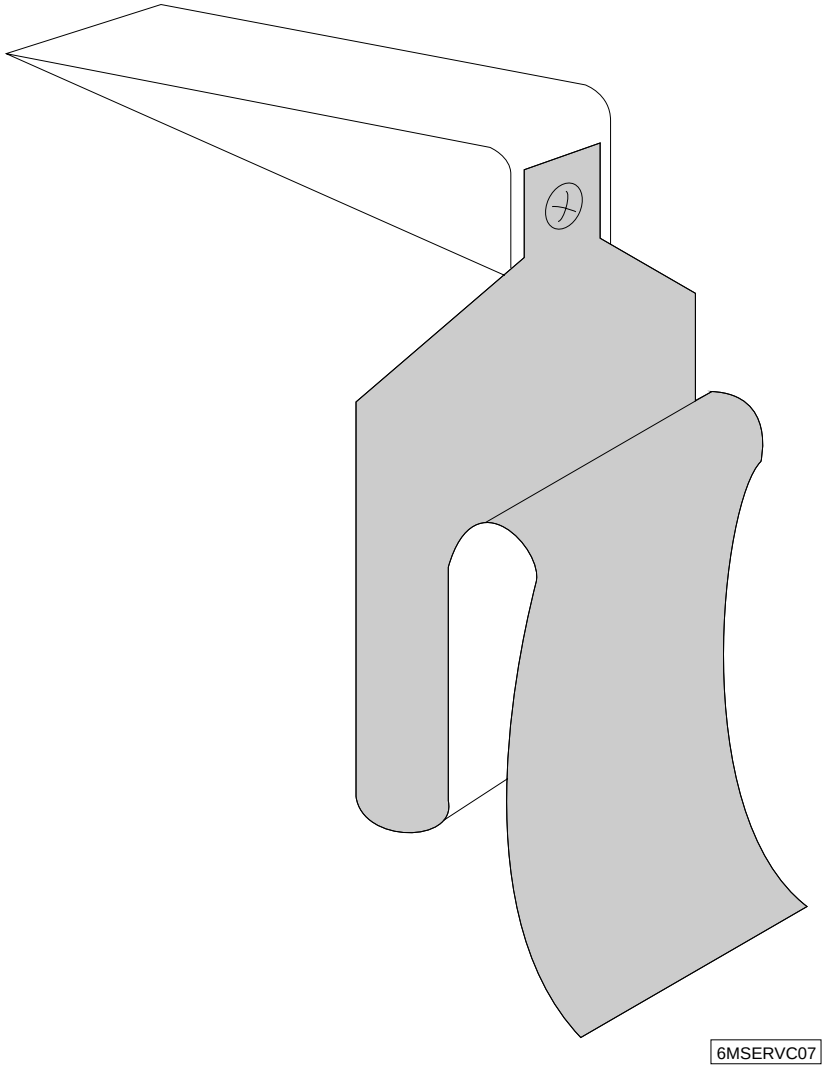
(1) The tester can be built into any size box that will accommodate the components. The particular tester shown in Figure 6 is 3 x 5 x 2. The jacks can be banana or pin type. The jumper is banana type. CLUTCH MA jacks must be 3/4 of an inch apart for use with this type jumper. The jumper must be installed when milliammeter is not connected to CLUTCH MA jacks. The WARN light should be red and should be 28 VDC rated and have replaceable bulb. The socket must not be case grounded and have two terminals. The two switches should be SP-DT miniature type. The WARN light and HORN/MUTE switch are provided to silence the stall warning when making adjustments to the SAS computer. When adjusting computer pot WARN HORN switch must be to HORN position. The SQT SW/OVRD switch is provided to override (cheat) the left main gear scissor switch during ground set-up and check of SAS computer. All wires should be 22 AWG.

MATERIAL LISTING TS27-02

ITEM NO.	NAME	PART NUMBER/ SPECIFICATION	DESCRIPTION	QTY.
1.	BOX	CU2106-A OR EQUIVALENT	3 in. X 5 in. X 2 in.	1
2.	ANCHOR WIRE (CLAMP)	ANY	TO PREVENT WIRES FROM BEING TORN LOOSE FROM CON-NECTIONS INSIDE BOX	2
3.	CONNECTOR (P1)	MS3126F18-32S	– – – –	1
4.	CONNECTOR (J1)	MS3121F18-32P	– – – –	1
5.	JACK, BANANA	ANY	BLACK	5
6.	JACK, BANANA	ANY	RED	5
7.	LIGHT	B355A OR MS25041-5 OR EQUIVALENT	TWO TERMINAL TYPE (RED LENS)	1
8.	BULB	327 OR MATCHING ITEM 7	28 VDC	1
9.	RESISTOR	ANY	200 OHM 10 WATT	1
10.	SWITCH	8A1011	SP-DT (MINIATURE)	2
11.	JUMPER, BANANA	ANY MATCHING ITEMS 5 AND 6	– – – –	1
12.	WIRE 22 AWG	ANY	STRANDED	AR

I. Squat Switch Cheater Tool (TF27-03)

(1) TF27-03 (32-82020-021) is used to depress the scissor switches on the left or right main landing gears during SAS ground set-up or other maintenance functions. The tool has a large and prominent red flag to prevent it from being inadvertently left in the scissor. It is recommended that the tool be used rather than a screwdriver. The tool is constructed from 1/2 inch bar stock aluminum cut to a wedge of approximately 15 degrees.



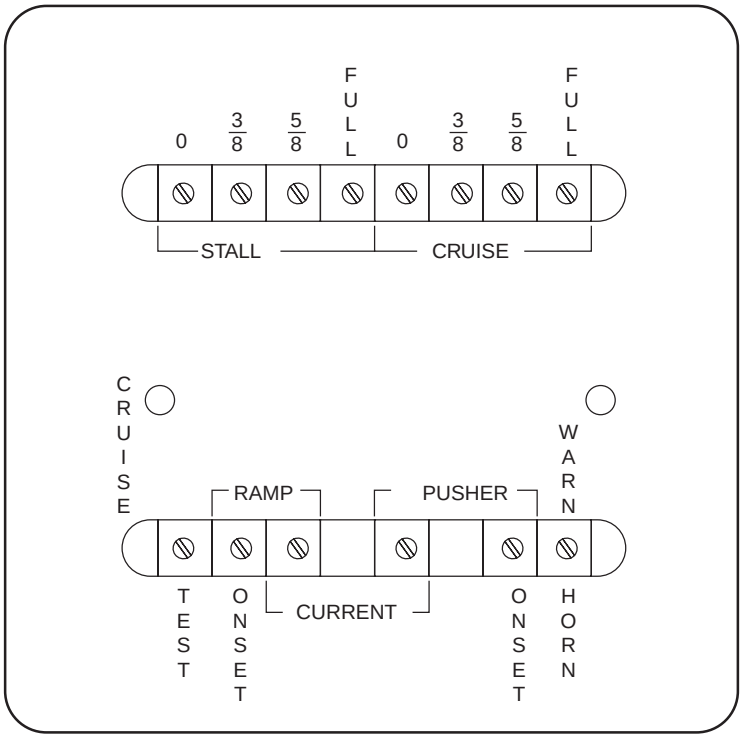
Squat Switch Cheater Tool
TF27-03 (32-82020-021)
Figure 7

J. Adjustment Location Plate (TF27-01)

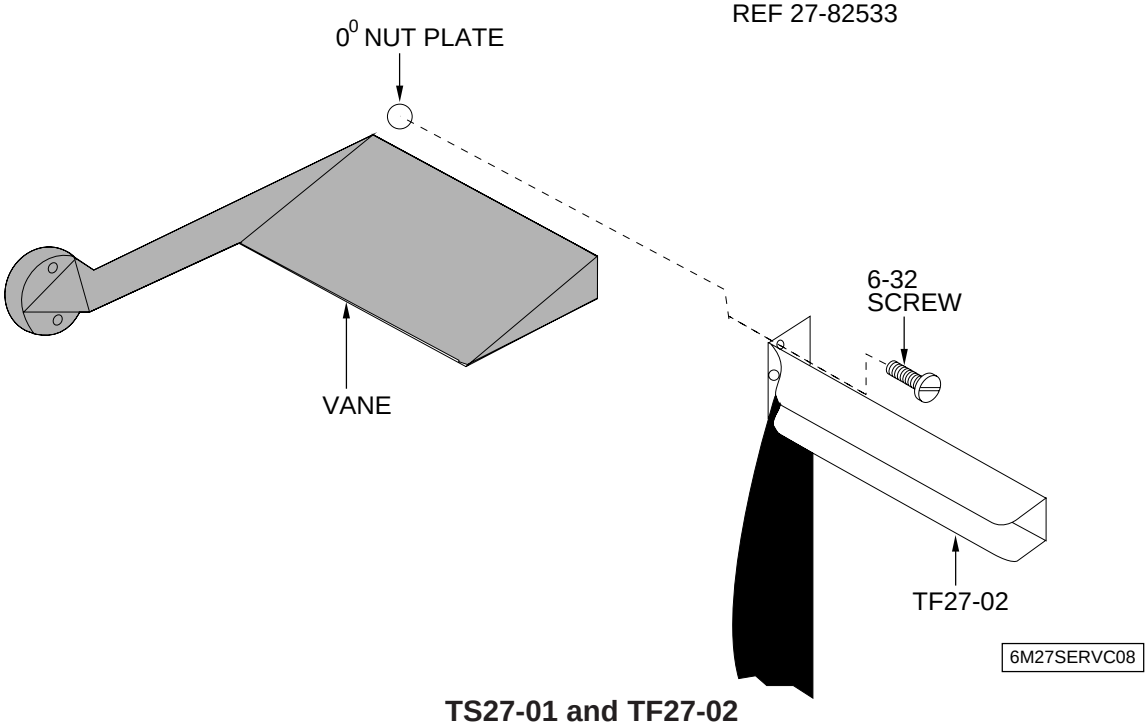
(1) TF27-01 is used to identify the trimmer pots on the CONRAC SAS COMPUTER. It can be constructed from aluminum or plastic sheet and engraved or cut directly from Figure 8 and glued to a piece of cardboard.

K. Vane Adjustment Tool (TF27-02)

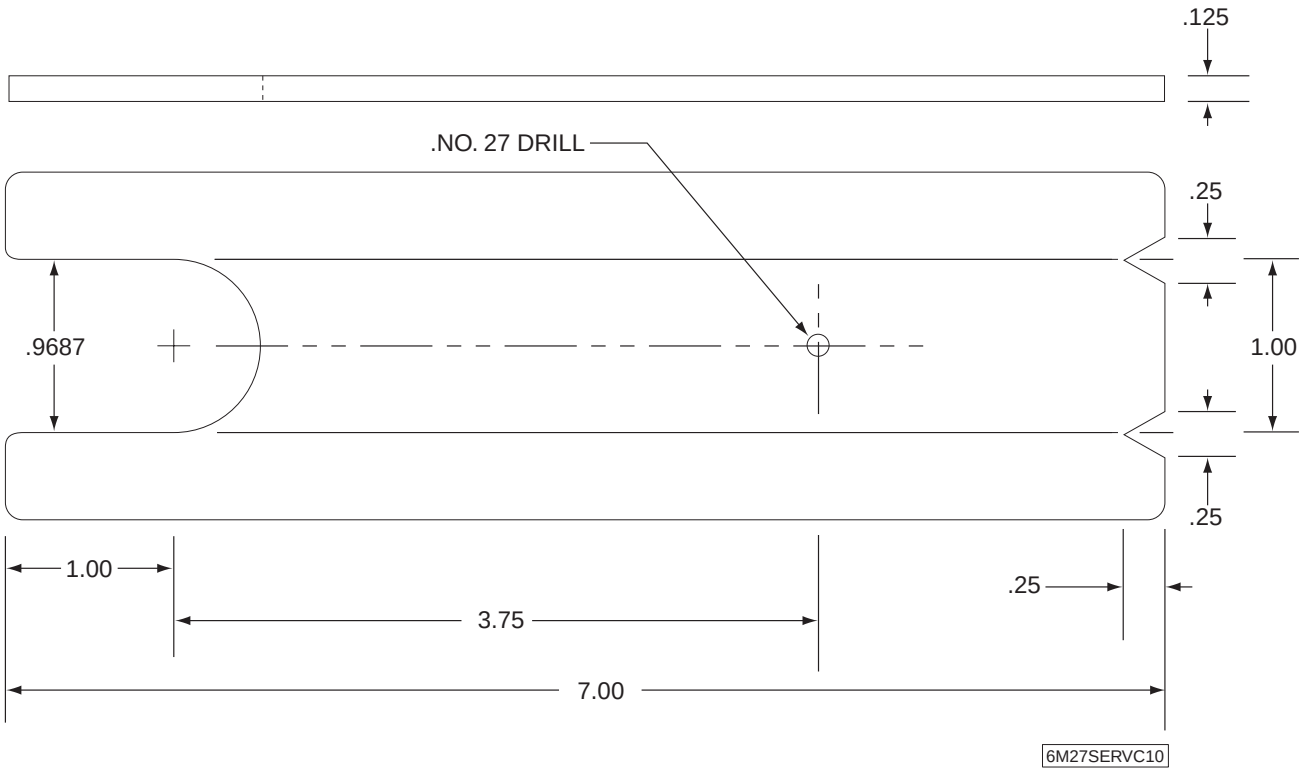
(1) TF27-02 is used when adjusting vane voltage for CONRAC SAS. The tool is slid onto the vane and attached to the zero nutplate using a 6-32 screw. The tool holds the vane stationary during the adjustment procedure. It can also be used as a vane protector. The tool is constructed from a stainless steel sheet.



COMPUTER ADJUSTMENT
LOCATION PLATE
TF27-01
REF 27-82533



TS27-01 and TF27-02
Figure 8



SAS Vane Alignment
Tool TF27-04
Figure 9

L. SAS Vane Alignment Tool (TF27-04)

(1) TF27-04 is used to locate the zero degree nutplate. The zero nutplate is used to accurately set vane voltages. The tool can be made of aluminum or any material of appropriate thickness, or it can be cut directly from Figure 9 and used.

M. Procedure for Installing Zero Degree Nutplate

(1) Jack aircraft.

(2) Place protective covering on left wing near fuselage.

(3) Obtain a 24 inch or longer carpenters level.

(4) Stand on left wing facing fuselage.

(5) Place level on centerline of top of fuselage.

(6) Level aircraft longitudinally by raising or lowering nose jack.

(7) After aircraft is leveled remove level.

(8) Place level below SAS vane and against pivot point.

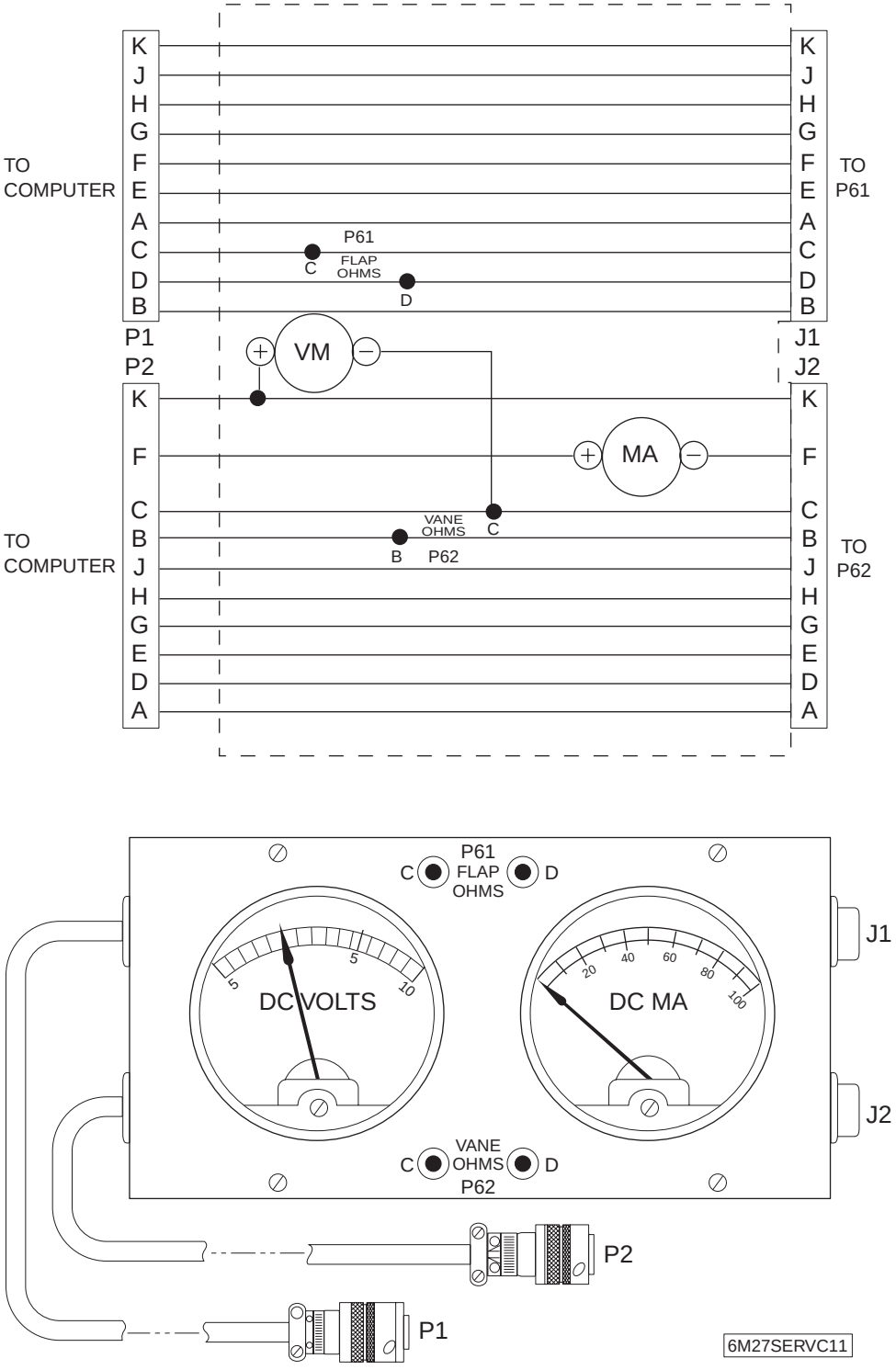
(9) Center bubble; using top of level as a straightedge draw a line approximately eight inches long from pivot point aft with a heavy black soft pencil.

(10) Place cutout on vane alignment tool TF27-04 over pivot point of vane. Align bottom V notch on tool with pencil line. Drill hole in fuselage at location of hole in tool.

(11) Drill hole with No. 27 bit and install nutplate NAS697A06 (6-32) at location with two countersunk rivets.

(12) Install No. 6-32 screw in nutplate.

(13) Down jack aircraft.



Rosemount SAS Calibrator
TS27-03
Figure 10

N. Rosemount SAS Calibrator (TS27-03)

(1) TS27-03 is required for ground and in-flight set-up of SAS on Merlin III aircraft T201-275, 277-291. The tester is used to set clutch voltage, clutch milliamps, and pusher milliamps. The tester is provided with a test jack to read flap and vane ohms. The tester is connected to the aircraft wiring, and the computer is disconnected when reading ohms. The tester can also be used on the Merlin IV and Metro; however, the voltmeter is not functional on these aircraft. Procedures for use of tester can be found in Section 27-31-06.

(2) There are other versions of this tester in use. Some of the testers have flap simulator circuitry for simulation of the flap. Simulator circuitry should not be used for ground or in-flight set-up of SAS. These testers should be modified per the schematic in Figure 10.

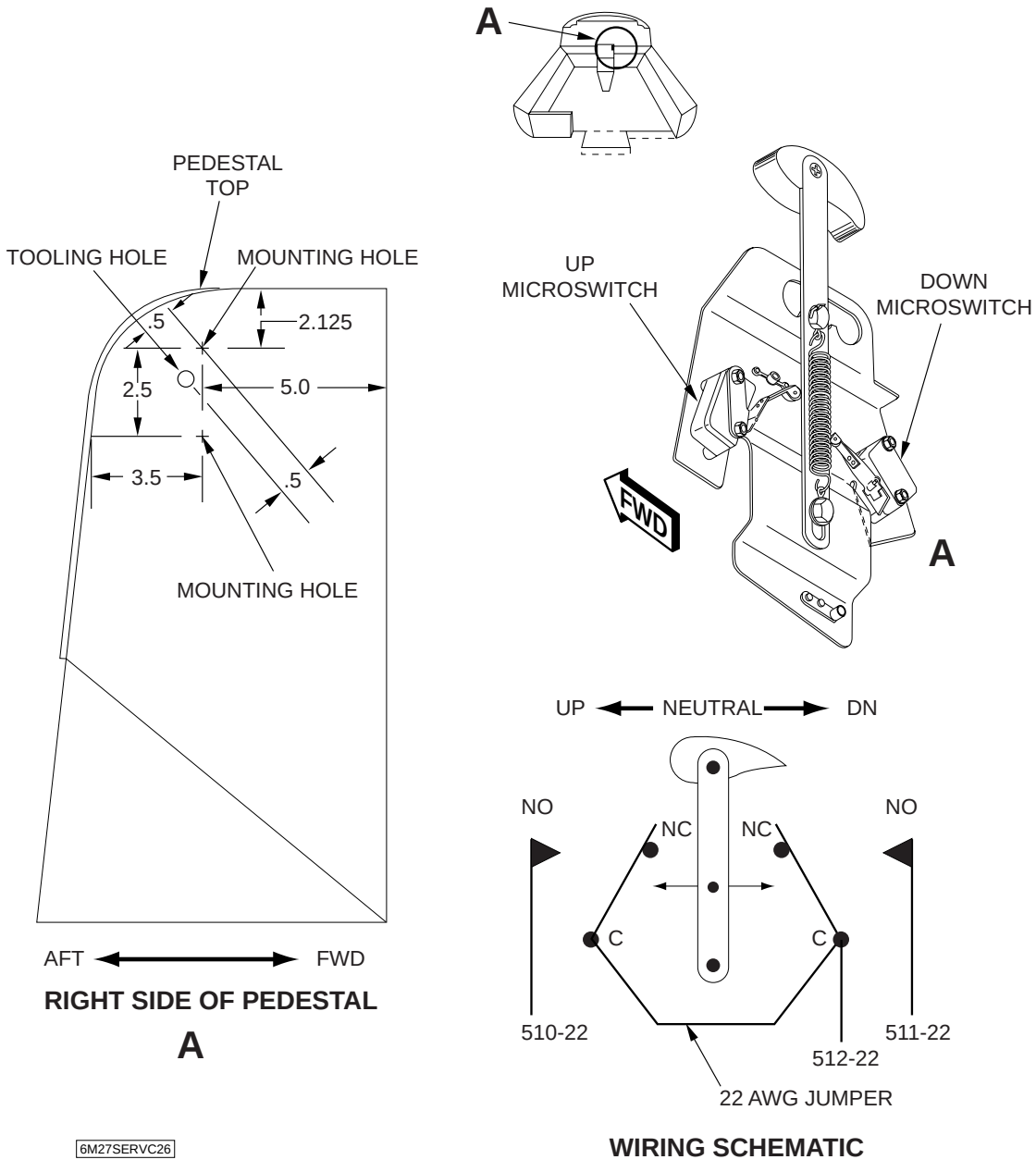
(3) The tester can be built into any size box that will accommodate the type of meters used. The voltmeter should be accurate to within .25 (1/4) of a volt and have a scale from minus five volts to plus ten. The milliammeter should be accurate to within one milliamp and have a full scale reading of at least 60 milliamps. The test jacks can be banana or pin type and should be marked clearly and indelibly as in Figure 10. The lead should be no longer than two feet. All wire should be 22 AWG.

MATERIAL LISTING TS27-03

ITEM NO.	NAME	PART NUMBER/ SPECIFICATION	DESCRIPTION	QTY.
1.	BOX	ANY	SIZE TO ACCOMO-DATE ITEMS 2 AND 3	1
2.	VOLTMETER	ANY	- 5 TO + 10 DC VOLTS	1
3.	MILLIAMMETER	ANY	0 TO 100 DC MILLIAMPS	1
4.	JACK	ANY	– – – –	4
5.	CONNECTOR (P1)	MS3116F-12-10SW	– – – –	1
6.	CONNECTOR (P2)	MS3116F-12-10S	– – – –	1
7.	CONNECTOR (J1)	MS3112E-12-10PW	– – – –	1
8.	CONNECTOR (J2)	MS3112E-12-10P	– – – –	1
9.	GROMMET	ANY	– – – –	1
10.	ANCHOR WIRE (CLAMP)	ANY	TO PREVENT WIRES FROM BEING TORN LOSE FROM CON-NECTIONS INSIDE OF BOX	2
12.	WIRE 22 AWG	ANY	STRANDED	AR

Z. Replacement of Flap Control Handle

(1) Two types of flap control handles are used on SA226 aircraft. A double microswitch type on aircraft T201-282, 298-999; AT001-061, 071-999; TC201-246, 280-999 and a single toggle switch type on aircraft T283-297, AT062-070, TC247-279. When replacement of single toggle switch control is required, it should be replaced by the double microswitch control (see Illustrated Parts Catalog for part numbers). Replacement will require drilling two mounting holes in the right side of the pedestal and a minor electrical wiring change. See Figure 11 for dimensions and wiring diagram.



Flap Control Handle Installation
Figure 11

(2) Removal – Flap Control Handle

(a) Ensure battery switches OFF.

(b) Remove knob from flap control handle.

(c) Remove center pedestal edge lighted panel/overlay.

(d) Remove center pedestal cover plate, if necessary.

(e) Remove two screws attaching flap control handle to right side of pedestal.

(f) Cut string ties or remove clamps, as necessary, to allow assembly to be brought out front of pedestal for easy access to wire terminals.

(g) Slide vinyl tubing back and remove wire terminals from switch terminals.

(3) Installation – Flap Control Handle

(a) Install wires on switch terminals of new assembly (see Figure 11). If replacing single toggle switch control with double microswitch control:

1 Replace #6 stud size wire terminals with #4 stud size wire terminals.

2 Cut a length of 22 AWG (gauge) wire approximately four inches long and attach #4 stud size wire terminals.

3 Cut four two inch lengths of 1/4 inch vinyl tubing and slide over wire terminals.

4 Connect wires to microswitches as shown in Figure 11.

(b) Slide vinyl tubing over terminals and tie. Service Bulletin 24-011 detailed adding vinyl tubing to landing gear and flap control switches (see Service Bulletin if necessary).

(c) Install flap control assembly with two screws. If replacing single toggle switch control with double microswitch control:

1 Drill and countersink two holes to accommodate #8 screws.

2 Locate holes to dimensions shown in Figure 11.

3 Install new flap control assembly with two screws previously used to attach old assembly to right side of pedestal.

4 Tie and clamp wires as necessary.

(d) Install pedestal center cover and edge lighted panel/overlay.

(e) Install knob on flap control handle.

(f) Perform flap extension and retraction check, see Section 32-50-00.

1. General

A. System Components

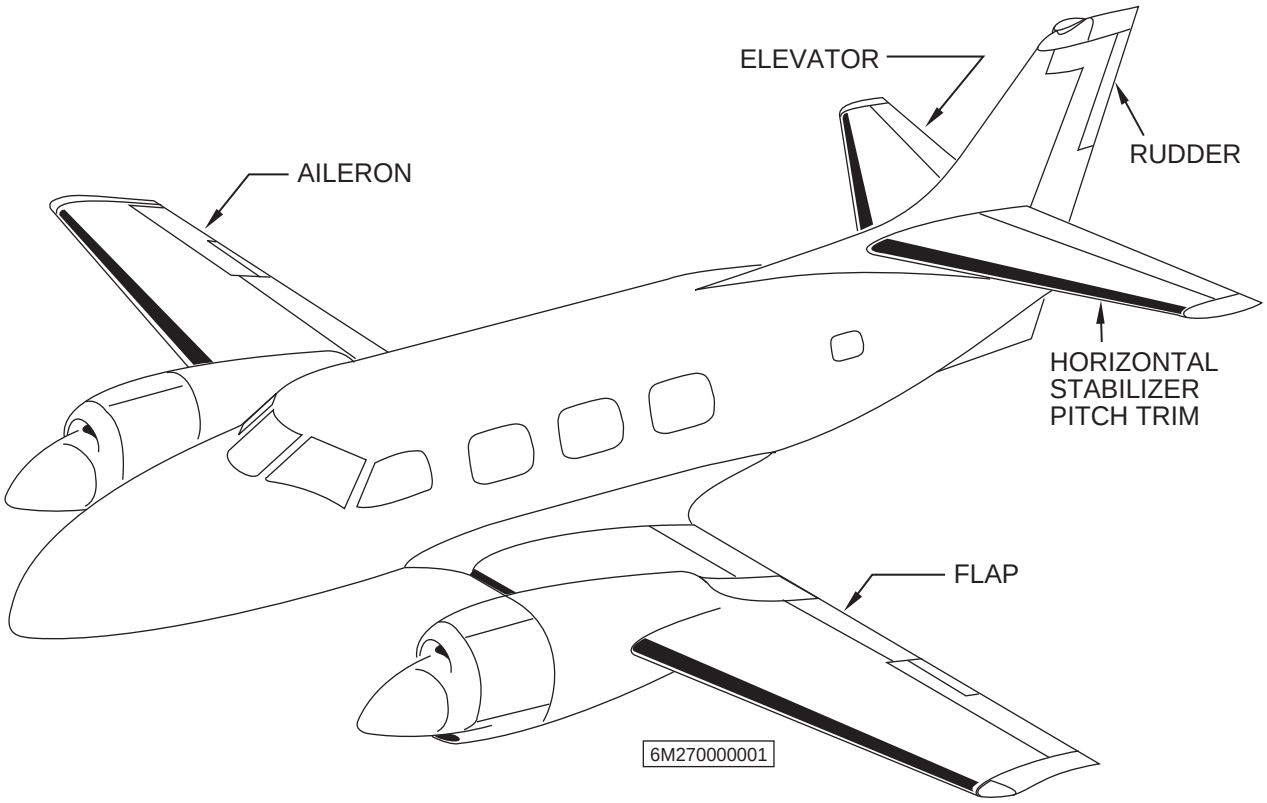
(1) The ailerons, rudder, and elevators are manually operated from either the pilot or copilot positions by a conventional yoke and rudder arrangement. The ailerons and rudder are interconnected to aid in coordinating turns. The ailerons and the rudder have moveable trim tabs operated by their respective control wheels located on the cockpit pedestal. A fixed tab is located on each aileron and on the vertical stabilizer cap.

(2) A stall avoidance system (SAS) is installed to both visually and aurally warn the pilot of an impending stall.

(3) Pitch trim is accomplished by electrically positioning the horizontal stabilizer.

(4) The flaps are electrically controlled and hydraulically activated. The flaps are also mechanically interconnected for symmetrical flap operation.

(5) A gust lock, located on the control pedestal, is interconnected with the power levers to prevent takeoff with the controls locked.



Flight Controls
Figure 1

B. Control Cables

(1) Flight control cables used on all SA226 series aircraft are made of MIL-W-83420, Type I, Composition A. All cables are pre-stretched prior to installation. Two types of cables are used: 7 X 7 and 7 X 19. A 7 X 7 cable is made up of seven bundles of seven strands of wire. A 7 X 19 cable is made up of seven bundles of 19 strands of wire.

(2) Three diameters of cables are used in all aircraft. Cables 1/8 inch in diameter are used for aileron, rudder, and elevator control. Cables 3/32 inch in diameter are used for the stick pusher, yaw damper, and control lock. The trim tabs use a 1/16 inch diameter cable.

(3) Cables are pre-stretched (before attaching end fittings) to the following tension loads:

1/8 inch diameter - 1,200 pounds

3/32 inch diameter - 600 pounds

1/16 inch diameter - 288 pounds

(4) The 1/8 inch and 5/32 inch cables are 7 X 19 type and the 1/16 inch cable is a 7 X 7 type cable. All control cables are cut to approximate length before installation and cut to exact specifications during final adjustments.

(5) Cable assemblies shall never be coiled to less than the minimum diameter shown in Table 1.

Cable Coiling Diameters (Inches)									
Cable Size	1/16	3/32	1/8	5/32	3/16	7/32	1/4	5/16	3/8
Coil Diameter	10	14	19	23	29	32	38	48	56

Control Cables
Table 1

1. Balancing – Controls Surfaces

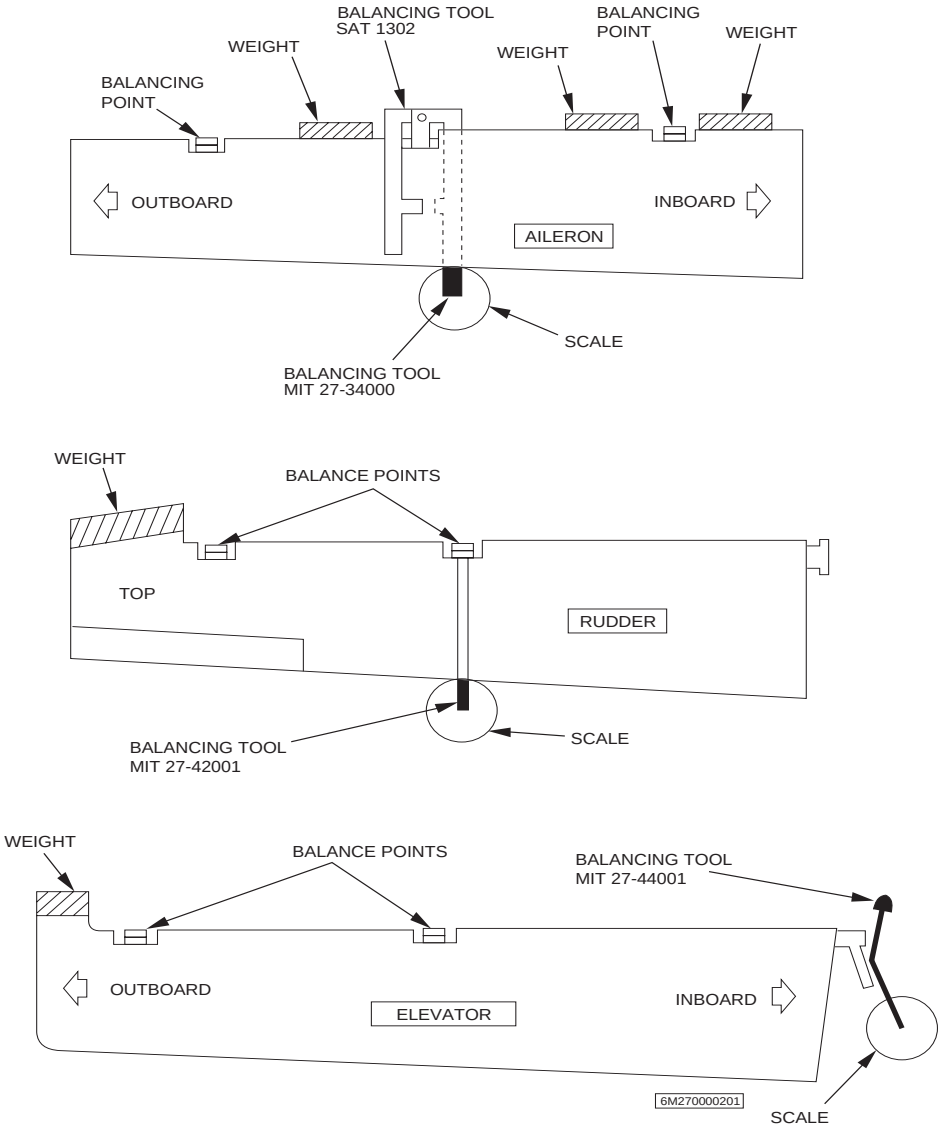
NOTE: Primary flight controls must be balanced to values within the "Flight Balance" range whenever they are painted or repainted, repaired, modified, or if static wick type, number, or location is changed.

A. Procedures

(1) Procedures used for balancing primary aerodynamic control surfaces of SA226–T, T(B), TC, and AT aircraft are divided into two (2) sections.

(a) Balancing procedures for use with factory tooling.

(b) Alternate balancing procedures.



Control Surface Balancing
Figure 201

(2) Either set of procedures used will provide the same result; however, step **1**., A., (1), (a) may be more suitable for large fleets and is similar to original factory procedures. Step **1**., A., (1), (b) provides procedures more suitable for field use.

NOTE: Balance data may differ depending on aircraft model/configuration. Use balance data provided for specific aircraft model/configuration, or improperly balanced flight controls may result.

B. Definitions

(1) Static Balance ..... Numerical value of the moment about the hinge line with control surface stationary and chord plane horizontal.

(2) Neutral Balance ..... Balance characteristic of a control surface whose center of gravity coincides with the hinge line (i.e. "zero moment").

(3) Static Underbalance..... Balance characteristic of a control surface whose center of gravity is aft of the hinge line (i.e. "trailing-edge heavy").

(4) Static Overbalance..... Balance characteristic of a control surface whose center of gravity is forward of the hinge line (i.e. "leading-edge heavy").

(5) Moment ..... Torque or rotational force about the control surface hinge line.

(6) Moment-arm..... The distance from the control surface hinge line to the point at which a weight or force is applied. This distance is measured at right angles to the hinge line and assumes that the chord plane of the surface is horizontal.

(7) Flight Balance ..... The range of balance values acceptable for flight use. Control surfaces shall be complete and in flight configuration including paint, static wicks, and repairs (except electrical bonding straps) prior to balancing.

C. Requirements – General

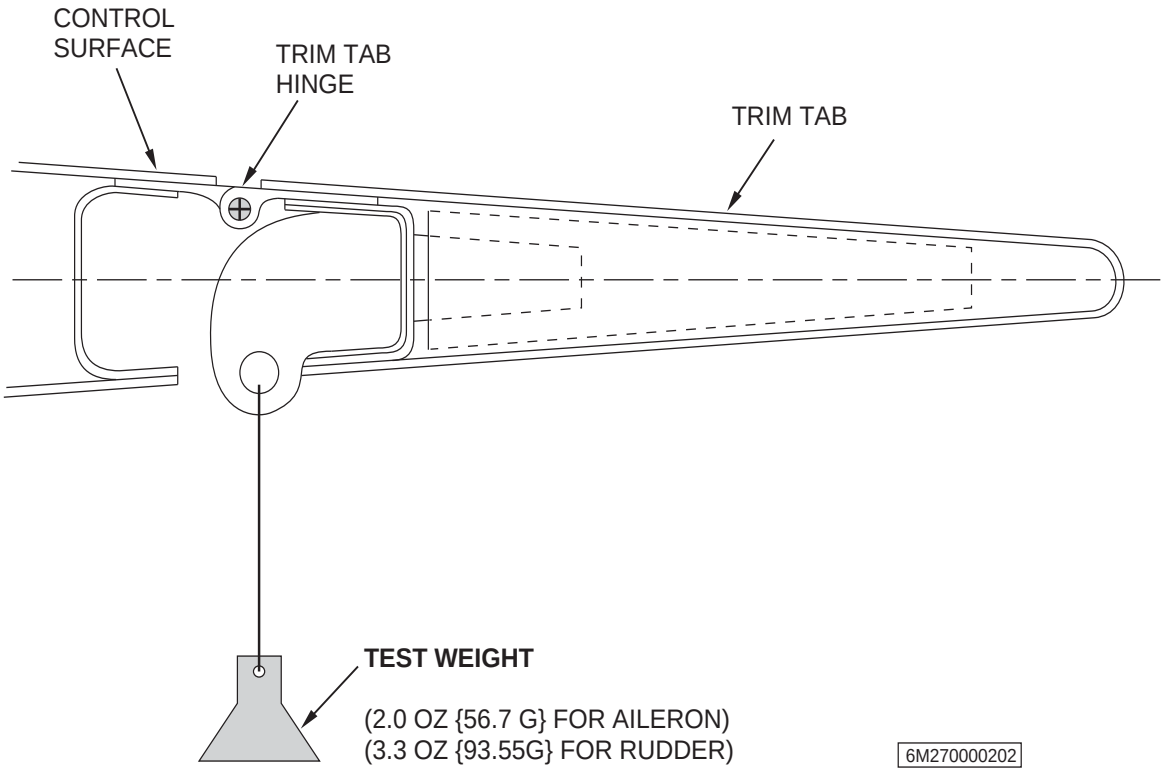
(1) Remove control surfaces from aircraft in complete configuration including cap, static wicks, repairs, and paint. Remove bonding straps.

NOTE: Control surfaces equipped with static discharge wicks must be balanced with wicks installed. Control surfaces require rebalance if there are changes in type, location, and number of static wicks installed. Aircraft may be operated with up to two wicks missing per surface provided all outboard wicks are in place.

(a) Remove trim actuator link and pushrods from ailerons.

(b) Remove lower section of torque tube, trim actuator pushrods, and attaching hardware from rudder.

(2) Suspend a test weight from the trim actuator attach point of ailerons or rudder to simulate trim actuator pushrod weight during balancing. Test weight for aileron is 2.0 oz (56.7g); test weight for rudder is 3.3 oz (93.55 g) (see Figure 202).



Aileron and Rudder Trim Actuator Compensation
Figure 202

(3) Carefully support control surfaces at hinge points. Ensure a minimum of rotational friction.

(4) Maintain level attitude of control surface during balancing measurement.

(5) Perform balancing procedures in a draft free room.

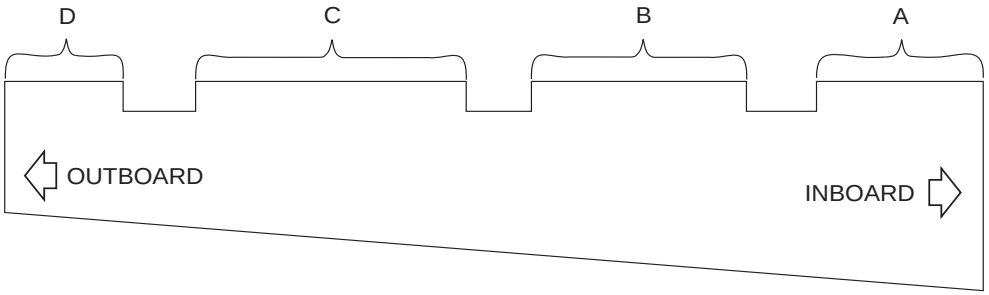
NOTE: Lead weights are used to attain balance of control surfaces. Some lead may need to be removed or added during balancing.

WARNING: INGESTION OR INHALATION OF LEAD PARTICLES POSES A HEALTH HAZARD. BE SURE TO CLEAN HANDS THOROUGHLY AFTER HANDLING LEAD WEIGHT.

(6) Remove weight from rudder by drilling-out counterweight as necessary to obtain correct scale reading. Should additional weight be required, one (1) 0.039 inch (0.99 mm) thick lead shim or one (1) 0.063 inch (1.60 mm) thick lead shim may be added between counterweight and structure. Use MS24694-86 screws in place of standard screws if shim is used.

(7) Remove weight from elevators by drilling-out counterweight as necessary to obtain correct scale reading. Should additional weight be required, lead may be added by inserting drilled-out pieces back into the holes and hammering down with a punch. Apply sealant to hole after hammering lead in place.

(8) Remove weight from ailerons by shaving material from either end of the three weights. Care must be taken when shaving material as it cannot be reattached. Should additional weight be required, replacement weight must be used. Remove material from weights or replace weights in the sequence shown in Figure 203. Add weight IAW Figure 204.



Replace weights in the following order:
Zone A, D, B, C

Remove weights in the following order:
Zone C, B, D, A

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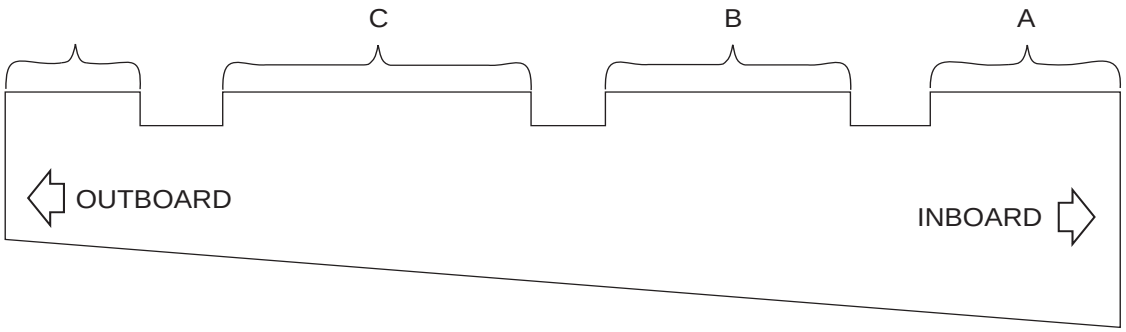
Aileron Weight
Removal/Replacement Sequence
Figure 203



Add weight as follows:
Zone A: Add 35,±5% of total weight
Zone B: Add 35,±5% of total weight
Zone C: Add remaining weight

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Aileron Weight Addition Sequence
Figure 204

D. Balancing Procedures For Use With Factory Tooling

(1) Equipment Required for Balancing:

(a) 2.0 oz (56.7 g) Aileron trim weight.

(b) 3.3 oz (93.55 g) Rudder trim weight.

(c) Bolts (0.250 inch diameter) to insert through control surface hinges.

(d) Pennsylvania Model 4020T scale or equivalent.

NOTE: If other scales are used, platforms of TSJ 27-10005 may require modification to allow control surfaces to be level during balancing. Scale must read to nearest 0.1 oz (2.84 g).

(e) TSJ 27-10005 Balance fixture.

(f) SAT 1302 Balance adapter, aileron (limited aircraft, see Table 202).

(g) MIT 27-34000 Balance adapter, aileron.

(h) MIT 27-42001 Balance adapter, rudder.

(i) MIT 27-44001 Balance adapter, elevator.

(2) Aileron Balancing:

(a) Remove aileron.

(b) Tape trim tab in a streamlined position. Use minimum amount of tape so as not to affect balance.

(c) Hang a 2.0 oz (56.7 g) test weight from the trim tab actuator pushrod bolt hole (see Figure 202).

(d) Install 0.250 inch diameter bolts and nuts through inboard and outboard aileron hinges securing SAT 1302 and MIT 27-34000 balance tools to aileron center hinge.

(e) Place aileron in TSJ 27-10005 balance fixture, trim hinge side up.

(f) Verify free movement of aileron.

(g) Position scale centered under point of MIT 27-34000 balancing tool. Hold MIT 27-34000 balancing tool to remove any load from scale, verify zero (0) setting. After zero (0) is verified allow the MIT 27-34000 balancing tool to contact scale. Ensure aileron is level.

(h) Read scale. Verify reading is within limits shown in scale reading column of Table 201 or Table 202 (depending on aircraft model).

(i) Shave or replace weight to achieve required reading IAW step **1**., C., (8).

(3) Rudder Balancing:

(a) Remove rudder.

(b) Tape trim tab in a streamlined position. Use minimum amount of tape so as not to affect balance.

(c) Hang a 3.3 oz (93.55 g) test weight from the trim tab actuator pushrod bolt hole (see Figure 202)

(d) Install 0.250 inch diameter bolt and nut through upper rudder hinge fitting securing MIT 27-42001 balancing tool to rudder center hinge.

(e) Place rudder horizontally on knife edges of TSJ 27-10005 fixture with left side of rudder facing down.

(f) Verify free movement of rudder.

(g) Position scale centered under point of MIT 27-42001 balancing tool. Hold MIT 27-42001 balancing tool to remove any load from scale, verify zero (0) setting. After zero (0) is verified allow the MIT 27-42001 balancing tool to contact scale. Ensure rudder is level.

(h) Read scale. Verify reading is within limits shown in scale reading column of Table 201 or Table 202 (depending on aircraft model).

(i) Add or remove weight to achieve required scale reading IAW step **1**., C., (6).

SURFACE	TOOLS	TRIM WT. (oz.)	MOMENT (in.-lb.)	SCALE READING (oz.)
Aileron	MIT 27-34000	2.0 (LH aileron only)	6.4 Underbalance to 4.4 Underbalance	12.3 to 9.6
Rudder	MIT 27-42001	3.3	46.0 Underbalance to 44.0 Underbalance	33.7 to 32.6
Elevator	MIT 27-44001	N/A	22.0 Underbalance to 20.0 Underbalance	29.3 to 26.7

Acceptable Flight Balance Values
SA226-T, T(B), AT, TC except S/N T-394, T397,
T-400 thru T-417, TC-398 thru TC-419
(For Use With Factory Tooling)
Table 201

(4) Elevator Balancing:

(a) Remove elevator.

(b) Install 0.250 inch diameter bolts and nuts through outboard and center elevator hinges. Place elevator horizontally on knife edges of fixture with upper surface facing up.

(c) Place nominal 0.250 inch diameter hole in MIT 27-44001 tool over the inboard hinge stud and secure to elevator inboard hinge with 0.250 inch diameter bolt and nut through slot in tool into elevator bellcrank.

(d) Verify free movement of elevator.

SURFACE	TOOLS	TRIM WT. (oz.)	MOMENT (in.-lb.)	SCALE READING (oz.)
Aileron	SAT 1302 MIT 27-34000	2.0 (LH aileron only)	0.0 Overbalance to 4.0 Overbalance	8.2 to 2.9
Rudder	MIT 27-42001	3.3	46.0 Underbalance to 20.0 Underbalance	33.7 to 18.8
Elevator	MIT 27-44001	N/A	22.0 Underbalance to 14.0 Underbalance	29.3 to 18.7

Acceptable Flight Balance Values
SA226-T(B) (S/N) T-394, T397,
T-400 thru T-417, TC-398 thru TC-419
(For Use With Factory Tooling)
Table 202

(e) Position scale centered under point of MIT 27-44001 balancing tool. Hold MIT 27-44001 balancing tool to remove any load from scale, verify zero (0) setting. After zero (0) is verified allow the MIT 27-44001 balancing tool to contact scale. Ensure elevator is level.

(f) Read scale. Verify reading is within limits shown in scale reading column of Table 201 or Table 202 (depending on aircraft model).

(g) Add or remove weight to achieve required scale reading IAW step **1**., C., (7).

E. Alternate Balancing Procedures

(1) Equipment Required for Balancing:

(a) 2.0 oz (56.7 g) Aileron trim weight.

(b) 3.3 oz (93.55 g) Rudder trim weight.

(c) Control Surface Supports (see Figure 206).

(d) Bolts (0.250 inch diameter) to insert through control surface hinges.

(e) Push-Pull Gage, Wagner Model FDK 32 or equivalent.

NOTE: FDK 32 is a P/N of Wagner Instruments Co., Greenwich, CT. Gage range is 0 to 32 oz (0 to 907.2 g) with 0.250 oz (7.09 g) graduations and ±1% accuracy.

(2) Aileron Balancing:

(a) Remove aileron.

(b) Tape trim tab in a streamlined position. Use minimal amount of tape so as not to affect balance.

(c) Hang a 2.0 oz (56.7 g) test weight from the trim tab actuator pushrod bolt hole on trim tab (see Figure 202).

(d) Install 0.250 inch diameter bolts and nuts through inboard and outboard aileron hinges. Place aileron on control surface supports.

(e) Verify free rotation of aileron. Apply push-pull gage to trailing edge of aileron at location shown in Figure 205. Raise trailing edge of aileron by lifting with push-pull gage. Ensure aileron is level.

(f) Read gage. Verify reading is within limits shown in gage reading column of Table 201 or Table 202.

(g) Shave or replace weight to achieve required gage reading IAW step **1**., C., (8).

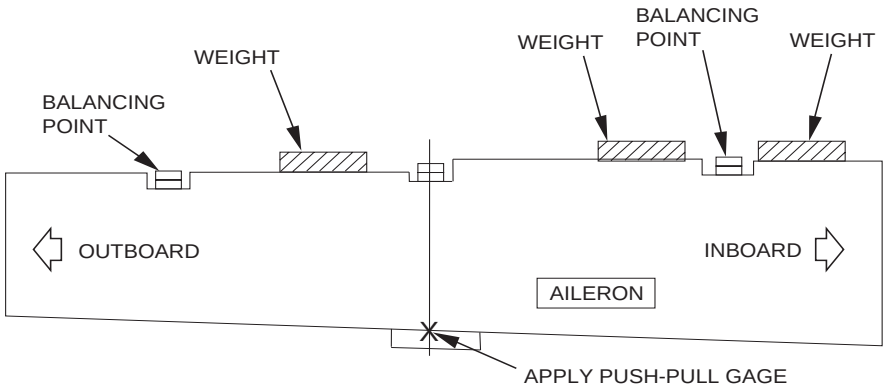
(3) Rudder Balancing:

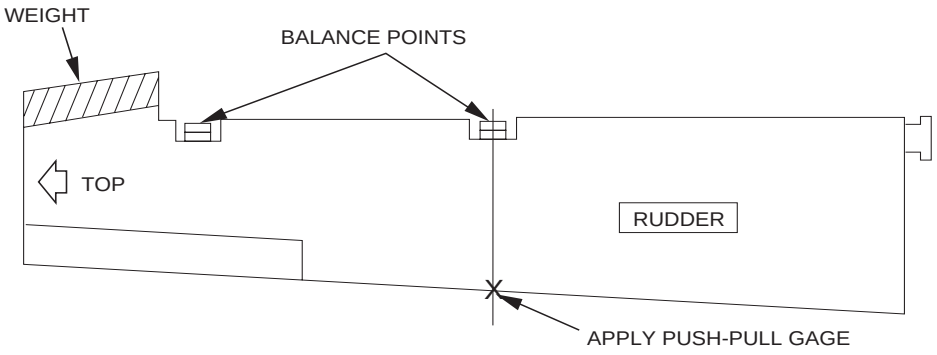
(a) Remove rudder.

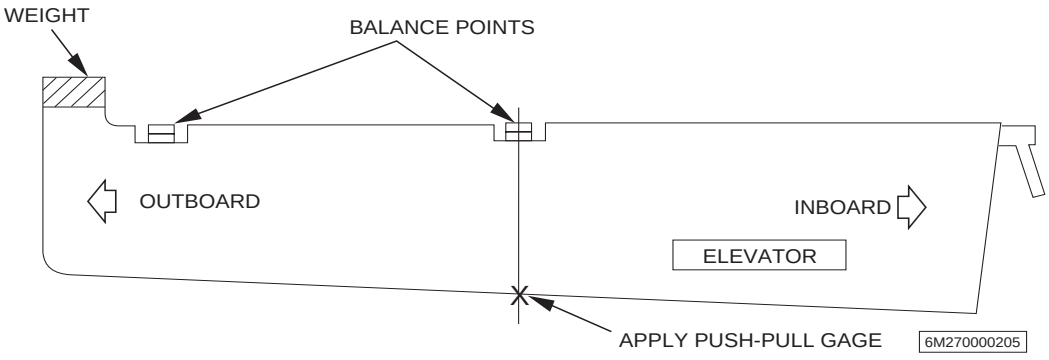
(b) Tape trim tab in a streamlined position. Use minimal amount of tape so as not to affect balance.

(c) Hang a 3.3 oz (93.55 g) test weight from the trim tab actuator pushrod bolt hole (see Figure 202).

(d) Install 0.250 inch diameter bolts and nuts through upper and center rudder hinge fittings. Place rudder horizontally on knife edges of control surface supports with left side of rudder facing down.







Control Surface Balancing
Figure 205

SURFACE	TRIM WT. (oz.)	MOMENT (in.-lb.)	GAGE READING (oz.)
Aileron	2.0 (LH aileron only)	6.4 Underbalance to 4.4 Underbalance	10.1 to 7.0
Rudder	3.3	46.0 Underbalance to 44.0 Underbalance	28.9 to 27.6
Elevator	N/A	22.0 Underbalance to 20.0 Underbalance	23.2 to 21.1

Acceptable Flight Balance Values
SA226-T, T(B), AT, TC except S/N T-394, T-397,
T-400 thru T-417, TC-398 thru TC-419
(Alternate Balancing)
Table 203

(e) Verify free rotation of rudder. Apply push-pull gage to trailing edge of rudder at location shown in Figure 205. Raise trailing edge of rudder by lifting with push-pull gage. Ensure rudder is level.

(f) Read gage. Verify reading is within limits shown in gage reading column of Table 203 or Table 204.

(g) Add or remove weight to achieve required gage reading IAW step **1**., C., (6).

(4) Elevator Balancing:

(a) Remove elevator.

(b) Install 0.250 inch diameter bolts and nuts through outboard and center elevator hinges. Place elevator horizontally on knife edges of control surface supports with upper surface facing up.

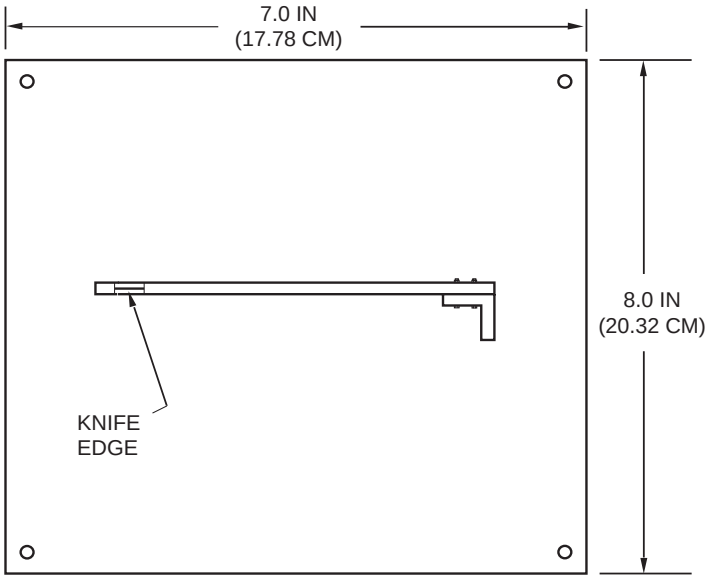
(c) Verify free rotation of elevator. Apply push-pull gage to trailing edge of elevator at location shown in Figure 205. Raise trailing edge of elevator by lifting with push-pull gage. Ensure elevator is level.

(d) Read gage. Verify reading is within limits shown in gage reading column of Table 203 or Table 204.

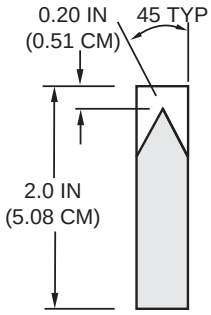
(e) Add or remove weight to achieve required gage reading IAW step **1**., C., (7).

SURFACE	TRIM WT. (oz.)	MOMENT (in.-lb.)	GAGE READING (oz.)
Aileron	2.0 (LH aileron only)	0.0 to 4.0 Overbalance	0.0 to 6.3
Rudder	3.3	46.0 Underbalance to 20.0 Underbalance	28.9 to 12.5
Elevator	N/A	22.0 Underbalance to 14.0 Underbalance	23.2 to 14.7

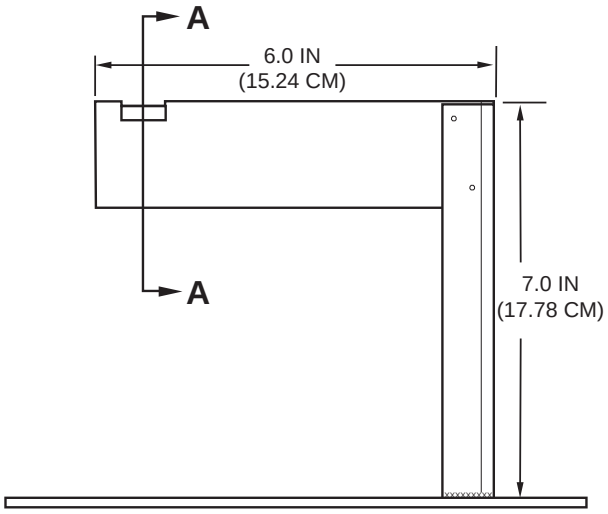
Acceptable Flight Balance Values
SA226-T(B) S/N T-394, T-397,
T-400 thru T-417, TC-398 thru TC-419
(Alternate Balancing)
Table 204



TOP VIEW



SECTION A - A



SIDE VIEW

6M270000206

NOTE: Dimensions shown in Figure 206 are for suggested configuration. Any functional equivalent will suffice, as long as control surfaces are supported on knife edges.

Control Surface Support
Figure 206

F. Control Surfaces – Free Play

(1) Check all control systems for free play.

(2) Apply only enough finger pressure to trailing edge to detect free play without deflecting any part of the system.

SURFACE	TOTAL NO. MOMENT DEFLECTION	MEASURED AT
Rudder	0.440 inch	Bottom end (T/E)
Rudder Tab	0.080 inch	Bottom end (T/E)
Aileron	0.240 inch	Outboard end (T/E)
Aileron Tab	0.070 inch	Outboard end (T/E)
Horizontal Stabilizer	0.250 inch	Outboard end (L or R)
Elevator	0.290 inch	Inboard end (T/E)
Flap	0.250 inch	Inboard end (T/E)

Free Play Check
Table 205

2. Removal/Installation – Flight Controls

A. Blocking – Control Cables

(1) Throughout this manual, reference is made to blocking cables. Cables must be blocked to keep tension on them and the cable taut on its pulleys. When a cable is allowed to hang free, there is a possibility that when the cable is retightened, it may not align with its pulleys.

(2) Cables should be blocked as close as possible to the area being worked to prevent unnecessary work and added maintenance time. Blocking of cables can be accomplished at numerous points throughout the aircraft, ideally at feed throughs in frames, spars, ribs, and at pulley brackets.

CAUTION: DO NOT USE ANY TOOL THAT WILL PINCH OR KINK CABLE (SUCH AS VISE GRIPS).

(3) A simple and effective cable block can be made from phenolic material approximately 1/8 inch thick. Fashion two (2) identical size pieces, one (1) inch by two (2) inches each, and connect together with a bolt, washer, and nut at each end. Clamp cable within the two (2) pieces of phenolic at a feedthrough or other appropriate location.

B. Corrosion Protection – Control Cables

(1) Prior to installation, treat each cable with a corrosion preventive compound (MIL-PRF-16173, Class I or II, Grade 4).

NOTE: Threaded terminal ends, bearings, chains and quick disconnects shall not be coated with corrosion preventive compound.

(2) Immerse control cables in corrosion preventive Grade 4 for one (1) to two (2) minutes, then slowly remove.

(3) Allow immersed control cables to dry for at least four (4) hours prior to installation.

NOTE: Control cables may be slightly tacky prior to installation.

(4) Spray control cables that cannot be immersed with corrosion preventive (Zip Chem D-5035NS or equivalent MIL-PRF-16173 aerosol).

C. Corrosion Protection – Control Chains

(1) Prior to installation, clean control chains thoroughly with a degreasing solvent (MIL-PRF-680 or equivalent). Once the chain has dried, lubricate the entire length of chain including master links with MIL-PRF-81322 grease. Work the grease into the chain by exercising the chain by hand

D. Temperature and Tension – Control Cable

(1) Rigging charts for temperature and tension are provided in Figures 207 and 209.

E. Replacement – Control Cables

(1) Control cables are cut to fit on installation. Refer to SA226 Illustrated Parts Catalog to find part numbers of desired cables.

NOTE: M7 Aerospace recommends a check of flight control cable tension at first 200 and 500 flight hours after installing new cable(s) due to cable stretch and/or thermal variances within the aircraft operating area, then every 1,000 hours.

F. Removal/Installation – Control Cables

(1) Removal and installation procedures of individual cable systems, aileron, rudder, elevator, etc. are covered in their respective sections.

(2) After installation, each cable is treated with a coat of rust preventative lubricant (MOBILARMA 777, 778, or equivalent).

(3) After swaging of end fitting to a cable section, pull test cable to the limits in Table 206.

CABLE DIAMETER	TYPE	PROOF LOAD (POUNDS)
1/16	7X7	290 plus 25 or minus 0
3/32	7X7	550 plus 25 or minus 0
1/8	7X19	1000 plus 50 or minus 0
5/32	7X19	1440 plus 50 or minus 0
3/16	7X19	2220 plus 50 or minus 0
7/32	7X19	3000 plus 50 or minus 0
1/4	7X19	3840 plus 50 or minus 0
5/16	7X19	5400 plus 50 or minus 0

Cable Limits
Table 206

NOTE: Turnbuckles and terminals are available in different lengths. If the cable has stretched to the point that proper cable tension cannot be obtained, select a turnbuckle or terminal of shorter length. Verify that cable is in fact stretched and not off a pulley before changing end fittings.

(4) Proof Test cable IAW MIL-C-5688 using the following load limits:

(a) Apply load to cable evenly by dead load or test machine for a minimum of three seconds.

(b) Release load evenly and gradually.

(c) Reject any terminal assemblies showing slippage noticeable to the unaided eye.

(d) Proof testing of cable may be used for pre-stretching of the cable.

G. Removal – Pressure Seals

(1) Remove two (2) retaining rings on the seal on the pressurized side.

(2) Remove one (1) retaining ring on the seal on the unpressurized side.

(3) Remove seal from bulkhead.

(4) Remove seal from cable by spreading open.

H. Installation – Pressure Seals

(1) Fill seal with graphite grease (MIL-G-7187 or equivalent).

(2) Lubricate cable with graphite grease at least six (6) inches in either side of seal. (Ensure grease covers length of cable travel within seal).

(3) Bend seal open and place it on the cable on the unpressurized side of bulkhead with the small end of the seal toward the bulkhead.

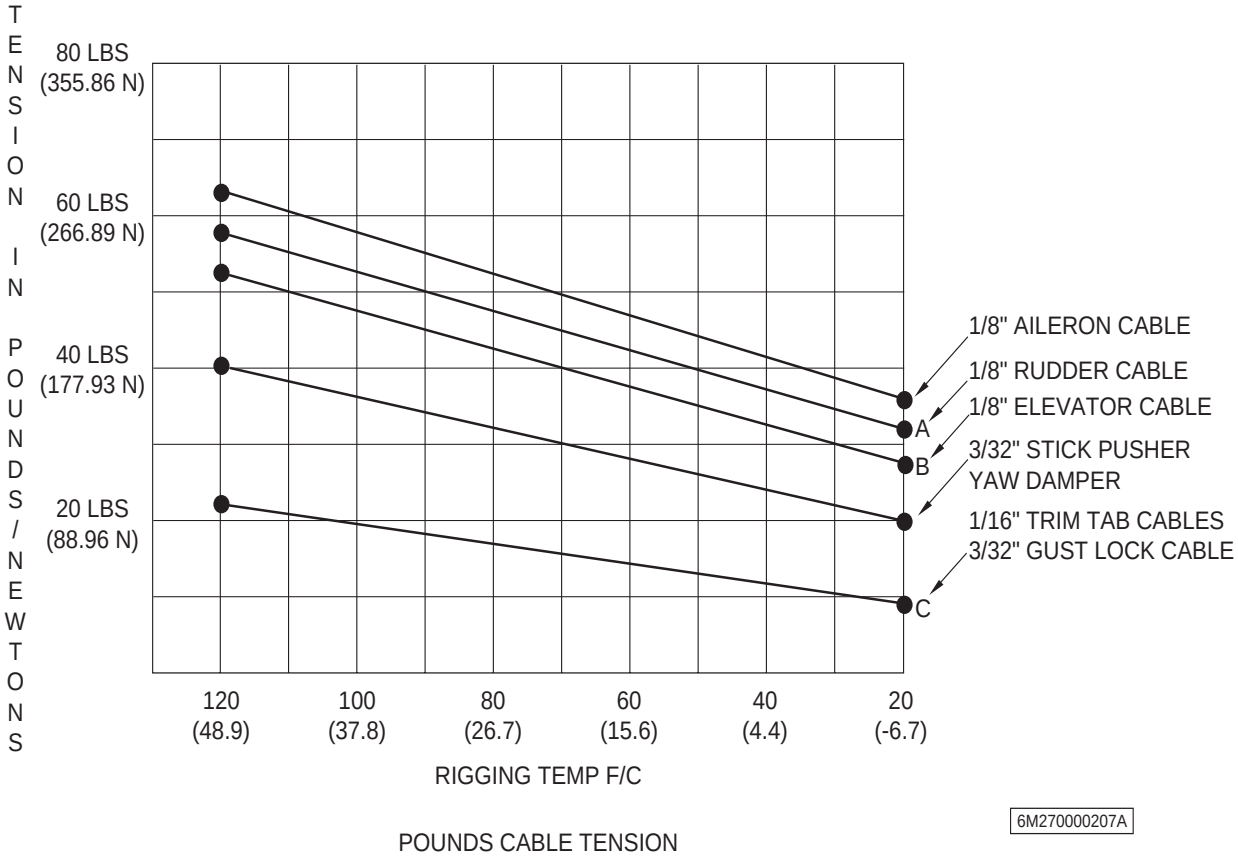
(4) Insert the seal into the bulkhead so that the bulkhead web is entirely within the retainer groove of the seal and the small end of the seal is in the pressurized section.

(5) Install two (2) retaining rings on the seal on the pressurized side and one (1) retaining ring on the unpressurized side of the bulkhead.

A. MEASURE RUDDER CABLE TENSION AFT
OF RUDDER INTERCONNECT AT F.S. 225.78.

B. 1/8" CABLE TENSION
TOLERANCE PLUS OR MINUS FIVE POUNDS

C. 1/16" AND 3/32" CABLE TENSION
TOLERANCE PLUS OR MINUS 2.5 POUNDS



Rigging Tension / Temperature
Chart SA226–T
Figure 207

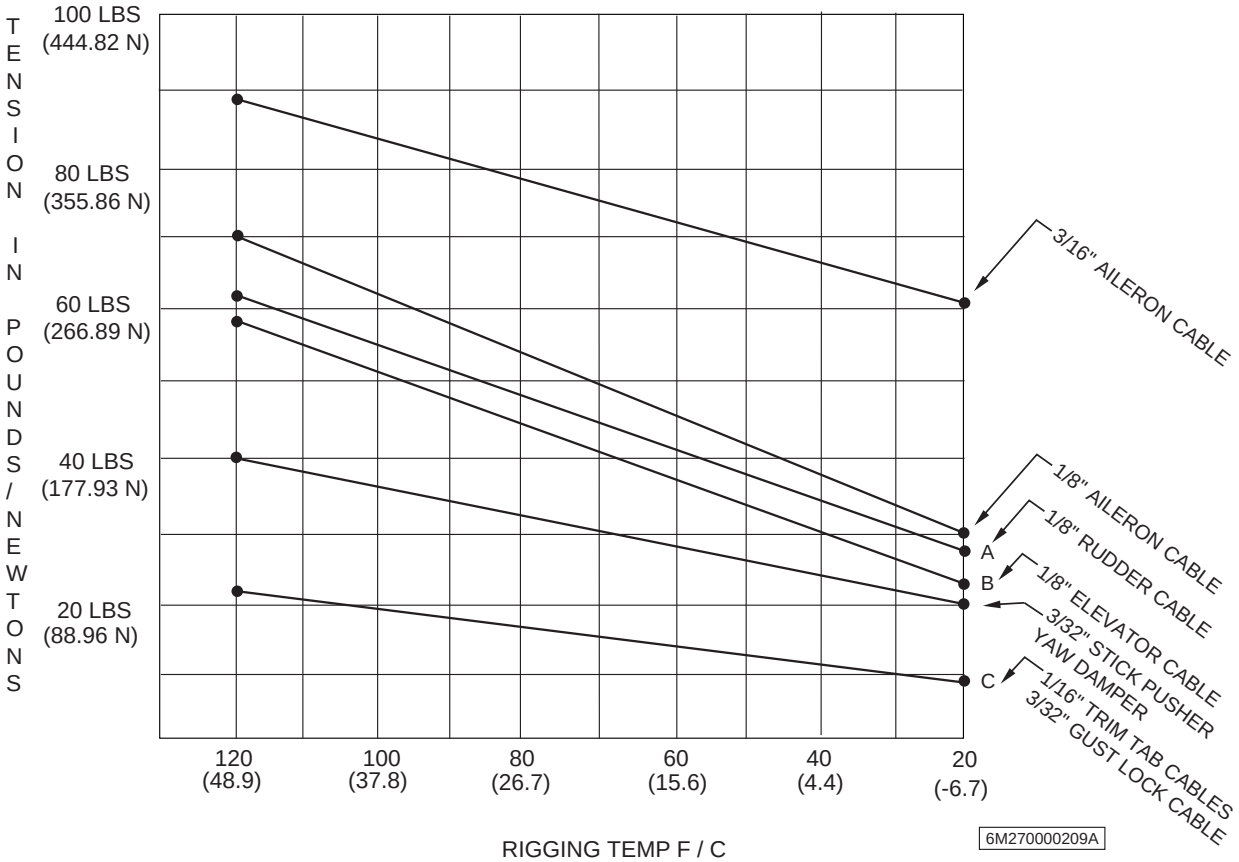
ITEM	DEGREES OF TRAVEL	CABLE SIZE	CABLE TENSION
AILERONS UP DOWN	 18 1/2º ± 1° 21 1/2º ± 1°	1/8	SEE FIGURE 207
AILERON TAB UP DOWN	 20º + 2 - 1° 20º + 2 - 1°	1/16	SEE FIGURE 207
AILERON TAB SERVO ACTION LEFT UP ONLY DOWN	 0º ± 2° 0º ± 2°		
RUDDER	25º ± 1°	1/8	SEE FIGURE 207
RUDDER TAB LEFT RIGHT	 25° ± 1 1/2° 25° ± 1 1/2°	1/16	SEE FIGURE 207
RUDDER TAB SERVO ACTION RIGHT LEFT	 0 ± 2° 0 ± 2°		
RUDDER FIXED TAB	MAXIMUM 30° LEFT OR RIGHT		
YAW DAMPER	SET MECHANICAL CLUTCH TO 150 ± 10 in. lbs.	3/32	SEE FIGURE 207
ELEVATORS AIRPLANE NOSE UP AIRPLANE NOSE DN	 30° ± 1° 15° ± 1°	1/8	SEE FIGURE 207
STICK PUSHER		3/32	SEE FIGURE 207
STABILIZER TRIM AIRPLANE NOSE UP AIRPLANE NOSE DN	MECH. STOP POSITION 8.00° TO 8.40° 1.90° TO 2.30°	ELECTRIC	SEE 27-40-01
STABILIZER TRIM AIRPLANE NOSE UP AIRPLANE NOSE DN	ELECT. STOP POSITION 0.2° ± .05° BEFORE MECHANICAL STOPS	ELECTRIC	SEE 27-40-01
FLAPS UP DOWN	 0° 36° ± 1°	HYDRAULIC	
GUST LOCK		3/32	SEE FIGURE 207

Control Cable Travel - SA226-T
Figure 208

A. MEASURE RUDDER CABLE TENSION IN THE AFT FUSELAGE AT F.S. 332.27.

B. 1/8” AND 3/16" CABLE TENSION
TOLERANCE PLUS OR MINUS FIVE POUNDS

C. 1/16" AND 3/32" CABLE TENSION
TOLERANCE PLUS OR MINUS 2.5 POUNDS



Rigging Tension / Temperature
Chart SA226-AT / TC
Figure 209

ITEM	DEGREES OF TRAVEL	CABLE SIZE	CABLE TENSION
AILERONS UP DOWN	 18 1/2º ± 1° 21 1/2º ± 1°	1/8	SEE FIGURE 209
AILERON TAB UP DOWN	 20º + 2 - 1° 20º + 2 - 1°	1/16	SEE FIGURE 209
AILERON TAB SERVO ACTION LEFT UP ONLY DOWN	 0º ± 2° 0º ± 2°		
RUDDER	19º ± 1°	1/8	SEE FIGURE 209
RUDDER TAB LEFT RIGHT	 25° ± 1 1/2° 25° ± 1 1/2°	1/16	SEE FIGURE 209
RUDDER TAB SERVO ACTION RIGHT LEFT	 0 ± 2° 0 ± 2°		
RUDDER FIXED TAB	MAXIMUM 30° LEFT OR RIGHT		
ELEVATORS AIRPLANE NOSE UP AIRPLANE NOSE DN	 30° ± 1° 15° ± 1°	1/8	SEE FIGURE 209
ELEVATOR DOWN SPRING			SEE 27-30-03
STICK PUSHER		3/32	SEE FIGURE 209
STABILIZER TRIM AIRPLANE NOSE UP AIRPLANE NOSE DN	<u>MECH. STOP POSITION</u> 7.60° TO 8.00° 2.20° TO 2.60°	ELECTRIC	SEE 27-40-01
STABILIZER TRIM AIRPLANE NOSE UP AIRPLANE NOSE DN	<u>ELECT. STOP POSITION</u> 0.2° ± .05° BEFORE MECHANICAL STOPS	ELECTRIC	SEE 27-40-01
FLAPS UP DOWN	 0° 36° ± 1°	HYDRAULIC	
GUST LOCK		3/32	SEE FIGURE 207

Control Cable Travel
SA226-AT / TC
Figure 210

3. Adjustment/Test — Flight Controls

A. Control Cables — Adjustment

(1) For adjustment and test of control cables see Figures 207 through 209 or individual chapters as they apply.

B. Aileron Cable Tension Adjustment

(1) Apply gust lock and work rudder pedals and aileron to engage gust lock and controls to neutral.

(2) Insert rig pin into rudder and aileron bellcranks.

(3) Locate aileron cable turnbuckles under floorboards, forward of rudder/aileron interconnect.

(4) Adjust turnbuckles to attain correct cable tension (see Figure 207 or 209).

NOTE: Adjust cables evenly so that when correct tension is met the control wheels are in neutral position.

C. Aileron Travel Check

(1) Ensure cable tension is at correct adjustment.

(2) Apply gust lock, and work rudder pedals to engage rudder gust lock.

(3) Insert rig pin into rudder and aileron bellcranks.

(4) Place aileron travel check gauge onto each wing. Hold in place using ten-pound shot bags.

(5) Center one aileron counterbalance tool onto each aileron.

(6) Adjust aileron trailing edge to read minus 0.19 plus or minus 0.19 degree droop on adjust plate of travel check gauge.

(7) Remove counterbalance tools from ailerons.

(8) Pull holding pin on adjust plate, and set plate so that zero (o) degree mark on plate is aligned with trailing edge of aileron.

(9) Remove rig pin and disengage gust lock.

(10) Using control wheel in cockpit check full up travel of left aileron and down travel of right aileron, see Figure 208 or 210.

(11) Check full down travel of left aileron, and up travel of right aileron.

(12) Adjustments are made as follows:

(a) Remove flap walking beam access panels (underside, center section of aircraft).

(b) Lengthen or shorten innermost aileron push-pull tubes, where tubes attach to rudder/aileron interconnect shaft, for desired aileron minus 0.19° droop.

(c) For correct travel adjust aileron bellcrank stop bolts located aft of the bellcrank.

NOTE: Upon completion of aileron rigging check all clearances throughout complete range of travel, aileron to wing tip and to flap. Ensure aileron does not strike wing trailing edge skin top or bottom when moved to extreme travel. Check all hinge cutouts for clearance of structure and for smooth movement.

D. Aileron Trim Tab Cable Tension Adjustment

(1) Apply gust lock and work control wheel to engage aileron gust lock.

(2) Set aileron trim indicator on pedestal to neutral.

(3) Locate aileron trim cable turnbuckles (under floorboards, forward of rudder/aileron interconnect).

(4) Adjust turnbuckles to obtain correct cable tension, see Figure 207 or 209.

NOTE: Adjust turnbuckles evenly so that when correct tension is met the trim indicator is in neutral position. If indicator is in neutral and tab is not streamlined adjust actuator push-pull rod (see Trim Tab Actuator Installation, 27-10-06).

E. Aileron Trim Tab Travel Check

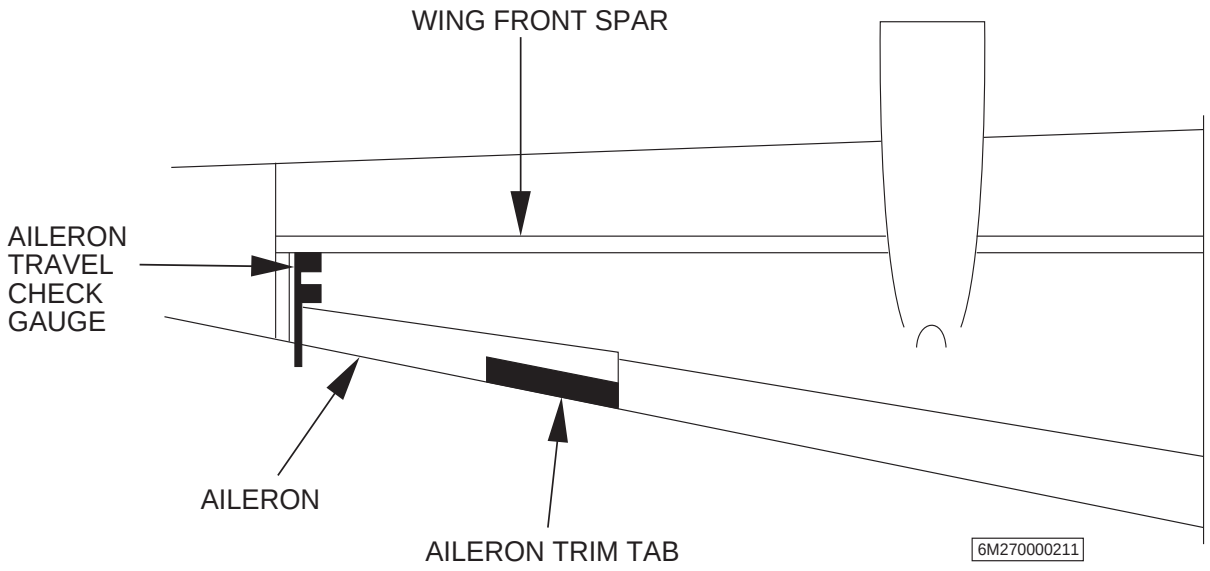
(1) Ensure cable tension is at correct adjustment.

(2) Engage aileron gust lock.

(3) Zero trim travel gauge or propeller protractor, and center onto trim tab.

(4) Check trim travel up and down, see Figure 208 or 210.

(5) Adjustments are made by adjusting cable stops located in left flap well.



Aileron Travel Check Gauge Positioning
Figure 211

F. Rudder Cable Tension Adjustment

(1) Apply gust lock, and work rudder pedals to engage rudder gust lock.

(2) Remove aft interior bulkhead.

(3) Locate rudder cable turnbuckles.

(4) Adjust turnbuckles to attain proper cable tension, see Figure 207 or 209.

NOTE: Adjust cables evenly so that when correct tension is met rudder is streamlined and pilot and copilot rudder pedals are aligned. If pedals need alignment see Rudder Pedal Operational Check, Section 27-20-02.

G. Rudder Travel Check

(1) Ensure cable tension is at correct adjustment.

(2) Apply gust lock and work rudder pedals to engage rudder dust lock.

(3) Remove tail cone.

(4) Remove two (of twenty) bolts (immediately left and right of top center bolt) that secure aft pressure bulkhead.

(5) Attach rudder travel gauge using the two bulkhead bolts.

(6) Check for rudder streamlined at zero (0) degree mark.

(7) Disengage gust lock.

(8) Check rudder travel left and right, see Figure 208 or 210.

(9) Adjustments are made as follows:

(a) Remove travel gauge.

(b) Gain access to rudder bellcrank (see Rudder Bellcrank Removal, 27-20-03).

(c) Install rudder travel gauge IAW step **3.**, G., (5).

(d) Correct rudder travel by adjusting rudder bellcrank stop bolts located forward of rudder bellcrank.

H. Rudder Trim Tab Cable Tension Adjustment

(1) Apply gust lock, and work rudder pedals to engage rudder gust lock.

(2) Set rudder trim wheel on pedestal to neutral position.

(3) Remove aft interior bulkhead.

(4) Locate rudder trim cable turnbuckles.

(5) Adjust turnbuckles to obtain correct cable tension, see Figure 207 or 209.

NOTE: Adjust turnbuckles evenly so that when proper tension is met the trim wheel is in neutral position. If wheel is in neutral and tab is not streamlined adjust actuator push-pull rod (see Trim Tab Actuator – Installation, Section 27-20-05). If adjustment to push-pull rod is made the trim tab travel must be check.

I. Rudder Trim Tab Travel Check

(1) Ensure cable tension is at correct adjustment.

(2) Engage rudder gust lock.

(3) Place rudder trim tab in streamlined position.

(4) Hold rudder trim protractor (No. 17, General Hardware Mfg. Co., N. Y.) on left side of rudder. Place protractor as close as possible to where actuator push-pull rod connects to trim tab.

(5) Rotate rudder trim wheel left and check tab travel see Figure 208 or 210.

(6) Hold protractor on right side of rudder. Place where push-pull connects tab.

(7) Rotate rudder trim wheel right and check tab travel, see Figure 208 or 210.

(8) Adjustments are made by adjusting cable stops (located under floorboards directly inside cargo door on long aircraft and passenger/crew door on short aircraft).

J. Rudder Fixed Tab Adjustment

(1) Rudder fixed tab should be adjusted (bent evenly the length of tab) left or right as necessary up to a maximum 30°. Bending tab to the right will trim aircraft yaw nose left.

K. Elevator Cable Tension Adjustment

NOTE: Turnbuckles and terminals are available in different lengths. If the cable has stretched to the point that proper cable tension cannot be maintained, select a turnbuckle or terminal of shorter length. Before changing end fittings, verify that cable is in fact stretched and not off a pulley.

(1) Remove aft interior bulkhead and locate elevator cable turnbuckles.

(2) If aircraft is autopilot equipped, disconnect autopilot bridle. Elevator cable tension cannot be properly adjusted if autopilot is connected.

(3) For all SA226–AT /TC aircraft (except those having an alternate elevator gust lock installed IAW SB27-041), engage gust lock to relieve tension of elevator down spring.

(4) Remove pilot seat, carpet and floorboard. Install 3/16 rig pin through elevator walking beam and bracket to position elevator in neutral, and relieve tension of elevator bob weight.

(5) Remove large access panel on right side of vertical stabilizer, and install 3/16 rigging pin through elevator control quadrant and quadrant lock.

(6) Take cable tension reading on elevator down cable (RH cable), and on elevator up cable (LH cable). Adjust turnbuckles, as necessary to attain correct cable tension, see Figure 207 (T) or 209 (AT / TC).

(7) Connect autopilot bridle and tension bridle IAW specifications for autopilot system.

NOTE: Elevator cable tensions change from recommended values when autopilot bridle has been tensioned. Bridle cable preload increases primary cable tension aft of bridle splice and decreases it forward of splice. Typically, cable tension aft of bridle increases by as much as 40% of recommended rigging tension, and forward of bridle decreases by as much as 15%.

CAUTION: SUPPORT CONTROL WHEEL WHEN DISENGAGING ELEVATOR GUST LOCK TO AVOID DAMAGE TO FLIGHT INSTRUMENTS.

(8) Remove walking beam and elevator quadrant rig pins. Disengage elevator gust lock.

(9) Perform Elevator Travel Check.

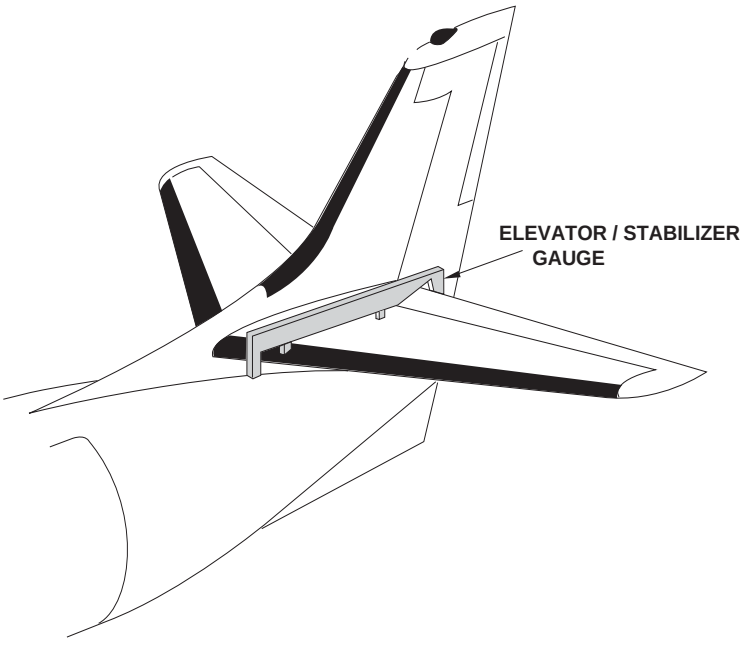
NOTE: Upon completion of rigging procedures, check distance between control column and instrument panel to be 2.25 inches, ± 0.25, with control column in the forward position. Adjust control rod located under RH cockpit floorboard, F.S. 98.56, if necessary.

L. Elevator Travel Check

(1) Ensure elevator tension is at correct adjustment.

(2) Run horizontal stabilizer to zero (0) rivet on LH side of vertical stabilizer. Position so that bottom edge of stabilizer is centered on zero (0) rivet.

(3) Remove quadrant access door (right side of vertical stabilizer) and insert 3/16 inch rig pin through quadrant lock and quadrant.



Elevator / Stabilizer Gauge Positioning
Figure 212

(4) Set elevator / horizontal stabilizer gauge onto left horizontal stabilizer, see Figure 212.

(5) Verify that elevator aligns with zero (0) degree mark on gauge. If elevator does not align, adjust length of LH elevator push-pull rod attaching elevator to quadrant. Retighten jam nut after adjustment.

(6) Center propeller protractor on most inboard row of rivets on LH elevator, and zero (0) adjust it.

(7) Place protractor on right elevator. Center on inboard row of rivets, and check zero (0) alignment. Adjust length of RH push-pull rod if required, and retighten jam nut.

(8) Remove quadrant rig pin.

(9) Using elevator / horizontal stabilizer gauge, and protractor, check LH and RH elevator up and down travel. Refer to Figures 208 (T), and 210 (AT / TC) for travel limits.

(10) To correct travel, adjust upper and lower quadrant stops. Stops are eccentric. Loosen bolt and turn stop to desired position. Retighten and safety.

(11) Operate system through full range of travel, and check for unusual noise, freedom of operation, positive contact on stops, clearance to adjacent structure and installations. Repeat checks with horizontal stabilizer in full nose up and full nose down positions.

(12) Replace access panels, floor panel, carpet, and pilot seat.

M. Pitch Trim Adjustment / Test

(1) For pitch trim adjustment / test, see Section 27-40-01.

N. Flap Travel Check

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP. REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

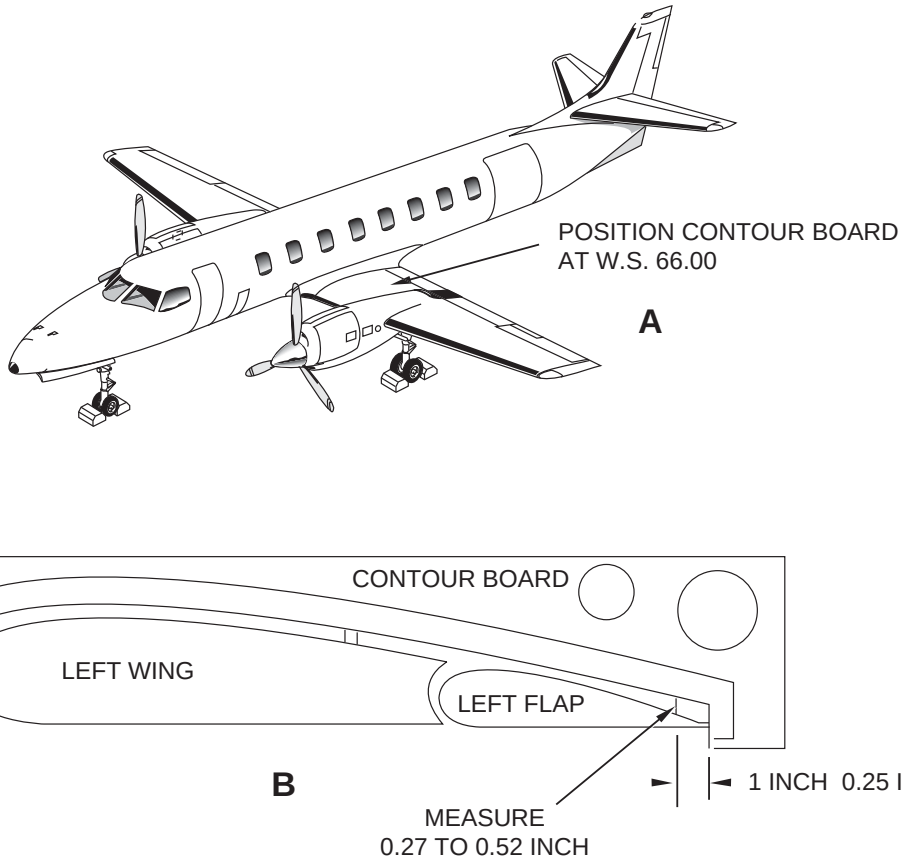
(1) If maintenance involves replacement of flap components or after major disassembly of the flap system, perform the following steps before checking flap travel:

(a) Begin with both flaps at approximately half travel. Remove flap interconnect bolt (see Figure 213, view D).

(b) Select flaps up and observe movement. If flaps move in opposite directions, hydraulic connections have been crossed. Correct connections before installing interconnect bolt.

(c) With interconnect bolt removed, verify that two halves of interconnect (walking beam) do not interfere with each other. If one flap begins to move faster than the other (with the interconnect bolt removed), flap may be slowed by applying hand pressure on the inboard trailing edge.

(d) When flaps have been run full up, check flap zero (0). Select flaps down. Modulate the flap down travel speed either by varying the hydraulic pressure applied to the system or by stationing an assistant in the cockpit to operate the flap handle.



Flap Travel Check
Figure 213

(e) As flaps near full down, observe the aft nacelle area and the flap fairing bubble. If necessary, stop the flaps before contact between these two structures is made.

(f) Trim or rework the flap bubble as necessary to gain clearance. This is usually only necessary if a new flap is being installed. Ensure that flap zero (0) setting is correct as interference may be caused by flap overtravel.

(g) After adequate clearance is assured, run flaps full up. Adjust the push-pull rod attaching the flap to its respective interconnect to allow the interconnect bolt to be installed with no force.

(2) Pressurize system 2,000 ± 50 psi.

(3) Actuate flap control lever to bring flaps to full up position.

(4) On right wing, set flap contour board on inboard side of wing leading edge access panel at W.S. 66.00, see Figure 213A.

(5) At trailing edge of right flap, measuring approximately one (1) inch forward on flap, space between flap and contour board should be between 0.27 and 0.52 inches (see Figure 213B). Achieve this distance by adjusting flap actuator shaft clevis connection in or out.

NOTE: After adjusting the actuator, at least three (3) full threads must be engaged between the actuator shaft end and the actuator clevis end (see Figure 213C).

(6) Tape a straightedge onto right flap on most inboard row of rivets (approximately twelve [12] inches outboard).

(7) Set propeller protractor onto straightedge, and set protractor on zero (0) mark.

(8) Tape a straightedge onto left flap at same position as right flap.

(9) Set propeller protractor onto left flap straightedge.

(10) Set left flap to zero (0) degree for SA226-T aircraft and to one degree (droop) for SA226-AT and SA226-TC aircraft. Set flap to zero (0) degree or one (1) degree, whichever applies, by adjusting left actuator clevis end.

(11) Starting with right flap run flap down to 1/4 on flap indicator in cockpit. Verify flap travel is nine (9) degrees ± one (1) degree.

(12) Verify left flap travel is nine (9) degrees, ± one (1) degree.

(13) Alternating from right flap to left flap, check flaps at 1/2, 3/4, and full down position as follows.

(a) 1/4 - 9°, ± 1°.

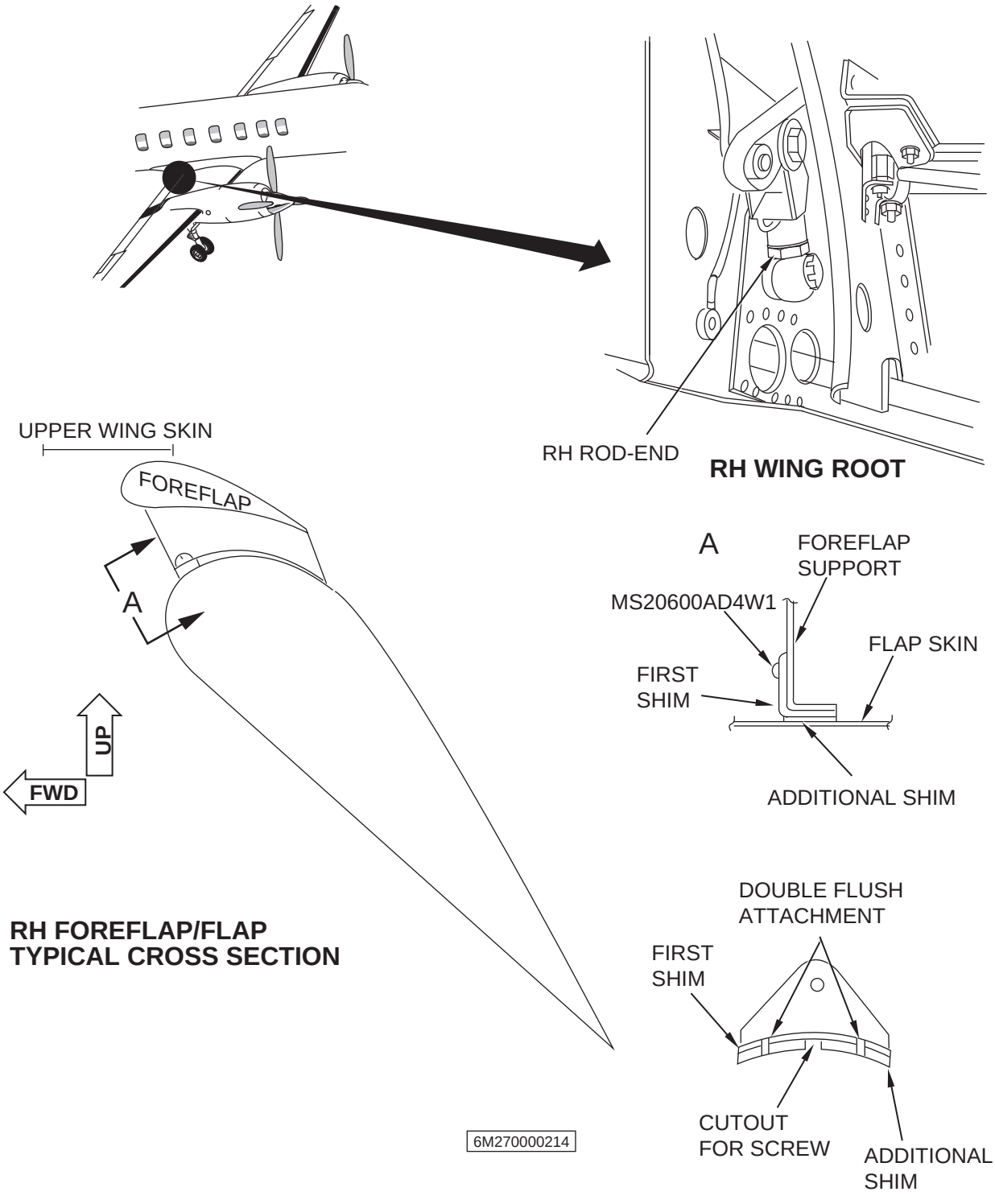
(b) 1/2 - 18°, ± 1°.

(c) 3/4 - 27°, ± 1°.

(d) Full Down - 36°, ± 1°.

(14) Adjust flap position indicating system IAW 27-51-06 or 27-52-06, whichever is applicable.

(15) Upon completion of flap travel check run flaps from full up position to full down position and from full down to full up position. Time lapse should be no more than 20 seconds.



Flap Roll Adjustments
Figure 214

O. Roll Adjustment/Test (Figure 214)

(1) Obtain pilot report following test flight (see Fig 215). This form may be duplicated and used as a worksheet to record flap roll conditions.

(2) Correction of roll is required if pilot report indicates a LWD trim setting greater than 2:00 o'clock or a RWD setting less than 10:00 o'clock.

NOTE: Aileron trim clock readings are taken from the outer trim wheel.

(3) Determine if roll is caused by ailerons or flaps.

(4) Verify symmetrical power is obtained at different flying speeds.

NOTE: At flight idle, full flap action is affected by fuel flow and propeller blade angle. Clean trim is affected by aileron droop and tabs. First travel of flaps is affected by preload while last travel (3/4 to full DOWN) is affected by foreflaps.

(5) Conduct a thorough visual inspection of flying surfaces. Check for loose erosion tape and other irregularities in aircraft skin that might cause sufficient turbulence to make LH and RH lift unequal.

(6) If aircraft rolls with flaps full up, verify that ailerons are properly adjusted. Refer to Aileron Travel Check, this section.

(7) If aircraft still requires trim for level flight with symmetrical power settings, bend aileron fixed tabs in increments of 5 degrees. For example:

(a) To correct a left wing down roll, move LH fixed tab 5 degrees up, and RH fixed tab 5 degrees down. For right wing down roll, move RH tab 5 degrees up, and LH tab 5 degrees down.

(b) Proper adjustment is achieved when aircraft maintains level flight with aileron trim wheel set for level flight.

(8) Fixed trim tabs should be adjusted in increments of 5 degrees to a maximum of 15 degrees.

(9) If aircraft does not roll with flaps fully retracted, but roll begins when flaps are between full up and 1/4 flaps, adjust flap preload as follows:

(a) Before making preload adjustments, perform Flap Travel Check, this section.

(b) Remove flap walking beam access panel.

(c) Select flaps full down.

(d) Remove flap interconnect assembly (walking beam). Refer to Maintenance Practices, Removal – Flap Interconnect, this chapter.

(e) Remove safety wire from rod end lock nut. Remove walking beam retaining nut and slide rod end off walking beam.

(f) For a left wing down roll of 3 o'clock, make one complete turn out on LH rod end, or one complete turn in on RH rod end (Figure 214). Reverse procedure for a right wing down roll of 9 o'clock.

(g) Connect rod end to flap walking beam stud. Secure with washer and nut and safety wire.

1 Tighten castellated nut until ball of rod end and washer are secure (no rotation allowed), but pre-load bearing on stud rotates freely.

NOTE: If the above condition cannot be obtained, contact Fairchild Service Engineering.

2 Secure with cotter pin.

(h) Install walking beam, if removed.

(i) Move flaps up and down two or three times. Verify operation is smooth and free of binding or chafing.

(j) Verify that walking beam does not contact aft spar. If contact is evident, split adjustment as follows:

1 For a left wing down roll of 3 o'clock, make a half turn out on LH rod end, and a half turn in on RH rod end. Reverse procedure for a right wing down roll of 9 o'clock.

(k) Install walking beam access panels, and obtain pilot report following next flight.

(l) If further adjustment is necessary, repeat step (j) 1.

(m) Enter all adjustments in aircraft log book for future reference.

(10) If aircraft does not roll when flaps are full up or between full up and 1/4, but rolls when flaps are between 3/4 and full down, shim fore flap as follows:

(a) Select FULL FLAPS.

(b) If roll occurs with flaps fully extended, shim fore flaps on low wing, or remove shims on high wing. Number of shims required will vary.

(c) To add shims, loosen all screws attaching fore flaps to flaps on side (LH or RH) requiring shims, and place shims between flap and fore flap at forward screw location only for each station. Refer to Figure 214 for shim location diagram. Refer to the Illustrated Parts Catalog for shim part numbers. (First shim is different type from additional shims).

(d) Tighten all screws.

(e) Cycle flaps up and down, and inspect for rubbing or chafing. Verify operation is smooth before flight.

(f) Add more shims in the same manner if further adjustment is necessary. Add no more than four (4) shims at any position. When trim is clean, rivet shims in place.

1 Rivet additional shims to first shim on each side of screw slot attaching foreflap support to flap (Figure 214). Use double flush rivets.

2 Rivet first shim to side of foreflap support (see Figure 214).

(11) If these adjustments do not cure flap roll, it may be necessary to droop a flap by adjusting flap actuator clevis end. If this is necessary, return flap preload (if adjusted) to original configuration, and recheck/adjust clean trim.

ROLL TEST FLIGHT REPORT SHEET								
	TRIM WHEEL CLOCK SETTINGS FLIGHT NUMBER							
	1	2	3	4	5	6	7	8
1. AILERON DEFLECTION								
2. CONTROL WHEEL ALIGNMEMNT								
3. AILERON TRIM								
A. 245 KIAS, CLEAN								
B. 215 KIAS, CLEAN								
C. 215 KIAS, 1/4 FLAP								
D. 180 KIAS, CLEAN								
E. 180 KIAS, 1/2 FLAP								
F. 150 KIAS CLEAN								
G.150 KIAS, 3/4 FLAP								
H.150 KIAS, FULL FLAP								
I. 110 KIAS, FULL FLAP								
J. 110 KIAS, FULL FLAP,FLT IDLE								
4. GEAR WARNING HORN AT 20 DEGREES FLAPS (MORE THAN 1/2, LESS THAN 3/4 INDICATED)								
5. RUDDER TRIM								

NOTE: ENSURE SYMMETRIC POWER SETTINGS THROUGHOUT CHECKS. AT EACH CHECK, TURN AILERON TRIM WHEEL UNTIL FLIGHT IS LEVEL. THEN READ AILERON WHEEL SETTING AS A CLOCK FACE. FOR EXAMPLE: A READING OF 2:00 O'CLOCK MEANS THE LEFT WING WAS LOW BEFORE ADJUSTMENT; A READING OF 10:00 O'CLOCK MEANS THE RIGHT WING WAS LOW BEFORE ADJUSTMENT.

6M270000215

Flap Roll Test Flight Worksheet
Figure 215

4. Inspection Check – Flight Controls

A. Inspection Check – Cables (Figure 216)

(1) Definition of Terms:

(a) Wire – Each individual cylindrical steel rod or thread.

(b) Strand – Each group of wires helically twisted or laid.

(c) Critical Fatigue Area – The working length of a cable where the cable runs over, under, or around a pulley, sheave, or through a fairlead, as well as any section where the cable is flexed, rubbed, or worked in any manner. A critical fatigue area is also located at any point within one (1) foot of a swaged-on fitting. The pressures exerted on the fitting during the swaging process sometimes pinch the wires in the cable, and cause a premature failure.

(2) Inspect all elevator, rudder, aileron, and aileron to rudder interconnect primary control cables, including those routed inside the control column as follows:

(a) Gain access to control cables IAW Adjustment/Test – Flight Controls in this section.

(b) Release tension on each cable.

(c) Pass a cloth along length of the cable to clean it for visual inspection, and to detect broken wires.

(d) Visually inspect the entire length for broken wires and general condition. Rotate the cable 180° where cable contacts any pulley/fairlead or pressure seal, birdcage cable (slightly untwist), and visually inspect for wear/broken wires and for general condition 24 inches either side of contact area when the controls are in the neutral position.

(e) Carefully note the location of any broken wires.

CAUTION: USE CLEAN, DRY CLOTH WHEN WIPING CABLE. USING VAPOR DEGREASING STEAM CLEANING, METHYL-PROPYL-KETONE (MPK) OR OTHER SOLVENTS MAY REMOVE CORROSION-PREVENTATIVE COMPOUNDS AND CABLE INTERNAL LUBRICANT.

(f) Make a careful visual inspection in the critical fatigue areas for wear and broken wires. A broken wire will not always protrude or stick out, but may lie in the strand and remain in the position in which it was manufactured. Broken wires of this type may show as a hairline crack in the wire. If a broken wire is suspected, further inspection with a magnifying glass of at least 7 power is recommended.

(3) Before further flight, replace with a new cable, any cable found to have any of the following conditions:

(a) Replace cables found to have any amount of corrosion.

(b) Replace a cable having more than three (3) broken wires within any one-foot section of the cable (any wire worn more than one-half its diameter is considered broken).

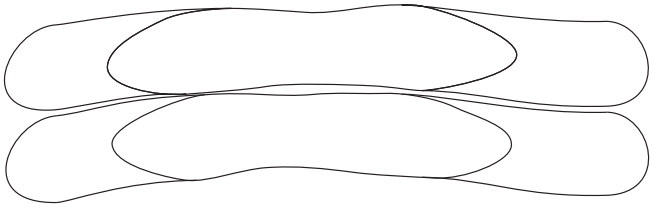
(c) Replace a cable having more than one (1) one-foot section with three (3) broken wires.

(d) Replace a cable with a total number of broken wires in the cable assembly exceeding the total number of feet in that cable (a cable assembly is defined as a length of cable with fittings on each end).

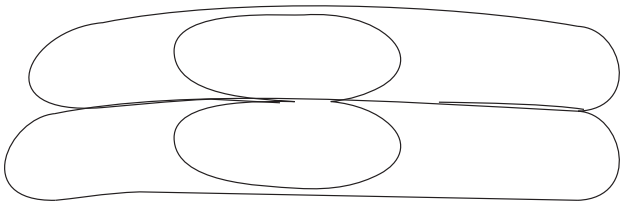
(e) Replace a cable having more than six (6) wires worn not more than half of the wire diameter, in any one (1) inch length of that cable.

(f) Replace a 7x7 cable if more than three (3) wires are worn more than half of the wire diameter in any three (3) inch length of cable. Any wire worn more than half of the wire diameter is considered broken, (see Figure 216). No two (2) worn spots may be adjacent in the same circumference area.

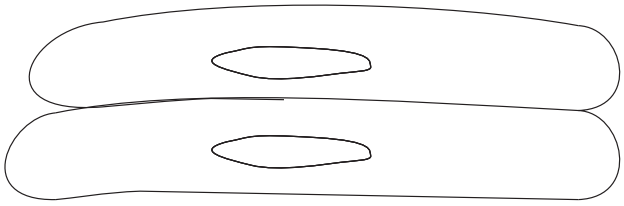
(g) Replace a 7x19 cable if more than six (6) wires are worn more than half of the wire diameter in any one (1) inch length of cable. Any wire worn more than half of the wire diameter is considered broken. No two (2) worn spots may be adjacent in the same circumference area.



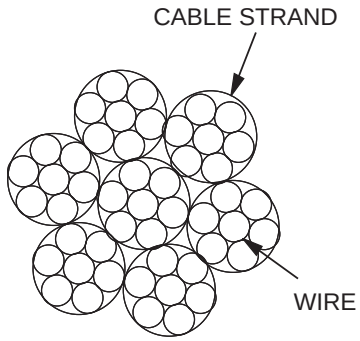
INDIVIDUAL OUTER WIRES
WORN MORE THAN 50%



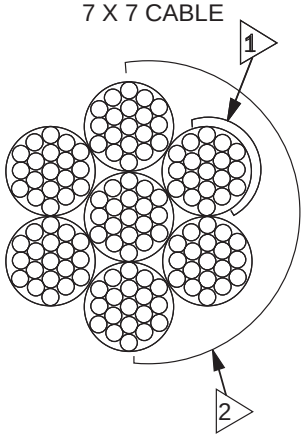
INDIVIDUAL OUTER WIRES WORN 40-50%
(NOTE BLENDING OF WORN AREAS)



INDIVIDUAL OUTER WIRES
WORN LESS THAN 40%
(WORN AREAS INDIVIDUALLY
DISTINGUISHABLE)



7 X 7 CABLE



7 X 19 CABLE

OUTER WIRES INDICATED BY SHADING

1 TYPICAL OUTER WIRE WEAR AREA ON CABLE STRAND.
HAIRLINE CRACKS BETWEEN WIRES OF FULLY BLENDED
SURFACE APPEARANCE OF APPROXIMATELY SIX WIRES PER
OUTER CABLE STRAND INDICATES 30 PERCENT WIRE WEAR.
IF THIS APPEARANCE IS VISIBLE, THE CABLE SHOULD BE REPLACED.

2 TYPICAL WIRE WEAR MAY OCCUR ON ONE SIDE ONLY OR
ON FULL CIRCUMFERENCE AND MAY EXTEND ALONG THE
CABLE FOR A DISTANCE EQUAL TO NORMAL CABLE MOVEMENT.

WHENEVER EXCESSIVE WEAR IS FOUND, CAUSE FOR WEAR
MUST BE FOUND AND CORRECTED.

6M27000216

Control Cable Wear
Figure 216

(4) Replace a cable that has been kinked. Even though the kink may be straightened so that the damage appears to be slight, the relative adjustment between strands has been disturbed so the cable cannot give maximum service. Inspect the cables for popped core, loose strands, or wires out of position. Replace cables with any of these conditions.

(5) Routing:

(a) Before next flight, inspect for incorrect cable routing at fairleads, pulleys, and guards. Cable should not chafe the flange of a pulley.

(b) Inspect cable for full travel, freedom of movement, and security of attaching hardware.

(c) Check for slack in cable system by attempting to move control column and / or pedals while gust locks are installed on control surfaces.

(d) With gust lock removed, actuate controls and check for friction or hard movement. These are indications that excess cable tension exists.

NOTE: If control movement is stiff after maintenance operations are performed on control surfaces, check for parallel cables twisted around each other or cables connected in reverse.

(e) Check that all control surfaces move in the correct direction when the cockpit controls are moved.

(f) Adjust cable tension (see Adjustment / Test – Flight Controls this chapter).

NOTE: Turnbuckles and terminals are available in different lengths. If the cable has stretched to the point that proper cable tension cannot be obtained, select a turnbuckle or terminal of shorter length. Verify that the cable is in fact stretched and not off a pulley before changing end fittings.

(6) Fittings:

(a) Check swaged terminal reference marks for an indication of cable slippage within fitting.

(b) Inspect fitting assembly for distortion and / or broken strands at the terminal.

(c) Ensure all bearings and swivel fittings pivot freely.

(d) Check turnbuckles for proper thread exposure and broken or missing safety wires/ clips.

B. Inspection Check – Pressure Seals

(1) Examine pressure seals for wear and / or material deterioration.

C. Inspection Check – Pulleys

(1) Turn turnbuckles to slack off on cable tension until cable rotates freely, or lift cable from pulley with finger.

(2) Rotate pulley by hand and check for binding or catching in bearing as it is turned.

(3) Check for excessive looseness which might indicate wear.

(4) Check for presence of corrosion. Bearing corroded on exterior is likely corroded internally.

(5) Inspect for cracks or broken flanges.

(6) Check cable groove for excessive wear. Analyze pulley wear patterns (see Figure 216).

(7) Check pulley alignment to prevent cable from riding on the flange and chafing against guards, covers, or adjacent structure.

(8) Check pulley brackets and guards for damage, alignment, and security.

(9) Inspect for roughness, sharp edges, and presence of foreign matter embedded in grooves.

D. Inspection Check – Fairleads, Guides, Anti-abrasion Strips

(1) Inspect fairleads for wear, breakage, alignment, cleanliness, and security.

(2) Determine that all guides and anti-abrasion strips are secure and in good condition.

E. Inspection Check – Push-Pull Rods, Bellcranks, Torque Tubes

(1) Check for alignment of mating parts.

(2) Check correctness of travel, clearance, and action of bellcranks. (The action of bellcranks and linkages must be such that at no time will it be possible for the bellcrank to go past center in such a way as to lock the control system).

CAUTION: INSTALL BOLTS WITH HEAD UP; ENSURE MINIMUM OF 0.25 INCH (6.35 MM) CLEARANCE BETWEEN BOLT TIP AND ACCESS PANEL.

IF 0.25 INCH (6.35 MM) CLEARANCE BETWEEN BOLT TIP AND ACCESS PANEL CANNOT BE ACCOMPLISHED, INSTALL BOLT WITH HEAD DOWN. ENSURE THAT COTTER PIN IS INSTALLED.

(3) Inspect attachment of rod ends, with specific attention to the amount of thread engagement, the tightness of jam nuts, and the torque of attaching bolts on clevis or universal joints which must be such as to allow freedom of movement.

(4) Inspect witness holes in push-pull rods with safety wire.

NOTE: Any defect found in push-pull tube or a torque tube is cause for replacement. It is recommended that replacement of these parts be made through Fairchild Aircraft and not locally purchased.

F. Free Play Measurement – Flight Controls

(1) Requirements – General:

(a) Position aerodynamic control surfaces in neutral (streamlined) position.

(b) Measure free play in direction perpendicular to streamline (90° to chord plane).

(c) Ensure gust lock is not engaged during free play measurement.

(d) Some control surfaces require immobilization with clamp or firm hand pressure for free play measurement.

(e) Ensure force applied to surface during free play measurement does not cause actuator displacement, cable stretching or surface deflection (bending).

(f) Excessive free play indicates wear in hinge bearings, interconnecting linkage, and/or actuating mechanism.

(g) If free play measurement exceeds limit, repair or replace as necessary.

(2) Equipment Required for Free Play Measurement:

(a) Dial indicator and stand (preferably with vacuum base), 0 - 1.000 inch range with 0.001 inch graduations.

(b) 27-95019 Rudder Pedal Lock Tube.

(3) Aileron Free Play Measurement (Figure 217):

(a) Immobilize left or right aileron to fixed structure.

(b) Attach dial indicator stand to supporting structure.

(c) Zero (0) dial indicator.

NOTE: Ensure indicator stylus is in positive contact with aileron.

(d) Apply upward force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

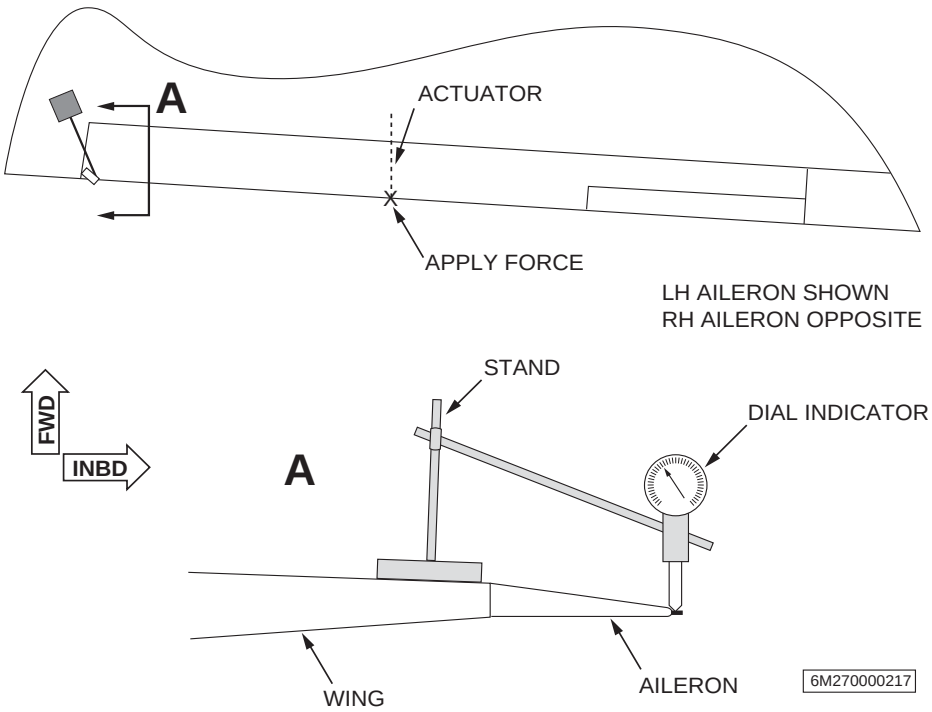
(e) Apply downward force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

(f) Add readings from steps **4.**, F., (3), (d) and (e). Verify free play does not exceed 0.240 inch (6.096 mm).

(g) Release aileron.

(h) If free play measurement exceeds limit, repair or replace as necessary.



Aileron Free Play
Figure 217

(4) Aileron Trim Tab Free Play Measurement (Figure 218):

(a) Immobilize left and right aileron to fixed structure.

(b) Immobilize left or right aileron trim tab.

(c) Attach dial indicator stand to supporting structure.

(d) Zero (0) dial indicator.

NOTE: Ensure indicator stylus is in positive contact with aileron trim tab.

(e) Apply upward force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

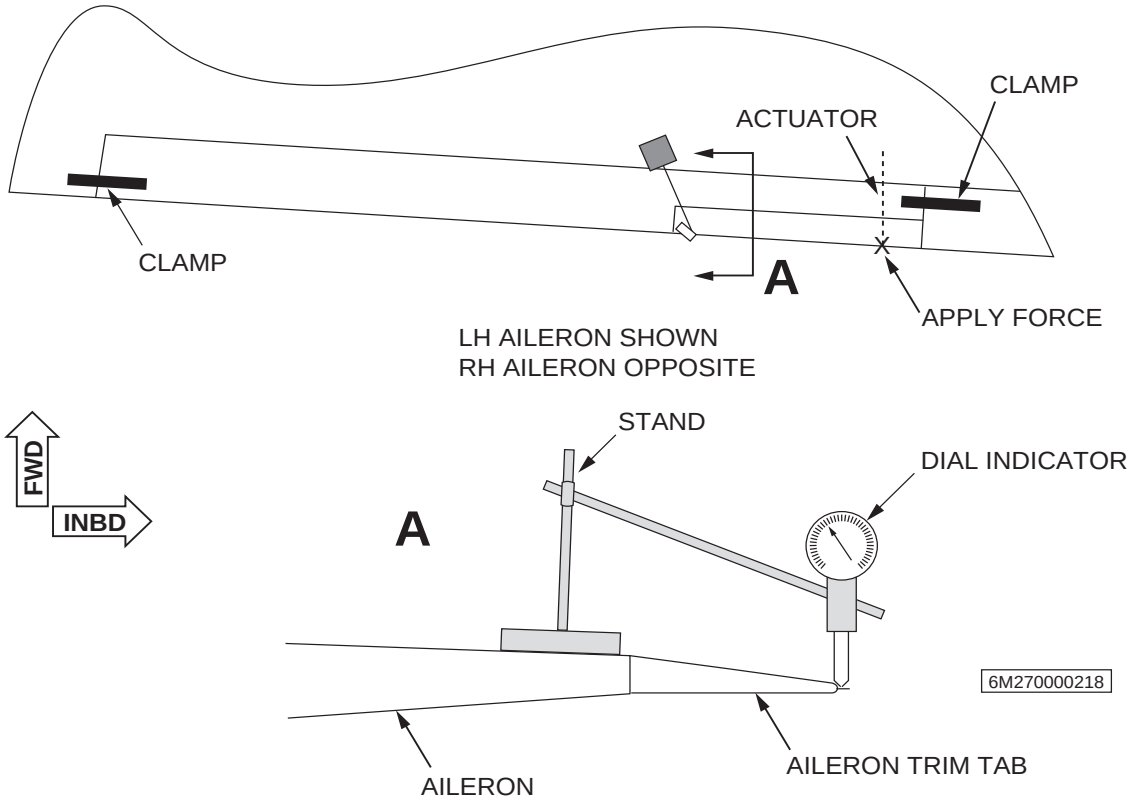
(f) Apply downward force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

(g) Add readings from steps **4.**, F., (4), (e) and (f). Verify free play does not exceed 0.070 inch (1.778 mm).

(h) Release aileron trim tab.

(i) If free play measurement exceeds limit, repair or replace as necessary.



Aileron Trim Tab Free Play
Figure 218

(5) Rudder Free Play Measurement (Figure 219):

(a) Secure rudder pedal assembly with 27-95019 pedal lock or equivalent.

(b) Attach dial indicator stand to supporting structure.

(c) Zero (0) dial indicator.

NOTE: Ensure indicator stylus is in positive contact with rudder.

(d) Apply (LH) force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

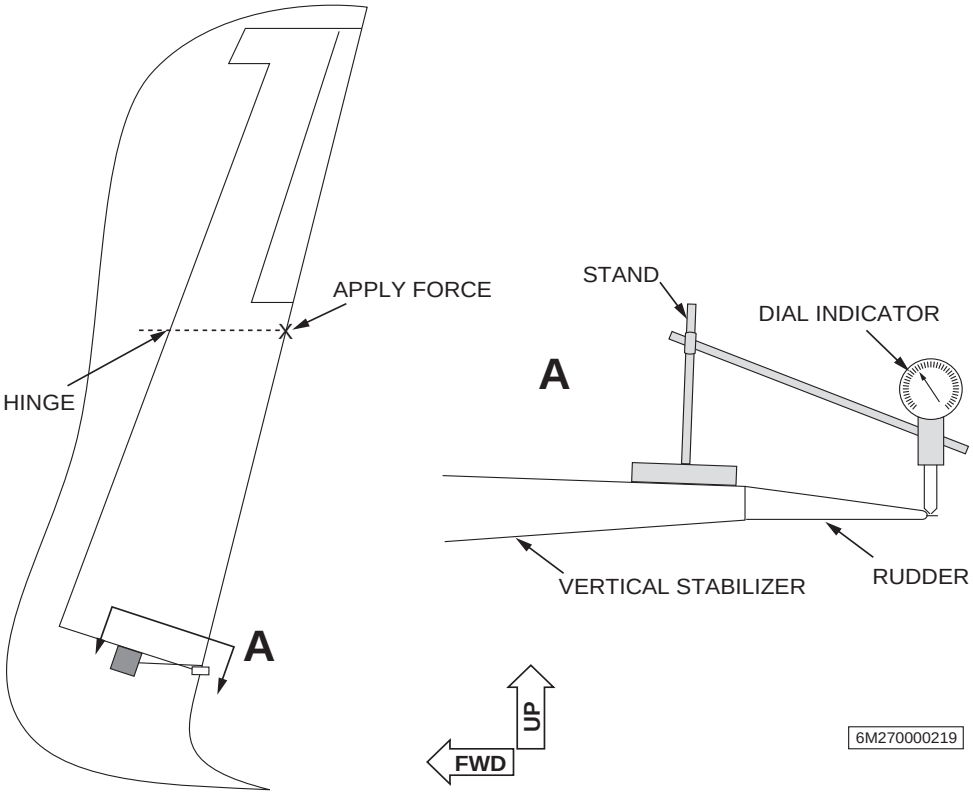
(e) Apply (RH) force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

(f) Add readings from steps **4.**, F., (5), (d) and (e). Verify free play does not exceed 0.440 inch 11.176 mm).

(g) Remove pedal lock.

(h) If free play measurement exceeds limit, repair or replace as necessary.



Rudder Free Play
Figure 219

(6) Rudder Trim Tab Free Play Measurement (Figure 220):

(a) Immobilize rudder to fixed structure.

(b) Attach dial indicator stand to supporting structure.

(c) Zero (0) dial indicator.

NOTE: Ensure indicator stylus is in positive contact with rudder trim tab.

(d) Apply (LH) force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

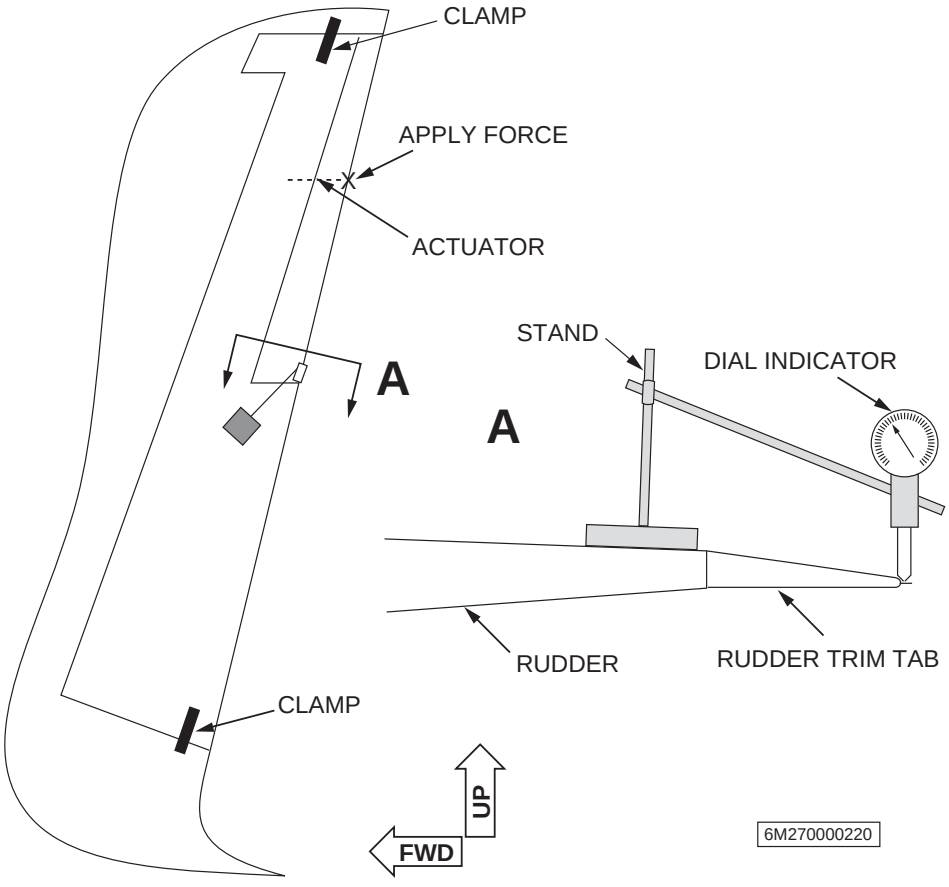
(e) Apply (RH) force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

(f) Add readings from steps **4.**, F., (6), (d) and (e). Verify free play does not exceed 0.080 inch (2.032 mm).

(g) Release rudder trim tab.

(h) If free play measurement exceeds limit, repair or replace as necessary.



Rudder Trim Tab Free Play
Figure 220

(7) Elevator Free Play Measurement (Figure 221):

(a) Immobilize left or right elevator to fixed structure.

(b) Attach dial indicator stand to supporting structure.

(c) Zero (0) dial indicator.

NOTE: Ensure indicator stylus is in positive contact with elevator.

(d) Apply upward force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

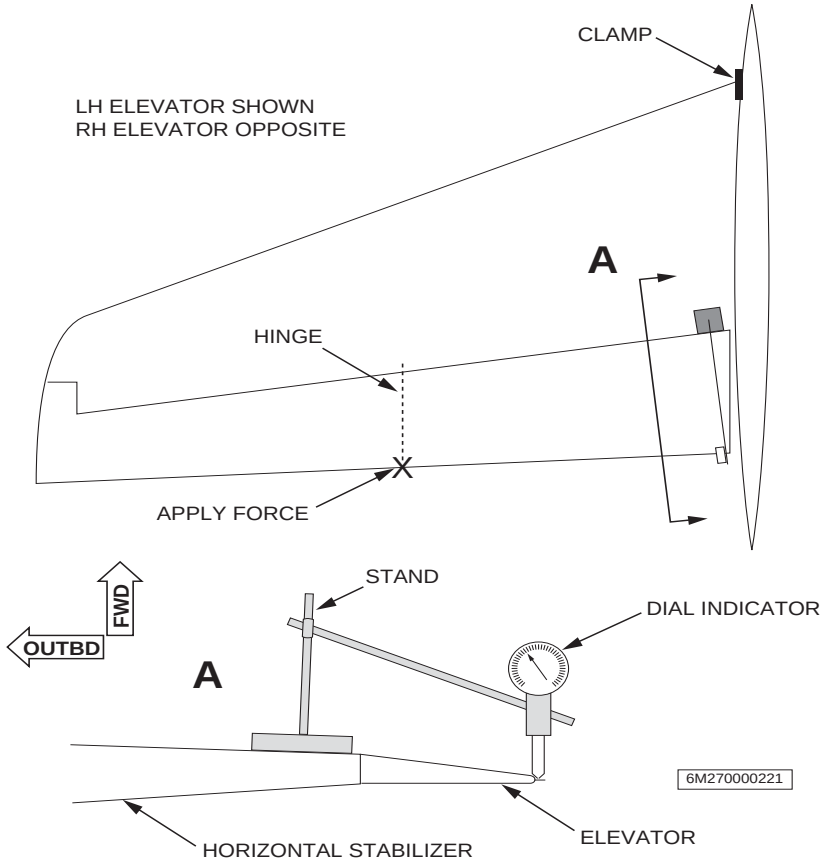
(e) Apply downward force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

(f) Add readings from steps **4.**, F., (7), (d) and (e). Verify free play does not exceed 0.290 inch (7.366 mm).

(g) Release elevator.

(h) If free play measurement exceeds limit, repair or replace as necessary.



Elevator Free Play
Figure 221

(8) Horizontal Stabilizer Free Play Measurement (Figure 222):

(a) Position workstand close to stabilizer tip to allow measurement of stabilizer relative to workstand.

(b) Apply upward force to surface and record travel.

NOTE: Apply only sufficient force to take-up free play.

(c) Apply downward force to surface and record travel.

NOTE: Apply only sufficient force to take-up free play.

(d) Add readings from steps **4.**, F., (8), (b) and (c). Verify free play does not exceed 0.250 inch (6.350 mm).

(e) If free play measurement exceeds limit, repair or replace as necessary.

(f) Apply forward force to surface and record travel.

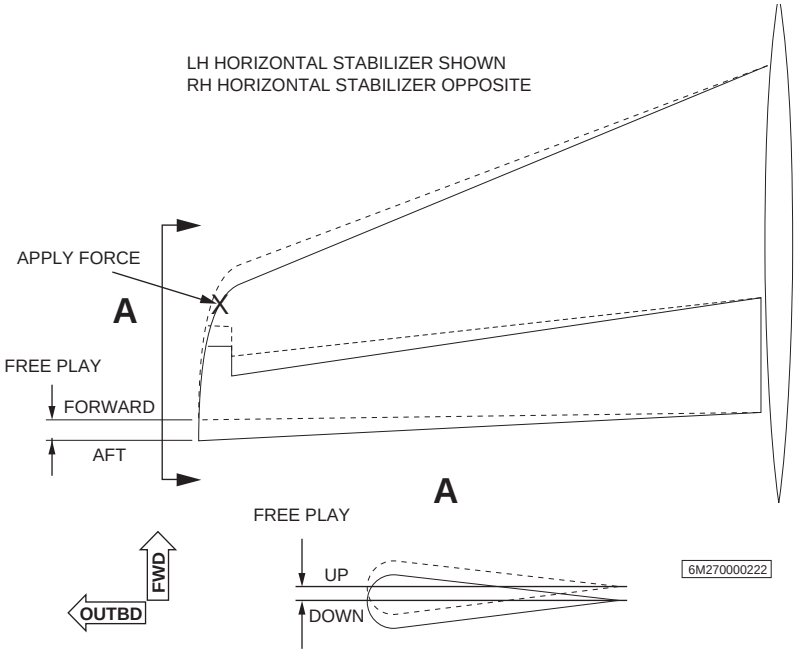
NOTE: Apply only sufficient force to take-up free play.

(g) Apply aft force to surface and record travel.

NOTE: Apply only sufficient force to take-up free play.

(h) Add readings from steps **4.**, F., (8), (f) and (g). Verify free play does not exceed 0.250 inch (6.350 mm).

(i) If free play measurement exceeds limit, repair or replace as necessary. Refer to 27-40-00 page 204, Inspection/Check – Horizontal Stabilizer Wear – Stabilizer Trunion Bolt.



Horizontal Stabilizer Free Play
Figure 222

(9) Flap Free Play Measurement (Figure 223):

(a) Position flaps to full UP.

(b) Immobilize left or right flap to fixed structure.

(c) Attach dial indicator stand to supporting structure.

(d) Zero (0) dial indicator.

NOTE: Ensure indicator stylus is in positive contact with flap.

(e) Apply upward force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

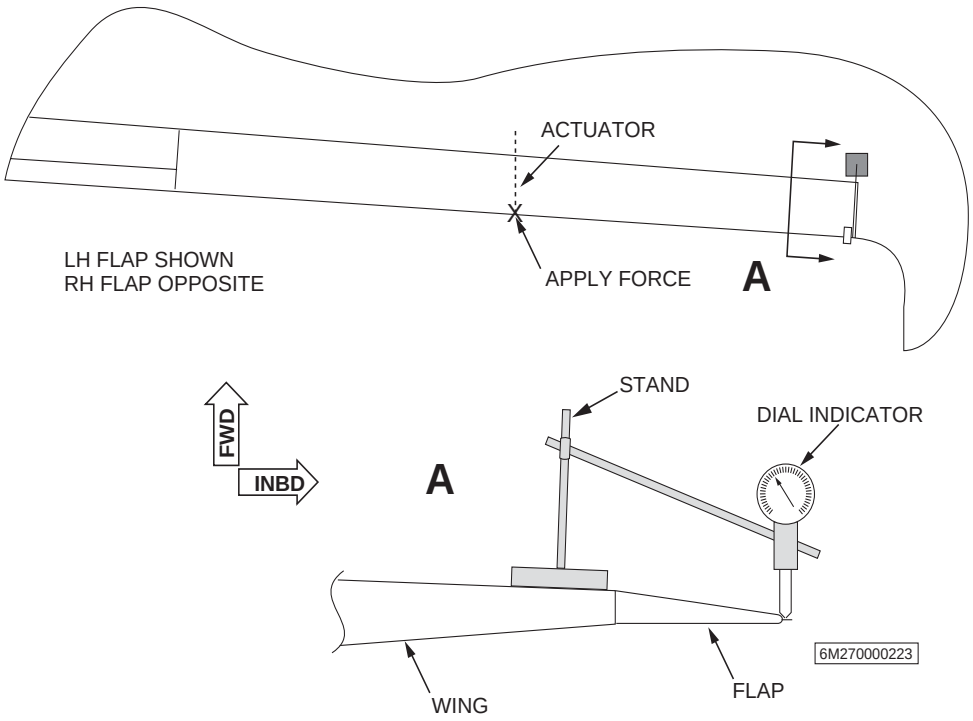
(f) Apply downward force to surface and record dial indicator reading.

NOTE: Apply only sufficient force to take-up free play.

(g) Add readings from steps **4.**, F., (9), (e) and (f). Verify free play does not exceed 0.250 inch (6.350 mm).

(h) Release flap.

(i) If free play measurement exceeds limit, repair or replace as necessary.



Flap Free Play
Figure 223

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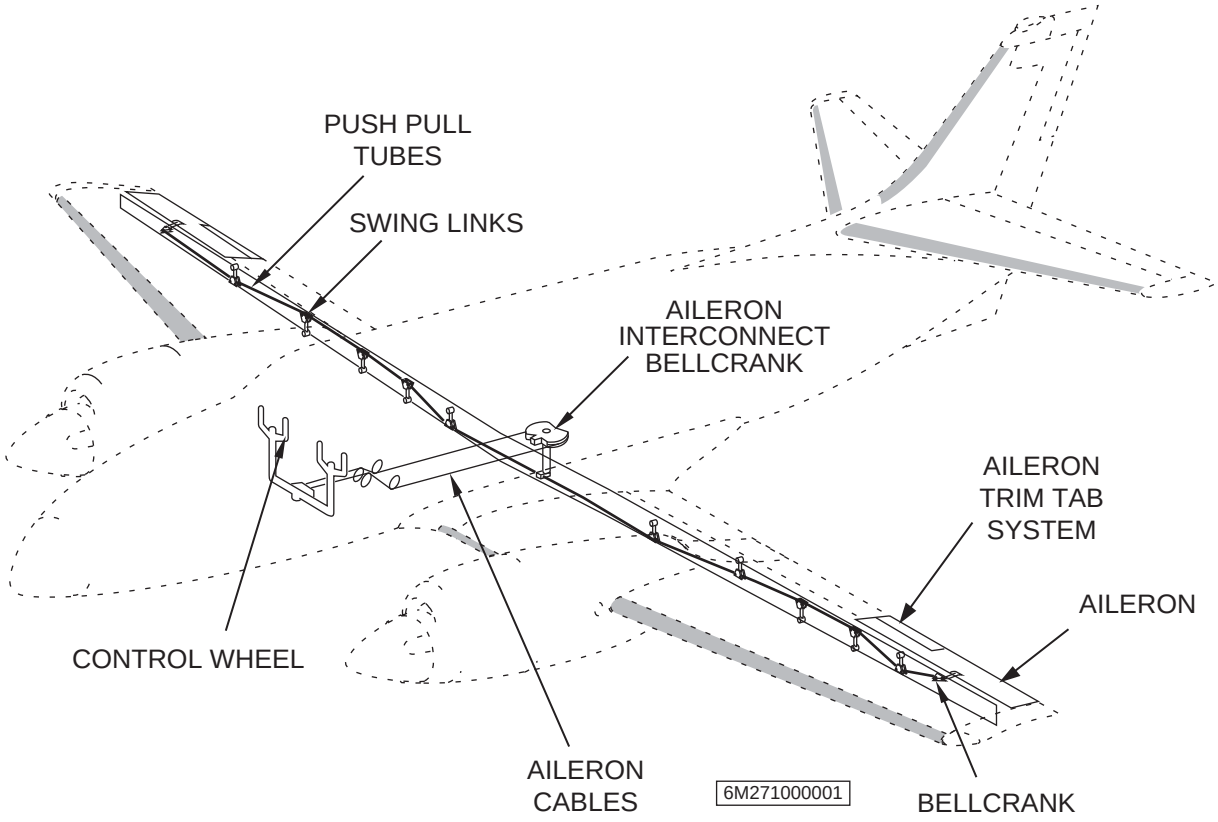
AILERON – DESCRIPTION AND OPERATION

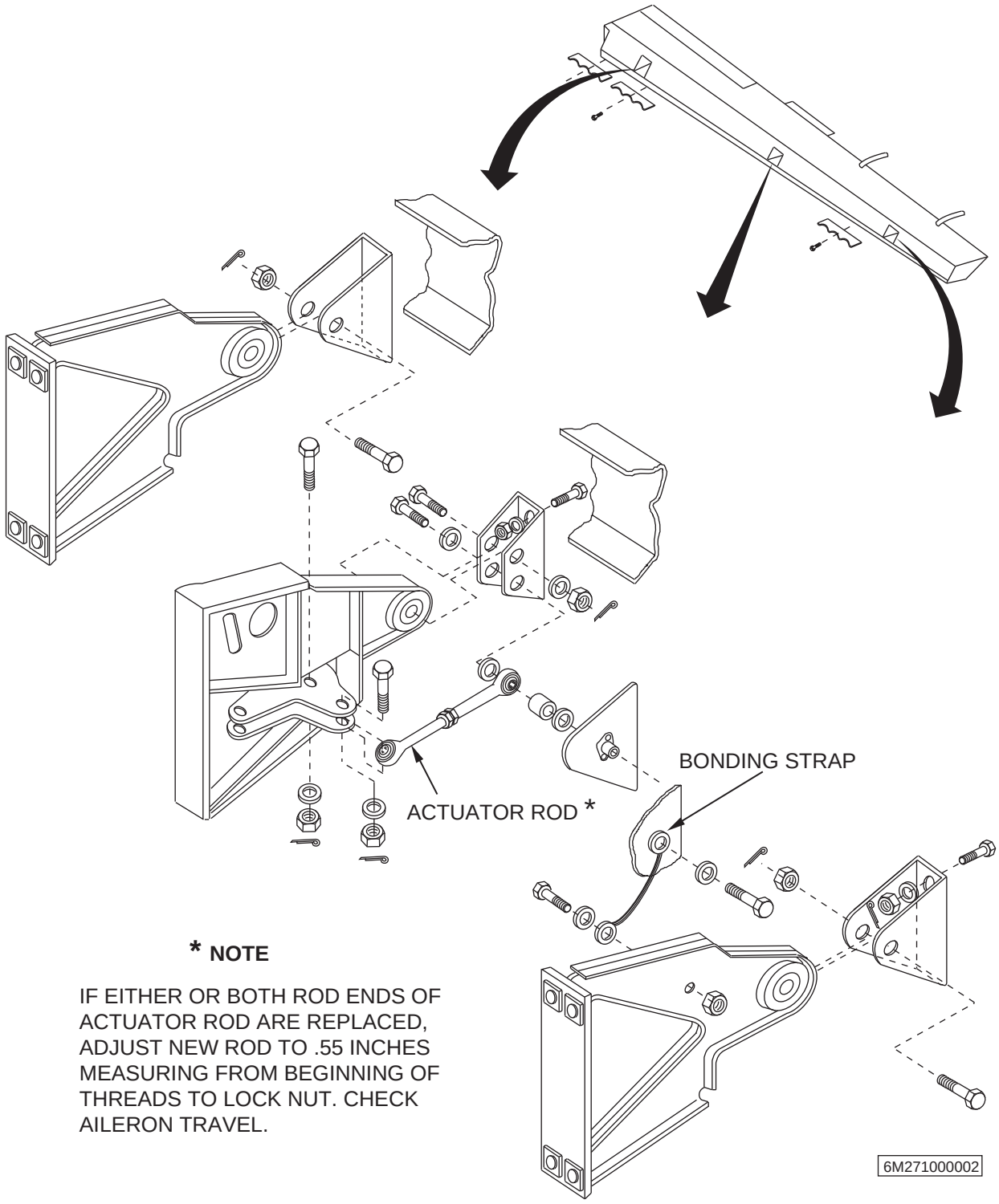
1. General

A. Ailerons are manufactured of aluminum skin riveted over a frame of spars and ribs. Each aileron is attached to the wing trailing edge at three hinge points. Small ground adjustable trim tabs are attached to each aileron.

B. Ailerons are interconnected to two control wheels by a sprocket, chain, and cable arrangement. A sprocket wheel is fixed to each control wheel and a cable is attached to the end of each chain. Cables are routed through the control wheel shaft under the aisle floorboards along the center aisle to the aileron interconnect bellcrank. The bellcrank turns the aileron shaft. The shaft actuates the push-pull tubes which are routed to the ailerons along the wing trailing edge. Swing links, mounted to the wing trailing edge, support a series of push-pull tubes. The push-pull tubes move the aileron bellcrank at the center hinge. The left aileron incorporates a flight adjustable trim tab.



Aileron Location
Figure 1



*** NOTE**

IF EITHER OR BOTH ROD ENDS OF ACTUATOR ROD ARE REPLACED, ADJUST NEW ROD TO .55 INCHES MEASURING FROM BEGINNING OF THREADS TO LOCK NUT. CHECK AILERON TRAVEL.

Aileron
Figure 2

AILERON – MAINTENANCE PRACTICES

1. Removal / Installation – Aileron

A. Removal – Aileron

(1) Disconnect trim tab actuator rod (left aileron).

(2) Disconnect one end of bond strap (located at outboard hinge).

(3) Remove safety wire and bolt securing aileron actuator rod at center hinge.

(4) Support aileron and remove cotter pins, nuts, and bolts at three aileron hinges.

(5) Remove aileron from wing.

B. Installation – Aileron

(1) Position aileron.

(2) Install bolts, nuts, and cotter pins at three hinges.

(3) Connect aileron actuator rod at center hinge and safety wire.

(4) Connect bond strap.

(5) Connect trim tab actuator rod (left aileron only).

(6) Check aileron through full travel for proper operation. Maintain .12 inch minimum clearance between aileron and flap and between aileron and wing tip. If any changes were made to linkages or adjustments, or if a new aileron was installed, aileron travel must be checked.

2. Adjustment/Test – Aileron

A. Operational Test

(1) Set pilot control wheel in neutral position. Aileron should be streamlined. Copilot wheel should be in neutral position.

(2) Turn wheel slowly full left. Wheel should turn smoothly. Left aileron should come up and right aileron should go down.

(3) Turn wheel slowly full right. Wheel should turn smoothly. right aileron should come up and left aileron should go down.

(4) Repeat procedures using copilot wheel.

3. Inspection/Check – Aileron

A. Inspection Aileron

(1) Open access panels as necessary to gain access to aileron components.

(2) Operate ailerons through full range of travel and check for unusual noise, freedom of movement, and correct travel of aileron and trim tab.

(3) Inspect cables, cable pulleys, fairleads, and pressure seals (see Section 27-00-00).

(4) Check swing links, push-pull tubes, bellcranks, and torque tube.

(5) Check cables for correct rigging tension (see Section 27-00-00).

(6) Check trim control in pedestal for damage, proper lubrication, and condition.

(7) Inspect trim actuating mechanism in aileron for damage, security of attachment, and proper lubrication.

(8) Check for aileron linkage rod end bearing wear IAW Table 201.

Rod End	Radial Wear Limit (inches)	Axial Wear Limit (inches)
HF-5M	0.008	0.024
HM-5M	0.008	0.024
HFR-5M	0.006	0.018
HMR-5M	0.006	0.018
H5-5	0.004	0.012
HE5-5	0.004	0.012
HFE-5M	0.0045	0.0135
HME-5M	0.0045	0.0135
KBD5-243	0.0045	0.0135
KBDE5-351	0.0045	0.0135
KR5-MA	0.0045	0.0135
REF10ATC10Z	0.006	0.018
REM10ATC10ZM	0.006	0.018

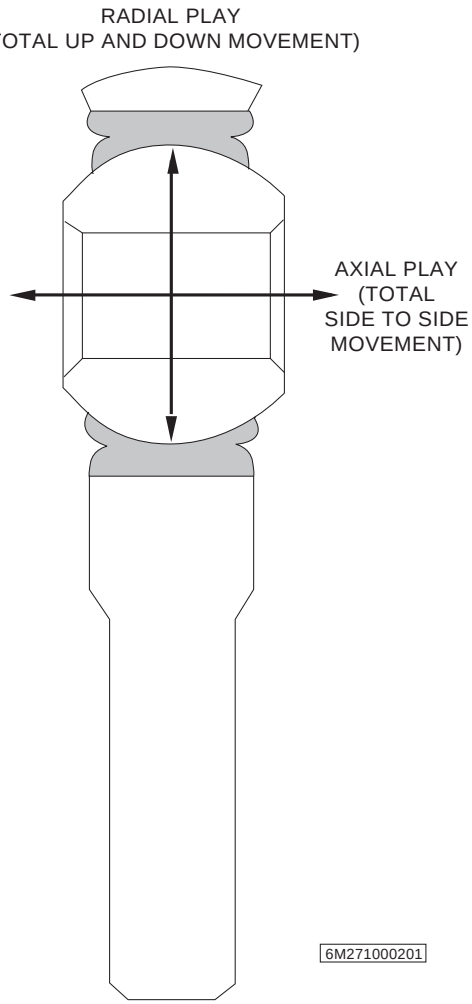
Aileron Linkage Rod End
Bearing Wear limits
Table 201

(9) Check roller chain master links for security.

(10) Inspect all hinges and hinge brackets for damage, cracks, wear, defective bearings, and security.

(11) Inspect aileron and tab structures for damage, cracks, corrosion, loose or missing rivets, cracked or buckled spars, ribs, or stringers, and separation of metal-bonded structure.

(12) Check aileron free play IAW 27-00-00.



Aileron Radial and Axial Play
Figure 201

4. Approved Repairs – Aileron

A. Hinge Rework – Aileron (see Figure 1)

(1) If the aileron bellcrank hinge hole wears excessively due to hinge bolt being too loose, perform the following:

(a) Remove aileron hinge bolt.

(b) Remove aileron hinge bellcrank.

(c) Measure diameter of hinge holes in hinge assembly (maximum service diameter 0.317).

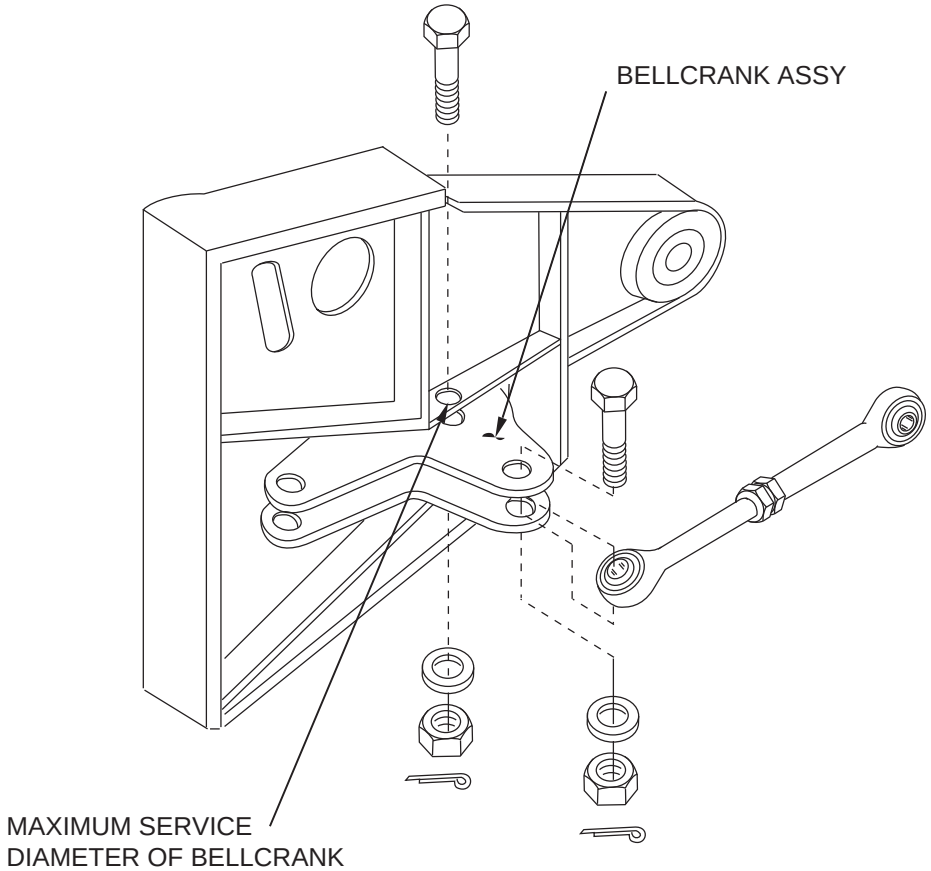
(d) If holes are over the maximum service diameter; line ream to 0.377 / 0.379.

(e) Install bushing in each hinge hole with loctite 609.

(f) Install bellcrank and new bolt – length to match.

(g) Torque bolts 20 to 25 inch pounds (2.26 to 2.82 N.M).

(h) Tighten nut to next castellation alignment and install cotter pin.



Aileron Bellcrank Pivot Repair
Figure 202

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AILERON CONTROL – DESCRIPTION AND OPERATION

1. General

A. The ailerons control the roll axis of the airplane, and the aileron control system provides the means of transmitting pilot/copilot commands to the ailerons.

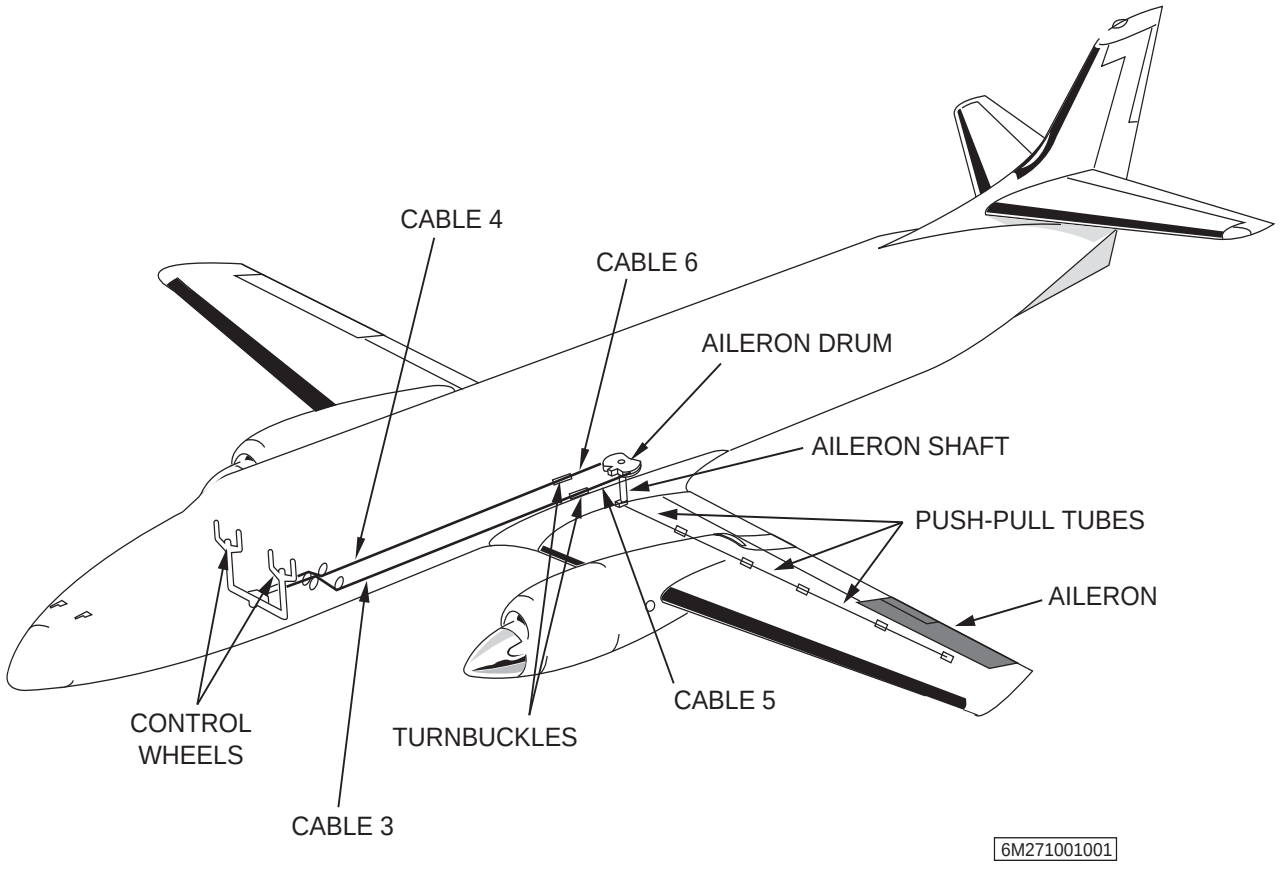
B. Aileron control is made up of the following components:

(1) Two control wheels (pilot and copilot) attached to bearing mounted sprocket and chain assemblies in the control column.

(2) Cables, turnbuckles, fairleads, and pulleys.

(3) An aileron drum, shaft, and bellcrank.

(4) Push-pull tubes and swing links.



Aileron Control
Figure 1

2. Control Wheels

A. Pilot and copilot control wheels are attached to bearing mounted sprocket and shaft assemblies in the control columns.

B. The pilot and copilot control wheels are interconnected through the control column to provide synchronized dual operation.

C. Control wheels may be removed individually without disturbing the rigging of the aileron or elevator control system.

3. Control Cables

A. The aileron control cable system consists of three sections, with two cables in each section, joined together with turnbuckles and chains to make one continuous control system.

(1) One cable section is housed within the control column (see Figure 202). Cable 1 (pilot) and Cable 2 (copilot) are joined together by a single turnbuckle. These cables are attached to aileron control chains engaging sprockets on the pilot and copilot control wheel shafts which enable the control wheels to move simultaneously.

(2) The next cable section (Cable 3 and Cable 4) connects to the other end of the aileron control chains (see Figure 202) and extends from the control columns, under the crew compartment and center aisle floorboards, and connects to turnbuckles near the wing rear spar (see Figure 1).

(3) The final section of cables (Cable 5 and Cable 6) connects to turnbuckles on the aft end of Cables 3 and 4, and connects to the aileron drum (aileron interconnect bellcrank on T models) by clevis attachments at their other end (See Figure 1).

4. Aileron Drum

A. The aileron drum is inside the pressure vessel under the center aisle floorboards between FS 311 and FS 322 and connects to the aileron shaft. The shaft extends downward outside the pressure vessel and drives push-pull tubes routed along the wing trailing edge, to a bellcrank at the aileron center hinge. The push-pull tubes are supported by a series of swing links (see Figure 203).

5. Push-Pull Tubes

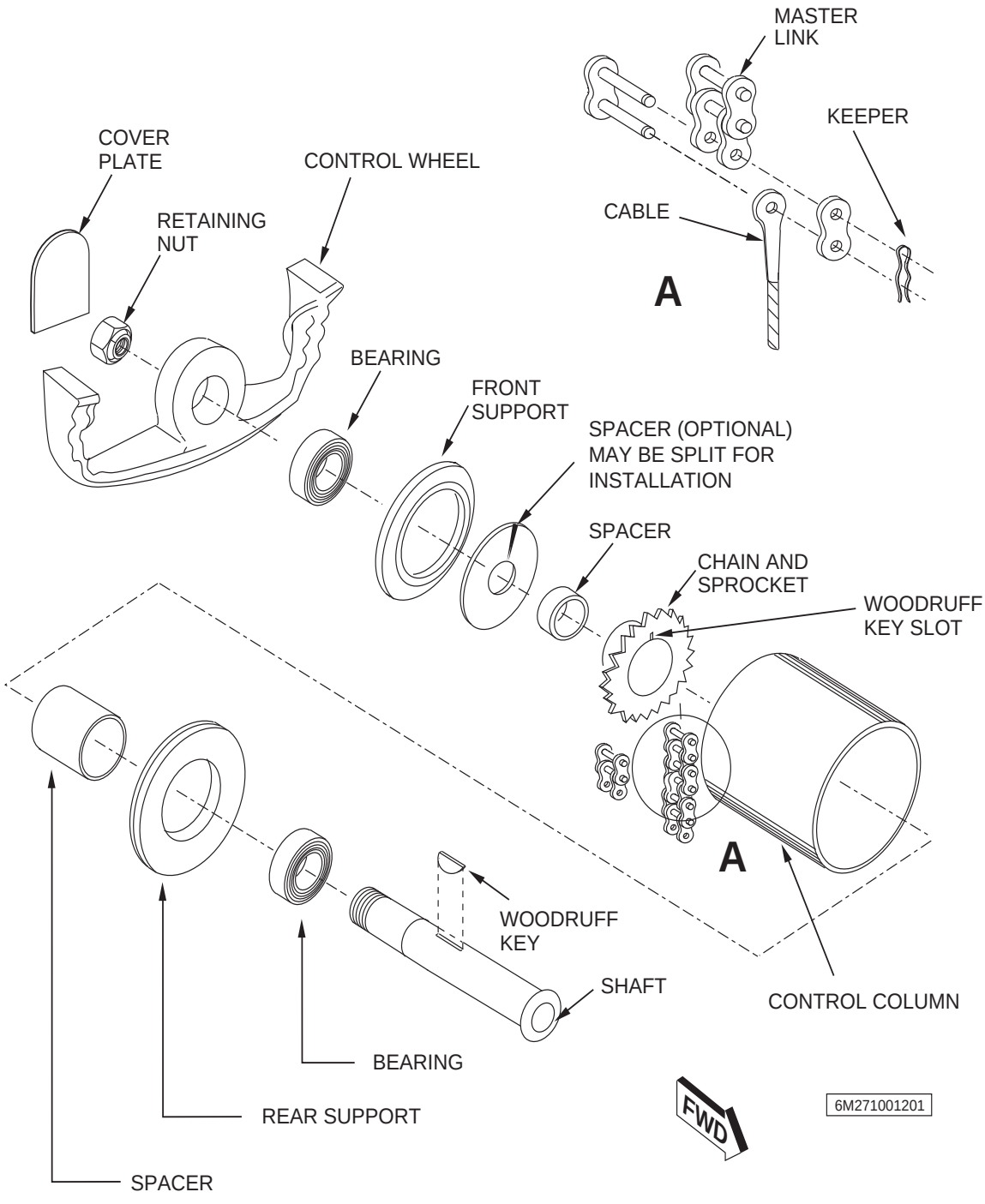
A. Seven adjustable push-pull tubes, attached to a series of swing links and routed along the wing trailing edge, activate the aileron (see Figure 203).

(1) The six outboard tubes have right-hand threads. The inboard (shortest) tube has opposing threads for adjustment.

(2) The two outermost tubes can only be removed by first removing the aileron. The inboard tube can be removed by first removing the flap walking beam access panel. The remaining tubes are located forward of the wing flap.

(3) Control tubes located forward of the flap may be removed without removing the flap by lowering flap, removing screws securing foreflap, and positioning foreflap clear of tubes.

1. Removal/Installation – Aileron Control



Control Wheel and Bearing Assembly
Figure 201

A. Removal – Control Wheel (Figure 201)

(1) Locate hole in lower center front of control wheel and remove cover plate setscrew.

(2) Remove cover plate.

(3) Mark/identify switch wiring for ease of reinstallation and correct location. Remove switches and switch housings. May include microphone, intercom, auto pilot disconnect, GA (go-around) and pitch trim.

(4) Remove switches. May require unsoldering wires from switches.

(5) Pull switch wires through control wheel.

(6) Remove retaining nut from shaft and remove control wheel.

B. Installation – Control Wheel (Figure 201)

(1) Apply thread compound IAW MIL-T-5544 to shaft splines. Thread compound may be purchased from Armite Labs, Los Angeles, California, or ESSO Standard Oil Co., New York, New York.

(2) Position control wheel onto shaft and secure with retaining nut. Torque nut 210 to 250 inch-pounds (23.73 to 28.25 N·m).

(3) Operate control wheel and check for roughness and binding. If necessary, add additional spacers to shaft until operation is smooth.

(4) Insert switch wires through control wheel shaft.

(5) Noting correct wire identification, connect wires to switches. May require soldering wires to switches. Ensure switches are installed at correct orientation for proper pitch trim travel.

(6) Install switches and housings.

(7) Position cover plate and secure with setscrew.

(8) Perform operational check of pitch trim and other control wheel switches as applicable.

C. Removal – Control Wheel Bearings (Figure 201)

(1) Apply gust lock and work control wheel to engage aileron lock.

(2) Remove control wheel IAW previous instructions in this section.

(3) Remove forward bearing support (with bearing inserted) from control column using screwdriver or similar tool. It may be necessary to temporarily install control wheel nut onto shaft and alternately strike nut and opposite end of shaft with a mallet until forward bearing support starts to separate from control column.

NOTE: It is not necessary to remove bearings from supports to check them for looseness or binding. Defective bearings can be detected by turning bearing with finger while bearing is still inserted in support.

(4) Remove forward spacer(s) from shaft.

(5) Mark a chain link and corresponding sprocket tooth for guidance when reassembling.

(6) Insert long heavy screwdriver or similar tool through center of control column wheel shaft to keep assembly from falling down control column when shaft is removed.

(7) Using a rubber mallet, drive control wheel shaft out far enough to free rear bearing support from control column, then pry rear bearing support from control column.

(8) Drive shaft out of sprocket and thread a wire through center of sprocket and chain. Wire should be long and stiff enough so if chain and sprocket fall down control column they can be retrieved.

(9) Remove shaft, rear bearing support (with bearing), spacer and Woodruff key from rear side of control column.

(10) Remove bearing(s) from support(s) by securing support and driving bearing out from inside support. Once bearing is removed, it must be discarded and replaced with a new bearing.

D. Installation – Control Wheel Bearings (Figure 201)

(1) Insert new bearing into rear support.

NOTE: Ensure force is applied to outside race of bearing when inserting bearing into support.

(2) Position rear support and bearing assembly onto shaft.

(3) Slide rear spacer(s) onto shaft and insert Woodruff key into slot in shaft.

(4) Hold sprocket and chain in place with screwdriver and remove wire from around chain and sprocket.

NOTE: Verify alignment between chain and sprocket and between sprocket and shaft has not slipped.

(5) Insert shaft with rear bearing and support assembly into control column from rear while positioning chain and sprocket onto shaft. Ensure groove in sprocket is aligned with Woodruff key and press sprocket onto shaft until up against rear spacer.

(6) Install forward spacer(s) onto shaft.

(7) Insert new bearing into front support. Install front bearing and support assembly onto shaft and seat into lip in column opening.

NOTE: If any looseness is evident in assembly after seating front support, a spacer may be added as required. To make installation easier, split one side of the spacer as shown in Figure 201.

(8) Install control wheel IAW previous instructions in this section.

E. Removal – Control Column Pulleys

NOTE: Top or bottom pulleys (see Figure 202).

(1) Gain access to floorboards immediately forward of and above aileron drum (rudder/aileron interconnect on T models) and remove floorboards. (Refer to 06-10-00 and 53-30-30)

(2) Disconnect aileron cable at either left or right turnbuckle, whichever side pulley is being removed. Tie safety wire to terminal of cable for ease in locating cable when reconnecting.

(3) Remove pilot and copilot seats. (Refer to 25-10-01)

(4) Remove boots from base of control column.

(5) Remove crew compartment floorboards at control column. (Refer to 53-30-20)

(6) Remove top pulley as follows:

(a) Remove bolts securing control column to base.

(b) Remove column by steadily twisting and pulling upward and lay column aside.

(c) Remove bolt securing pulley in base of control column and remove pulley.

(7) Remove bottom pulley as follows – pilot side only:

(a) Remove access door cover plate located on base of control column.

(b) At base of control column, locate rod end connecting elevator walking beam to control column and disconnect rod from column.

(8) Pull column aft far enough to enable removal of bolt securing pulley.

(9) Remove pulley through access door in base of control column.

(10) Remove long bolt holding pulley guard at base of control column.

(11) Working through access door remove nut, washer, spacer, and bolt from within guard.

(12) Remove pulley through access door.

F. Installation – Control Column Pulleys

(1) Install top pulley as follows:

(a) Position cable onto pulley and position pulley into place.

(b) Insert pulley bolt through control column and pulley. Secure with nut.

(c) Install control column into column base and secure with bolts.

(2) Install bottom pulley as follows – pilot side only:

(a) Through access door, position cable onto pulley and insert pulley into place.

(b) Insert pulley bolt through control column and pulley and secure with nut.

(3) For ease in future removals, reverse installation of pulley bolt and nut. Insert bolt from forward side of column with nut on aft side.

(a) Insert bolt, spacer, washer, and nut within pulley guard.

(b) Insert long bolt holding pulley guard at base of control column.

(c) Install access door cover plate.

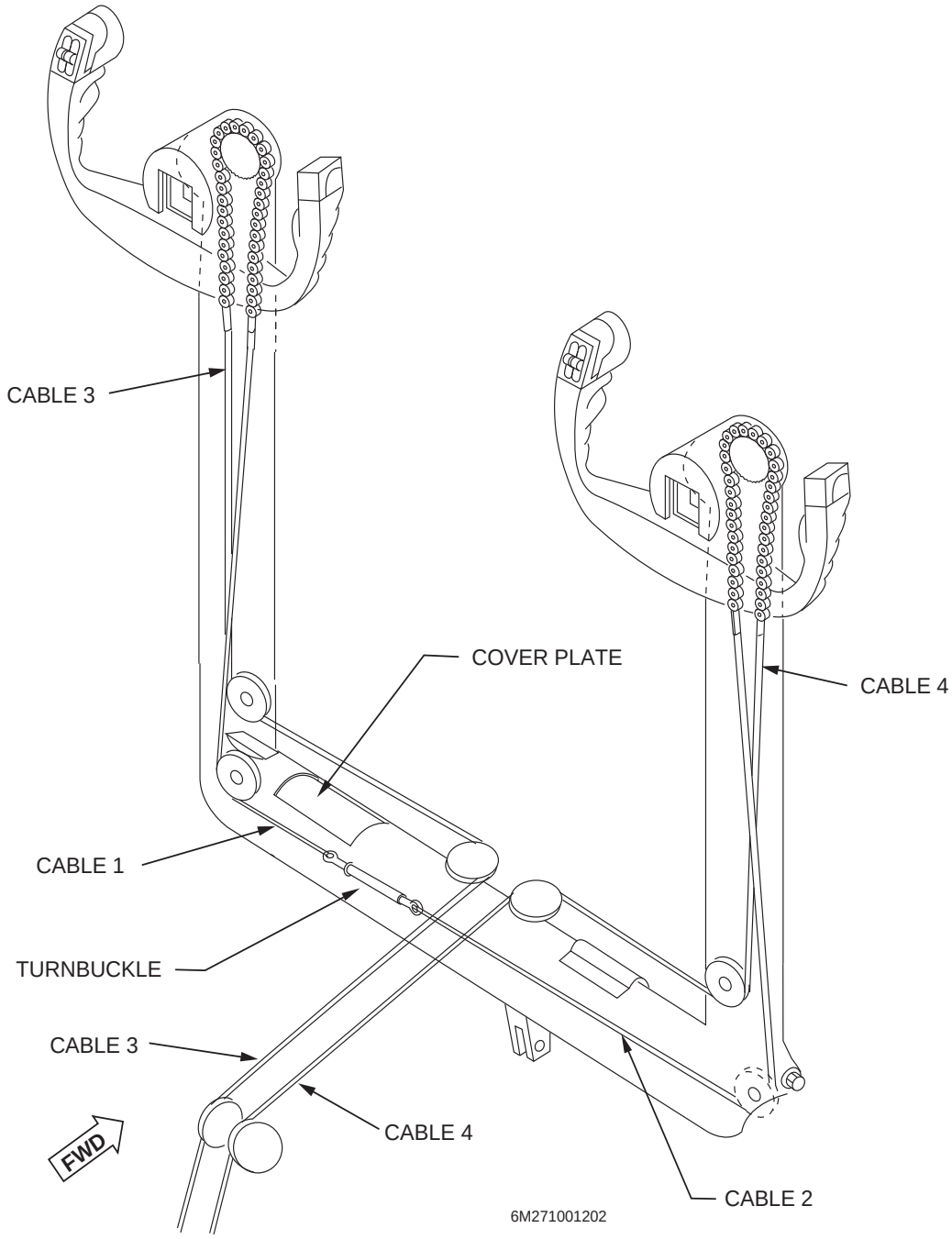
(d) Attach walking beam rod to base of control column.

(4) Connect aileron cable at turnbuckle at rudder/aileron interconnect. Adjust cable tension. (Refer to 27-00-00)

(5) Perform Aileron Operational Test. (Refer to 27-10-00)

(6) Install floorboards and control column boots. (Refer to 53-30-20)

(7) Install pilot and copilot seats. (Refer to 25-10-01)



Aileron Control Cables
Figure 202

G. Removal - Cables 1, 2, 3 and 4 (see Figure 201 and Figure 202)

(1) Gain access to floorboards immediately forward of and above aileron drum (rudder/aileron interconnect on T models) and remove floorboards. (Refer to 53-30-30)

(2) For Cable 1 or Cable 2 removal, disconnect either Cable 3 (left) or Cable 4 (right) at turnbuckle. For Cable 3 or Cable 4 removal, disconnect cable being removed.

(3) Remove pilot and copilot seats. (Refer to 25-10-01)

(4) Remove boots from base of control columns and remove crew compartment floorboards at control column. (Refer to 53-30-20)

(5) Perform Control Wheel Bearing Removal procedure in this section, but do not remove bearings from supports unless bearings require replacement at this time. For sprocket to chain alignment reference during reinstallation, be sure to secure chain to sprocket with one end of the wire that was threaded through center of sprocket and chain during the Control Wheel Bearing Removal procedure.

(6) Remove screwdriver securing sprocket and chain to top of control column. Guide sprocket and chain down and out through bottom of control column while allowing retrieval wire to be pulled down through top of control column.

NOTE: Control column removal is optional, although it maybe more convenient to replace chain and cables. If column is not removed, disregard steps **2.**, A. (7) and (8), and **2.**, B. (8).

(7) Remove bolts securing control column to base. Bolt located in center of groups of four bolts hold pulley and need not be removed to remove column.

(8) Remove column by steadily twisting and pulling upward and lay column aside.

(9) For Cable 1 or Cable 2 removal:

(a) Remove cover plate located on base of control column and separate cables at turnbuckle.

(b) Cut terminal off of cable being replaced.

(c) Attach ball end of new cable to cut end of cable being replaced with safety wire or tape.

(d) Separate cable being replaced from chain master link.

(e) At base of column, locate rod end connecting elevator walking beam to control column and disconnect rod from column.

(f) Rotate column far enough to enable removal of bolts securing lower pulley and pulley cable guard.

(g) Pull cable being replaced through base of control column working ball end of new cable through pulley guard and around lower pulley.

(h) Separate old and new cable.

(10) For Cable 3 or Cable 4 removal:

(a) Separate cable being replaced from aileron control chain master link.

(b) Remove hardware securing upper pulley to column.

(c) Pull cable being replaced from control column.

(d) Remove cable guards and fairleads in crew compartment and along center aisle as necessary and remove cable.

(11) If chain and/or sprocket will be replaced, keep removed chain wired to sprocket for use as alignment reference during installation of new chain and/or sprocket.

H. Installation – Cables 1, 2, 3 and 4 (see Figure 201 and Figure 202)

NOTE: Aileron control cable, control wheel chain, control wheel sprocket and control wheel bearings are time limited components. Verify time has not expired on any of these components prior to installation.

(1) Clean the aileron control wheel chain thoroughly with a degreasing solvent (MIL-PRF-680 or equivalent). Once the chain has dried, lubricate the entire length of chain including master links with MIL-PRF-81322 grease. Work the lubricant into the chain by exercising the chain by hand.

(2) If new chain and/or sprocket are being installed:

(a) Mesh sprocket to chain so sprocket teeth are toward front of aircraft and woodruff key slot aligns with center of chain. Woodruff key slot should be at 12 o'clock position with sprocket teeth forward after sprocket and chain are pulled up through control column.

(b) Wrap end of a wire around chain and sprocket sufficiently to maintain the same chain to sprocket alignment during the following steps.

(3) Attach ball end of Cable 1, 2, 3 and/or 4 to master link at end of aileron control chain.

(4) For Cable 1 or Cable 2 installation, connect terminal end of new cable to remaining control column cable at turnbuckle inside base of control column.

(5) For Cable 3 and Cable 4 installation, route cable along center aisle and crew compartment through pulley guards and fairleads as necessary.

(6) Pull opposite end of retrieval wire securing chain to sprocket. Guide chain and sprocket through control column.

(7) Hold sprocket and chain in place by inserting screwdriver or similar tool through top of control column and through sprocket.

(8) If removed, install control column into base and secure with bolts.

(9) Perform Control Wheel Bearing Installation procedure in this section. If bearings were not replaced, disregard step installing new bearings into supports.

(10) Install all removed pulleys and guards on control column, and all removed pulley guards and fairleads between control column and aileron drum (rudder/aileron interconnect on T models).

(11) Connect terminal end of Cable 3 to turnbuckle on Cable 5 and/or connect Cable 4 to turnbuckle on Cable 6 as applicable.

(12) Adjust cable tension to proper setting. Adjust cables at turnbuckle within control column until both pilot and copilot wheels are evenly centered. (Refer to 27-00-00)

(13) Check rigging of aileron control system to verify proper control wheel to aileron direction and aileron travel is within limits. (Refer to 27-00-00)

(14) Install cover plate on base of control column.

(15) Install crew compartment floorboards and control column boots (Refer to 53-30-20).

(16) Install center aisle floorboards. (Refer to 53-30-30)

(17) Install pilot and copilot seats. (Refer to 25-10-01)

I. Removal – Cable 5 and Cable 6 (see Figure 1)

(1) Apply gust lock and work control wheel to engage aileron lock.

(2) Gain access to floorboards immediately forward of and above aileron drum (rudder/aileron interconnect on T models) and remove floorboards. (Refer to 53-30-30)

(3) Block cables as appropriate and disconnect aileron Cable 5 from turnbuckle on Cable 3 and/or disconnect Cable 6 from turnbuckle on Cable 4.

(4) Disconnect Cable 5 and/or Cable 6 from aileron drum and/or rudder/aileron interconnect.

J. Installation – Cable 5 and Cable 6 (see Figure 1)

(1) Connect Cable 5 and/or Cable 6 ball end to aileron drum and/or rudder/aileron interconnect.

(2) Connect Cable 5 terminal end to turnbuckle attached to Cable 3, and/or connect Cable 6 terminal end to turnbuckle attached to Cable 4.

(3) Remove cable blocking.

(4) Adjust cable tension to proper setting. (Refer to 27-00-00)

(5) Check rigging of aileron control system to verify proper control wheel to aileron direction and aileron travel is within limits. (Refer to 27-00-00)

(6) Install cabin center aisle floorboard. (Refer to 53-30-30)

K. Removal – Push-Pull Tubes (Figure 203)

NOTE: Push-Pull tubes 1 and 2 can only be removed by first removing the aileron. Push-Pull tubes 6 and 7 can be removed by removing the flap walking beam access panel. The remaining tubes are located behind the wing flap. It is possible to remove tubes located behind the flap, without removing the flap, by lowering flap, removing screws securing foreflap, and positioning foreflap clear of tubes.

(1) Remove aileron IAW section 27-10-00, foreflap IAW 27-50-01, and/or walking beam access panel as necessary.

CAUTION: AILERON CAN BE MOVED BEYOND NORMAL LIMITS OF TRAVEL WITH PUSH-PULL TUBES DISCONNECTED. SECURE AILERON IN A NEUTRAL POSITION TO HELP PREVENT DAMAGE.

(2) Remove cotter pin, nut, bolt, and washer from swing link and disconnect tube.

NOTE: Do not remove tube by loosing jam nut at threads and unscrewing tube. Tube should be kept at original length so that new tube can be measured to same length as replacement tube prior to installation.

L. Installation – Push-Pull Tubes (Figure 203)

(1) Adjust length of replacement tube to match length of removed tube. If this cannot be done, approximate length of tubes are as follows (Figure 203):

(a) Tube 1 – 37.46 inches (95.15 cm).

(b) Tube 2 – 36.99 inches (93.95 cm).

(c) Tube 3 – 37.00 inches (93.98 cm).

(d) Tube 4 – 32.88 inches (83.52 cm).

(e) Tube 5 – 31.96 inches (81.18 cm).

(f) Tube 6 – 25.81 inches (65.56 cm).

(g) Tube 7 – 12.48 inches (31.70 cm).

WARNING: TUBE 7 TO RUDDER/AILERON INTERCONNECT BELLCRANK BOLT MUST BE INSTALLED WITH HEAD DOWN TO ENSURE BOLT TO ACCESS PANEL CLEARANCE; ENSURE COTTER PIN IS INSTALLED.

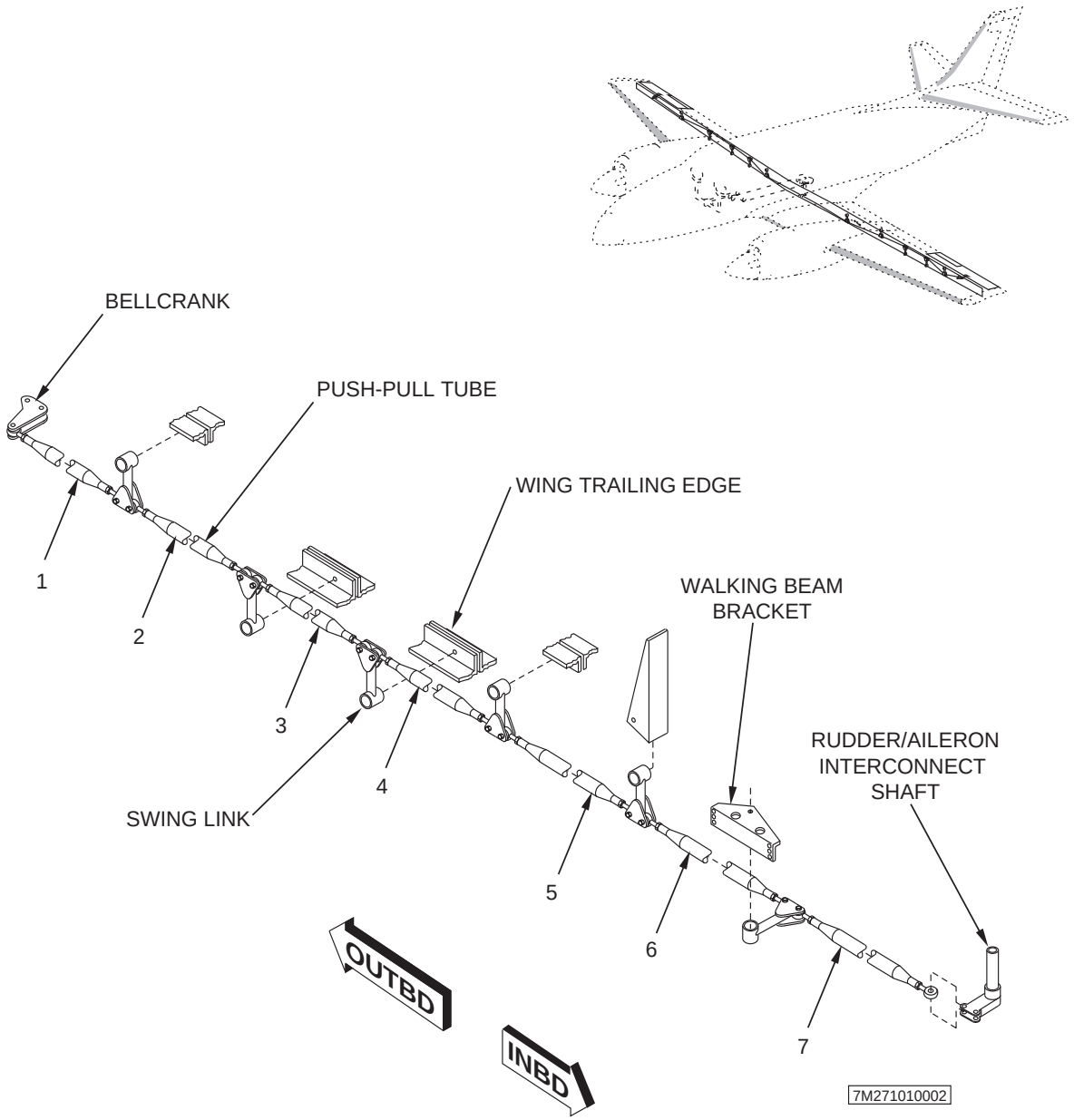
INSTALL TAB LOCK WASHER ON INBOARD TUBE (ITEM 7, FIGURE 203) WITH TAB POINTING AFT TO ENSURE ADEQUATE CLEARANCE BETWEEN TAB AND WALKING BEAM.

(2) Position tube(s) and connect each end with bolt, nut, washer, and cotter pin.

NOTE: The swing link between push-pull tubes 6 and 7 has a small angle. It can be installed with the angle up or down. To prevent rubbing on access panels and ensure proper alignment, install swing link with angle up.

(3) Install aileron IAW section 27-10-00, foreflap IAW 27-50-01, and/or walking beam access panel.

(4) Check rigging of aileron control system to verify proper control wheel to aileron direction and aileron travel is within limits (Refer to 27-00-00).



Aileron Push-Pull Tubes
Figure 203

2. Inspection/Check – Aileron Control

A. Inspection – Control Wheel Operation (Pilot and Copilot)

(1) Grab control wheel and disengage gust locks.

(2) With both hands on control wheel, rotate control wheel full deflection several times in both directions.

(3) While rotating control wheel, feel for smooth operation of control wheel bearings and control chains with no evidence of binding. Listen for unusual noises indicating a defective bearing, chain or sprocket in control column.

(4) Any discrepancy noted must be isolated and resolved prior to further flight.

B. Inspection - Aileron Control Chain

(1) If a discrepancy is noted during the Control Wheel Operation Inspection above and the isolation process requires visual inspection of the aileron control chain inside the control column, disassemble affected control column as necessary per instructions in this section and inspect as follows:

(a) Inspect control chain links for corrosion, security, cracks or other damage.

(b) Inspect control chain for proper position on sprocket and freedom of movement with aileron control cable set to proper tension.

(2) Any defect noted during this chain inspection/check requires chain replacement.

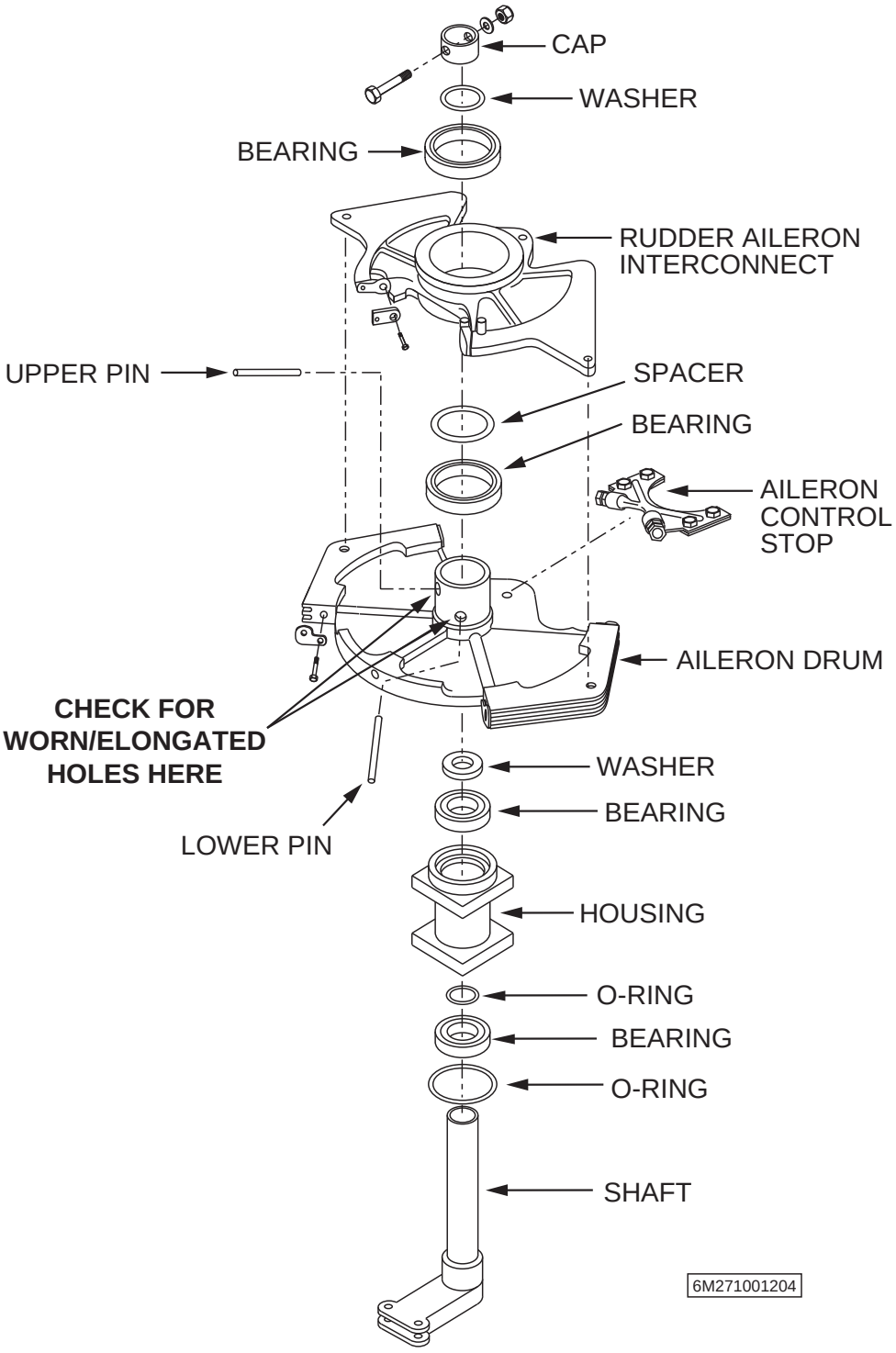
3. Approved Repair – Aileron Control

A. Repair – Aileron Drum (Figure 204)

SA226 airplanes use pins to attach aileron drum to shaft in order to provide clearance for the rudder aileron interconnect bellcrank to fit over the upper portion of aileron drum.

Excessive free play at the aileron, or slop in the system, may be traced back to elongated holes at the aileron drum lower and upper pins (Refer to Section 27-00-00 for Free Play Measurement - Flight Controls).

(1) If upper or lower pin holes in aileron drum boss are worn or elongated, replace drum or contact M7 Aerospace Support Engineering at MetroTech@M7Aero.com with details for repair advice.



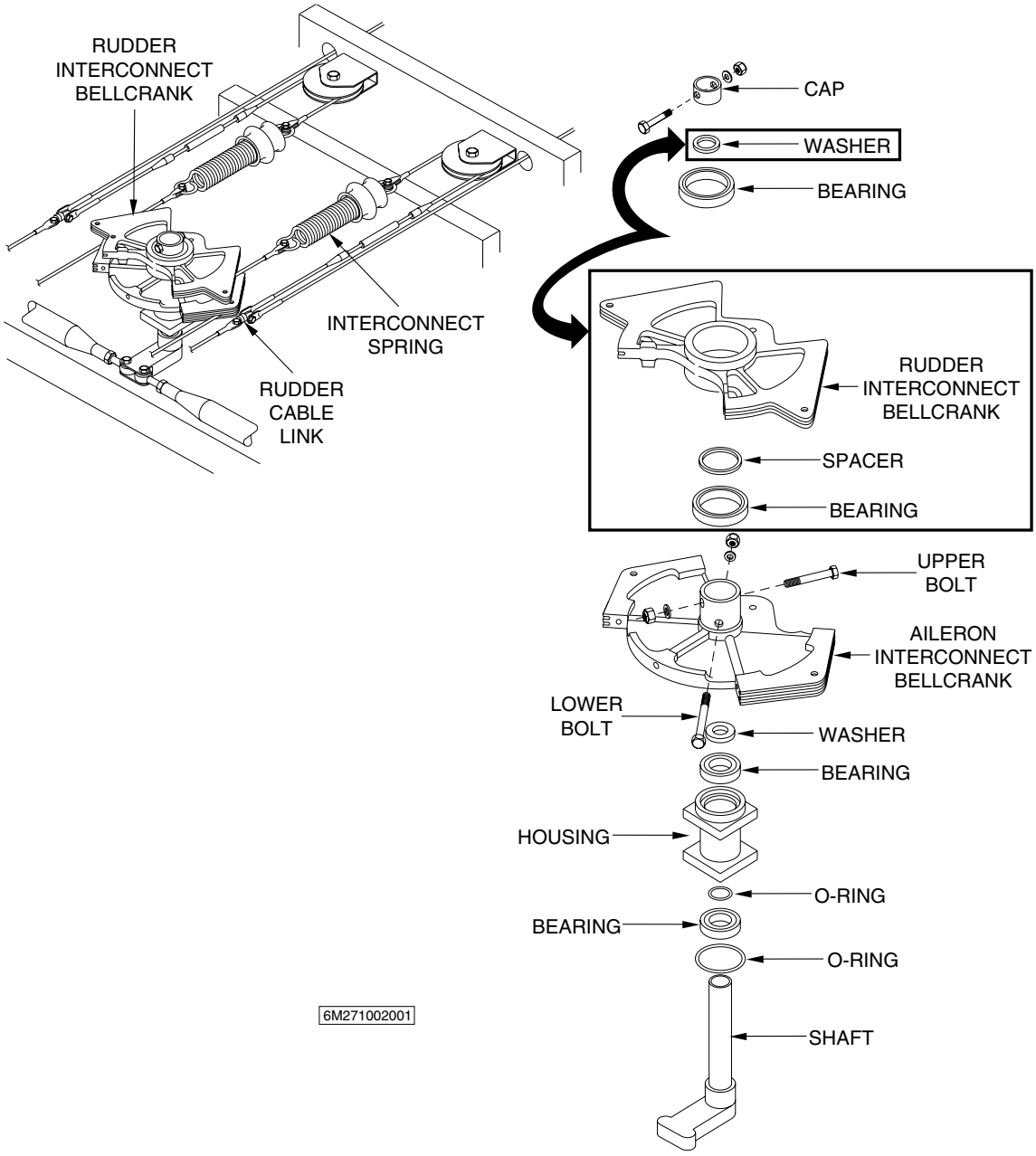
Aileron Drum
Figure 204

RUDDER/AILERON INTERCONNECT – DESCRIPTION AND OPERATION

1. General

A. A bellcrank (rudder interconnect bellcrank) mounted above the aileron drum, is interconnected through a spring assembly to the rudder cables (see Figure 1).

(1) The rudder cables continue on from the spring assembly, under the center aisle floorboards, to the rudder bellcrank located in the tail section, just forward of the aft pressure bulkhead. The spring assembly allows the pilot to physically override the interconnect when cross control is desired.



Rudder/Aileron Interconnect
Figure 1

RUDDER/AILERON INTERCONNECT – MAINTENANCE PRACTICES

1. Removal/Installation – Rudder/Aileron Interconnect

NOTE: The rudder interconnect bellcrank is mounted above the aileron drum and is interconnected through a spring assembly to the rudder cables.

A. Removal – Rudder Interconnect Bellcrank

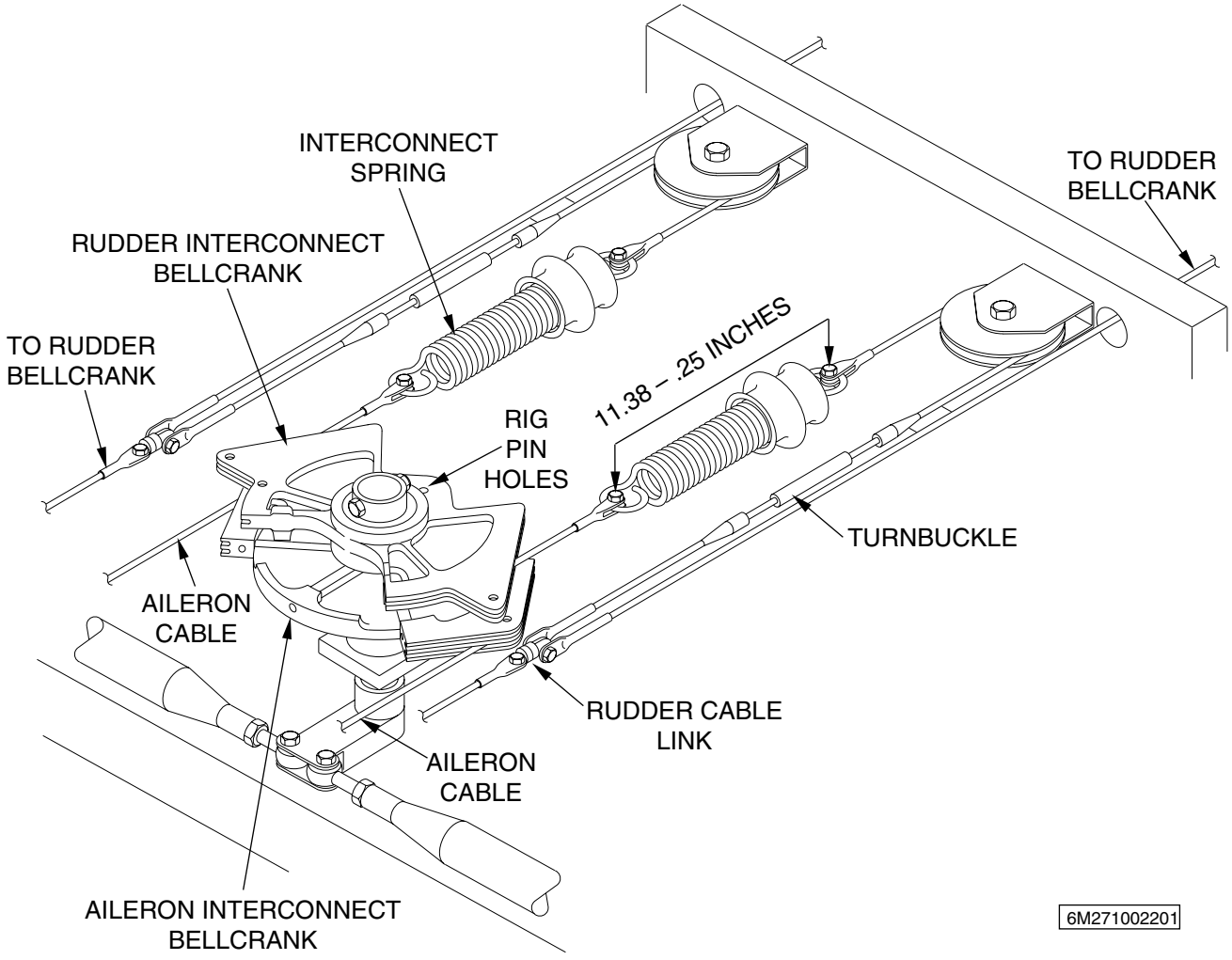
(1) Remove aft center aisle floorboard.

(2) Have assistant cross control FULL LEFT rudder and FULL RIGHT aileron.

NOTE: In this configuration, one of the rudder interconnect springs will stretch and the second spring will relax.

(3) Remove bolt securing cable at forward end of relaxed spring.

(4) Cross controls in the opposite direction and remove bolt securing cable at forward end of remaining spring.



Interconnect Spring Adjust
Figure 201

(5) Disconnect rudder cables from rudder interconnect bellcrank.

(6) Remove bolt, nut, and washer securing cap to aileron shaft.

(7) Remove cap and washer from aileron shaft.

(8) Drive out retaining pins securing rudder interconnect bellcrank to aileron shaft.

(9) Lift rudder interconnect bellcrank off of aileron shaft.

B. Installation – Rudder Interconnect Bellcrank

(1) Position rudder interconnect bellcrank onto aileron shaft. Orient bellcrank with rig pin hole aft.

(2) Secure bellcrank to aileron shaft with pins.

(3) Install cap onto aileron shaft and secure with bolt, washer, and nut.

(4) Attach short cables to rudder interconnect bellcrank and secure with retainers, bolts, washers, and nuts.

(5) Cross control FULL LEFT rudder and FULL RIGHT aileron.

(6) Stretch spring and secure interconnect cable clevis to spring eye with bolt and nut.

(7) Cross controls in the opposite direction and secure interconnect cable clevis to forward end of remaining spring.

(8) Check interconnect spring length is 11.38, ± 0.25 inches (289, ± 6.35 mm), as measured from center of attaching bolts at each end of spring. Spring length is measured with rudder and aileron in locked neutral position with 27-71040 drum - rudder aileron positioned so that 3/16 rig pin holes aligns with rig pin hole in 27-71046 drum aileron.

NOTE: After establishing each spring length, adjust aft rudder cable tension. Continue to alternate between spring length and cable tension until both the cable tension and spring length are within specified values. Subsequent tensioning of aft cables will cause tightening of rudder forward cables.

(9) Adjust aileron cables at turnbuckles IAW 27-00-00.

(10) Functional check aileron operation.

(11) Functional check rudder operation.

(12) Install aft center aisle floorboard.

C. Removal – Aileron Drum

NOTE: Rudder/aileron interconnect must be removed prior to removal of aileron drum.

(1) Remove rudder/aileron interconnect bellcrank.

(2) Engage aileron gust lock or secure control column with seat belt extension.

(3) Tape ailerons in streamlined position.

(4) Remove flap walking beam access panel.

(5) Disconnect both left and right aileron control tubes from aileron bellcrank.

(6) Remove aft center aisle floorboards (T) or center aisle floorboard between F.S. 311 and F.S. 322 (AT, TC).

(7) Block aileron cables at or FWD of F.S. 311.

(8) Disconnect aileron cables from aileron drum.

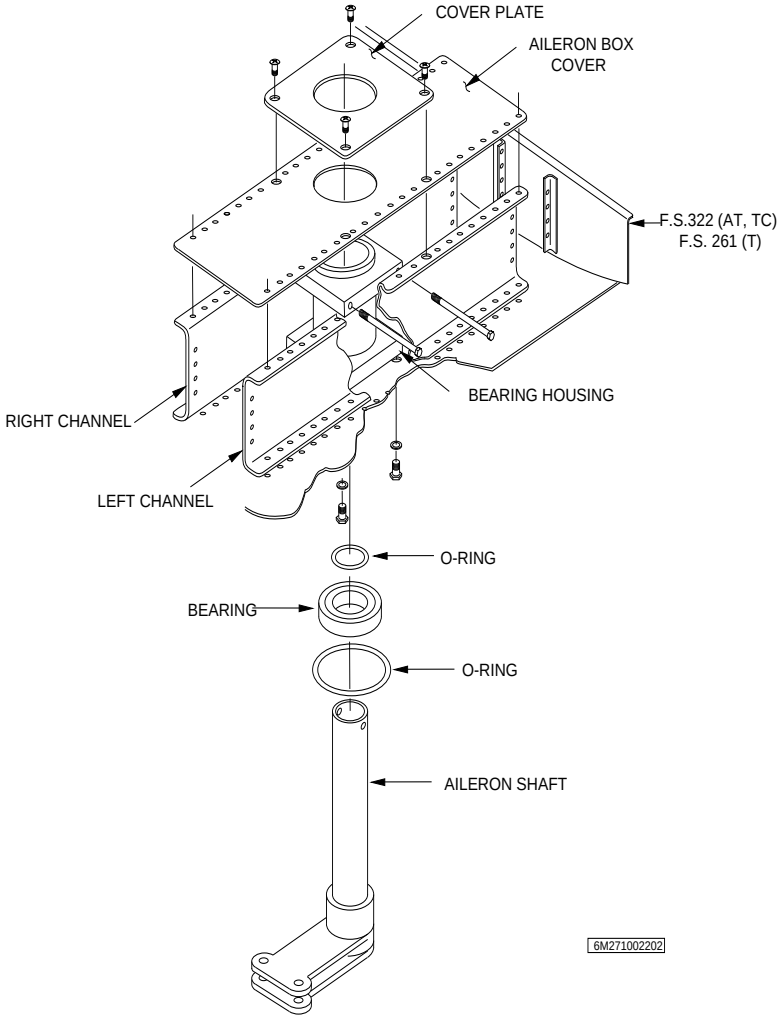
(9) Remove upper and lower bolts, washers, and nuts securing aileron drum to shaft and lift aileron drum off of shaft.

NOTE: Identify upper and lower attaching hardware for installation.

(10) Remove washer from bearing housing.

(11) Remove aileron shaft and bellcrank from bearing housing through bottom of the fuselage.

NOTE: Removal of bearing housing requires disassembly of aileron control box.



Aileron Control Box Assembly
Figure 202

D. Installation – Aileron Drum

(1) Position aileron shaft and bellcrank through bearing housing from the bottom of the fuselage. Orient bellcrank forward.

(2) Install washer into bearing housing.

CAUTION: INSTALL BOLTS WITH HEAD UP; ENSURE MINIMUM OF 0.25 INCH (6.35 MM) CLEARANCE BETWEEN BOLT TIP AND ACCESS PANEL.

IF 0.25 INCH (6.35 MM) CLEARANCE BETWEEN BOLT TIP AND ACCESS PANEL CANNOT BE ACCOMPLISHED, INSTALL BOLT WITH HEAD DOWN. ENSURE THAT COTTER PIN IS INSTALLED.

(3) Connect aileron control tubes to aileron bellcrank with bolts, washers, and nuts. Secure nuts with cotter key.

(4) Position aileron drum onto aileron shaft with rig pin aft. Secure drum with upper and lower bolts, washers, and nuts.

(5) Install rudder interconnect bellcrank.

(6) Install washer and cap on shaft and secure with bolt, nut, and washer.

(7) Connect aileron cables to aileron drum.

(8) Remove cable block from aileron cables.

(9) Check and adjust aileron cable tension.

(10) Conduct a functional check of the aileron system.

(11) Install center aisle floorboards.

E. Removal – Aileron Bellcrank Shaft

(1) Remove rudder/aileron interconnect drum.

(2) Remove aileron drum.

(3) Remove aileron bellcrank shaft.

(4) Remove aileron control stop block.

(5) Remove gust lock plunger assembly.

(6) Remove bolts securing bearing housing to fuselage skin.

(7) Remove bearing housing plate.

(8) Drill out rivets securing cover to channels and remove cover.

(9) Remove through bolts securing bearing housing to left and right channels.

(10) Free bearing housing from fuselage skin.

NOTE: Bearing housing is installed with wet sealer. Breaking the seal may be accomplished by rocking the bearing housing in a FWD and aft direction.

F. Installation – Aileron Bellcrank Shaft

(1) Clean old sealant from fuselage surface where bearing housing mates.

(2) Apply a coat of sealer (PR-1422 or equivalent) to fuselage where bearing housing mates.

(3) Secure bearing housing to left and right channel with bolts, washers, and nuts.

(4) Secure bearing housing to fuselage skin and doubler with bolts and washers. Safety wire bolts in pairs.

(5) Install aileron control box cover and secure with rivets.

(6) Position bearing housing plate over bearing housing and secure to left and right channels with screws, washers, and nuts.

(7) Install gust lock plunger assembly.

(8) Install aileron control stop block.

(9) Install aileron bellcrank shaft. Orient bellcrank forward.

(10) Install aileron drum.

(11) Install rudder/aileron interconnect drum.

(12) Conduct an operational check of the aileron system.

2. Approved Repairs – Rudder/Aileron Interconnect

A. Out of round bolt holes in the aileron shaft may be enlarged and oversized bolts installed as follows:

(1) Remove bolts, washers and nuts securing interconnect bellcrank to shaft. Discard old bolts; retain washers and nuts for installation of oversize bolts.

CAUTION: IF YOU SUSPECT REAMING WILL WEAKEN STRUCTURE, CONSULT FAIRCHILD SERVICE ENGINEERING BEFORE PROCEEDING WITH REPLACEMENT. IF IN DOUBT, REPLACE INTERCONNECT BELLCRANK.

(2) For upper bolt, install either a 0.016 inch oversize bolt, (ream size 0.203 to 0.204 inch), or a 0.031 inch oversize bolt (ream size 0.218 to 0.219 inch).

(3) Secure bolt with original washer and nut.

(4) For lower bolt, install either a 0.016 inch oversize bolt, (ream size 0.265 to 0.266 inch), or a 0.031 inch oversize bolt, (ream size 0.281 to 0.282 inch).

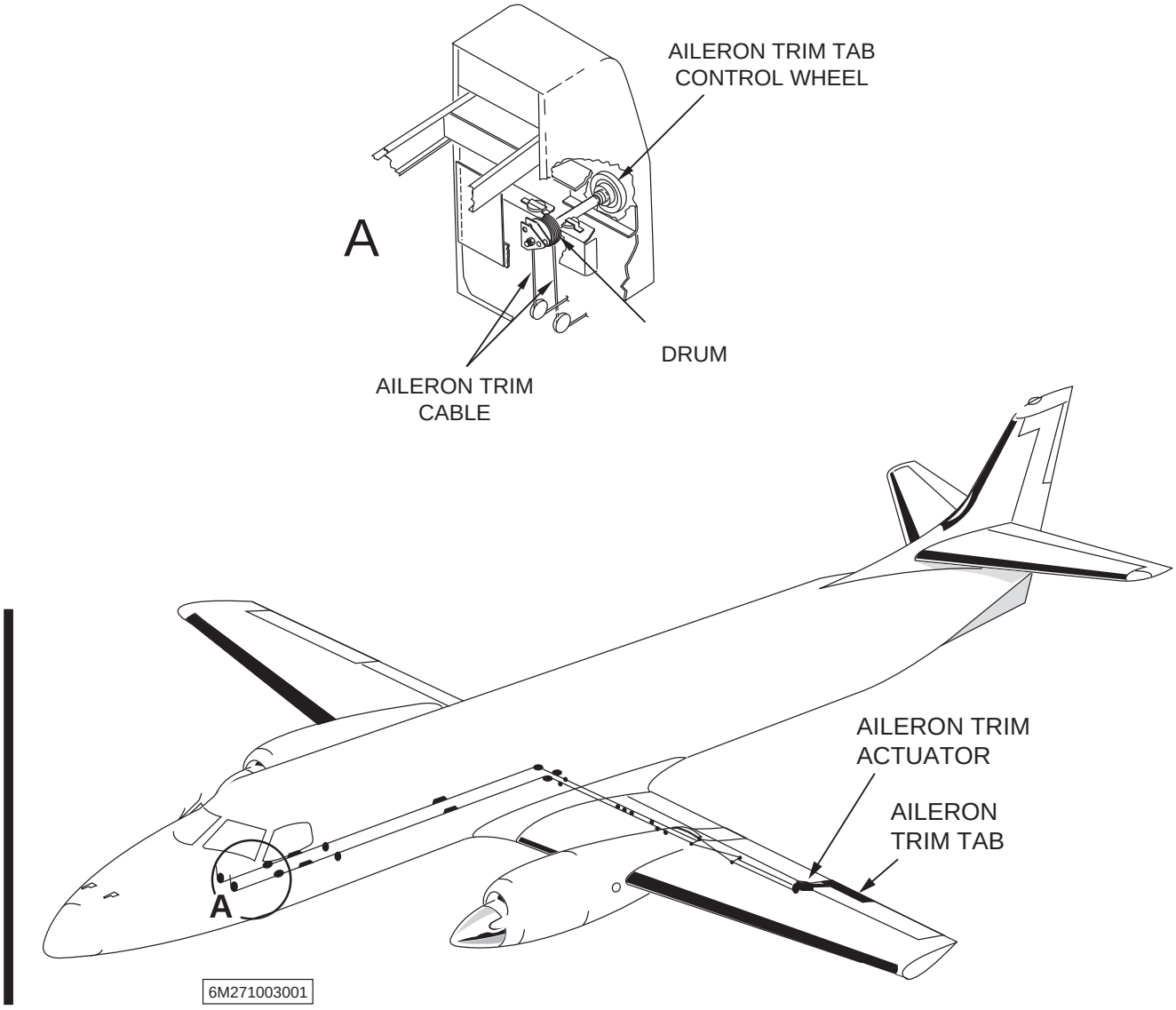
(5) Secure bolt with original washer and nut.

MAINTENANCE MANUAL

AILERON TRIM TAB – DESCRIPTION AND OPERATION

1. General

A. A cable operated trim tab is incorporated into the inboard portion of the left aileron. It is controlled by a trim tab wheel on the pedestal through a cable system which actuates the trim actuators mounted on the wing trailing edge. The actuators move a push-pull rod through the aileron to the tab.



Aileron Trim Tab
Figure 1

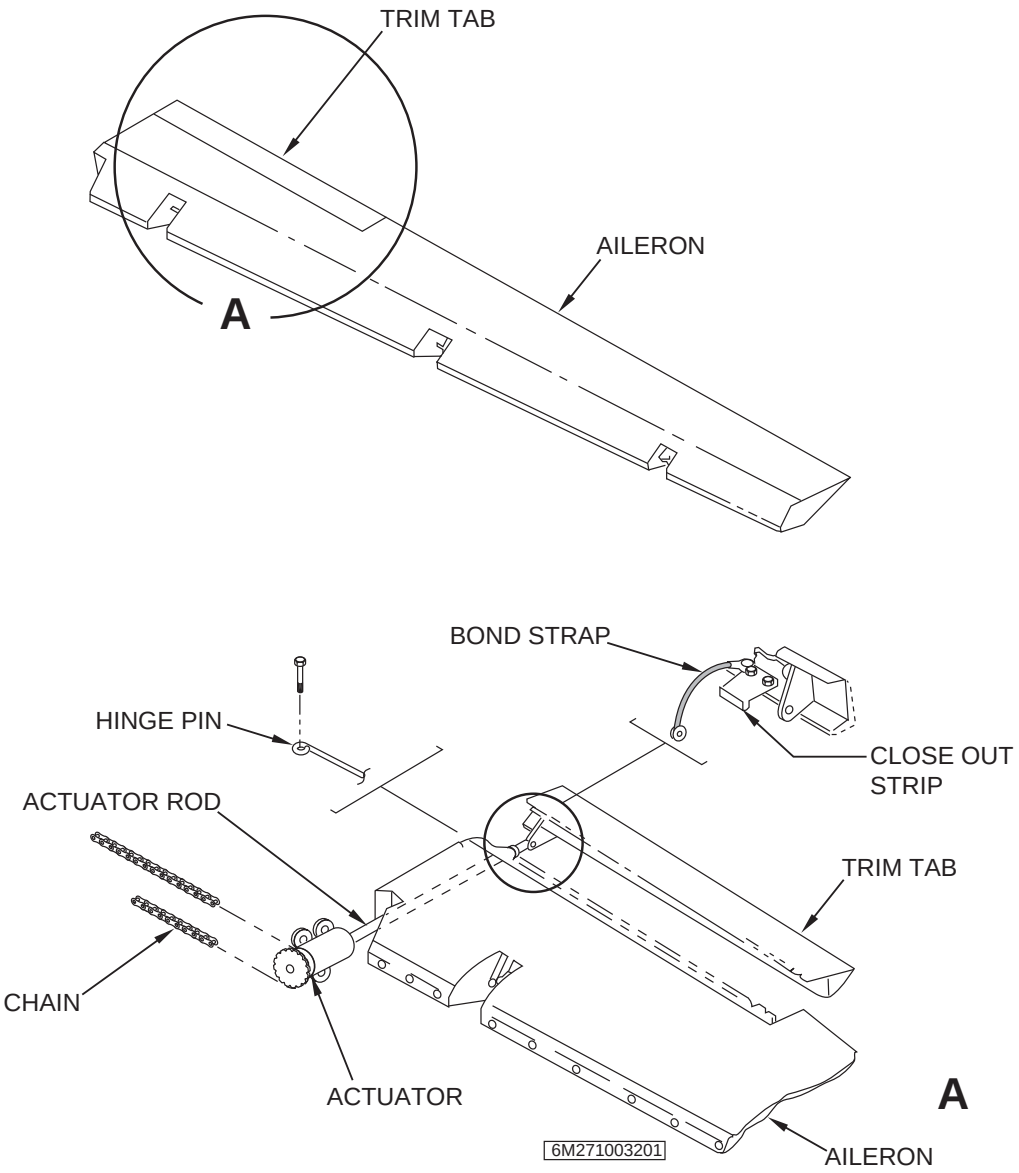
1. Removal / Installation - Aileron Trim Tab

A. Removal - Aileron Trim Tab (Figure 201)

(1) Remove two screws (one screw attaching bond strap) securing close-out strip and remove strip.

(2) Disconnect trim tab actuator rod (located at inboard end of tab).

(3) Remove screw, nut and washer securing hinge pin.



Aileron Trim Tab
Figure 201

(4) Remove hinge pin.

(5) Remove trim tab from aileron.

B. Installation - Aileron Trim Tab (Figure 201)

(1) Position trim tab.

(2) Insert hinge pin.

(3) Install screw, nut and washer to looped end of hinge pin.

(4) Connect actuator rod and cotter pin.

(5) Replace close-out strip and bond strap end.

(6) Check trim tab through full travel for proper operation.

(7) Check trim tab for excessive freeplay and proper travel IAW Section 27-00-00.

C. Removal - Trim Tab Actuator (Figure 202)

(1) Place trim tab in neutral position.

(2) Disconnect actuator push-pull rod from trim tab (located at inboard end of trim tab).

(3) Remove aileron (see Section 27-10-10).

(4) Lower flaps.

(5) Remove floorboard at main spar.

(6) Relieve tension of trim tab cables at turnbuckles and block cables. Mark cables for reference point when reinstalling.

(7) Mark chain link (for reference point when reinstalling) and remove chain from actuator sprocket.

(8) Remove three mounting bolts securing actuator and remove actuator.

(9) Remove push-pull rod from actuator.

D. Installation - Trim Tab Actuator (Figure 202)

(1) Prior to installing actuator, check to ensure full up and full down travel of tab will be achieved.

(a) Turn sprocket until shaft is fully retracted into housing.

(b) Measure length of exposed shaft.

(c) Turn sprocket until shaft is fully extended.

(d) Measure length of shaft.

(e) Subtract retracted shaft length from extended shaft length and divide by 2.

(f) Retract shaft, then extend shaft to length obtained in step (e).

(2) Attach push-pull rod to actuator.

(3) Position actuator and secure with three bolts.

(4) Center chain on sprocket using marked link as reference.

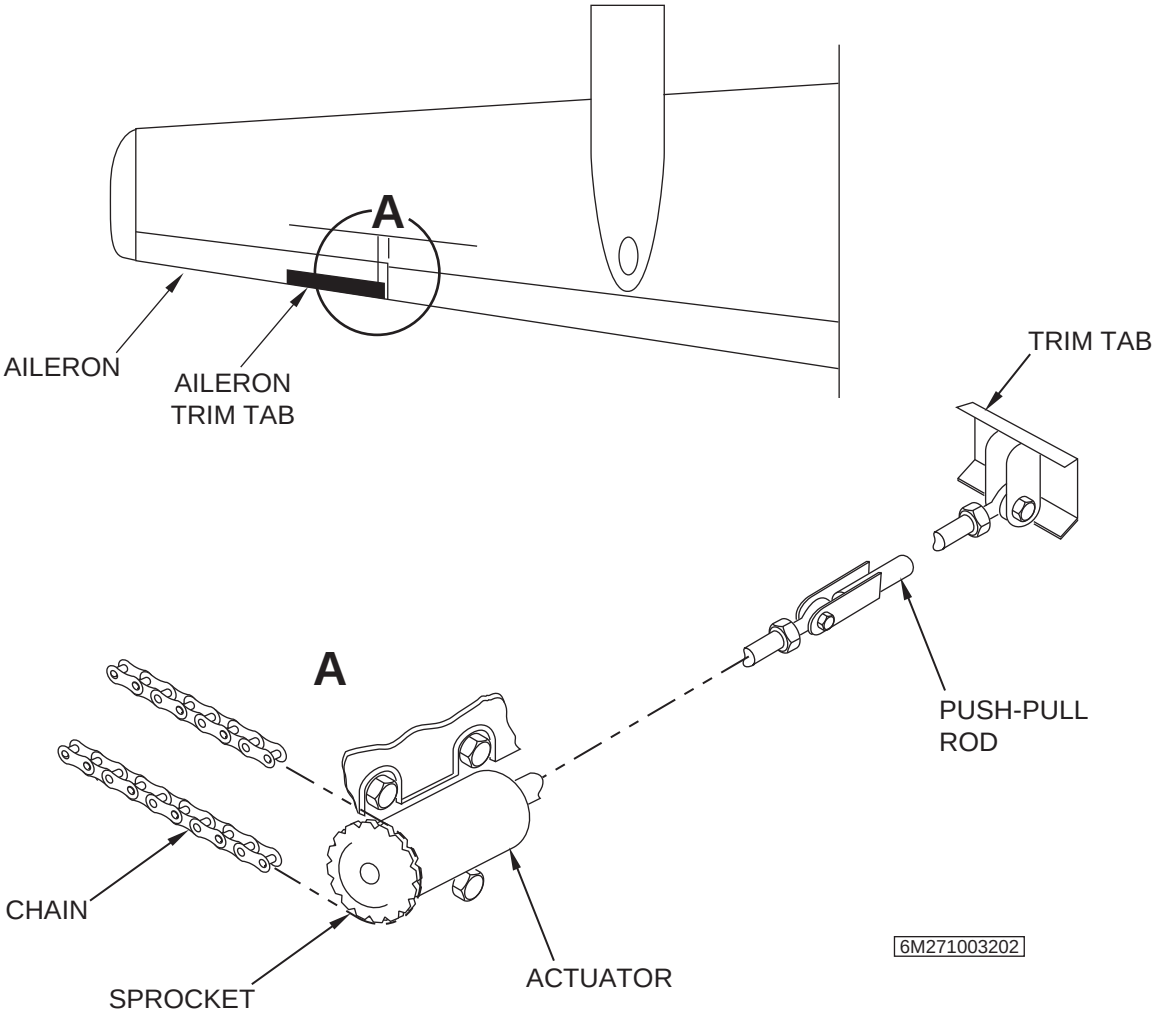
(5) Install aileron (see Chapter 27-10-01). Ensure push-pull rod is positioned through opening in aileron.

(6) Attach push-pull rod to aileron trim tab.

(7) Attach aileron trim tab cables to turnbuckles and adjust tension IAW Section 27-00-00.

(8) Check trim tab for excessive freeplay and proper rigging IAW Section 27-00-00.

(9) Replace floorboard.



Aileron Tab Actuator
Figure 202

2. Adjustment/Test Aileron Trim Tab

A. Operational Test - Aileron Trim Tab

(1) On pedestal, set aileron trim tab wheel so line on indicator knob and control wheel are aligned. Verify trim tabs are streamlined.

(2) Rotate trim wheel full right (clockwise). Observe that trim tab comes up.

(3) Align indicating lines on wheel. Observe that trim tabs are streamlined.

(4) Rotate trim wheel full left (counterclockwise). Observe that trim tab goes down.

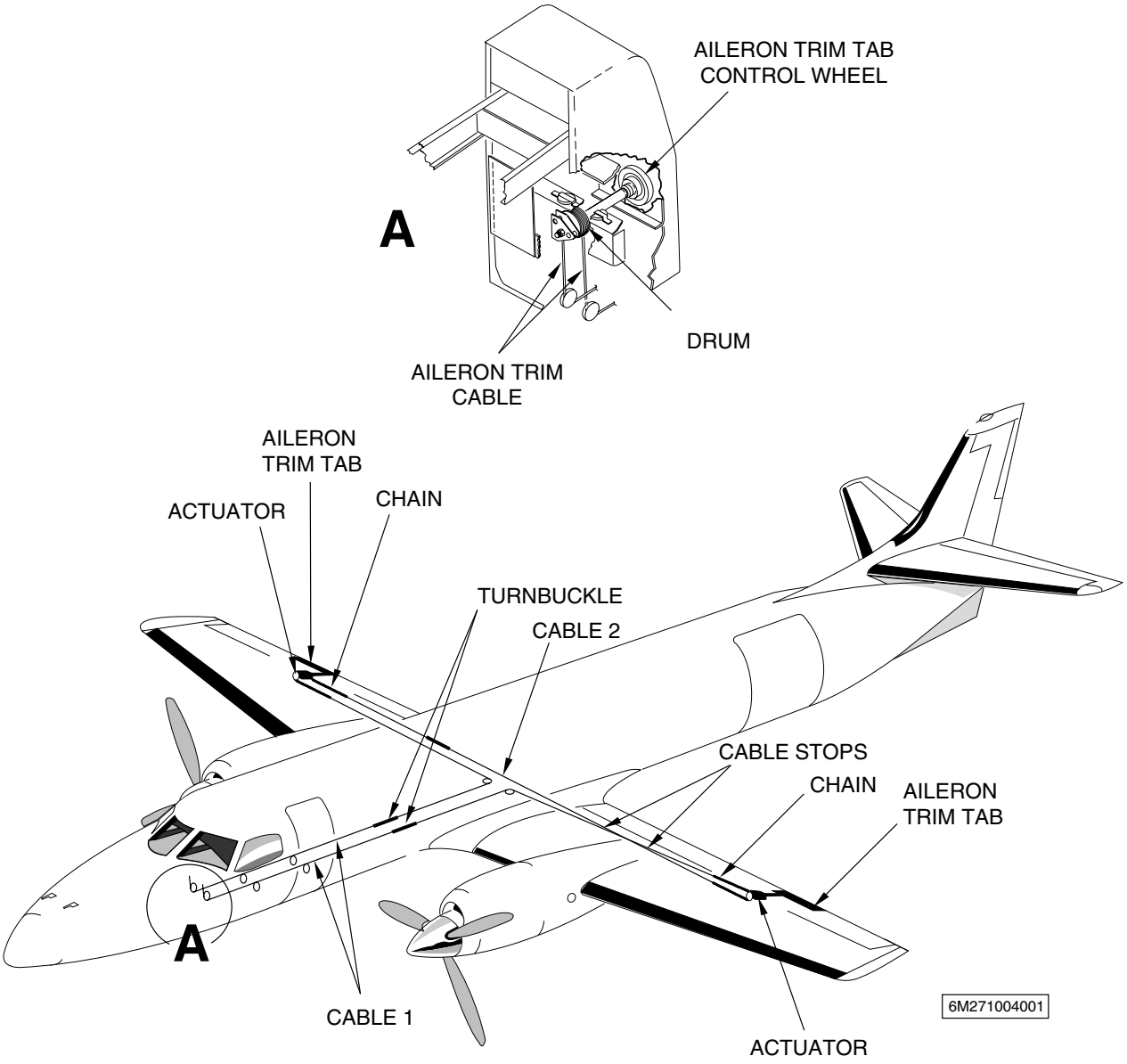
(5) Align indicating lines on wheel. Observe that trim tabs are streamlined.

AILERON TRIM TAB CABLES – DESCRIPTION AND OPERATION

1. General

A. The aileron trim tab cable system is comprised of three cables which are joined together by two turnbuckles and a drive chain. The cable system is divided into two sections.

(1) The first section consists of one cable. The cable runs from a turnbuckle located at the wing rear spar, under the center aisle floorboards, through the aileron trim mechanism in the pedestal, and back along the center aisle to terminate at another turnbuckle at the wing rear spar (see Figure 1, cable 1).



Aileron Trim Tab Cables
Figure 1

(2) The next section of cables (2) continues from the turnbuckles, travels through pressure seals at the fuselage, along the trailing edge of the wing where it interconnects with a chain that drives the trim tab actuator. Aileron trim cable stops are attached to cable 2 and are located between the left flap and the wing rear spar.

AILERON TRIM TAB CABLES – MAINTENANCE PRACTICES

1. Removal and Installation – Aileron Trim Tab Cables

A. Removal – Cable Section 1

(1) Ensure battery switches are OFF.

(2) Place aileron trim in neutral position.

(3) Remove pilot and copilot seats IAW Chapter 25.

(4) Remove crew compartment floorboards at control column.

(5) Remove cabin center aisle floorboards to wing rear spar.

(6) Locate aileron trim cable ends at turnbuckles (just forward of rudder / aileron interconnect) and block cables aft of turnbuckles.

(7) Disconnect cable ends at turnbuckles.

(8) Remove center overlay/edge lighted panel from pedestal.

(9) Remove center pedestal panel.

(10) Remove auxiliary trim switch from panel.

(11) Support cover out of way taking stress off of wires.

(12) Remove trim control gear cover from bracket (see Figure 2, Section 27-10-05).

(13) Remove locknut from rear of indicator shaft.

(14) Unscrew indicator and pull rod out through shaft.

(15) Remove nut securing wheel on shaft and remove wheel from shaft.

(16) Remove woodruff key from shaft.

(17) Remove snap ring from shaft.

(18) Remove lower panel from pedestal.

(19) Remove two rudder trim pulley guards (or pulleys if necessary) and position rudder trim cables off of bracket assembly.

(20) Remove six screws (three from each side of pedestal) securing bracket assembly to pedestal.

(21) Remove bracket assembly with shaft, gears, and cable drum through center panel access.

(22) Remove cotter pins and washers from clevis pins (cable keepers).

(23) Remove channel (in front of drum) and slide off of shaft.

(24) Remove snap ring securing drum to shaft and remove drum from shaft.

(25) Unwind cable from drum.

(26) Remove pin securing cable in drum and remove cable from drum.

(27) Remove cable through pulleys and fairleads as necessary to remove cable from aircraft.

B. Installation – Cable Section 1

(1) Locate center of cable being installed.

(2) Insert cable onto drum and secure with pin.

(3) Wind cable onto drum starting each side of cable simultaneously and winding in opposite directions. Cable is to be wound inward from each side and terminate at center of drum (approximately two and one-half turns). Drum is fully wound when cables meet and woodruff key slot in drum is at opposite end.

(4) Slide drum onto shaft aligning slot in drum onto woodruff key on shaft. Ensure woodruff key is at 12 o'clock position and cable ends are extended down.

(5) Insert snap ring into groove in shaft securing drum onto shaft.

(6) Slide channel onto shaft, through clevis pins, narrow end down. Ensure cables ends are on each side of base of channel and secure channel to bracket.

(7) Insert washers and cotter pins onto clevis pins.

(8) Position bracket assembly inside pedestal and secure to pedestal with six screws.

(9) Position cable ends inside pedestal and remove cable block.

(10) Position rudder trim cables onto pulleys on top of bracket assembly. Both cables run along left side of pulleys.

(11) Replace lower panel onto pedestal.

(12) Replace snap ring onto shaft.

(13) Insert woodruff key onto shaft.

(14) Insert wheel onto shaft and secure with nut.

(15) Insert indicator rod through shaft and screw down until flush with wheel.

(16) Replace nut onto rear of shaft.

(17) Replace gear cover, if applicable.

(18) Position cable ends through pulleys and fairleads and attach ends to turnbuckles.

(19) Take up slack in cable at turnbuckles and remove cable blocks aft of turnbuckles.

(20) Work trim wheel to test cable for smoothness of operation.

(21) Adjust cable tension. See charts, Section 27-00-00.

(22) Replace auxiliary trim switch onto panel.

(23) Replace center pedestal panel.

(24) Replace center overlay/edge-lighted panel onto panel.

(25) Replace center aisle floorboards.

(26) Replace crew compartment floorboards.

(27) Replace pilot and copilot seats.

(28) Check trim tabs for proper travel.

C. Removal – Cable 2

(1) Place trim tabs in neutral position.

(2) Remove cabin center aisle floorboards at wing rear spar (rudder / aileron interconnect).

(3) Remove floorboard aft of wing rear spar.

(4) Remove left flap walking beam access panel.

(5) Remove left fuselage to wing access plate aft of wing rear spar.

(6) Remove left aileron (see 27-10-01).

(7) Lower left flap.

(8) Disconnect cable at turnbuckle located at wing rear spar.

(9) Remove cable stop from cable being removed.

(10) Remove chain from sprocket of trim tab actuator.

(11) Disconnect cable from chain at master link.

(12) Remove pressure seal at fuselage.

(13) Remove pulley guards and fairleads as necessary and remove cable. Note exact travel of cable for proper reinstallation.

D. Installation – Cable 2

(1) Position cable along center aisle and through fairleads and pulley guards as necessary.

(2) Reinstall pressure seal around cable and into fuselage.

(3) Connect cable to chain.

(4) Attach chain to actuator sprocket and take up slack on cable from inside.

(5) Attach cable to turnbuckle.

(6) Check travel of cable.

(7) Check cable stop and adjust as necessary.

(8) Replace aileron.

(9) Check travel of trim tab. Zero trim tab by adjusting actuator push-pull rod.

(10) Replace fuselage to wing access plate.

(110 Replace flap walking beam access panel.

(12) Replace floorboards.

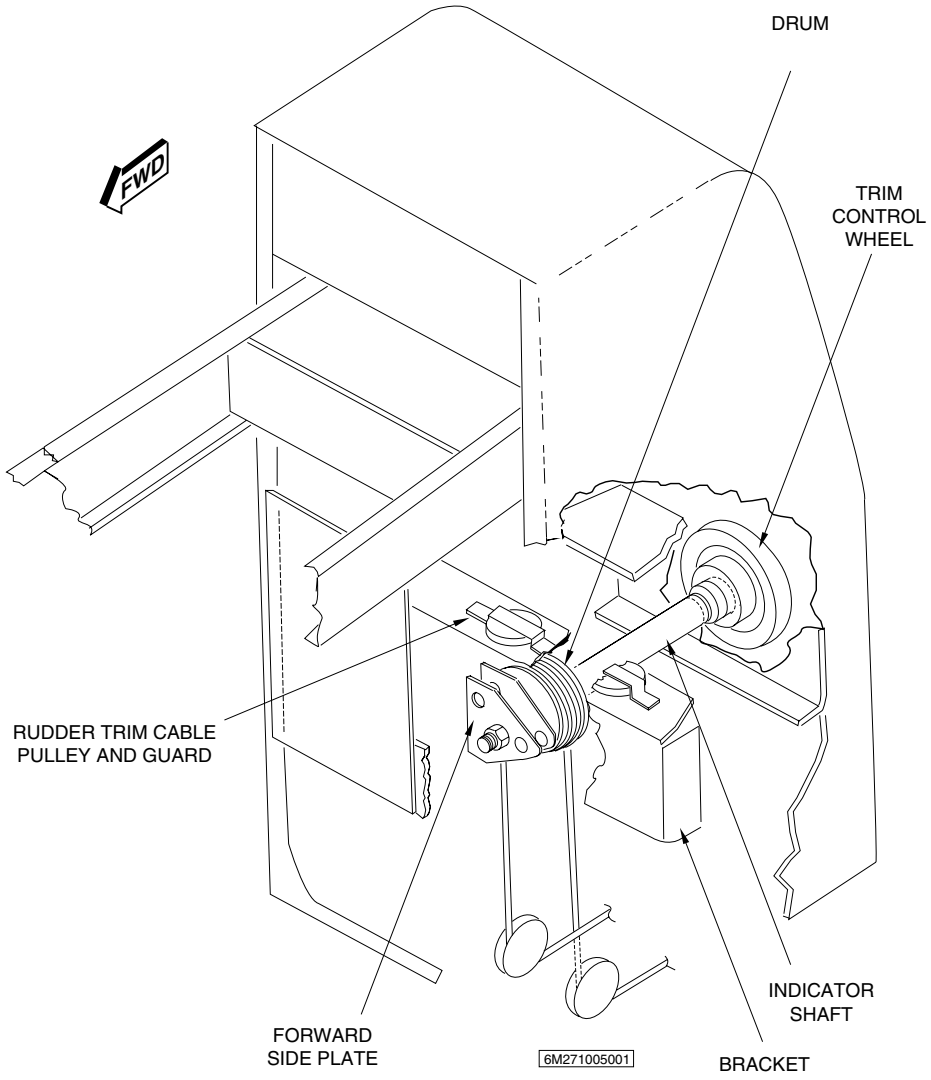
(13) Check trim tab for proper travel.

AILERON TRIM TAB CONTROL – DESCRIPTION AND OPERATION

1. General

A. The aileron trim control is located on the lower front panel of the cockpit pedestal. The control consists of an outer knob and an inner wheel. The outer knob shows left and right direction and has an index line. The inner wheel also has an index line. When the line on the knob is positioned straight up (12 o'clock position) and aligned with the index line on the inner wheel, the aileron trim is in the neutral position.

B. By actuating the aileron trim control knob, cable movement rotates the sprocket, which actuates a push-pull rod through the aileron to the tab, deflecting it from neutral to the desired direction. Turning the trim knob to the right (tab up) will trim the aileron so that the right wing drops and the airplane banks right. Likewise, when the knob is turned left (tab down) the left wing drops and the airplane banks left. Gears within the control mechanism enable the outer knob to make a complete revolution while the inner wheel moves only a few degrees. Therefore, if the outer knob index line is at 12 o'clock and the inner wheel index line is at 11 o'clock or 1 o'clock, the aileron trim tab is not in the neutral position.



Aileron Trim Tab Control
Figure 1

AILERON TRIM TAB CONTROL – MAINTENANCE PRACTICES

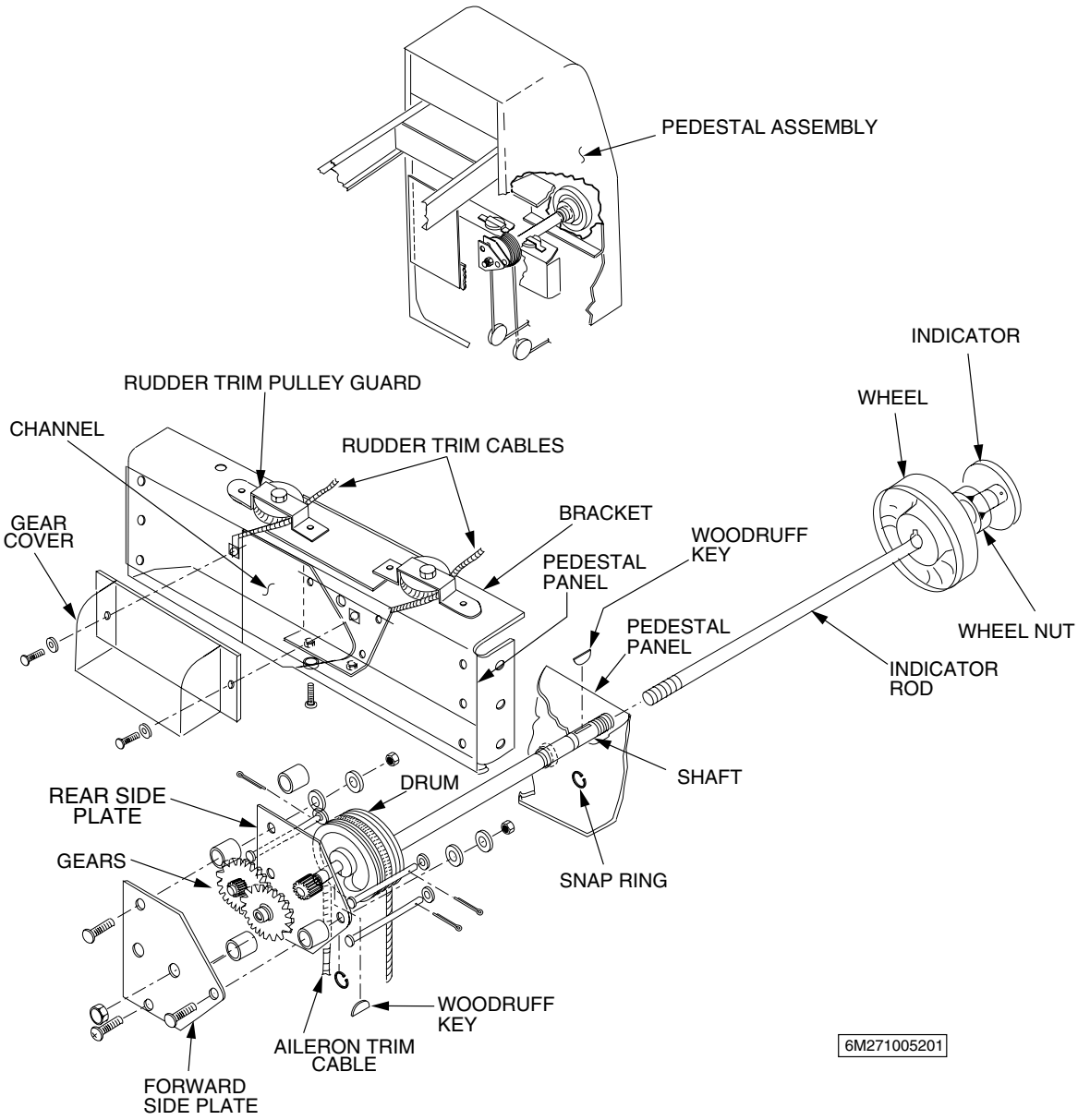
1. Removal / Installation – Aileron Trim Tab Control

A. Removal – Control Wheel Assembly

(1) Ensure battery switches OFF.

(2) Place aileron trim in neutral position.

(3) Remove cabin center aisle floorboard just forward of rudder/ aileron interconnect (at wing rear spar).



Aileron Trim Tab Control
Wheel Assembly
Figure 201

(4) Locate aileron trim cable ends at turnbuckles and block cables aft of turnbuckles.

(5) Accomplish steps (7) through (21) of REMOVAL – CABLE 1, Section 27-10-04.

(6) Remove trim control gears as necessary (see Figure 201).

(7) For removal of cable drum, see REMOVAL – CABLE 1, Section 27-10-04.

B. Installation – Control Wheel Assembly

(1) Position bracket assembly inside pedestal and secure to pedestal with six screws.

(2) Accomplish steps (10) through (25) of INSTALLATION – CABLE 1, Section 27-10-04.

1. Adjustment / Test – Aileron Trim Tab Control

A. Adjustment – Control Wheel Indicator

NOTE: When the aileron trim tab is streamlined with the aileron and the marks on the trim indicator and wheel on the pedestal do not align, the indicator may be realigned as follows:

(1) Place trim tab in streamlined position.

(2) Locate indicator locknut behind pedestal and loosen locknut (see Figure 201).

(3) Position indicator to align with mark on wheel.

(4) Tighten locknut. Do not over torque.

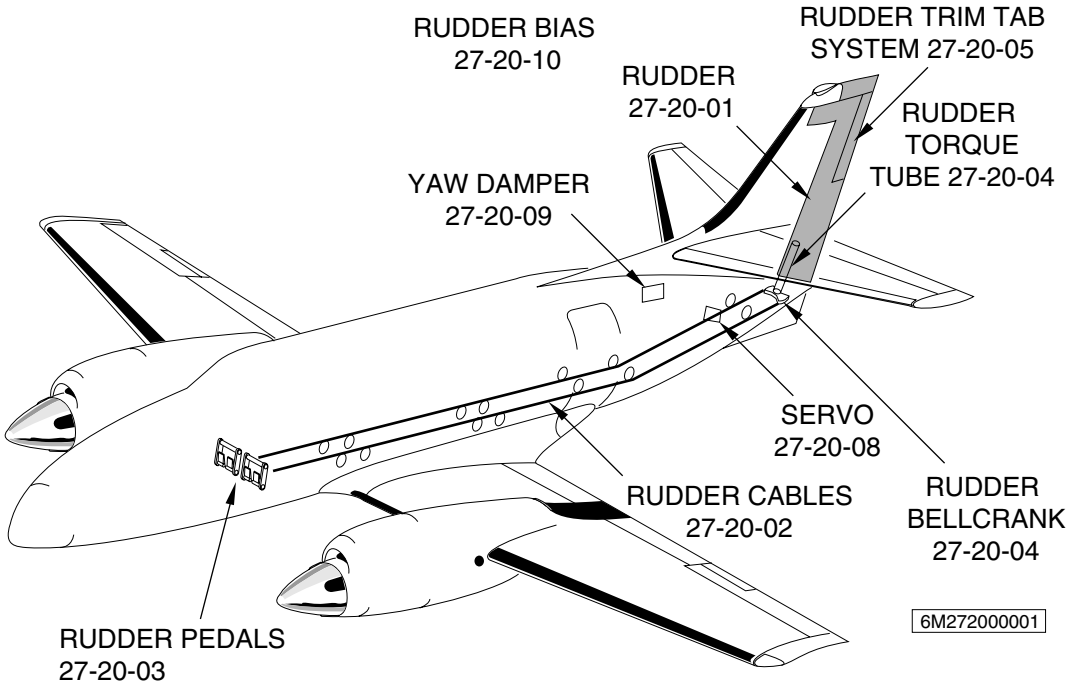
RUDDER – DESCRIPTION AND OPERATION

1. General

A. The rudder, manufactured of aluminum skin riveted over a frame of spars and ribs is attached to the vertical stabilizer at three hinge points. A hinged metal trim tab is attached to the trailing edge of the rudder. The rudder trim is controlled by a trim wheel on the pedestal.

B. The rudder is controlled by cables from the cockpit. Pilot and copilot rudder pedals, interconnected by push-pull rods and bellcranks, actuate the cables from the cockpit to the rudder bellcrank, located just forward of the aft pressure bulkhead. The bellcrank is attached to the rudder torque tube which passes through the pressure bulkhead. The torque tube rotates within the pressure vessel. On earlier models, T201-257, AT001-042, TC201-221, the rudder torque tube is sealed at the fuselage by a hose and clamp arrangement. On later models, T258-999, AT043-999, TC222-999, the torque tube is sealed at the fuselage by a pressure seal.

C. All SA226-T aircraft are equipped with a yaw damper system. Later model SA226-T aircraft incorporate a rudder bias system. The yaw damper / rudder bias system uses a single electro mechanical servo with two modes of operation: the yaw damper mode and the rudder bias mode. The yaw damper mode of operation is controlled by a signal from the pilot turn and bank indicator. This signal is processed by the yaw damper amplifier which controls the servo linked to the rudder control system. The rudder bias mode becomes activated by low engine torque pressure when the power levers are in the advanced position. The rudder bias system which is the master system, provides a rudder force input of approximately 40 pounds in the event of an engine failure. A mechanical clutch in the drive linkage provides override capability in the event of a system malfunction.



Rudder Cables
Figure 1

1. Removal / Installation – Rudder

A. Removal – Rudder (Reference Section 27-20-04, Figure 1)

(1) Engage gust lock and push rudder pedals until rudder locks.

(2) Remove cap on vertical stabilizer.

(3) Disconnect electrical connection to rotating beacon.

(4) Remove left and right rudder fairings.

(5) Detach rubber pressure seals from rudder.

(6) Remove tail cone. If applicable, remove JATO bottle and rack (see Chapter 71-00-01).

(7) Remove bolts securing aft pressure bulkhead and tilt bulkhead aft.

(8) Remove and cap lines to outflow valve.

(9) Separate handshake disconnects in tail light wiring and remove bulkhead. For JATO equipped aircraft, see Chapter 71-00-01.

(10) Remove bolts securing rudder control lock to rudder torque tube assembly.

(11) For aircraft T201-257, AT001-042, TC201-221:

(a) Loosen rudder tube seal hose clamps.

(12) For aircraft T258-999, AT043-999, TC222-999:

(a) Remove screws attaching rudder torque tube seal.

(b) Break seal loose and slide it down torque tube.

(c) Separate seal by unscrewing and separating two halves and remove seal, if necessary.

(13) Disconnect trim tab actuator from push-pull rod at rudder trim tab.

(14) Connect hoisting tool and appropriate crane to rudder. Rudder can be lifted by three people but is rather clumsy because of top heaviness.

(15) Disconnect one end of bond strap at both attaching points.

(16) Disconnect top hinge at rudder side of hinge, and bottom hinge at vertical stabilizer spar side of hinge.

NOTE: Hinge bolts are harder to remove than hinge attachments.

(17) Remove bolts from attaching points.

(18) Place gust lock in UNLOCK position.

(19) Tilt rudder aft to clear top of rudder and lift torque tube out and rudder clear.

NOTE: On aircraft T201-257, AT001-042, TC201-221, be sure to retrieve seal housing and packing located at bottom of rubber hose should it fall out when torque tube is removed.

B. Installation – Rudder

(1) Position rudder at vertical stabilizer.

(2) Insert torque tube into place and tilt rudder forward.

NOTE: On aircraft T201-257, AT001-042, TC201-221, before seating torque tube into rudder control lock, insert seal housing onto torque tube.

On aircraft T258-999, AT043-999, TC222-999, before seating torque tube into rudder control lock assembly, insert O-ring and gasket onto torque tube. Position two shims between control lock weldment and base of torque tube.

(3) Attach rudder at hinges and connect both bonding straps.

(4) Connect rudder trim tab push-pull rod to trim tab actuator.

(5) Aircraft T201-257, AT001-042, TC201-221:

(a) Lubricate seal housing assembly with Parker O-Lube (Barium Base) or equivalent.

(b) Position seal housing into rubber hose. Ensure groove in seal is facing aft.

(c) Secure rubber tube seal hose clamps.

(6) Aircraft T258-999, AT043,999, TC222-999:

(a) If torque tube seal was removed, join both halves of seal and insert O-ring and gasket.

(b) Lubricate seal housing assembly with Parker O-Lube (Barium Base) or equivalent.

(c) Slide seal up torque tube into position and secure with screws.

(7) Secure rudder torque tube assembly to control lock with bolts.

(8) Engage gust lock and push rudder pedals until rudder locks.

(9) Connect handshake connects for tail light.

(10) Position bulkhead in place and secure with original hardware. Longer bolts go in holes that pass through the stiffening channels on aft side of bulkhead.

(11) If applicable, install JATO bottle and rack.

(12) Apply sealant to all places as necessary. Use Coast Pro-Seal 890B2 or equivalent.

(13) Install tailcone.

(14) Connect rubber pressure seals to vertical stabilizer.

(15) Replace left and right rudder fairings.

(16) Connect electrical connection for rotating beacon.

(17) Replace cap on vertical stabilizer.

(18) Place gust lock in UNLOCK position.

2. Inspection / Check – Rudder

A. Inspection – Rudder

(1) Open access panels necessary to gain access to system components.

(2) Operate system through full range of travel and check for unusual noise, freedom of operation, and correct travel of rudder and trim tab.

(3) Inspect cables, cable pulleys, fairleads, and pressure seals (see Section 27-00-00).

(4) Check push-pull rods, bellcranks, and torque tubes.

(5) Check cables for correct tension.

(6) Check trim control in pedestal for damage, proper lubrication, and condition.

(7) Inspect trim actuating mechanism in vertical stabilizer for damage, security of attachment, and lubrication. Check actuator chain master links for security.

(8) Inspect all hinges and hinge brackets for damage, cracks, corrosion, wear, defective bearings, and security.

(9) Inspect rudder and tab structures for damage, cracks, corrosion, loose or missing rivets, cracked or buckled spars, ribs, or stringers, and separation of metal-bonded structure.

RUDDER CONTROL CABLES – DESCRIPTION AND OPERATION

1. General

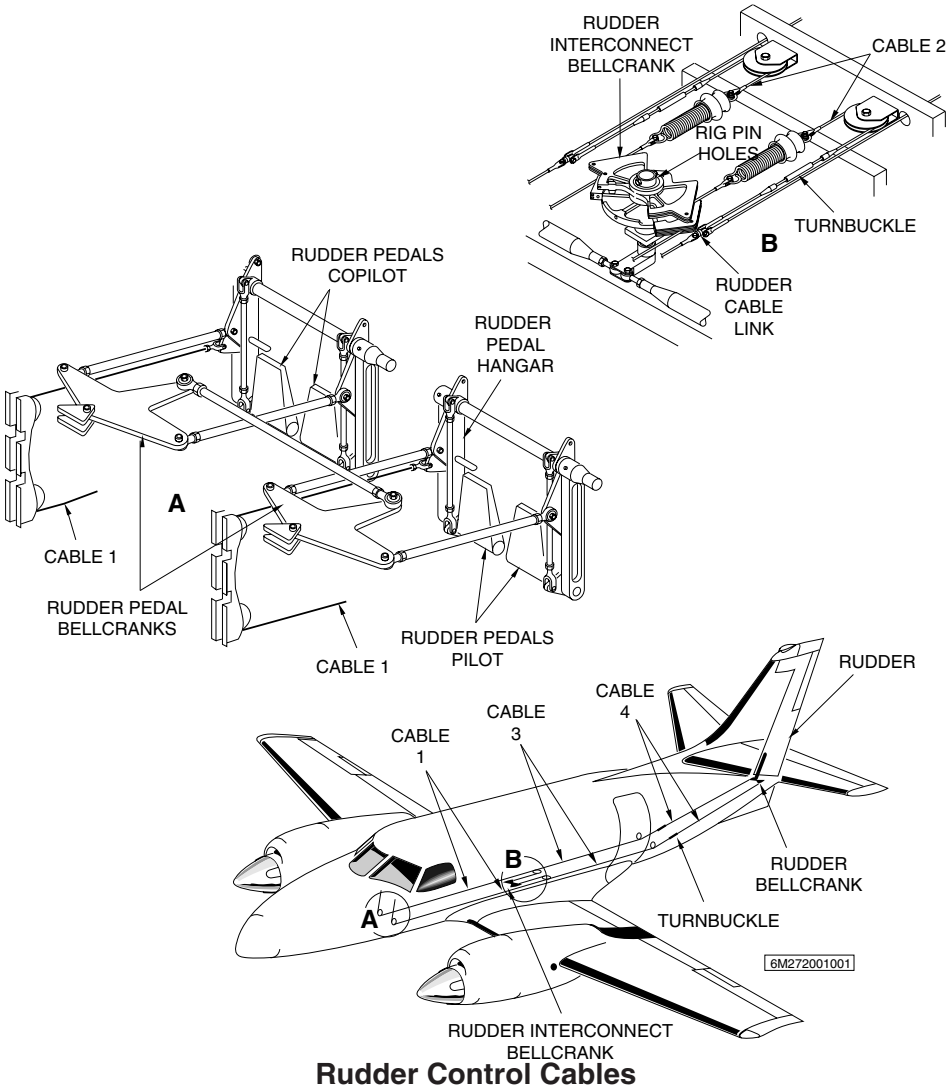
A. The rudder control cable system is comprised of 12 separate cables joined together by a series of links, turnbuckles, and springs to make one continuous cable system. It is divided into four sections with each section having its own separate function.

B. One section of the cables runs from the rudder pedal, under the center floorboards, to a link located adjacent to the rudder interconnect bellcrank (see A and B, Figure 1).

C. The next section of cables (2) branches off from the first section and travels to the rudder interconnect bellcrank. This section of cables consists of six separate cables (three each side) joined by a turnbuckle and a spring and terminates at the rudder interconnect bellcrank.

D. The third section of cables continues from the link adjacent to the interconnect, runs along the floorboard, and terminates at the turnbuckles aft of the interior bulkhead.

E. Section four continues from the turnbuckles aft of the bulkhead and terminates at the rudder bellcrank located forward of the aft pressure bulkhead.



Rudder Control Cables
Figure 1

RUDDER CONTROL CABLES – MAINTENANCE PRACTICES

1. Removal and Installation – Rudder Control Cables

A. Removal – Cable Section 1 (Forward of Rudder Interconnect)

(1) Remove pilot or copilot seat (whichever cable is being removed).

(2) Remove cabin floorboards to rudder / aileron interconnect.

(3) Remove aft cabin bulkhead.

(4) Locate rudder cable (by moving rudder pedals).

(5) Engage gust lock and pull control column back until elevators lock. Work rudder pedals until rudder locks.

(6) Lock rudder pedals by inserting a rod or tube through both rudder pedal shafts.

(7) Insert three-sixteen inch rig pin into rudder and aileron interconnect bellcranks (see view B, Figure 1).

(8) Release tension of rudder cable at turnbuckle located aft of link located at rudder / aileron interconnect.

(9) Unfasten cable from rudder pedal hangar located at rudder pedal assembly.

(10) Unfasten cable at link located at rudder / aileron interconnect.

(11) Remove pulley guards and fairleads as necessary and remove cable.

B. Installation – Cable Section 1

(1) Position cable and replace fairleads and pulley guards.

(2) Attach cable to rudder pedal hangar at rudder pedal.

(3) Attach cable to link at rudder / aileron interconnect.

(4) Take up slack at turnbuckle at rudder / aileron interconnect.

(5) Adjust cable tension using aft turnbuckle (see charts, Section 27-00-00).

(6) Remove rig pin.

(7) Disengage gust lock.

(8) Replace interior bulkhead.

(9) Replace floorboards.

(10) Replace seat.

C. Removal – Cable 2 (At Rudder Interconnect Bellcrank)

(1) Remove floorboard at rudder / aileron interconnect (located aft of rear wing spar).

(2) Engage gust lock and pull control column back until elevators lock. Work rudder pedals until rudder locks.

(3) Insert three-sixteen inch rig pin into rudder and aileron interconnect bellcranks (see view B, Figure 1).

(4) Pull interconnect spring (using looped piece of cable with handle or similar tool) to release tension on spring while removing bolt that fastens cable to aft end of spring.

(5) Individual cables within interconnect may now be removed as necessary.

D. Installation – Cable Section 2

(1) Replace and attach cables as necessary.

(2) Using spring stretching tool, pull spring to cable and attach bolt.

(3) Adjust length of interconnect springs. Length is 11.38 ± .25 inches measuring from center of attaching bolts at each end of spring. Adjust spring length by adjusting turnbuckle at interconnect.

(4) Remove rig pin.

(5) Support control column and disengage gust lock.

(6) Replace floorboard.

E. Removal – Cable Section 3 (Aft of Rudder / Aileron Interconnect)

(1) Engage gust lock and work rudder pedals until rudder locks.

(2) Remove all floorboards from rudder / aileron interconnect (located aft of rear wing spar) to aft interior bulkhead.

(3) Remove aft interior bulkhead.

(4) Insert three-sixteen inch diameter rig pin into rudder and aileron interconnect bellcranks (see view B, Figure 1).

(5) Disconnect cable at link at rudder / aileron interconnect.

(6) Disconnect cable at aft turnbuckle.

(7) Remove pulley guards and fairleads as necessary and remove cable.

F. Installation – Cable Section 3

(1) Position cable and replace fairleads and pulleys as necessary.

(2) Attach cable to link at interconnect.

(3) Attach cable to aft turnbuckle.

(4) Adjust cable tension (see charts, Section 27-00-00).

(5) Remove rig pin.

(6) Disengage gust lock.

(7) Replace interior bulkhead.

(8) Replace floorboards.

G. Removal – Cable Section 4 (Aft of Aft Turnbuckle)

(1) Engage gust lock and work rudder pedals until rudder locks.

(2) Remove aft interior bulkhead.

(3) Locate rudder cables (by removing rudder pedals) and block cables forward of rudder turnbuckle.

(4) Disconnect cable at aft turnbuckle.

(5) Remove tailcone. If applicable, remove JATO bottle and rack (see Chapter 71-00-00).

(6) Remove bolts securing aft pressure bulkhead and tilt bulkhead aft.

(7) Remove and cap lines to outflow valve.

(8) Separate handshake disconnects to tail light.

(9) Swing bulkhead to either left or right side and secure. For JATO equipped aircraft, do not allow bulkhead weight to strain JATO wires.

(10) Disconnect rudder cable at bellcrank and remove cable.

H. Installation – Cable Section 4

(1) Position cable through bulkhead.

(2) Attach rudder cable to bellcrank.

(3) Attach remaining cable end to turnbuckle and adjust tension (see charts, Section 27-00-00).

(4) Replace lines to outflow valve.

(5) Connect tail light wires.

(6) Position aft pressure bulkhead in place and secure with original hardware. (Longer bolts go in holes that pass through the stiffening channels on aft side of bulkhead.

(7) Apply sealant to pressure bulkhead as necessary. Use Coast Pro-Seal 890B2 or equivalent.

(8) Replace JATO bottle and rack if applicable (see Chapter 71).

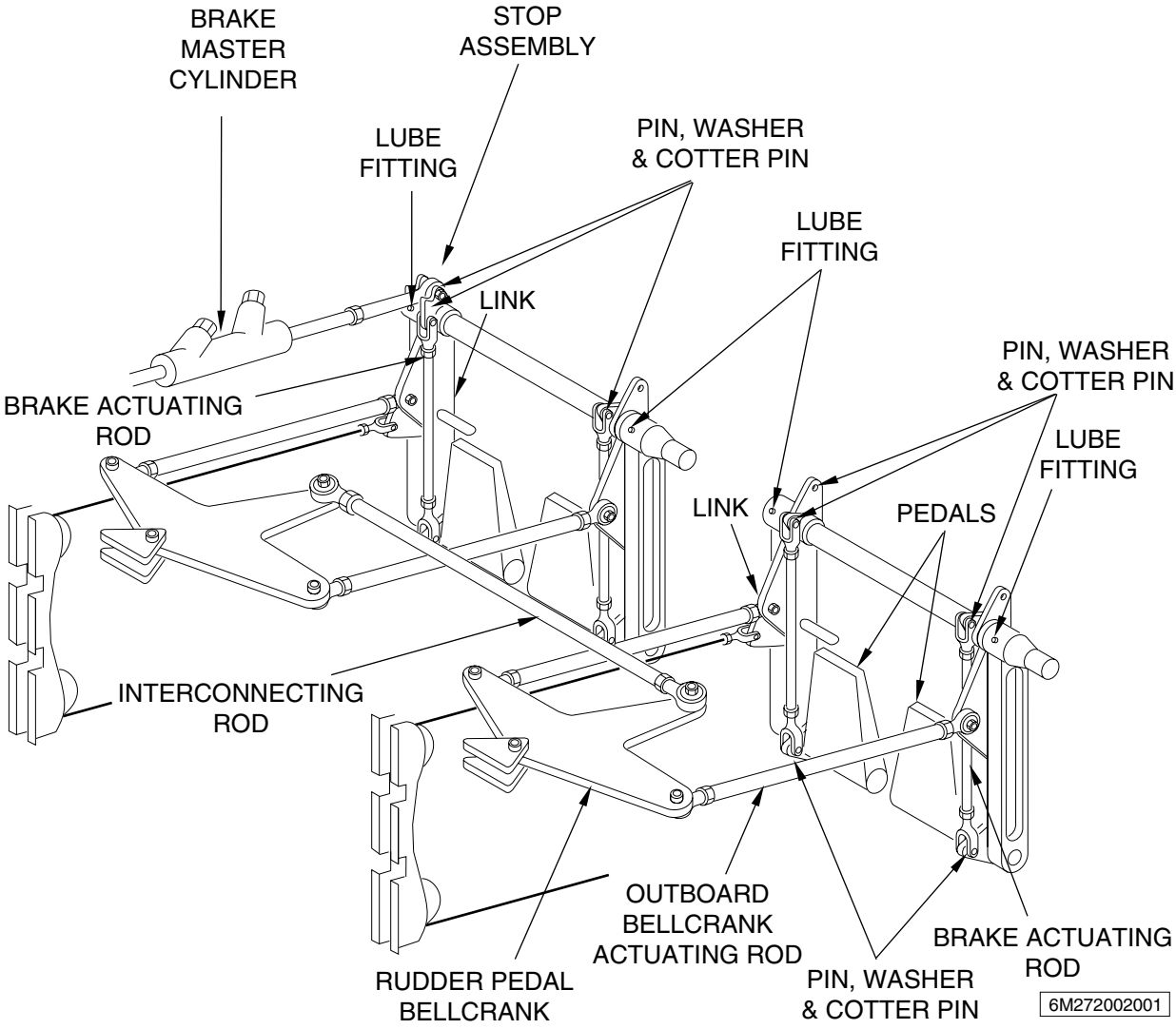
(9) Replace tailcone.

(10) Replace aft interior bulkhead.

RUDDER PEDAL ASSEMBLY – DESCRIPTION AND OPERATION

1. General

A. The pilot and copilot stations are provided with a pair of pedals that are used for rudder and brake control. Each pair of pedal assemblies are attached to a torque tube mounted between fittings on the pedestal and fuselage frame. The pedals are secured to, and pivot from frames hanging from the torque tube, Attaching arms for the rudder control are secured to the upper part of the frames, and the fittings for the brake control are on the forward end of the pedals. Forward movement of the rudder pedals control the rudder.



Rudder Pedal Assembly
Figure 1

RUDDER PEDAL ASSEMBLY – MAINTENANCE PRACTICES

1. Servicing – Rudder Pedal Assembly

A. The only maintenance required of the redder pedal assembly is periodic lubrication. Each pedal assembly has two lubrication fittings. Each is located where the pedal arm attaches and rides on the torque tube hangar assembly. Lubricate with general type grease (MIL-G-23827 or equivalent).

2. Removal / Installation – Rudder Pedal Assembly

NOTE: To remove the entire rudder pedal assembly as a unit, both the control column and the pedestal must be removed. Therefore, removal and installation instructions include only those parts that can be readily removed.

A. Removal – Rudder Pedal

(1) Remove snap ring from end of pedal shaft.

(2) Disconnect lower end of brake actuating rod.

(3) Slide pedal and washer off of shaft.

B. Installation – Rudder Pedal

(1) Position washer and pedal onto shaft.

(2) Connect brake actuating rod.

(3) Install snap ring to shaft to secure pedal.

(4) Perform operational check of rudder pedals.

C. Removal – Outboard Actuating Rod

NOTE: Before installing any rudder pedal rod, measure new rod to length of old rod. With all rods installed, the four rudder pedals should be even.

CAUTION: THE PILOT AND COPILOT RUDDER PEDAL ASSEMBLIES EACH HAVE AN OUTBOARD AND AN INBOARD ACTUATING ROD. THESE RODS CONNECT TO THE RUDDER PEDAL BELLCRANK OF EACH ASSEMBLY. PROCEDURES FOR REMOVAL OF THE OUTBOARD ROD OF EACH ASSEMBLY ARE A SIMPLE MATTER OF UNBOLTING THE ROD AT EACH END AND REMOVING THE ROD. HOWEVER, THE INBOARD ACTUATING RODS OF EACH ASSEMBLY AND THE INTERCONNECTING ROD ARE UNDER TENSION AND CAUTION MUST BE TAKEN DURING REMOVAL.

(1) Remove seat (pilot or copilot) for easy access.

(2) Unbolt and disconnect rod at bellcrank.

(3) Unbolt and disconnect rod at hangar and remove rod.

D. Installation – Outboard Actuating Rod

(1) Position rod end at hangar and secure.

(2) Position rod end at bellcrank and secure.

(3) Replace seat.

E. Removal – Inboard Actuating Rod

(1) Remove seat (pilot or copilot) for easy access.

(2) Pull both inboard rudder pedals forward and secure pedals at base and around front of pedestal tightly with strap. (Pedals are secured to prevent rudder pedals from thrusting forward when inboard actuating rod is disconnected.

(3) While assistant depresses outboard rudder pedal on opposite side, firmly hold inboard pedal and unbolt and disconnect inboard actuating rod at rudder pedal bellcrank.

(4) Unbolt and disconnect rod from rudder pedal hangar and remove rod.

(5) Release outboard pedal pressure.

F. Installation – Inboard Actuating Rod

(1) Attach inboard actuating rod to hangar and secure.

(2) While assistant depresses outboard rudder pedal on opposite side, connect actuating rod to rudder pedal bellcrank.

(3) Release outboard pedal pressure.

(4) Unstrap pedals.

(5) Replace seat.

G. Removal – Interconnecting Rod

(1) Remove seat (pilot or copilot) for easy access.

(2) Pull both inboard rudder pedals forward and secure pedals at base and around front of pedestal tightly with strap. (Secure pedals to prevent rudder pedals from thrusting forward when interconnecting rod is disconnected.

(3) Unbolt and disconnect interconnecting rod at both rudder pedal bellcranks.

H. Installation – Interconnecting Rod

(1) Position rod at one of the pedal bellcranks and attach rod.

(2) Connect interconnecting rod to other bellcrank and secure.

(3) Unstrap pedals.

(4) Replace seat.

I. Removal – Rudder Pedal Bellcrank

(1) Remove seat (pilot or copilot) for easy access.

(2) Remove outboard actuating rod IAW **2.**, C.

(3) Remove inboard actuating rod IAW **2.**, E.

(4) Remove interconnecting rod IAW **2.**, G.

(5) Unbolt bellcrank from pressure bulkhead plate and remove bellcrank.

NOTE: Observe assembly order of spacer washer on bolt securing bellcrank to bulkhead.

(6) Those aircraft equipped with rudder pedal variable authority nose gear steering:

(a) Prior to unbolting bellcrank from pressure bulkhead, disconnect rod end(s) of pot arm(s) and electrical connector from bellcrank.

(b) Unbolt bellcrank from pressure bulkhead plate and remove bellcrank.

NOTE: Observe assembly order of spacer washer on bolt securing bellcrank to bulkhead.

J. Installation – Rudder Pedal Bellcrank

(1) Position bellcrank and bolt to pressure bulkhead.

(2) Those aircraft equipped with rudder pedal variable authority nose gear steering:

(a) Position rod end(s) of pot arm(s) and secure.

(b) Connect electrical connector.

(c) For adjustment and test of potentiometers, see Section 32-50-04.

(3) Install interconnecting rod IAW **2.**, H.

(4) Install inboard actuating rod IAW **2.**, F.

(5) Install outboard actuating rod IAW **2.**, D.

(6) Unstrap pedals.

(7) Replace seat.

3. Inspection / Check – Rudder Pedal Assembly

A. Operation Check – Rudder Pedal

(1) Place pilot and copilot rudder pedals as same leg adjustment.

(2) Engage gust lock and work rudder pedals until rudder locks.

(3) Check to ensure all four rudder pedals are aligned. Should one pedal be lower or higher than its mate, adjustment is made by lengthening or shortening the bellcrank actuating rod relative to the pedal (see Figure 1).

(4) Disengage gust lock.

(5) Press left pilot rudder pedal. Pedal should move smoothly. Copilot pedal should go down simultaneously. Rudder should swing left.

(6) Realign pedals. Copilot pedal should align simultaneously. Rudder should resume streamlined position.

(7) Repeat procedure using right pedal. Rudder should swing right and return to streamlined position when pedal is realigned.

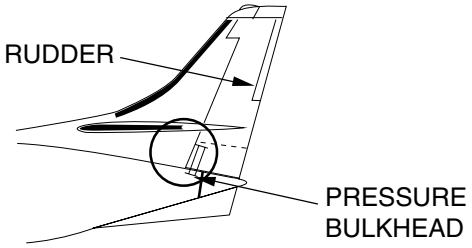
(8) Repeat process using copilot pedals.

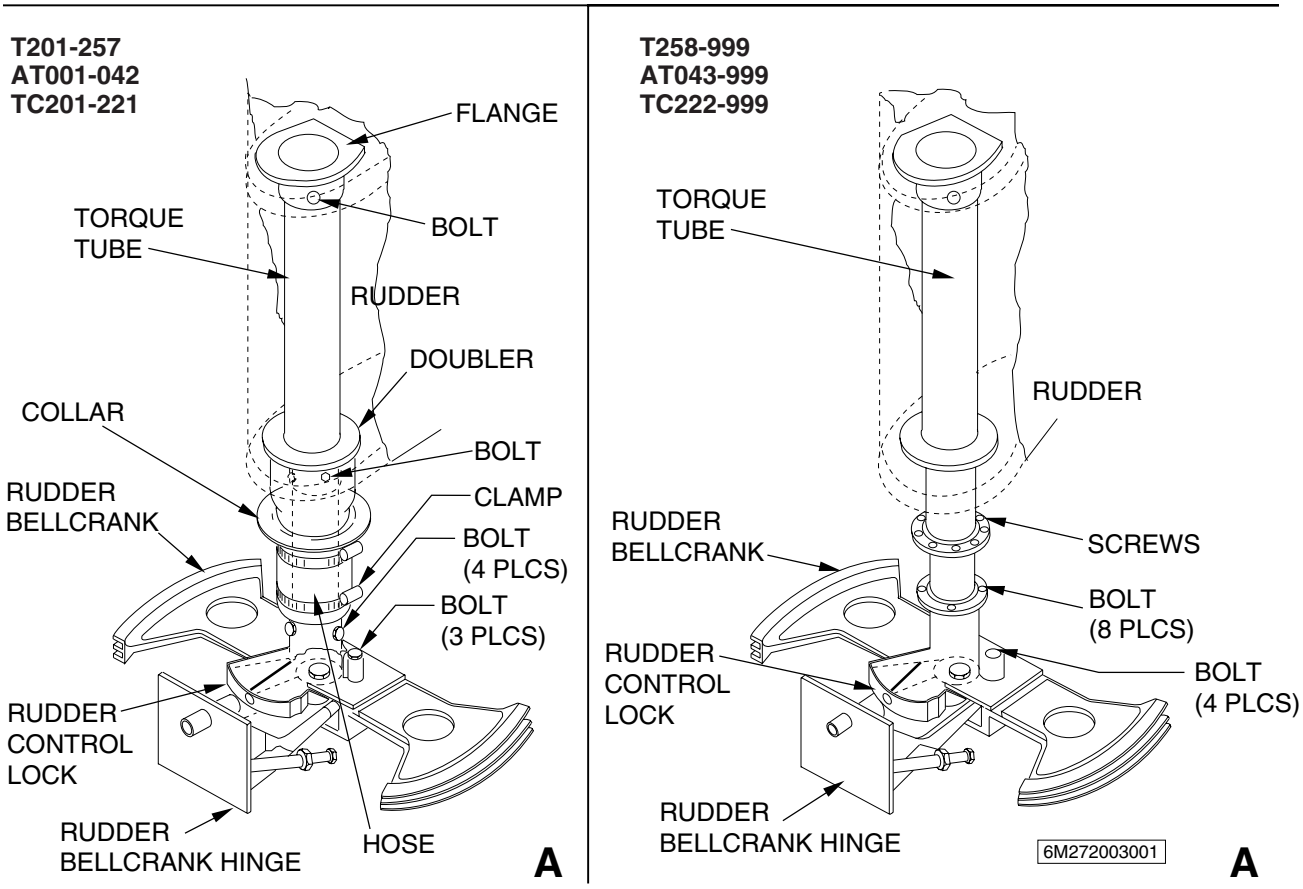
RUDDER BELLCRANK AND TORQUE TUBE – DESCRIPTION AND OPERATION

1. General

A. The rudder bellcrank is located in the fuselage tail section just forward of the aft pressure bulkhead. The torque tube is attached to the upper side of the bellcrank and extends vertically through a pressure seal at the top of the fuselage tail section onto the rudder. Removal and installation procedures of the torque tube are provided under removal and installation of the rudder.

NOTE: A jig is required to replace the torque tube in the rudder.





Rudder Bellcrank and Torque Tube
Figure 1

RUDDER BELLCRANK AND TORQUE TUBE – MAINTENANCE PRACTICES

1. Removal/Installation – Rudder Bellcrank and Torque Tube

A. Removal – Rudder Bellcrank

(1) Remove tail cone. If applicable, remove JATO bottle and rack (see Chapter 71-00-01).

(2) Remove bolts securing aft pressure bulkhead and tilt bulkhead aft.

(3) Remove and cap lines to outflow valve.

(4) Separate handshake disconnects to tail light.

(5) Swing bulkhead to either left or right side and secure. For JATO equipped aircraft, turn bulkhead 180° and secure. Do not allow bulkhead weight to strain JATO wires.

(6) Engage gust lock and push rudder pedals until rudder locks.

(7) Loosen rudder cables at turnbuckle (aft fuselage). Block cables forward of turnbuckles.

(8) Disconnect rudder cables from rudder bellcrank.

(9) Secure rudder cables to bulkhead located just forward of bellcrank.

(10) If applicable, block yaw damper cables to bulkhead and disconnect cables from rudder bellcrank.

(11) Remove three bolts securing bellcrank to rudder control lock.

(12) Remove six bolts securing rudder bellcrank hinge to bulkhead and remove hinge and bellcrank as an assembly.

(13) Remove bolt securing hinge to bellcrank and remove bellcrank.

B. Installation – Rudder Bellcrank

(1) Position rudder bellcrank hinge into bellcrank, aligning bearing with center hole in bellcrank, and secure with bolt and nut.

(2) Position hinge/bellcrank assembly to rudder control lock and secure with three bolts.

(3) Secure hinge to bulkhead with six bolts.

(4) Unblock rudder cables and attach to rudder bellcrank.

(5) Rig rudder system with pedals centered, rudder streamlined and cable tension correct IAW 27-00-00 Maintenance Practices.

(6) If applicable, unblock yaw damper cables and attach cables to rudder bellcrank.

(7) Remove caps and connect lines to valve.

(8) Connect tail light wires.

(9) Position aft pressure bulkhead in place and secure with original hardware. (Longer bolts go in holes that pass through the stiffening channels on aft side of bulkhead.)

(10) Replace JATO bottle and rack if applicable (see Chapter 71-00-01).

(11) Replace tail cone.

C. Removal – Rudder Torque Tube

(1) Remove rudder assembly as described in Maintenance Manual.

NOTE: Rudder torque tube and lower rib are to be removed together.

(2) Remove rivets joining lower five rudder nose ribs to leading edge fairings. Remove rivets joining fairings to ribs in order to gain access to rudder forward spar.

(3) Remove rivets securing lower rudder rib to intercostals on each side.

(4) Remove rivets from torque tube upper flange clip assembly and forward spar.

(5) Remove double row of rivets from torque tube center flange clip assembly and forward spar.

(6) Remove torque tube and lower rib from rudder.

(7) Remove remaining rivet from upper torque tube clip assembly to separate clip from torque tube.

(8) Remove all remaining rivets holding torque tube to lower rib.

D. Installation – Rudder Torque Tube

(1) Install center rivet on upper nose rib and clip assembly first.

NOTE: It is important to do item (1) now.

(2) Install upper nose rib and clip assembly to torque tube flange.

(3) Fit torque tube with clip and nose rib into rudder lower rib. Loose install lower rib and torque tube in rudder.

(4) Clamp torque tube into position in rudder and mount the rudder to the vertical stabilizer using bolts through upper and center hinges.

E. Alignment of Replacement Torque Tube and Rudder with Aircraft

NOTE: If misalignment of vertical stabilizer hinges is suspected, streamline all three hinges. This will require removal of rudder bellcrank. If misalignment of greater than 0.030 inch is found, contact Fairchild Service Engineering Department.

(1) Lock rudder pedals together in centered position.

(2) Engage gust lock and streamline rudder with vertical stabilizer.

(3) Clamp torque tube upper flange so that the flat side of upper flange rests on forward face of rudder spar.

(4) Clamp torque tube center flange to rudder lower rib.

(5) Clamp torque tube lower flange to bellcrank. A flat shim may be fabricated to take up axial gap between lower flange and bellcrank.

(6) Back drill two holes on lower flange from bellcrank.

(7) Back drill two holes on center flange from lower rib.

(8) Remove rudder.

(9) Install rivets in upper clip to forward spar.

(10) Install double row of rivets at center torque tube clip assembly and lower rib.

(11) Install rivets at center torque tube flange and nose rib assembly.

(12) Install rivets on lower rib to rudder skin and intercostals.

(13) Install rudder leading edge fairing.

(14) Install rudder assembly on the aircraft.

(15) Drill remaining holes between center flange and lower rudder rib. Install rivets.

(16) Drill remaining holes between lower flange and bellcrank. Install hardware.

(17) Rig rudder system with pedals centered, rudder streamlined, and cable tension correct per 27-00-00 Maintenance Practices.

NOTE: If unable to get proper alignment of torque tube and rudder with aircraft, contact Fairchild Service Engineering Department.

2. Approved Repairs – Rudder Bellcrank and Torque Tube

A. Approved Repairs – Rudder Torque Tube, Control Lock

NOTE: On early model aircraft, T-201-257, AT001-042, TC201-221, wear may occur between the rudder torque tube and the control lock causing play between the parts. It is recommended that the four long bolts that hold the torque tube and control lock weldment together be replaced with eight short bolts. Repair procedures are as follows:

(1) Gain access to the rudder torque bellcrank/torque tube by completing steps (2) through (12) of REMOVAL – RUDDER BELLCRANK procedures.

(2) Remove bolts securing control lock to torque tube.

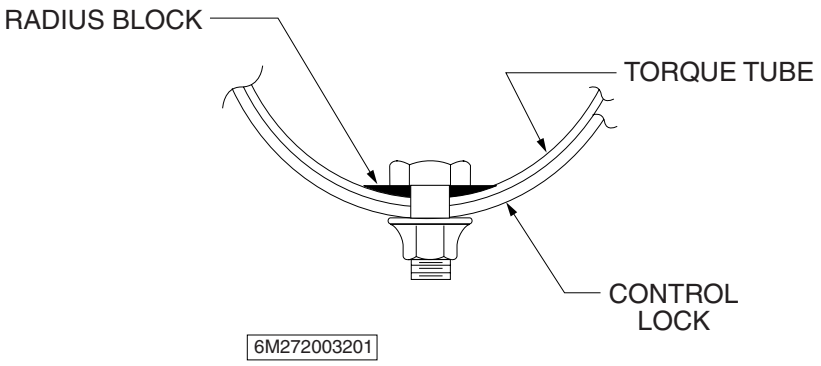
(3) Ream elongated holes to next repair bolt size, eight places in torque tube and eight places in control lock.

NOTE: If the width of material between the edge of a bolt hole and the end of the control lock is less than .125 inch, an additional bolt should be added at 45° from original bolt hole.

(4) Make one radius block for each bolt. Radius block can be made from a 2.0 inch diameter round bar stock (.75 x .75 aluminum alloy primed).

(5) Position control lock onto torque tube and install new short bolts, radius block, and nut. Install with wet primer (zinc chromate primer, MIL-P-85A-1) with bolt head and radius block inside (see Figure 201).

(6) Complete all INSTALLATION – RUDDER BELLCRANK procedures.

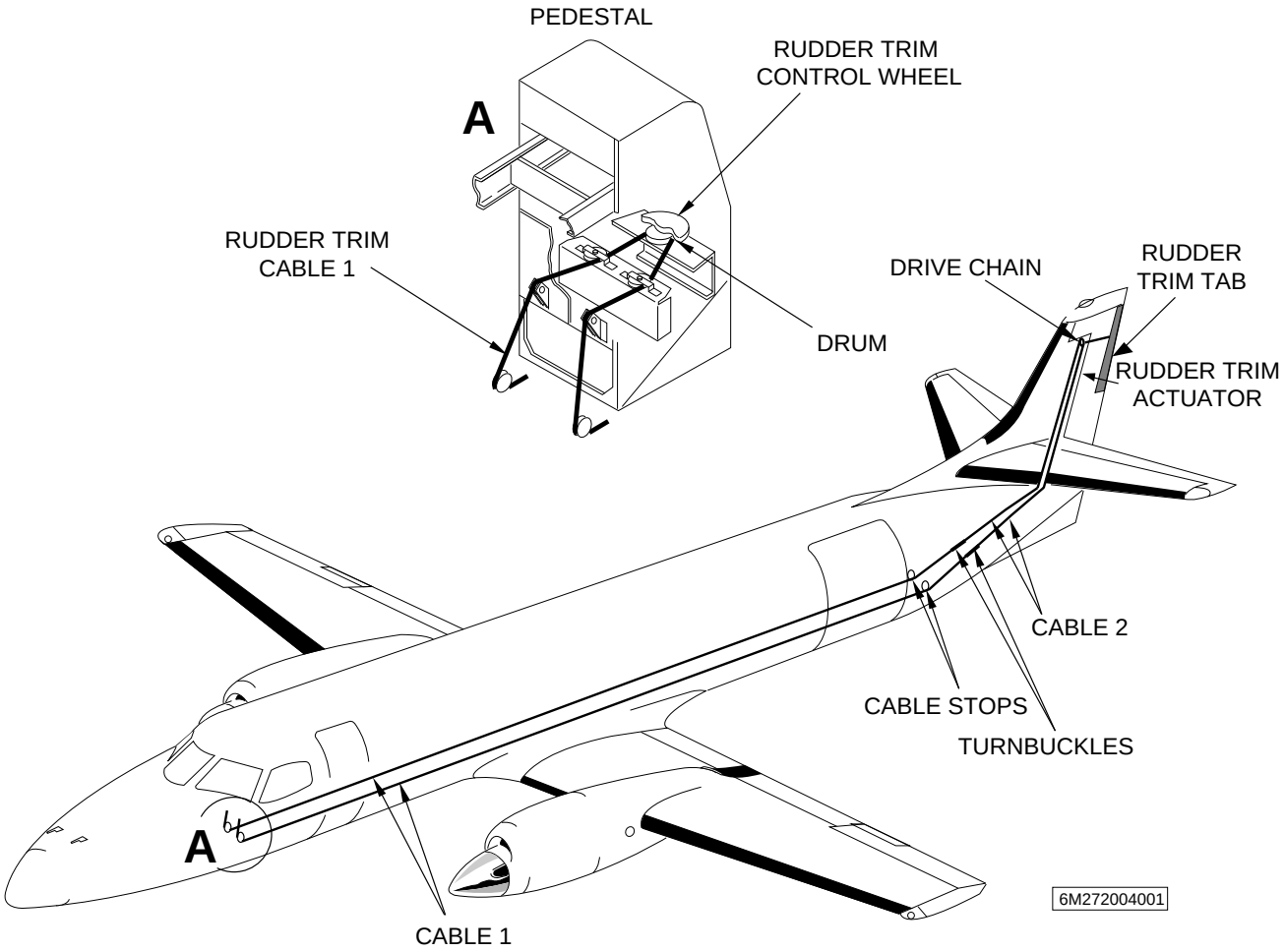


Bolt, Radius Block, Nut Installation
Figure 201

RUDDER TRIM TAB – DESCRIPTION AND OPERATION

1. General

A. A cable operated trim tab is incorporated into the aft end of the rudder. It is controlled by a trim tab wheel on the pedestal through a cable system which activates the trim actuator mounted on the vertical stabilizer trailing edge. The actuator moves a push-pull rod through the rudder to the tab.



Rudder Trim Tab
Figure 1

RUDDER TRIM TAB – MAINTENANCE PRACTICES

1. Servicing – Rudder Trim Actuator

A. In order to relubricate, access to the actuator is obtained through the inspection door in the vertical fin, left side, just in front of the trim push rod. With the shaft fully extended (trim tab full right-left rudder), introduce about 1/4 cu. in. (.14 fluid oz.) suitable grease (MIL-G-81322 or MIL-G-23827 are approved greases) into the axial hole in the center of the sprocket. Then, with a suitable container to catch the excess grease, retract the shaft fully, thereby pushing out the excess. Since this hole is the only pressure relief for the actuator shaft, it must not be fully closed.

2. Removal / Installation – Rudder Trim Tab

A. Removal – Rudder Trim Tab

(1) Move tab, using tab trim located on pedestal, to gain access to actuator rod end. Disconnect rod end.

(2) Remove rudder cap.

(3) Remove screw and nut securing hinge pin clip and remove clip.

(4) Remove hinge pin.

(5) Remove trim tab from rudder.

B. Installation – Rudder Trim Tab

(1) Position trim tab to rudder and insert hinge pin.

(2) Slip clip into hinge and secure with screw and nut.

(3) Attach actuator rod end to trim tab.

(4) Replace rudder cap.

(5) Perform operational check of rudder trim tab. If new tab was installed, check trim tab travel.

C. Removal – Trim Tab Actuator

(1) Position trim tab control wheel indicator on pedestal in neutral position.

(2) Remove left rudder fairing.

(3) Remove aft interior bulkhead.

(4) Locate rudder trim cables (by operating trim wheel on pedestal).

(5) Release tension of cables at turnbuckles. Mark cables for reference when reinstalling.

(6) Gain access to chain and sprocket by removing screws securing vertical stabilizer access door (round door near top on left side of vertical stabilizer).

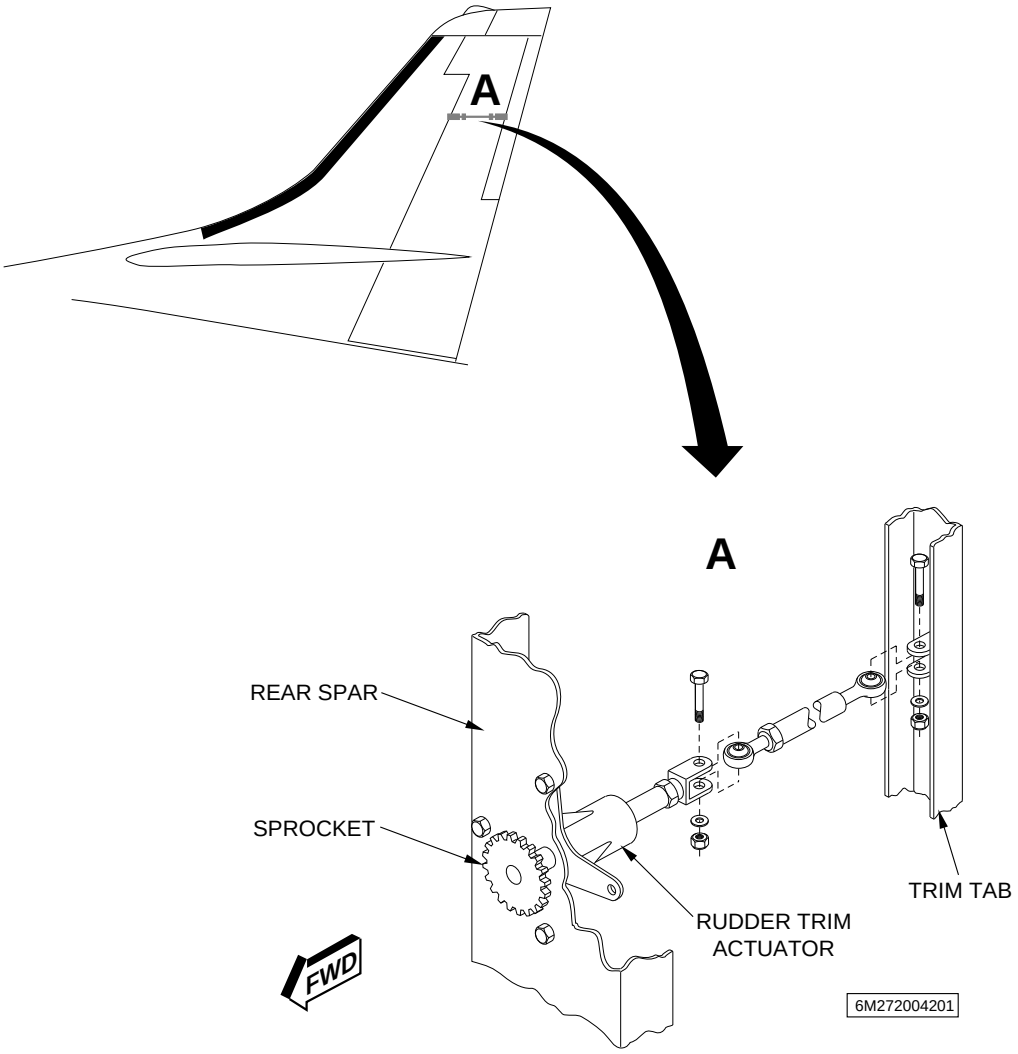
(7) Through vertical stabilizer access opening, mark chain link for reference point and remove chain from actuator sprocket.

(8) Remove actuator sprocket.

(9) Disconnect push-pull rod from actuator.

(10) Remove bolts securing actuator and remove actuator.

RUDDER TRIM TAB – MAINTENANCE PRACTICES



Rudder Trim Actuator
Figure 201

D. Installation – Trim Tab Actuator

(1) Prior to installing actuator, check to ensure full travel of tab will be achieved.

(a) Turn sprocket until shaft is fully retracted into housing.

(b) Measure length of exposed shaft.

(c) Turn sprocket until shaft is fully extended.

(d) Measure length of shaft.

(e) Subtract retracted shaft length from extended shaft length and divide by two.

(f) Retract shaft, then extend shaft to length obtained in step (e).

(2) Position actuator and secure with bolts.

CAUTION: WHEN INSTALLING THE BOLT THAT CONNECTS THE FORWARD END OF THE RUDDER TRIM TAB ROD TO THE ACTUATOR, ENSURE THAT THE BOLT IS INSTALLED VERTICALLY WITH THE BOLTHEAD ON TOP.

(3) Attach push-pull rod to actuator.

(4) Attach actuator sprocket.

(5) Center chain onto actuator sprocket using marked link as reference.

(6) Adjust turnbuckles to reference points and safety.

(7) Check tab through full travel for proper operation.

NOTE: Tab travel left or right is 25° ± 1 1/2°.

(8) Install aft interior bulkhead.

(9) Install vertical stabilizer access door.

(10) Install rudder fairing.

3. Inspection/Check – Rudder Trim Tab

A. Operational Check – Trim Tab

(1) On pedestal, set rudder trim tab wheel on neutral. Verify trim tab is streamlined.

(2) Rotate trim wheel left. Verify wheel rotates smoothly. Observe trim tab rotates to the right. Travel is 25° ± 1 1/2°.

(3) Center trim wheel on neutral. Verify trim tab is streamlined.

(4) Rotate trim wheel right. Verify wheel rotates smoothly. Observe trim tab rotates to the left. Travel is 25° ± 1 1/2°.

(5) Center trim wheel on neutral. Verify trim tab is streamlined.

(6) Check rudder trim tab shaft play:

(a) Remove left rudder fairing.

(b) Hold rudder in a fixed position and deflect rudder tab from left to right.

(c) Measure shaft movement at the clevis bolt using a rule calibrated in thousandths, and align parallel with the shaft.

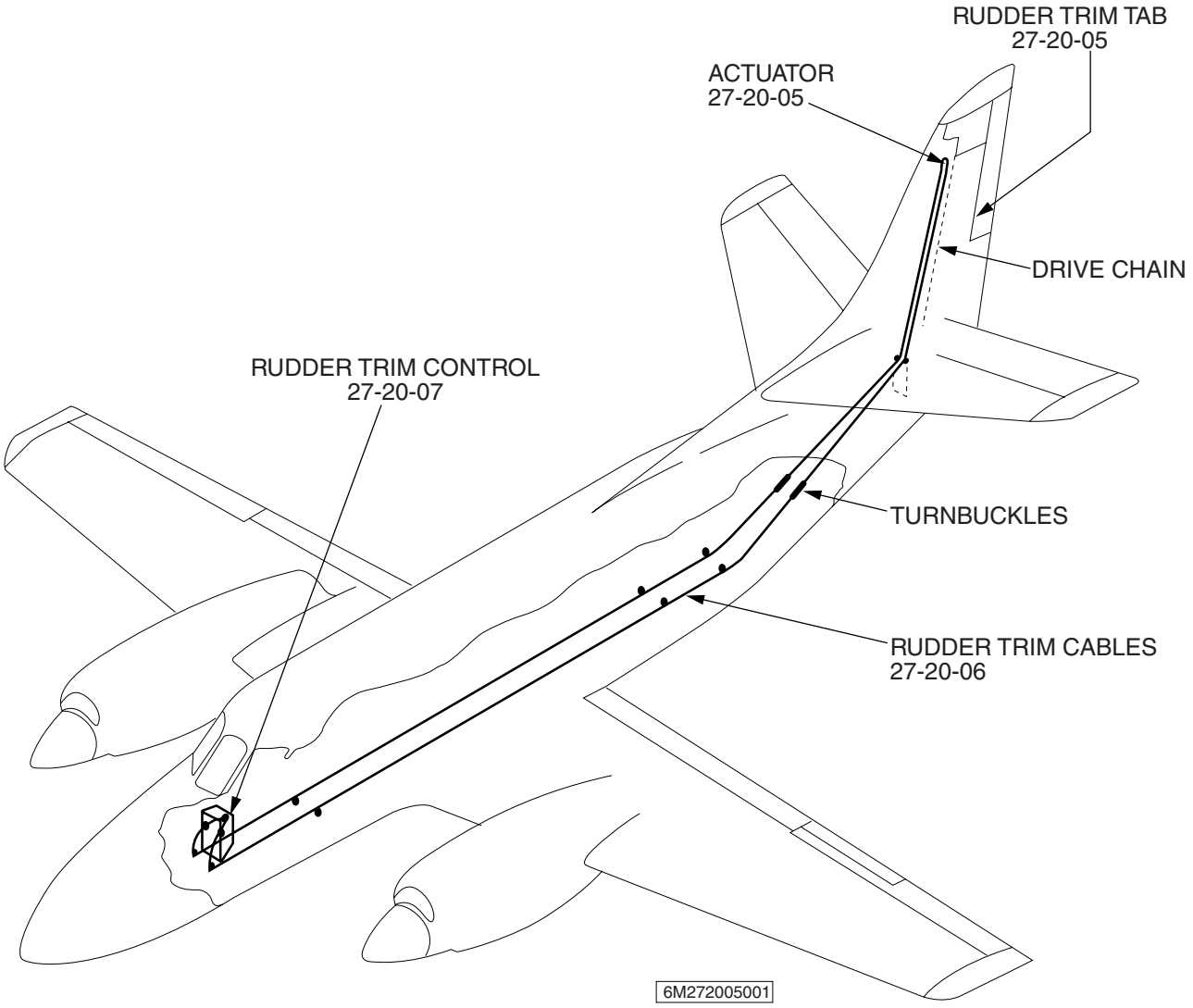
(d) If the shaft freeplay exceeds 0.080 inch (2.03 mm), replace rudder trim tab actuator IAW this chapter.

(e) Install rudder fairing.

RUDDER TRIM TAB CABLES – DESCRIPTION AND OPERATION

1. General

A. The rudder trim cable system is comprised of three cables which are joined together by two turnbuckles and a drive chain. The cable system is divided into two sections. The first section cable (1), consists of one cable. The cable runs from a turnbuckle aft of the aft interior bulkhead, under the center aisle floorboards, through the rudder trim mechanism in the pedestal, and back along the center aisle to terminate at another turnbuckle aft of the aft interior turnbuckle. The next section of cables (2) continues from the turnbuckles, travels up along the trailing edge of the vertical stabilizer where it interconnects with a chain that drives the trim tab actuator. Rudder cable stops are attached to cable (1) and are located under the cabin floorboards directly inside the cargo door on the SA226-AT and SA226-TC aircraft and directly inside the passenger/crew door on the SA226-T aircraft.



Rudder Trim Tab Cables
Figure 1

RUDDER TRIM TAB CABLES – MAINTENANCE PRACTICES

1. Removal/Installation – Rudder Trim Cables

A. Removal – Cable Section Forward of Turnbuckles

(1) Ensure battery switches are OFF.

(2) Place rudder trim in neutral position.

(3) Remove pilot and copilot seats.

(4) Remove crew compartment floorboards at control column.

(5) Remove cabin center aisle floorboard at aft interior bulkhead.

(6) Remove aft interior bulkhead.

(7) Locate rudder trim cables and block aft of turnbuckles.

(8) Disconnect cables at turnbuckles.

(9) Remove cable stops.

(10) Remove center overlay/edge lighted panel from pedestal.

(11) Remove center pedestal panel.

(12) Remove auxiliary trim switch from panel.

(13) Support cover out of way taking stress off of wires.

(14) Remove screws securing cross bracket to pedestal.

(15) Pull bracket complete with drum, cable and gears up so that shaft is clear and pull assembly out through opening.

(16) Remove drum with cable intact from shaft.

(17) Remove pin securing cable in drum and remove cable.

(18) Remove cables from pulleys located on aileron drum/gear bracket.

(19) Remove cotter pins from pulleys located in rear of pedestal and remove cables.

(20) Remove cable through pulleys and fairleads as necessary.

B. Installation – Cable Section Forward of Turnbuckles

(1) Ensure battery switches are OFF.

(2) Find center of cable and insert into slit in drum.

(3) Secure cable with holding pin.

(4) Wind cable onto drum by starting each side of cable simultaneously and winding in opposite directions. Wind cable inward from each side and terminate at center of drum. Drum is fully wound when cables meet and slot in drum is at opposite end.

(5) Block cable tightly so that cable does not unwind from drum.

(6) Slide drum onto shaft and over woodruff key. Drum is centered for alignment when slot is facing up or forward.

(7) Insert cables through pedestal opening and insert shaft of assembly (consisting of cross bracket, gears, control wheel, and drum) into pedestal. Insert shaft through bushing in lower bracket.

(8) Secure cross bracket to pedestal with screws.

(9) Guide cables through pulleys on aileron drum/gear assembly (both cables go on copilot side of each pulley). Secure pulleys with screws.

(10) Guide cables over pulleys located in rear of pedestals and secure with cotter pins.

(11) Insert cable through pulleys and fairleads as necessary.

(12) Remove cable block at drum. Keep enough tension on cable so that it does not unwind from drum.

(13) Attach cable ends to turnbuckles.

(14) Attach cable stops.

(15) Apply cable tension (see chart, Section 27-00-00).

(16) Adjust tab travel (see chart, Section 27-00-00) and set cable stops.

(17) Replace panels on pedestal.

(18) Replace aft bulkhead.

(19) Replace floorboards.

(20) Replace pilot and copilot seats.

C. Removal – Cable Section Aft of Turnbuckles

(1) Position trim tab control wheel indicator on pedestal in neutral position.

(2) Remove aft interior bulkhead.

(3) Locate rudder trim cables.

(4) Block cables forward of turnbuckles.

(5) Disconnect cables at turnbuckles in bulkhead.

(6) Gain access to cable where cable is attached to actuator chain by removing access door (oval shaped) midway up on right side of vertical stabilizer. Disconnect cable from chain at master link.

(7) Remove access door on right side of vertical stabilizer at elevator bellcrank and round door near bottom of stabilizer.

(8) Remove cable through pulleys and fairleads as necessary.

D. Installation – Cable Section Aft of Turnbuckles

(1) Center actuator chain on sprocket and attach cable to chain.

(2) Feed cable down through vertical stabilizer and attach at turnbuckle aft of bulkhead.

(3) Remove cable blocks.

(4) Adjust cable tension (see chart, Section 27-00-00).

(5) Check trim tab travel.

(6) Replace access doors.

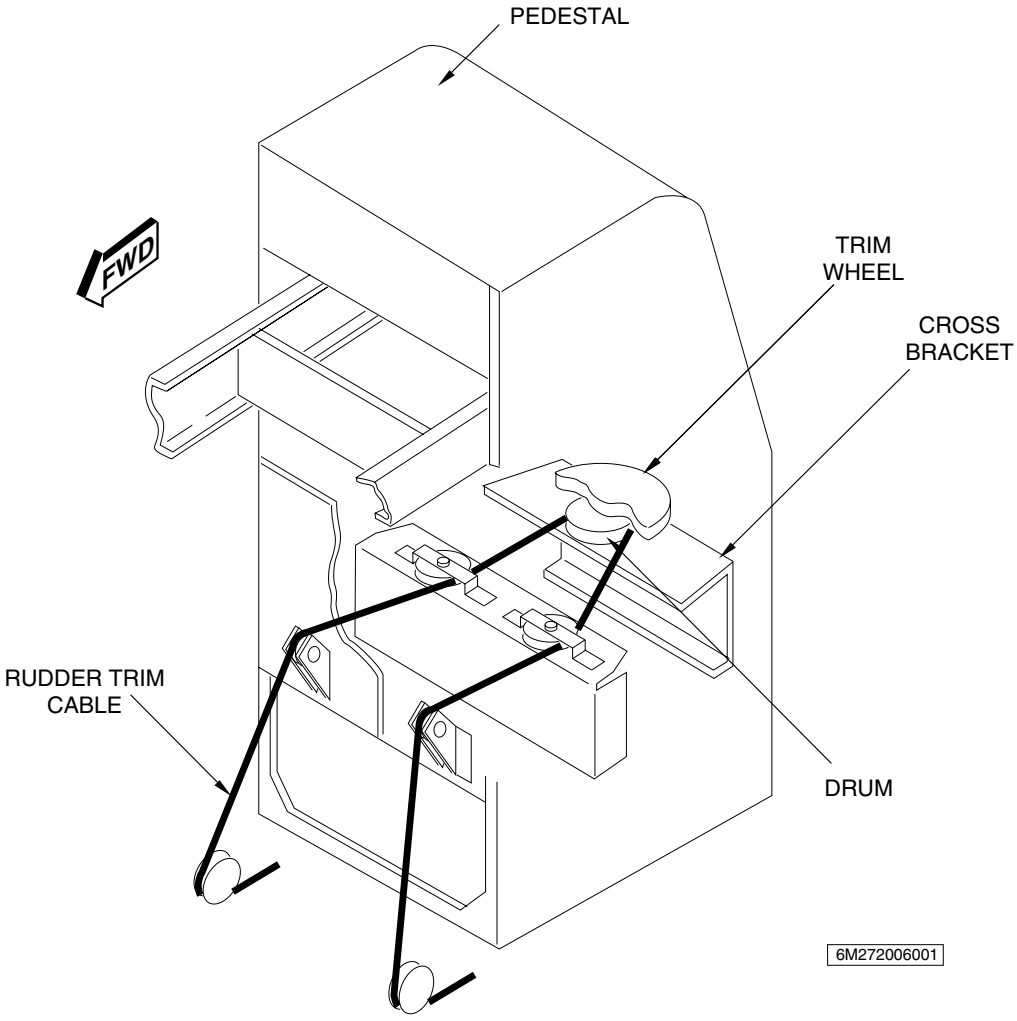
(7) Replace aft bulkhead.

RUDDER TRIM TAB CONTROL – DESCRIPTION AND OPERATION

1. General

A. The rudder trim control is located on the lower front of the cockpit pedestal. The control consists of a large thumb-wheel and a smaller indicator wheel. The smaller indicator wheel is positioned above the larger thumb-wheel. Both wheels are set horizontally within the pedestal so that approximately 180° of each wheel is visible. Both wheels have index marks painted in black for ease in readability. The small indicator wheel is notched for touch trim position indication. When the index mark on the thumb-wheel is positioned straight out (six o'clock position) and aligned with the notch on the smaller indicator wheel, the rudder trim is in neutral position.

B. By actuating the rudder trim control wheel, cables movement rotates a sprocket which actuates a push-pull rod through the rudder to the tab, deflecting it from neutral to the desired direction. Turning the thumb-wheel to the right (tab left) will trim the rudder so that the airplane turns right. Likewise, when the thumb-wheel is turned left (tab right) the airplane turns left. Gears within the control mechanism enable the thumb-wheel (lower wheel) to make a complete revolution while the smaller (upper wheel) moves only a few degrees. Therefore, if the lower wheel index mark is at six o'clock and the upper wheel notch is at five o'clock or seven o'clock, the rudder trim tab is not in the neutral position.



Rudder Trim Tab Control
Figure 1

1. Removal/Installation – Rudder Trim Tab Control

A. Removal – Rudder Trim Control Wheel

(1) Ensure battery switches are OFF.

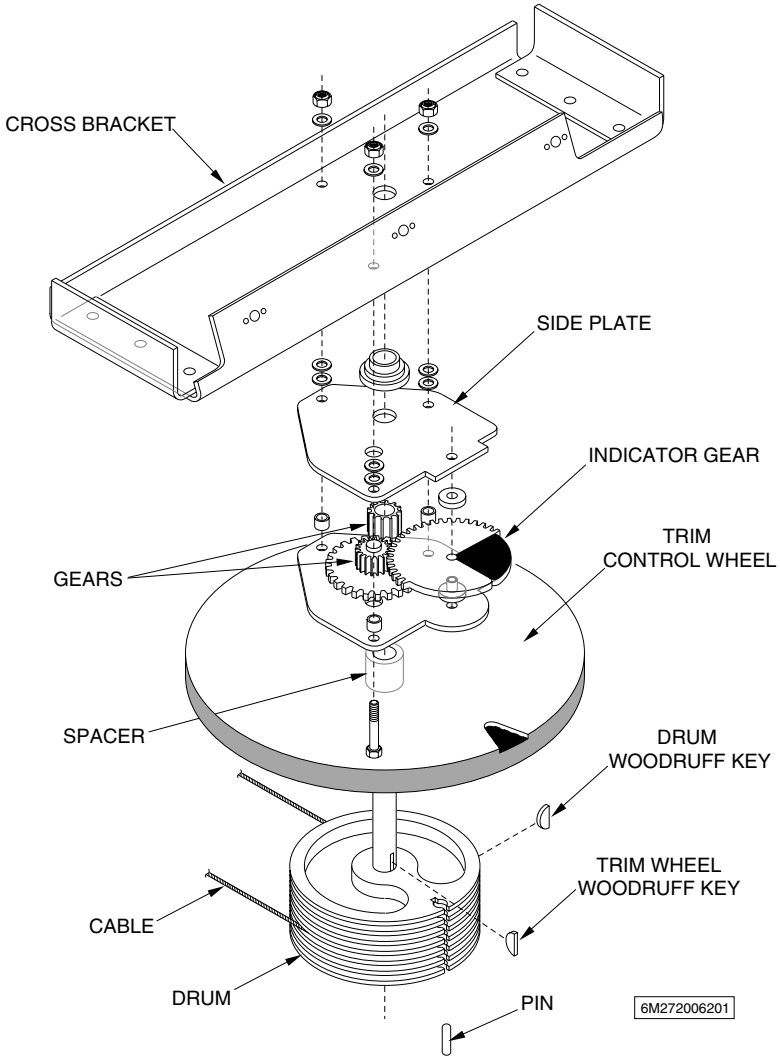
(2) Place rudder trim in neutral position.

(3) Remove aft interior bulkhead.

(4) Locate rudder trim cable end at turnbuckles and block cables aft of turnbuckles.

(5) Disconnect cable ends at turnbuckles.

(6) Remove center overlay/edge-lighted panel from pedestal.



Rudder Trim Control
Figure 201

(7) Remove center pedestal panel.

(8) Remove auxiliary trim switch from panel.

(9) Support cover out of way taking stress off wires.

(10) Remove screws securing cross bracket to pedestal.

(11) Pull bracket complete with drum, cable, and gears up so that shaft is clear of cross bracket.

(12) Remove drum, with cable intact, from shaft.

(13) Pull assembly out through opening in pedestal.

(14) Remove drum woodruff key and remove trim wheel.

(15) Remove trim wheel woodruff key and spacer.

(16) Remove screws securing gear housing and slide bracket off of shaft.

(17) Pry side plate off of shaft to expose gear assembly and note arrangement of gears (see Figure 201).

(18) Remove gears as necessary.

B. Installation – Rudder Trim Control Wheel

(1) Position two large gears onto bottom side (plate with round tab). Ensure indicator gear is aligned evenly across front of plate (see Figure 210).

(2) Insert shaft with gear, long side through side plate, and position onto gear so that upper woodruff key slot is forward and lower slot is on the right side (see Figure 201).

(3) Position spacers, top side plate, washers, and bracket and secure with screws.

(4) Slide spacer up shaft and secure with woodruff key.

(5) Position trim control wheel onto shaft (indicator side up) and slide over woodruff key.

(6) Insert bottom woodruff key.

(7) Slide drum onto shaft over woodruff key. Drum is centered for alignment when slot is facing up or forward.

(8) Insert shaft of assembly (consisting of cross bracket, gears, control wheel, and drum) onto pedestal and insert shaft through bushing in lower bracket.

(9) Secure cross bracket to pedestal with screws.

(10) Attach auxiliary trim switch onto panel.

(11) Replace center pedestal panel.

(12) Replace center overlay/edge-lighted panel onto panel.

(13) Connect cable ends at turnbuckles and take up slack.

(14) Remove cable blocks.

(15) Remove aft interior bulkhead.

(16) Perform operational check of rudder trim control wheel.

YAW DAMPER / RUDDER BIAS – DESCRIPTION AND OPERATION

1. General

A. A yaw damper system is incorporated into the SA226–T model aircraft rudder system to dampen transient motion about the yaw axis of the aircraft. The system utilizes a turn and bank indicator with a rate gyro. The gyro sends information to an amplifier that processes this information to control a servo. The servo activates a capstan which in turn works a cable system that causes the rudder to deflect to eliminate the yaw motion.

B. On later model SA226-T aircraft (T292 and up) a rudder bias system is incorporated. The system allows a lower rotation speed in the event of a loss of engine power during takeoff. The system uses pressure switches to sense torque in each engine. The torque switches control relays. Should an engine lose torque these relays override the yaw damper control of the servo and allows the rudder bias system to kick the rudder in the opposite direction of the low torque engine.

YAW DAMPER – DESCRIPTION AND OPERATION

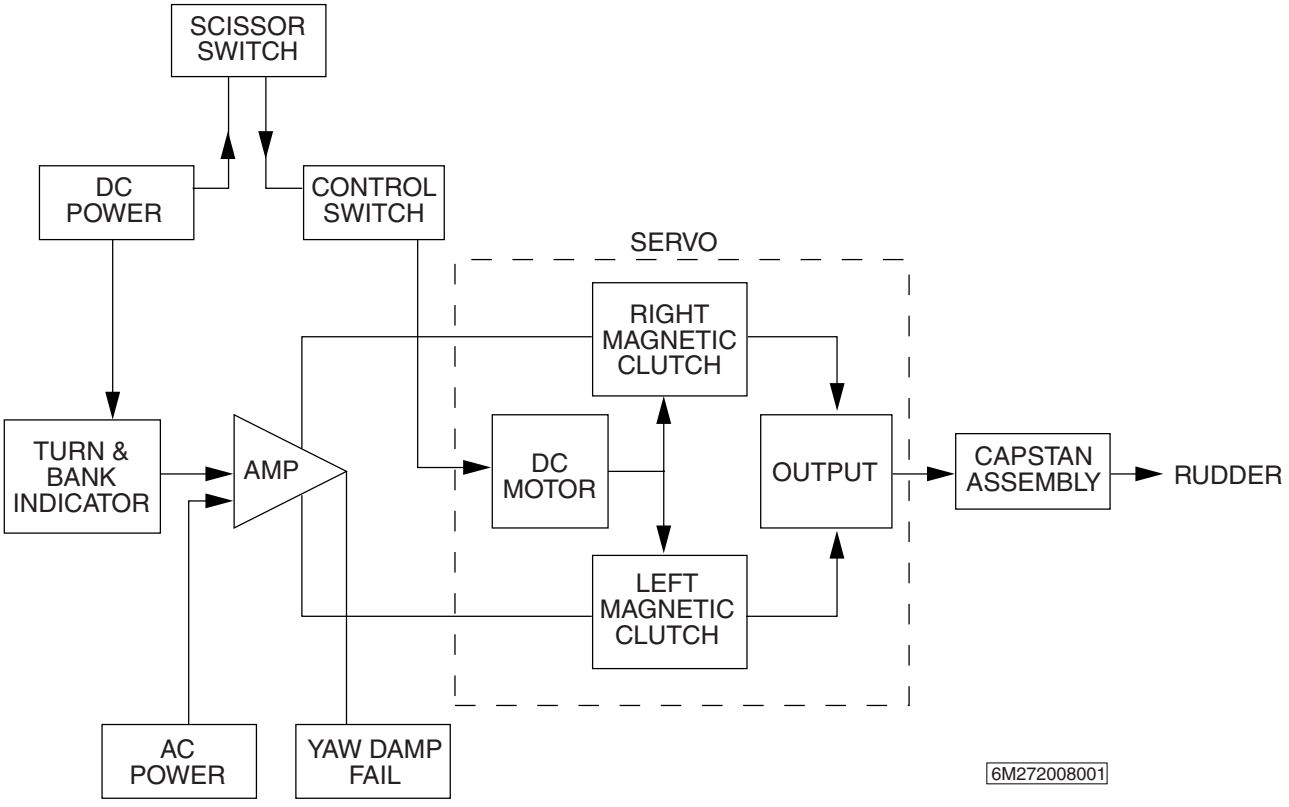
1. General

A. The yaw damper operates during takeoff, the landing gear strut, upon being extended, engages the landing gear scissor switch. The yaw damper control/test switch, located on the pedestal, turns the yaw damper on and off and provides a preflight test capability. By holding the control switch in the TEST position and inducing a yaw motion in the aircraft during taxi, a preflight test of the system can be performed.

B. A turn and bank indicator, located on the instrument panel, provides the pilot a visual display of the aircraft's rate of turn and bank attitude. The indicator also provides the yaw rate signal to the yaw damper system. The indicator is powered by 28 VDC. A bus transfer switch is provided to allow the turn and bank indicator to be powered by the right essential bus should failure of the left essential bus occur. The yaw amplifier is powered by ll5 VAC from the yaw damper circuit breaker from the left essential bus. An AC voltage signal from a rate gyro, contained within the turn and bank indicator, reflecting the rate of change of the aircraft's yaw attitude is fed to the yaw amplifier. The amplifier electronically engages clutches in the servo. The servo activates the capstan, which in turn works a cable system that causes the rudder to eliminate the yaw motion.

C. Yaw Amplifier

(1) The yaw amplifier is powered from the ll5 volt left AC bus and is located on the right side of the tail section interior at F.S. 324. The amplifier receives signals from the gyros in the pilot's turn and bank indicator. It processes these signals and feeds rudder movement signals to the servo and capstan assembly located at the upper center section of the adjacent bulkhead.



Yaw Damper
Figure 1

D. Turn and Bank Indicator

(1) The turn and bank indicator, located on the pilot's instrument panel, provides a visual display of the aircraft's rate of turn. It also provides the yaw rate signal to the yaw damper system. A bus transfer switch is provided to allow the instrument to be powered from the left essential bus or the right essential bus. It normally is selected to the left essential bus.

E. Yaw Damper/Test Switch

(1) The yaw damper test switch, located on the cockpit pedestal, is used to turn the yaw damper on and off as well as to provide a preflight test capability. Normally, power is removed from the yaw damper by the inboard left main gear safety switch when the airplane is on the ground. However, by holding the control switch in the test position and inducing a yaw motion in the aircraft during taxi, a preflight test of the system can be performed. If functioning correctly, the yaw damper will push back on the rudder pedal depressed during the test to oppose the yaw.

F. Fault Light

(1) The yaw damper fault light will illuminate anytime power to the pilot's turn and bank indicator or yaw amplifier is interrupted.

G. Servo and Capstan Assembly

(1) The servo and capstan assembly, located aft of the interior bulkhead, converts the electrical output of the amplifier into mechanical energy to position the rudder.

H. Servo

(1) The servo consists of a continuous DC powered motor connected to a magnetic clutch, which contains two discs with powdered iron particles between them. One disc is connected to the motor and the other to an output shaft. When current passes through these particles they grab or stick to each other; thus torque from the motor is coupled to the output shaft. The amount of output torque is directly related to the amount of current allowed to flow through the clutch. For the yaw damper, the current is controlled by the yaw amplifier. For rudder bias, the amount of current is set by resistors on the terminal strip. The output shaft of the servo turns a pulley drum in the capstan assembly.

I. Capstan

(1) The capstan consists of an inner gear (driven by the servo), a mechanical slip clutch (which allows the pilot to override the servo), and a pulley drum. Cables on the pulley drum are connected to the rudder.

YAW DAMPER – MAINTENANCE PRACTICES

1. Servicing – Yaw Damper

A. No regularly scheduled maintenance is required on the servo or the capstan assembly. During regularly scheduled inspections the servo and capstan assembly should be checked for security and condition. The cables should be inspected and maintenance performed in accordance with procedures in Section 27-00-00.

2. Removal/Installation – Yaw Damper

A. Removal – Yaw Damper Servo

(1) Remove aft interior bulkhead.

(2) Remove safety wire and remove electrical connector from servo.

(3) Remove bolts attaching servo to capstan.

NOTE: When removing bolts, turn bolts only from front. Do not attempt to turn off from rear as bolts are secured by clinch nuts.

(4) Remove servo.

B. Installation – Servo

(1) Position servo and secure with bolts. Torque bolts 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(2) Connect electrical connector and safety wire.

(3) Replace aft interior bulkhead.

(4) Perform yaw damper functional check.

C. Removal – Yaw Damper Capstan Assembly

(1) Remove servo.

(2) Remove screws attaching three legged guard over pulley drum; remove guard.

(3) Remove bolts and nuts attaching mounting plate to side channels.

(4) Separate drum and plate.

(5) Notice how cable is wound on drum; turn drum to unwind cable.

D. Installation – Yaw Damper Capstan Assembly

(1) Install cable onto pulley drum as follows (see Figure 201):

(a) Position cable over pulley with swaged ball (item 3) aligned with pulley socket.

(b) Seat swaged ball (item 3) into pulley socket.

(c) Grasp RH cable (item 4) and wrap around pulley drum (item 2) one turn in a counterclockwise direction as viewed from the rear looking forward.

(d) Grasp LH cable (item 1) and wrap around pulley drum (item 2) one turn in a clockwise direction as viewed from the rear.

(e) Verify that cable completes a minimum of two full wraps around drum when socket and ball (item 3) are positioned at twelve o'clock position.

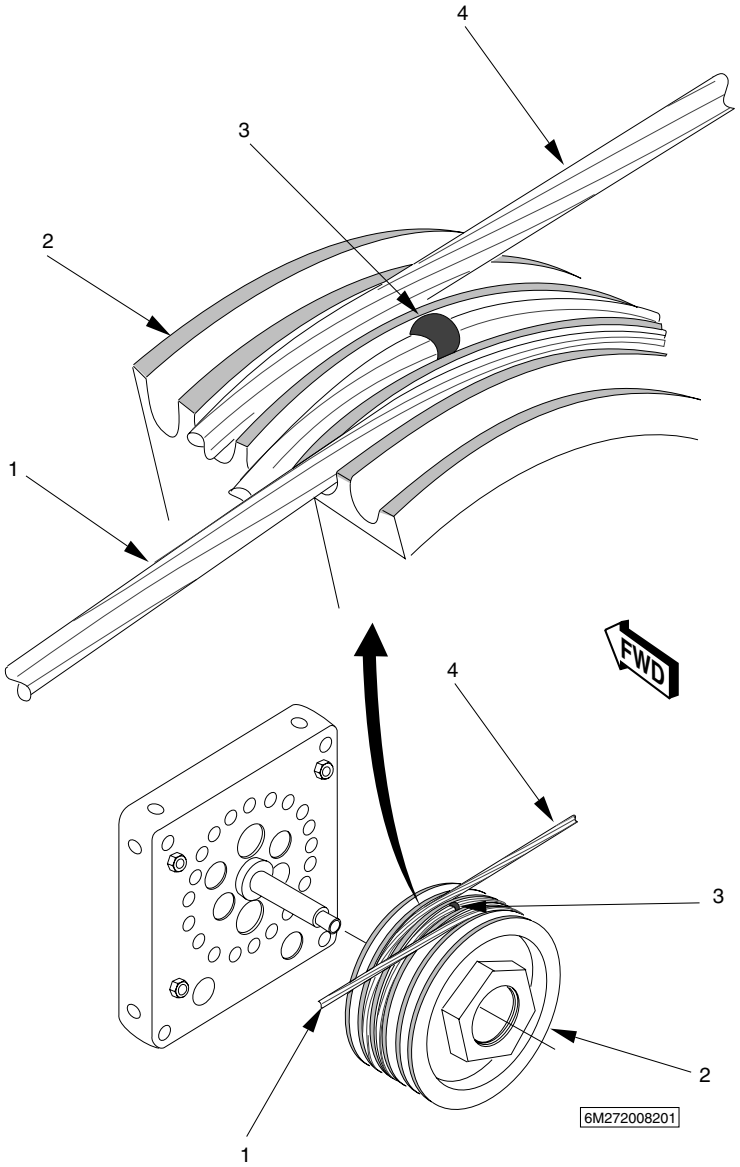
(2) Position drum on shaft.

(3) Install bolts and nuts attaching plate to side channels.

(4) Position guard and secure with screws.

(5) Work rudder pedals through a few times to check for binding or improper cable installation.

(6) Install servo.



Yaw Damper Servo
Pulley Cable Installation
Figure 201

YAW DAMPER – MAINTENANCE PRACTICES

1. Adjustment/Test – Yaw Damper

A. Servo Capstan Assembly Adjustment

(1) Refer to 27-30-11, Adjustment /Test Stick Pusher Assembly for capstan and mechanical clutch adjustment. Torque for yaw damper clutch is 180 plus or minus 10 inch-pounds.

B. Yaw Damper Functional Check

(1) The yaw damper may be functionally tested by taxiing the aircraft at low speeds while holding the yaw damper switch in TEST. Turn the aircraft by either nose wheel steering or brakes, and note right rudder pedal deflection during left turns and left rudder pedal deflections during right turns.

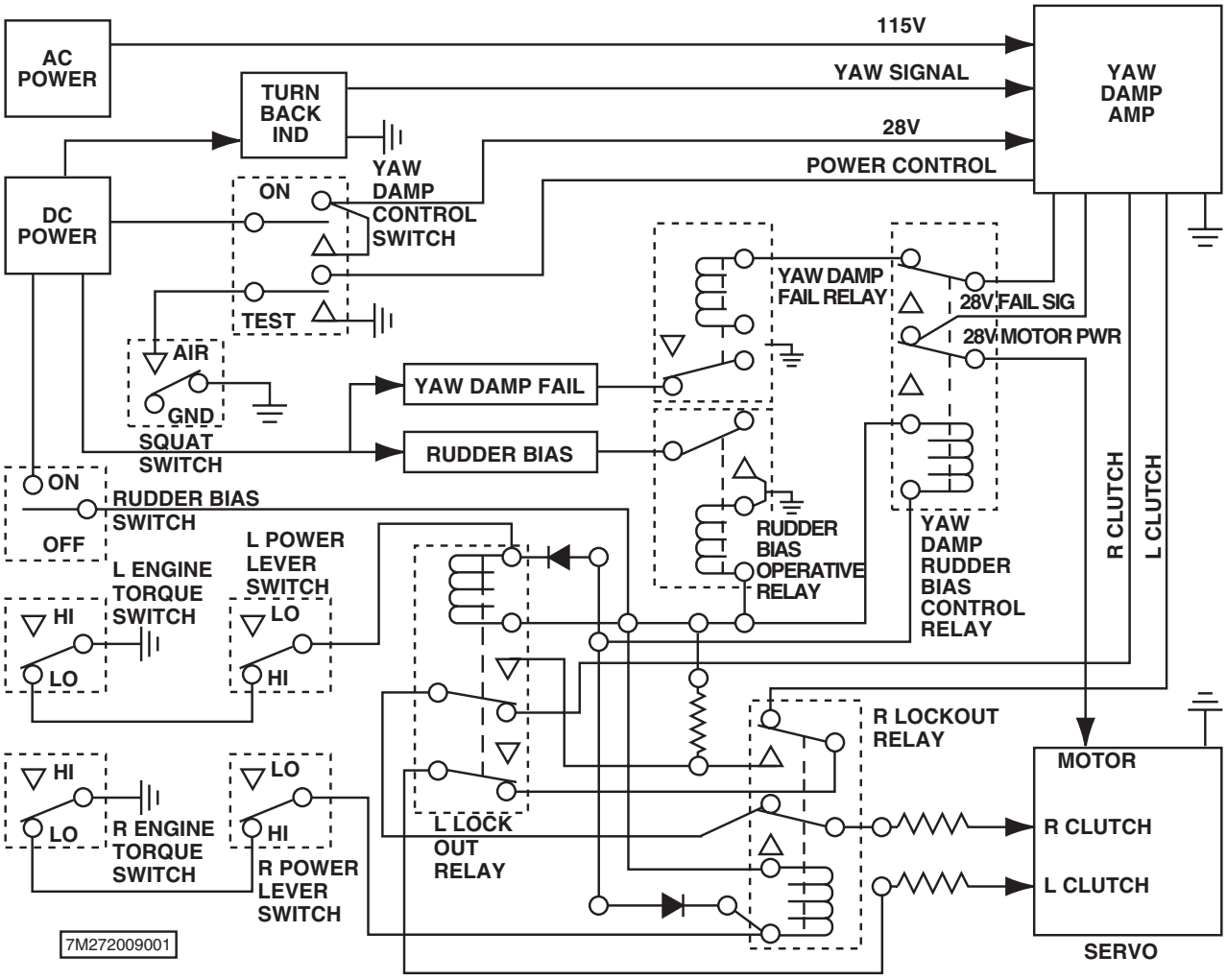
(2) To check the yaw damper in the hangar will require that the turn and bank indicator be removed from the instrument panel or that a separate turn and bank indicator be available. Moving the turn and bank should produce rudder deflection in the opposite direction to the movement of the turn and bank indicator.

RUDDER BIAS – DESCRIPTION AND OPERATION

1. General

A. The rudder bias is installed to provide more rudder control in the event of engine failure during takeoff. The system consists of switches installed in each engine compartment that receive engine raw torque pressure as a means to sense engine power output. Switches are installed in the pedestal to sense the position of the power levers, which must be advanced to 50% of their travel before the switches close and arm the system. Various control relays are installed to actually provide the bias signal and inform the pilot that the rudder bias system is functioning. The actual mechanical displacement of the rudder is caused by the yaw damper servo and cable system.

B. As shown in Figure 1 electrical power is supplied to the system through the rudder bias switch. The rudder bias power relay will close and cause illumination of the RUDDER BIAS light installed in the annunciator panel. Power is also supplied to the bias relays and the indication relays. The ground path for each of the bias relays consist of a series path made up of the respective engine power lever switch and the engine torque switch. The yaw damper amplifier controls, through the bias relays, the output of the yaw servo.



Rudder Bias Schematic
Figure 1

RUDDER BIAS – DESCRIPTION AND OPERATION

C. During takeoff, with the rudder bias switch on, closure of each power lever switch arms the system (since the power levers must be advanced beyond 50% of their travel to close the switches, engine torque will have already increased sufficiently to open the torque switches). The system is now armed, the rudder bias light is illuminated, and the YAW DAMPER FAIL light extinguished (indicating that the yaw damper circuit is controlling the yaw servo).

D. Should an engine failure occur, the low torque output will cause the torque switch to close and supply a ground to the respective bias relay and the indication relay. The bias relays are wired so that closure of one of the relays disconnects the yaw damper amplifier from both clutch connections of the yaw servo. Bias current, through a limiting resistor is supplied to the respective clutch in the yaw servo to cause rudder deflection. The wiring of the bias relays will not permit a simultaneous bias signal to be supplied to both clutches of the yaw servo, even if the second engine failed as well.

E. The ground supplied through the torque switch will also cause the indication relay to close and supply power to the servo. Thus, if the yaw damper circuit were inoperative power would still be provided to the servo. Closure of the indication relay also removes power from the fail relay and thereby causes the YAW DAMPER FAIL light in the annunciator to illuminate. The illumination of the fail light serves to inform the pilot that the bias system has sensed the engine failure.

F. The bias system will continue to supply a force input into the rudder until the power lever for the failed engine is retarded or the rudder bias switch is turned off.

RUDDER BIAS – MAINTENANCE PRACTICES

1. Servicing – Rudder Bias

NOTE: The following check may be performed without the engines operating.

A. Rudder Bias

(1) Position battery switches to ON.

(2) Position inverter switch to ON.

(3) Set power levers at flight idle.

(4) Position yaw damper switch to ON, check that yaw damper fail light is off.

(5) Position rudder bias switch to ON, check that rudder bias light is on.

(6) Check that no force is being applied to the rudder pedals.

(7) Advance the left power lever full forward. Note that the left rudder pedal is forced rearward, and the yaw damper fail light is on.

(8) Retard the left power lever to flight idle. Check that no force is being applied to the rudder pedals, and that the yaw damper light is extinguished.

(9) Advance the right power lever full forward. Note that the right rudder pedal is forced rearward, and the yaw damper fail light is on.

(10) Retard the right power lever to flight idle. Check that no force is being applied to the rudder pedals, and that the yaw damper light is extinguished.

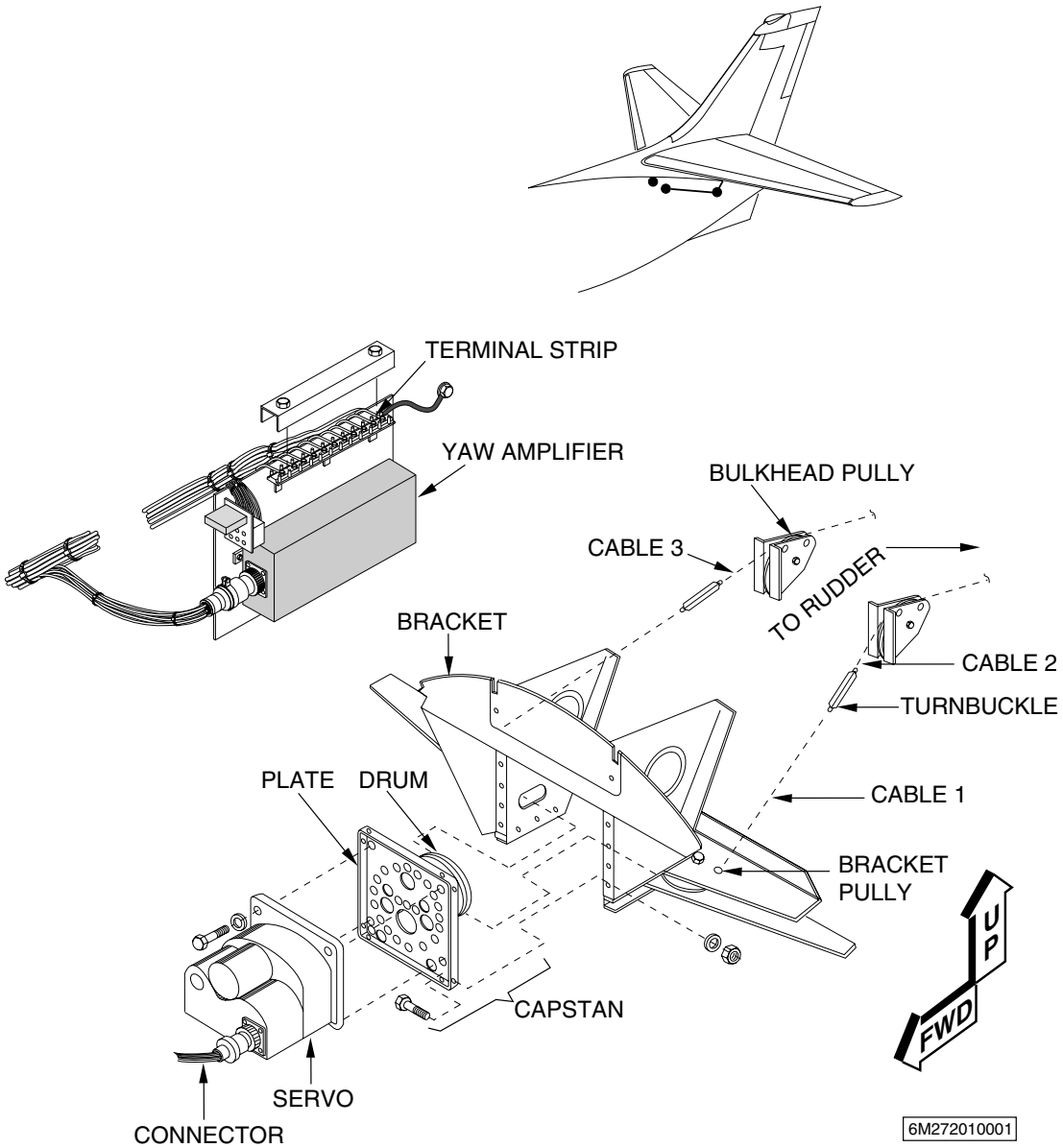
(11) Position inverter switch to OFF.

(12) Position battery switches to OFF.

YAW DAMPER / RUDDER BIAS CABLE – DESCRIPTION AND OPERATION

1. General

A. The yaw damper / rudder bias cable system consists of three individual cables joined together by turnbuckles to make one continuous cable system (see Figure 1). One section of cable winds around the capstan drum, and travels through two pulleys located in the servo/capstan bracket, with each end terminating at turnbuckles (see cable 1, Figure 1). Each end of cable 1 is joined by a cable (cables 2 and 3) with a turnbuckle. These cables pass through a pulley and attach to opposite ends of the rudder bellcrank. Cables 2 and 3 differ only in that cable 2 has a left hand threaded terminal joining the turnbuckle and cable 3 has a right hand threaded terminal joining its turnbuckle.



Yaw Damper / Rudder Bias Cable
Figure 1

YAW DAMPER / RUDDER BIAS CABLE – MAINTENANCE PRACTICES

1. Servicing – Yaw Damper

NOTE: No regularly scheduled maintenance is required of the yaw damper / rudder bias servo. During regularly scheduled maintenance servo should be checked for security and overall condition.

2. Removal/Installation – Yaw Damper

A. Removal – Servo

(1) Remove aft interior bulkhead.

(2) Disconnect electrical connector from servo.

(3) Remove bolts attaching servo to capstan.

NOTE: When removing bolts, turn bolts only from front. Do not attempt to turn off from rear as bolts are secured by clinch nuts.

(4) Remove servo.

B. Installation – Servo

(1) Position servo and secure with bolts. Torque bolts 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(2) Connect electrical connector.

(3) Install aft interior bulkhead.

(4) Perform yaw damper / rudder bias functional check, see 27-20-09.

C. Removal – Capstan

NOTE: Capstan consists of a cable retainer bracket, a plate, and a cable drum.

(1) Remove servo.

(2) Remove bolts, washers, and nuts securing capstan to bracket.

(3) Remove nuts and screws securing cable retainer bracket to capstan plate and remove plate.

(4) Remove cable retainer bracket from drum, Unwind cable from drum. Note position of cable for installation.

D. Installation – Capstan

(1) Disassemble new capstan assembly.

(2) Wind drum onto drum.

(3) Position cable retainer onto drum and attach bracket to plate.

(4) Position capstan assembly to bracket and secure with bolts, washers, and nuts.

(5) Install servo.

E. Removal – Cable Section 1

(1) Remove aft interior bulkhead.

(2) Block cables at bulkhead aft of turnbuckles.

(3) Remove electrical connector from servo.

(4) Remove servo.

(5) Remove nuts, washers, and bolts securing capstan assembly to bracket.

(6) Remove capstan assembly.

(7) Unwind cable from drum. Note position of cable for installation.

(8) Disconnect cable ends at both turnbuckles.

(9) Remove cable keepers from two pulleys in a bracket and remove cable through bracket.

F. Installation – Cable Section 1

(1) Locate shank ball at center of cable.

(2) Insert ball into drum and wind cable onto drum.

(3) Assemble drum, cable retainer bracket, and capstan plate with three nuts and screws.

(4) Feed cable ends through pulleys in bracket and replace cable keepers at pulleys.

(5) Position capstan assembly into bracket and secure with bolts, washers, and nuts.

(6) Attach cable ends to turnbuckles.

(7) Remove cable blocks and adjust cable tension, see Section 27-00-00.

(8) Replace servo and electrical connector

(9) Replace bulkhead.

G. Removal – Cables Section 2 or 3

(1) Engage gust lock and work rudder pedals until rudder locks.

(2) Remove aft interior bulkhead.

(3) Disconnect cable at turnbuckle.

(4) Remove tailcone. If applicable remove JATO bottle and rack (see Chapter 71-00-00).

(5) Remove bolts securing aft pressure bulkhead and tilt bulkhead aft.

(6) Remove and cap lines to outflow valve.

(7) Separate handshake disconnect to tail light.

(8) Detach bulkhead and swing it to either left or right side and secure. For JATO equipped aircraft, do not allow bulkhead weight to strain JATO wires.

(9) Disconnect cable at bellcrank.

NOTE: Upper cable in bellcrank is yaw damper / rudder bias cable. Lower cable is rudder cable.

(10) Remove cable keeper on pulleys at bulkhead and remove cable.

H. Installation – Cables Section 2 or 3

(1) Position cable through pulley and insert cable keeper.

(2) Attach cable to rudder bellcrank.

(3) Attach cable end to turnbuckle and adjust tension, see 27-00-00.

(4) Connect lines to outflow valve.

(5) Connect tail light wires.

(6) Position aft pressure bulkhead in place and secure with original hardware. (Longer bolts go in holes that pass through the stiffening channels on aft side of bulkhead.)

(7) Apply sealant to pressure bulkhead as necessary. Use Coast Pro-Seal 890B2 or equivalent.

(8) Replace JATO bottle and rack if applicable (see Chapter 71-00-01).

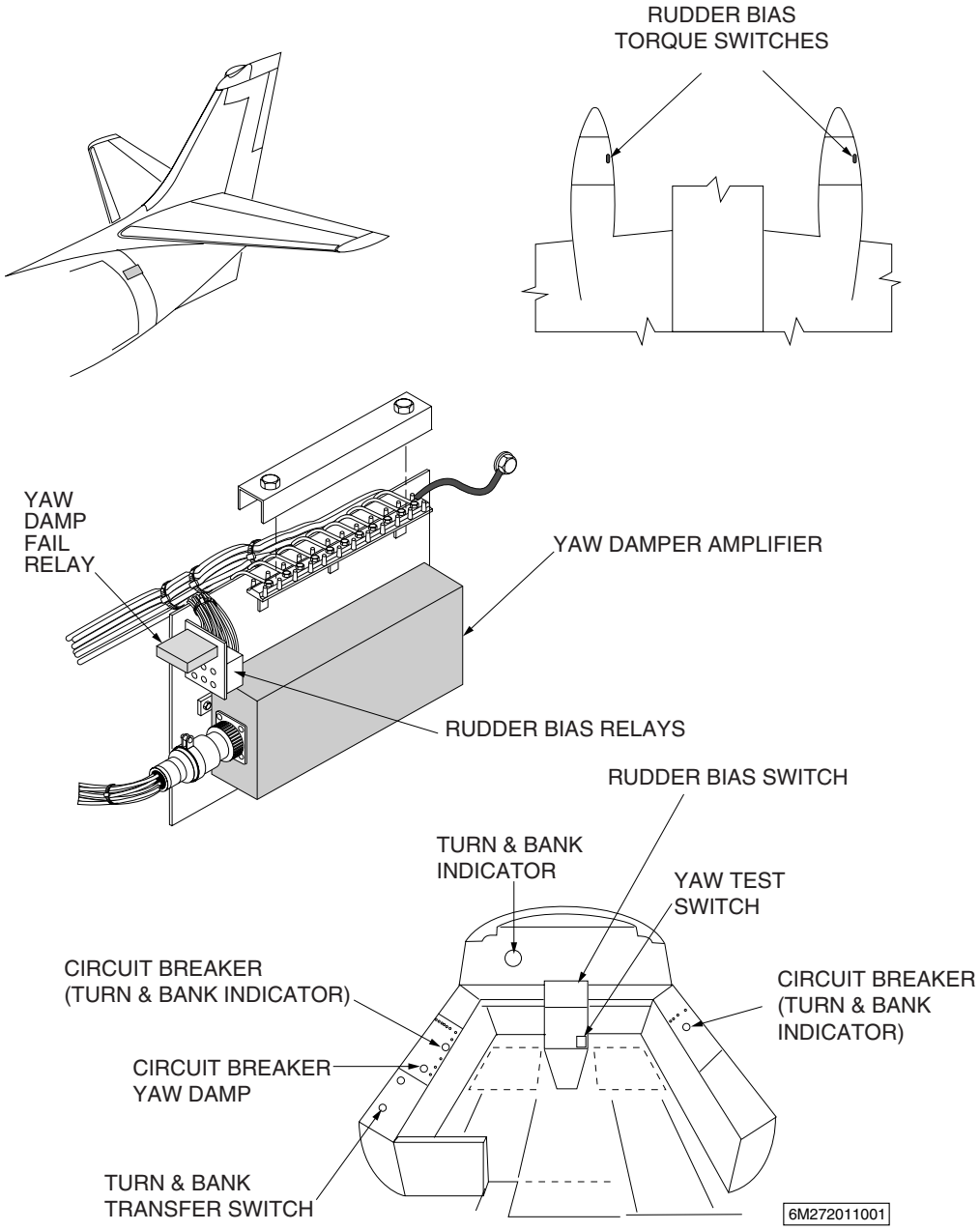
(9) Install tailcone.

(10) Install aft interior bulkhead.

YAW DAMPER / RUDDER BIAS ELECTRICAL COMPONENTS – DESCRIPTION AND OPERATION

1. General

A. Yaw damper / rudder bias electrical components are installed in the cockpit and tail cone area. Components of the yaw damper system located in the cockpit include the three position yaw damper switch installed in the pedestal, a turn and bank indicator installed in the instrument panel, a transfer switch located on the left hand side console, and circuit breakers installed on both essential busses.



Electrical Components
Figure 1

B. The turn and bank indicator, as delivered from the factory, is installed on the pilot's side of the instrument panel. In some avionics configurations the indicator may have been moved to the right side of the instrument panel, or the output normally provided by the yaw damper may be provided through some portion of the auto pilot. Components installed in the tail cone area include yaw damper amplifier and the yaw damper servo and capstan assembly.

C. Aircraft equipped with rudder bias also feature a rudder bias switch installed in the center pedestal and additional relays installed near the yaw damper amplifier. A torque switch is installed on the right hand side of each engine, near the engine torque transducer. The torque switches are set to close by 26 psi increasing pressure and open by 19 $\pm$ 2 psi decreasing pressure.

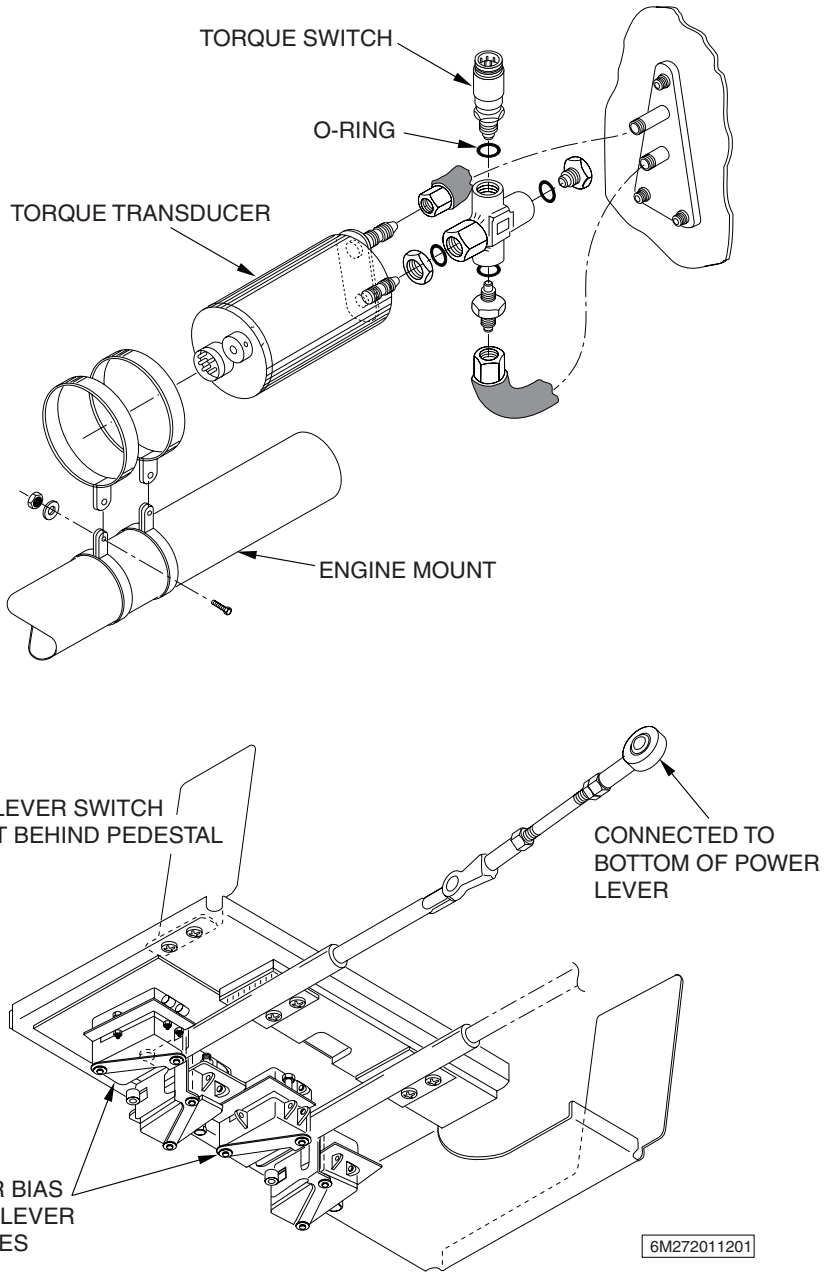
1. Removal / Installation – Yaw Damper / Rudder Bias Electrical Components

A. Removal – Torque Switch

(1) Gain access to the switch by removing the right hand side cowl from the engine.

(2) Cut the safety wire and remove the electrical connector to the switch.

(3) Unscrew the switch from its mounting boss.



Switches
Figure 201

B. Installation – Torque Switch

(1) Install a new O-ring (P/N MS29561-012) on the new torque switch. Install the new switch in its mounting boss and torque to a maximum of 30 inch-pounds (3.39 N·m).

(2) Install and safety the electrical connector to the switch.

(3) Operate and test the system.

2. Adjustment / Test – Yaw Damper / Rudder Bias Electrical Components

A. Adjustment / Test – Power Lever Switches

NOTE: Prior to adjusting pitch out of trim power lever switches, adjust landing gear warning power lever switches in accordance with Chapter 32-60-00.

(1) Pull power levers to flight idle gate.

(2) Loosen screws in slotted ends of switches.

(3) Push switches against rods until actuators are fully depressed.

(4) Move power levers forward and aft. Switches should actuate approximately one inch before the flight idle gate when the power levers are pulled back. This is measured from the gate to the lift latch, near the word LOCKED.

(5) If switches are installed on vertical bracket, loosen screws attaching bracket to plate. Adjust bracket forward or aft as necessary. If switches are attached directly to plate, move switches forward and aft as necessary.

(6) Tighten screws.

(7) Perform functional check.

ELEVATOR – DESCRIPTION AND OPERATION

1. General

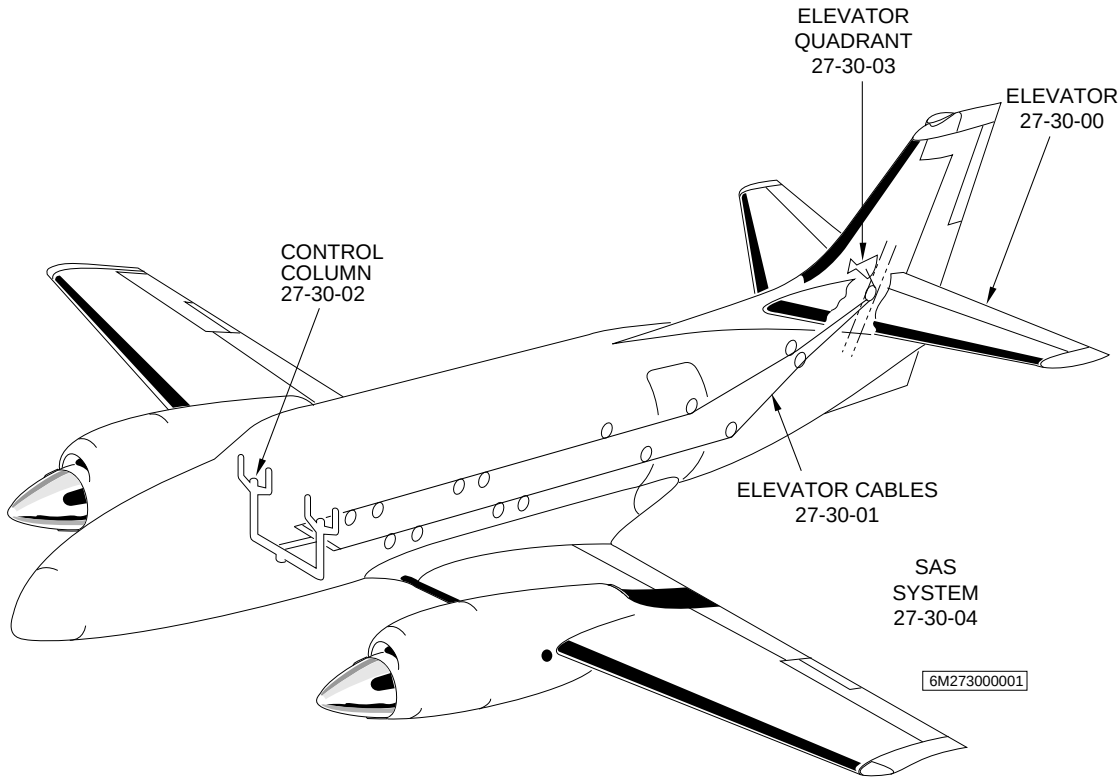
A. Elevators manufactured of aluminum skin riveted over a frame of spars and ribs are controlled by forward and backward movement of the control column. Each of the two elevators are attached at three hinge points to the trailing edge of the horizontal stabilizer.

B. The elevators are actuated from an arm mounted on the interconnecting torque tube between the two control columns. A push-pull rod is connected between the arm and a walking beam located centrally under the cockpit floor. Cables are routed around the walking beam, through a pulley arrangement, beneath the cabin floor to the aft fuselage section. The cables are then routed through pressure seals to a bellcrank arrangement (quadrant) installed in the vertical stabilizer. Two push-pull rods are attached from the quadrant to the elevators for operation of the elevators.

C. The SA226-AT, and SA226-TC airplanes have an elevator-down spring connected from the lower part of the quadrant to a bracket on the lower section of the vertical stabilizer. This spring is adjusted to place a preload on the elevator system to keep the elevators, which tend to fly in an upward position, in a neutral position.

D. The SA226-T has no elevator-down spring. Instead, a 20 pound bob weight extends forward from an arm attached to the base of the control column, is used to provide the desired stability assistance. Also the SA226-T has an elevator damper located forward of the pedestal. Its purpose is to prevent sudden movements of the elevator.

E. All SA226 model aircraft are equipped with a stall avoidance system commonly referred to as the SAS. In SA226-T model aircraft the system, with some modification, also serves as a stability augmentation system (SAS^2).



Elevator
Figure 1

1. Removal/Installation – Elevator

A. Removal – Elevator

(1) Remove elevator fairing.

(2) Detach one end of bond strap at outboard elevator hinge.

(3) Remove bolts securing elevator hinge support to horizontal stabilizer at center elevator hinge.

(4) Detach one end of bond strap at outboard hinge.

(5) Remove two and loosen two (leave two for support) of four bolts securing outboard elevator hinge support.

(6) Remove elevator quadrant access door located on right side of vertical stabilizer.

(7) Remove bolt securing elevator torque tube push-pull rod to elevator quadrant. Check that elevator quadrant clears threaded stud on torque tube fittings. Trim stud if interference is noted. Leave 1 1/2 threads extended through nut.

(8) Raise elevator to gain access to bottom bolt of cap which clamps elevator torque tube to aft spar fitting and remove cotter pin, nut, washer, and bolt.

(9) Lower elevator to gain access to top bolt of cap and remove cotter pin, nut, washer, and bolt.

(10) Remove remaining two bolts from outboard hinge support and remove elevator.

B. Installation – Elevator

(1) Position elevator. Secure center elevator hinge support with bolts.

(2) Mounted on the elevator torque tube fitting is a bearing and matching pair of spacers (one spacer on each side of the bearing), secured with a nut. With center elevator hinge support secured, ensure bearing on torque tube fitting is seated within groove of aft spar fitting. If bearing does not line up with groove, proceed as follows:

(a) Remove bolts securing center hinge support. Remove elevator.

(b) Remove nut, spacers and bearing from torque tube fitting.

(c) Slide bearing onto torque tube fitting. Install nut on fitting to prevent bearing from falling off.

NOTE: Check alignment of horizontal stabilizer hinges prior to installation of elevators. Maximum misalignment of center hinge is 0.030 inches (0.762 mm), measured perpendicular to the axis through inboard and outboard hinge centerlines.

(d) Position elevator. Secure center hinge support with bolts, while positioning bearing into groove in aft spar fitting.

(e) With center elevator hinge secured and bearing seated within groove in aft spar fitting, measure the distance between inboard edge of torque tube and outboard edge of bearing. Note dimension.

(f) Remove bolts securing center hinge support. Remove elevator.

(g) Remove nut and bearing from torque tube fitting.

(h) Install a spacer of the dimension measured in step **1.**, B., (2), (e) onto the torque tube fitting (see Illustrated Parts Catalog for spacer part number).

(i) Install bearing onto torque tube.

(j) Install matching spacer onto torque tube (see Illustrated Parts Catalog for spacer part number). Secure with nut.

(3) Position elevator. Secure center elevator hinge support with bolts.

(4) With center elevator hinge support secured, ensure bearing on torque tube fitting is seated within groove of aft spar fitting. If bearing does not line up with groove, repeat step **1.**, B., (2) as required.

(5) Secure outboard elevator hinge support with bolts.

(6) Connect bond straps to both hinges.

CAUTION: PUSH-PULL ROD ATTACHING BOLT MUST BE INSTALLED WITH BOLT HEAD INBOARD.

(7) Connect push-pull rod to elevator quadrant and safety wire.

(8) Position cap and attach bolts (top and bottom) securing elevator torque tube into fitting. Safety wire bolts.

(9) Operate elevator through complete travel. Inspect for interference with surrounding structure and operating mechanisms. If new elevator was installed, check travel.

(10) Verify that elevator quadrant clears threaded stud on torque tube fittings. Trim the stud if interference is noted. Leave 1 1/2 threads extending through nut.

(11) Install elevator quadrant door.

(12) Install elevator fairing.

C. Removal – Elevator Torque Tube

(1) Remove elevator (see Removal/Installation – Elevator).

(2) Remove nuts, washers, and bolts securing torque tube to elevator. Remove torque tube.

D. Installation – Elevator Torque Tube

(1) Position torque tube on elevator. Install bolts, washers, and nuts securing torque tube to elevator. Torque to 20 to 25 inch-pounds (2.26 to 2.82 N·m),

(2) Install elevator.

2. Adjustment/Test – Elevator

A. Adjustment – Elevator Downspring

(1) Adjust elevator downspring to obtain column loads measured as follows:

(a) Place a Hand Held Force Gauge with a range between 0 and 100 Lbs., directly behind on of the hand grips on the pilot or copilot yoke.

(b) Place the palms of both hands around the gauge to pull the yoke aft, so that the force required to move the column is transmitted evenly from hands to gauge. Do not grip front of yoke with thumbs, or a squeezing force will also register on the gauge.

(c) Pull control column smoothly aft. Record force at neutral (vertical) position.

(d) Pull control column all the way aft. Then let spring pull column smoothly forward. Record force at neutral (vertical) position.

(e) Add forces determined in steps (a) and (b) above. Divide by two. Result shall be 41 Lbs., plus or minus 4 Lbs., for aircraft not complying with SB27-032; and 37 Lbs., plus 1 minus 0 Lbs., for aircraft complying with SB27-032.

(f) If force is below required range, adjust tension at the downspring / quadrant link adjustment.

4. Inspection/Check – Elevator

A. Inspection – Elevator

(1) Open all access panels necessary to gain access to system components.

(2) Operate system through full range of travel and check for unusual noise, freedom of operation, and correct travel of elevator.

(3) Inspect cables, cable pulleys, fairleads, and pressure seals (see Section 27-00-00).

(4) Inspect quadrant (see Section 27-30-04).

(5) Check push-pull rods and torque tubes.

(6) Check cables for correct tension.

(7) Check control column for looseness of bearings at pivot points by moving column left and right.

(8) Inspect all hinges and hinge brackets for damage, cracks, corrosion wear, defective bearings, and security.

(9) Inspect elevators for damage, cracks, corrosion, loose or missing rivets, cracked or buckled spars, ribs or stringers, and separation of metal-bonded structure.

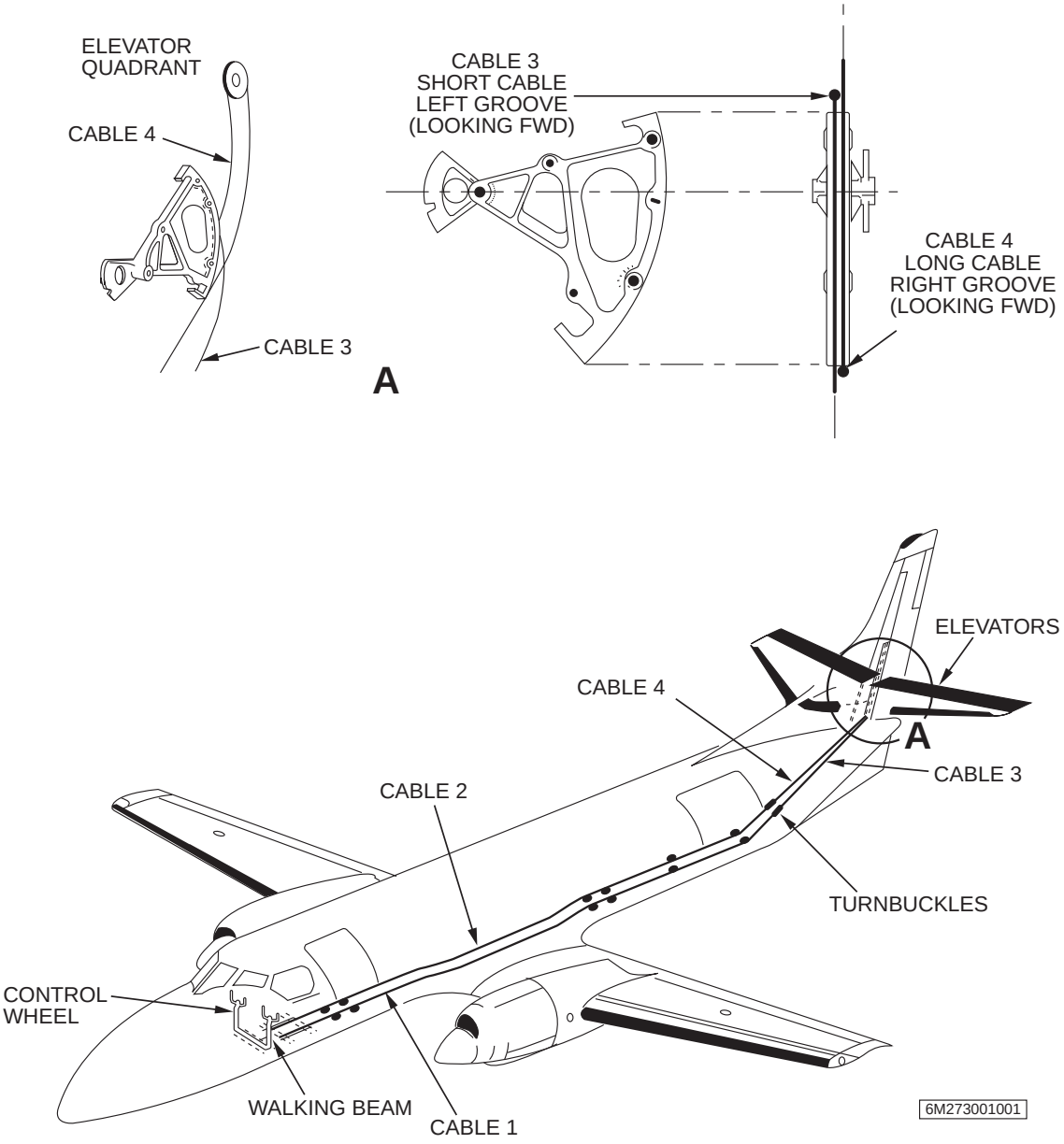
(10) Check elevator free play. Verify that deflection does not exceed 0.29 inches measured at inboard end.

ELEVATOR CONTROL CABLES – DESCRIPTION AND OPERATION

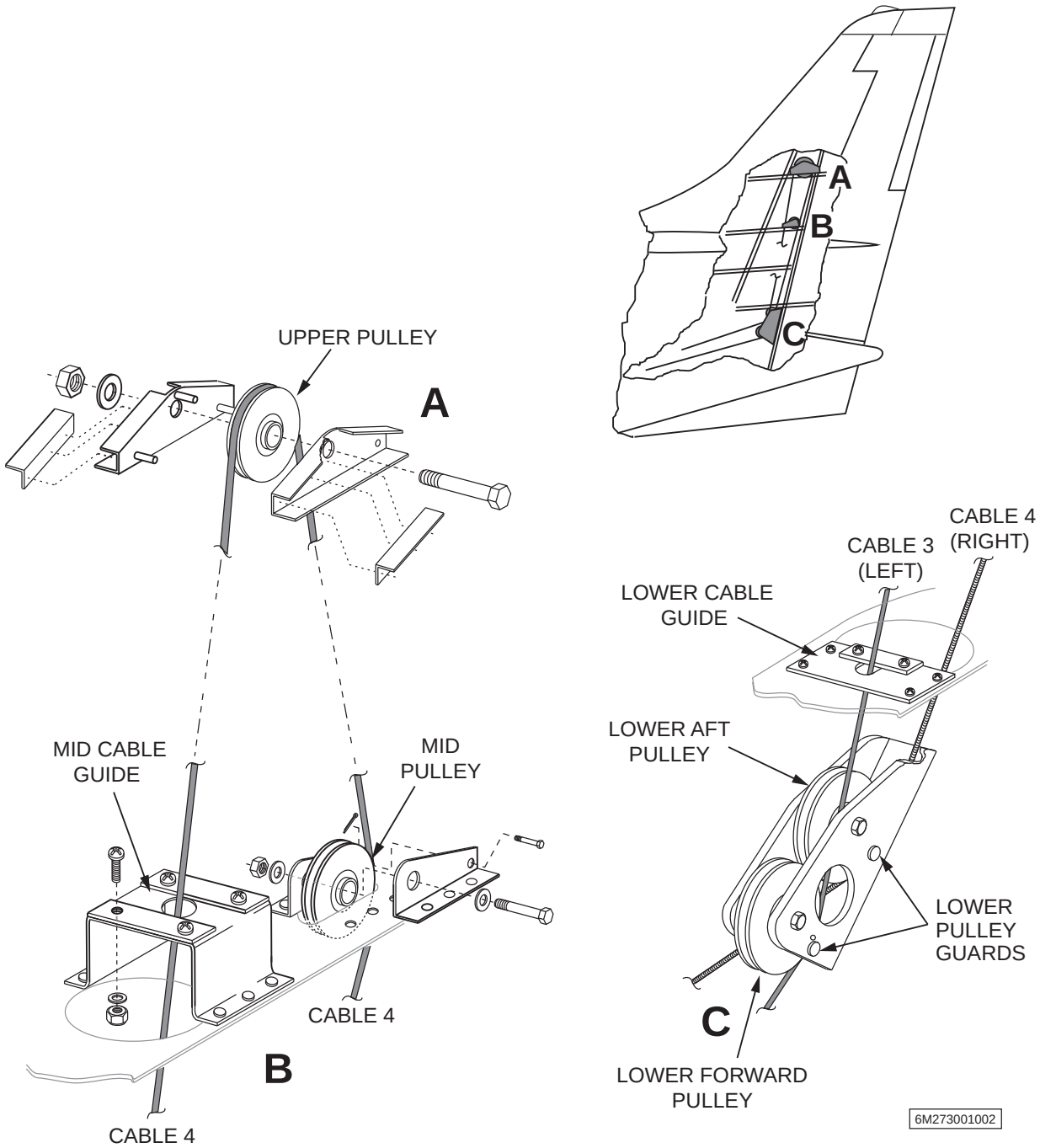
1. General

A. The elevator control system is comprised two sections, forward and aft. The forward section of the cable system (left cable 1 and right cable 2) are routed from the walking beam (located under the cockpit flooring) to turnbuckles in the tail section aft of the interior bulkhead.

B. The aft section (left cable 3 and right cable 4) continue from the turnbuckles aft of interior bulkhead and terminate at the elevator quadrant located in the vertical stabilizer.



Elevator Control Cables
Figure 1



Elevator Control Cables 3 and 4 Routing
Figure 2

ELEVATOR CONTROL CABLES – MAINTENANCE PRACTICES

1. Removal/Installation – Elevator Control Cables

A. Removal – Elevator Cables 1 and 2, Forward of Turnbuckles (Figure 1)

(1) Remove pilot or copilot seat (whichever cable is being removed).

(2) Remove pilot/copilot compartment floorboard to gain access to walking beam.

(3) Remove floorboards to aft interior bulkhead.

(4) Remove aft interior bulkhead (T Model) or aft baggage compartment aft bulkhead (AT, TC Models) as applicable, to gain access to turnbuckle.

(5) Block cables aft of turnbuckle.

(6) Disconnect cable at turnbuckle.

(7) Disconnect cable at walking beam.

(8) Remove pulley guard and fairleads as necessary and remove cable.

B. Installation – Elevator Cables 1 and 2, Forward of Turnbuckles (Figure 1)

NOTE: Cut the flight control cable to length by adding or subtracting the terminal allowance.

(1) Swage fitting on to cut end of cable IAW standard industry practices.

(2) Be sure swaged fittings show no visual evidence of cracking, and surface of swaged area is free of sharp dimensional changes.

(3) Position cable and replace fairleads and pulleys as necessary. Be sure cable is not twisted.

(4) Connect cable to walking beam.

(5) Connect cable to turnbuckle.

(6) Remove cable blocks and adjust tension (see chart, Section 27-00-00).

CAUTION: AFTER ALL CABLE RIGGING IS COMPLETED, ENSURE DISTANCE BETWEEN CONTROL COLUMN AND INSTRUMENT PANEL IS 2.25 ± 0.25 INCHES (57.15 ± 6.35 MM) WITH COLUMN IN FULL FORWARD POSITION. ADJUST CONTROL ROD LOCATED UNDER RH COCKPIT FLOORBOARD, F.S. 98.56, IF NECESSARY.

(7) Perform elevator travel check IAW 27-00-00.

(8) Install aft interior bulkhead (T Model) or aft baggage compartment aft bulkhead (AT, TC Models) as applicable.

(9) Install pilot/copilot compartment floorboards.

(10) Install pilot/copilot seats(s).

C. Removal – Elevator Cables 3 and 4, Aft of Turnbuckles (Figures 1 and 2)

(1) Remove aft interior bulkhead (T Model) or aft baggage compartment aft bulkhead (AT, TC Models) as applicable, to gain access to turnbuckle.

(2) Block elevator cable to be replaced forward of turnbuckle.

(3) Remove elevator quadrant access door located on right side of vertical stabilizer.

NOTE: Looking forward, cable on left side of aircraft is in left groove of quadrant and is fastened to top of quadrant (see Figure 1 Item A, cable 3). Cable from right side of aircraft travels up to pulley above quadrant, comes down into right groove of quadrant and is attached to bottom of quadrant (see Figure 1 Item A, cable 4).

(4) Unfasten cable to be replaced at turnbuckle.

(5) Removal cable 3 (left)

(a) Remove pulley guard from pulley (lowest of two in bracket just under quadrant).

(b) Unbolt cable keeper at top end (left groove) of quadrant.

(c) Lower cable and remove.

(6) Removal cable 4 (right)

(a) Remove pulley guard from pulley (uppermost of two in bracket just under quadrant).

(b) Unbolt cable keeper at bottom end (right groove) of quadrant.

(c) Work cable through upper pulley guide, over pulley located above quadrant, through lower pulley guide and through lower pulley.

(d) Lower cable and remove.

D. Installation – Elevator Cables 3 and 4, Aft of Turnbuckles (Figures 1 and 2)

(1) Preparation

(a) Cut the flight control cable to length by adding or subtracting the terminal allowance.

(b) Swage fitting on to cut end of cable per standard industry practices.

(c) Be sure swaged fittings show no visual evidence of cracking, and surface of swaged area is free of sharp dimensional changes.

(d) Position cable and replace fairleads and pulleys as necessary. Be sure cable is not twisted.

(2) Installation – Cable 3 (left)

(a) Connect cable 3 to top end (left groove) of quadrant with cable keeper.

(b) Position cable through lower cable guide and behind lower forward pulley (lowest of two in bracket immediately under quadrant).

CAUTION: ENSURE THAT CABLE IS ROUTED UNOBSTRUCTED IN THE GROOVE OF THE PULLEY. CABLE MUST NOT COME IN CONTACT WITH ANY HARDWARE OR AIRCRAFT STRUCTURE.

(c) Install pulley guards for Cable 3 lower forward pulley guard.

(d) Attach other end of cable to turnbuckle.

(e) Insert 3/16 inch diameter rig pin (through elevator stop control and into quadrant).

(f) Remove cable block.

(g) Adjust cable tension (see chart, Section 27-00-00).

CAUTION: AFTER ALL CABLE RIGGING IS COMPLETED, ENSURE DISTANCE BETWEEN CONTROL COLUMN AND INSTRUMENT PANEL IS 2.25 ± 0.25 INCHES (57.15 ± 6.35 MM) WITH COLUMN IN FULL FORWARD POSITION. ADJUST CONTROL ROD LOCATED UNDER RH COCKPIT FLOORBOARD, F.S. 98.56, IF NECESSARY ADJUST CONTROL ROD LOCATED UNDER RH COCKPIT FLOORBOARD, F.S. 98.56.

(h) Perform elevator travel check IAW 27-00-00.

(i) Install elevator quadrant door.

(j) Replace aft interior bulkhead (T Model) or aft baggage compartment aft bulkhead (AT, TC Models) as applicable.

(3) Installation – Cable 4 (right)

(a) Position Cable 4 through lower aft pulley and mid pulley, then over upper pulley and down through mid cable guide.

(b) Connect cable to lower end (right groove) of quadrant and secure to quadrant with cable keeper.

CAUTION: ENSURE THAT CABLE IS ROUTED UNOBSTRUCTED IN THE GROOVE OF THE PULLEYS. CABLE MUST NOT COME IN CONTACT WITH ANY HARDWARE OR AIRCRAFT STRUCTURE.

(c) Install pulley guards for Cable 4 at lower aft pulley, mid pulley and upper pulley.

(d) Attach other end of cable to turnbuckle.

(e) Insert 3/16 inch right pin (through elevator stop control and into quadrant).

(f) Remove cable block.

(g) Adjust cable tension (see chart, Section 27-00-00).

CAUTION: AFTER ALL CABLE RIGGING IS COMPLETED, ENSURE DISTANCE BETWEEN CONTROL COLUMN AND INSTRUMENT PANEL IS 2.25 ± 0.25 INCHES (57.15 ± 6.35 MM) WITH COLUMN IN FULL FORWARD POSITION. ADJUST CONTROL ROD LOCATED UNDER

RH COCKPIT FLOORBOARD, F.S. 98.56, IF NECESSARY ADJUST CONTROL ROD LOCATED UNDER RH COCKPIT FLOORBOARD, F.S. 98.56.

(h) Perform elevator travel check IAW 27-00-00.

(i) Install elevator quadrant door.

(j) Replace aft interior bulkhead (T Model) or aft baggage compartment aft bulkhead (AT, TC Models) as applicable.

CONTROL COLUMN – DESCRIPTION AND OPERATION

1. General

A. The control column for all aircraft is basically the same. The only difference between the control column installed in the SA226-T series (short) aircraft and the SA226-AT and TC (long) aircraft is the addition of an elevator arm, a bob weight, and a control damper. The bob weight is added to the short aircraft to provide stability. The damper is installed to retain the control column and prevent any sudden movement of the elevators.

B. Removal and installation of the control wheel, aileron control cables and control column pulleys is covered in Section 27-10-01.

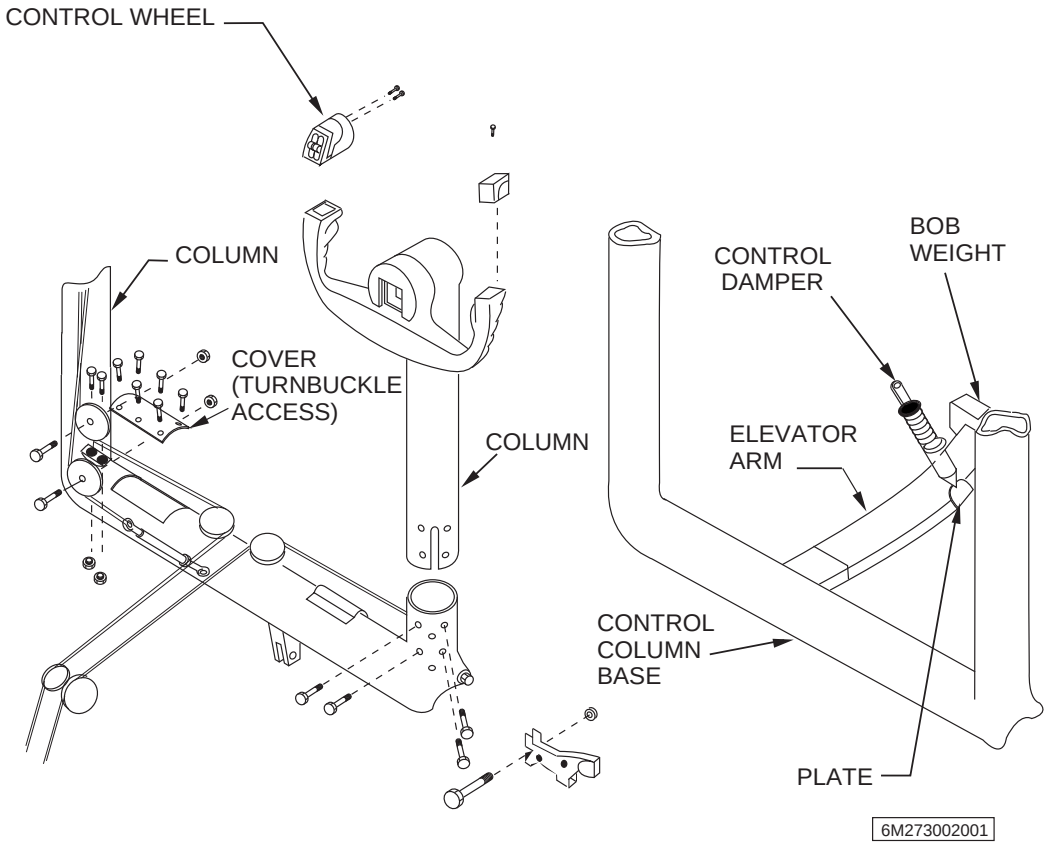
C. Control Column Damper

(1) The SA226-T series aircraft damper is connected to the elevator arm and the pedestal (see Figure 201). Its purpose is to prevent sudden movements of the elevators.

(2) The piston of the damper is attached to a bracket on the pedestal. The cylinder of the damper is attached to the elevator arm between the base of the yoke and the bob weight.

D. Control Column Roller Bearing

(1) A control column roller bearing is located at each end of the control column base (see Figure 203). Its purpose is to hold the column in position while allowing the column to rotate backward and forward, thereby activating the elevator cable system.

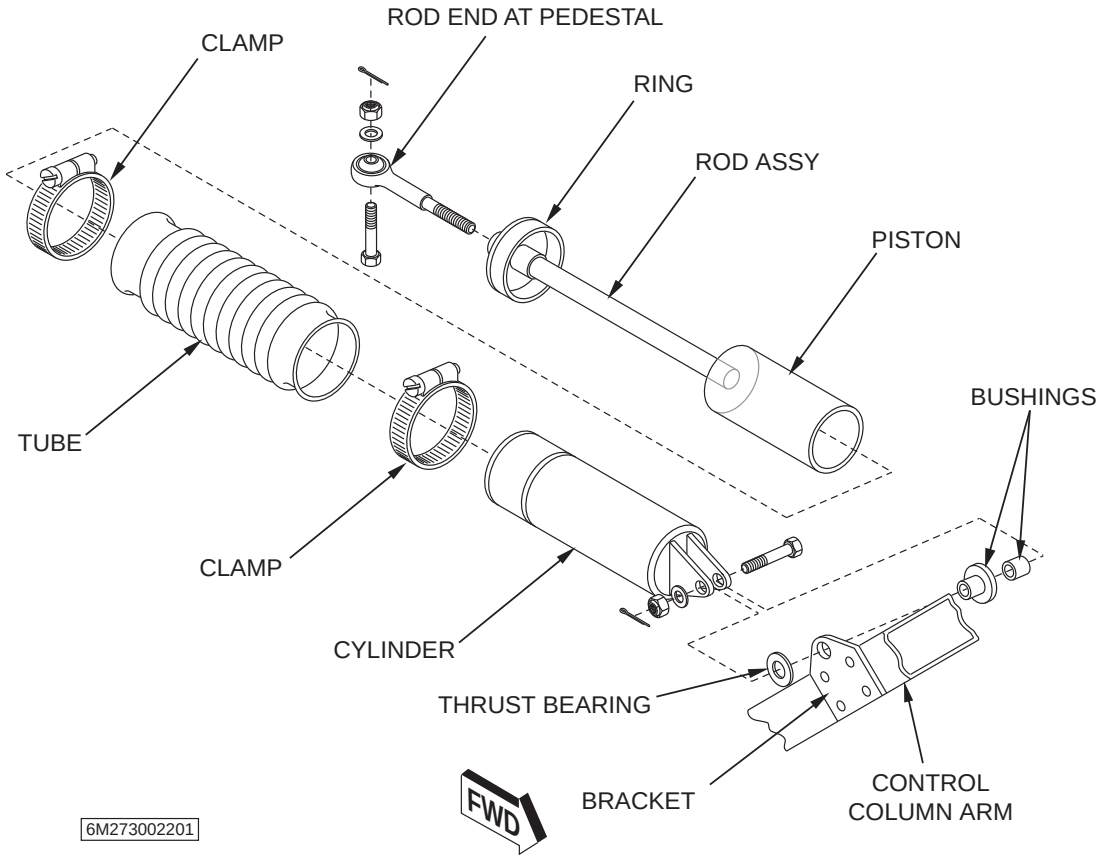


Control Column
Figure 1

1. Servicing – Control Column

A. Servicing – Control Column Damper

(1) Fill cylinder with Dow Corning Xiameter PMX-200, Dow Corning Xiameter PMX-0001, or ESCO E200 silicon fluid, viscosity grade at 25°C, 1000 centistokes (or equivalent). Fluid level is 1.50 inches (38.10 mm) from top of cylinder with piston inserted.



Control Column Damper
Figure 201

2. Removal / Installation – Control Column

A. Removal – Control Column Damper (Figure 201)

(1) Remove cotter pin, nut, washer, and bolt from rod end at pedestal.

(2) Maintain damper assembly in upright position so that fluid does not run out.

(3) Remove cotter pin, nut, washer, thrust bearing, and bushing attaching cylinder end to control column arm bracket.

(4) Remove damper assembly from behind pedestal.

B. Installation – Control Column Damper (Figure 201)

(1) Position damper assembly at bracket on control column arm with bushing and thrust bearing between cylinder end and bracket as shown.

(2) Insert bolt and washer; secure with nut and cotter pin.

(3) Position rod end at bracket on pedestal.

(4) Insert bolt and washer; secure with nut and cotter pin.

C. Removal – Elevator Walking Beam (Figure 202)

(1) Remove pilot and copilot seats, cockpit carpet and floorboards.

(2) Locate elevator walking beam between F.S. 98.06 and 107.06.

(3) Remove cotter pins, nuts, washers and bolts (item 15) securing elevator cables to eyelets (item 16) at F.S. 116.56.

(4) Remove cotter pins, nuts, washers and bolts securing stick pusher cables to eyelets at F.S. 116.56.

(5) Remove screws (item 1), spacers (item 13), nuts (item 2), washers and cable guards (item 4) from walking beam (item 6). Remove stick pusher cables from pulleys (item 12).

(6) Remove screws (item 3), spacers (item 14) and cable guards (item 5) from walking beam (item 6). Remove elevator cables from pulleys (item 11).

(7) Remove control rod (item 9) retaining hardware (item 10) and disconnect control rod from walking beam (item 6).

(8) Remove nut (item 17), washer (item 18), bushing (item 19) and pivot bolt (item 8) from walking beam mounting bracket (item 7).

(9) Making sure to retain the two bearings (item 20) and spacer assembly (item 21) inside the walking beam (item 6) pivot bore, remove walking beam from aircraft. Avoid damaging aileron control cables while removing walking beam.

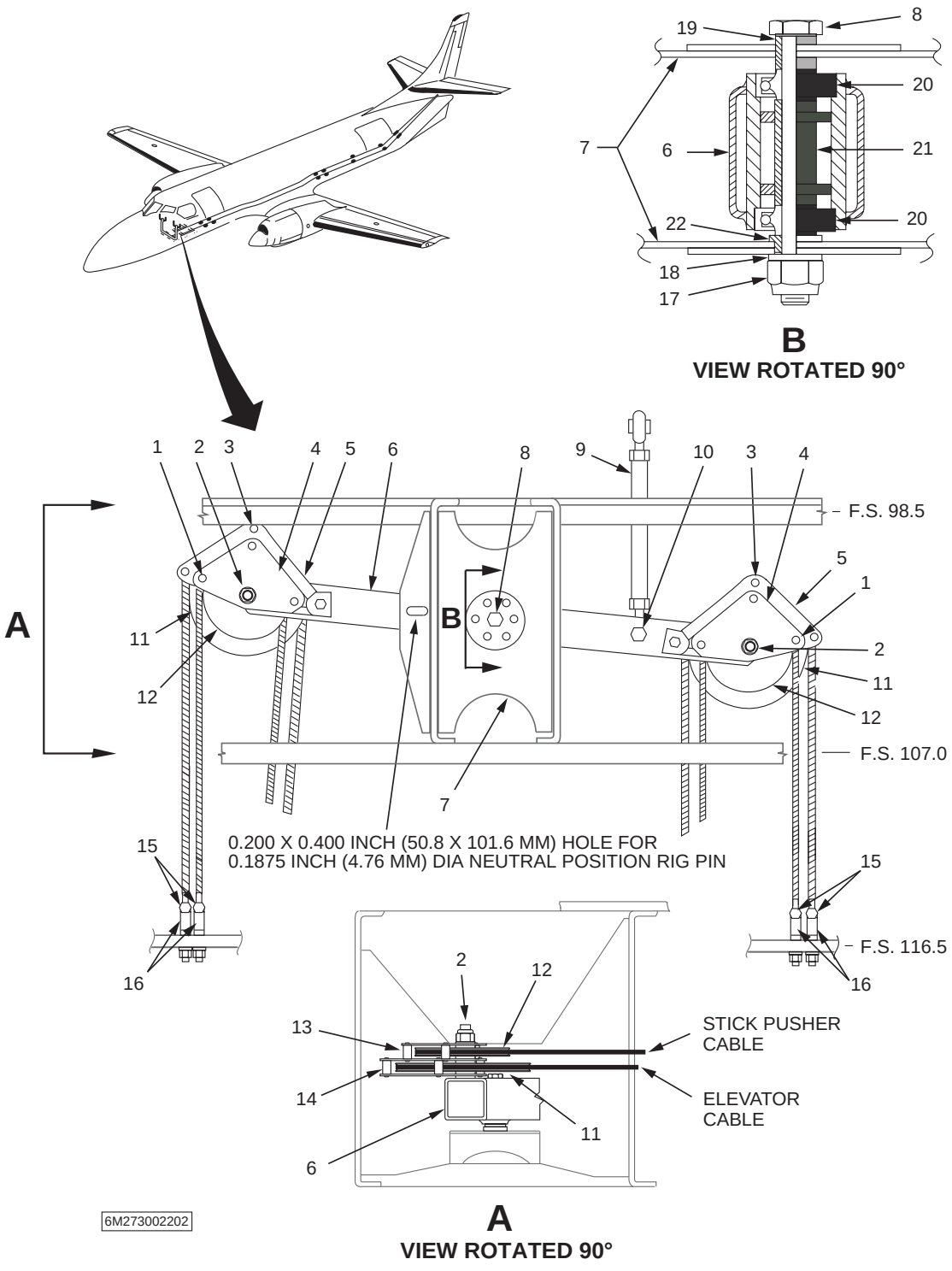
(10) The bushing (item 22) in bottom flange of mounting bracket (item 7) should remain with bracket.

D. Installation – Elevator Walking Beam (Figure 202)

(1) Verify bushing (item 22) in bottom flange of mounting bracket (item 7) shows no signs of wear. Replace if necessary.

(2) Position walking beam (item 6) into mounting bracket.

(3) Secure walking beam (item 6) to mounting bracket (item 7) by inserting pivot bolt (item 8) through bushing (item 19), mounting bracket top flange, upper bearing (item 20), spacer assembly (item 21), lower bearing (item 20), bushing in mounting bracket bottom flange (item 22), washer (item 18) and nut (item 17). Torque nut 50 to 70 inch-pounds (5.65 to 7.91 Nm).



Elevator Walking Beam Assembly
Figure 202

(4) Connect control rod (item 9) to walking beam (item 6) and secure with hardware (item 10) specified in SA226 IPC, 27-30-02 or SB226-27-060 (latest revision) as applicable.

(a) For Pre SB226-27-060 aircraft, torque walking beam control rod retaining nut 50 to 70 inch pounds (5.65 to 7.91 Nm).

(b) For Post SB226-27-060 aircraft, use washer(s) on either side of rod end bearing as required for clamp up, while ensuring nut does not ride on shank of bolt. Torque nut 50 to 70 inch pounds (5.65 to 7.91 Nm) and install cotter pin.

(5) Install elevator cables onto pulleys (item 11). Position cable guards (item 5) over pulleys (item 11). Position spacers (item 14) and secure with screws (item 3).

(6) Install stick pusher cables onto pulleys (item 12). Position cable guards (item 4) over pulleys (item 12). Position spacers (item 13) and secure with screws (item 1).

(7) Install pulley retaining nuts (item 2) and washers. Torque retaining nuts (item 2) 50 to 70 inch-pounds (5.65 to 7.91 Nm).

(8) Secure terminal fork end of stick pusher and elevator cables to eyelets (item 16) at F.S 116.56 with bolts (item 15), washers, nuts and cotter pins.

(9) Install cockpit floorboards, carpet, pilot seat and copilot seat.

E. Removal/Installation - Control Column Pivot Bearings

(1) Refer to Service Bulletin 226-27-060 or 226-26-061 as applicable to your aircraft configuration for control column pivot bearing removal and installation instructions.

3. Adjustment/Test – Elevator Control

A. Control Column Clearance

(1) Adjust pilot and copilot seats to full forward and full up position. Pull control column full aft. Column must not contact either seat frame. Column may contact seat covering as long as travel is not limited.

(2) Allow control column to travel full forward and verify distance between forward top surface of control column and instrument panel is 2.25 ± 0.25 inches (57.15 ± 6.35 mm).

(3) If necessary, adjust control rod (Figure 202, Item 9) located under RH cockpit floorboard, at F.S. 98.56 to obtain specified clearances and verify elevator travel is within tolerances specified in section 27-00-00.

B. Control Column Side Play (See Figure 203)

NOTE: Service Bulletin 226-27-061 improves the control column pivot design that adds strength and maintainability by replacing the life limited pivot bearings pins a and mount fitting which provide terminating action for Airworthiness Directive 2015-19-10.

(1) Grab either control wheel and apply force in both lateral directions to check for side play at control column pivot attachment.

(2) If side play is evident, shim either side of control column IAW the following steps:

(a) Peel back carpet at base of pilot or copilot control column.

(b) Remove screws securing column closeout boot and slide boot clear.

(c) Remove industrial tape from base and access slot.

(d) Remove floorboard panel around control column IAW Section 53-30-20.

(e) For Pre SB 226-27-061 aircraft:

1 Remove hardware securing doubler to frame.

2 Remove bearing and doubler through access slot.

3 Remove nut and washer from bearing and remove doubler.

NOTE: To remove nut from bearing insert common screwdriver into slot at large end of bearing (to keep bearing from rotating) while removing nut.

4 Shim bearing by adding AN960C416 and/or AN960C416L washer(s) to shaft of bearing.

5 Install bearing onto doubler and secure with washer and nut.

6 Install bearing and doubler assembly by inserting assembly into access slot.

7 Position bearing into control column and secure doubler to frame with attach bolts and washers.

(f) For Post SB 226-27-061 aircraft:

1 Remove attach bolts and washers securing fitting to frame.

2 Remove pin and fitting through access slot.

3 Shim pin by adding AN960C816 and/or AN960C816L washer(s) to shaft of pin.

4 Position fitting onto pin and secure fitting with attach bolts and washers.

5 Install pin and fitting assembly by inserting assembly into access slot.

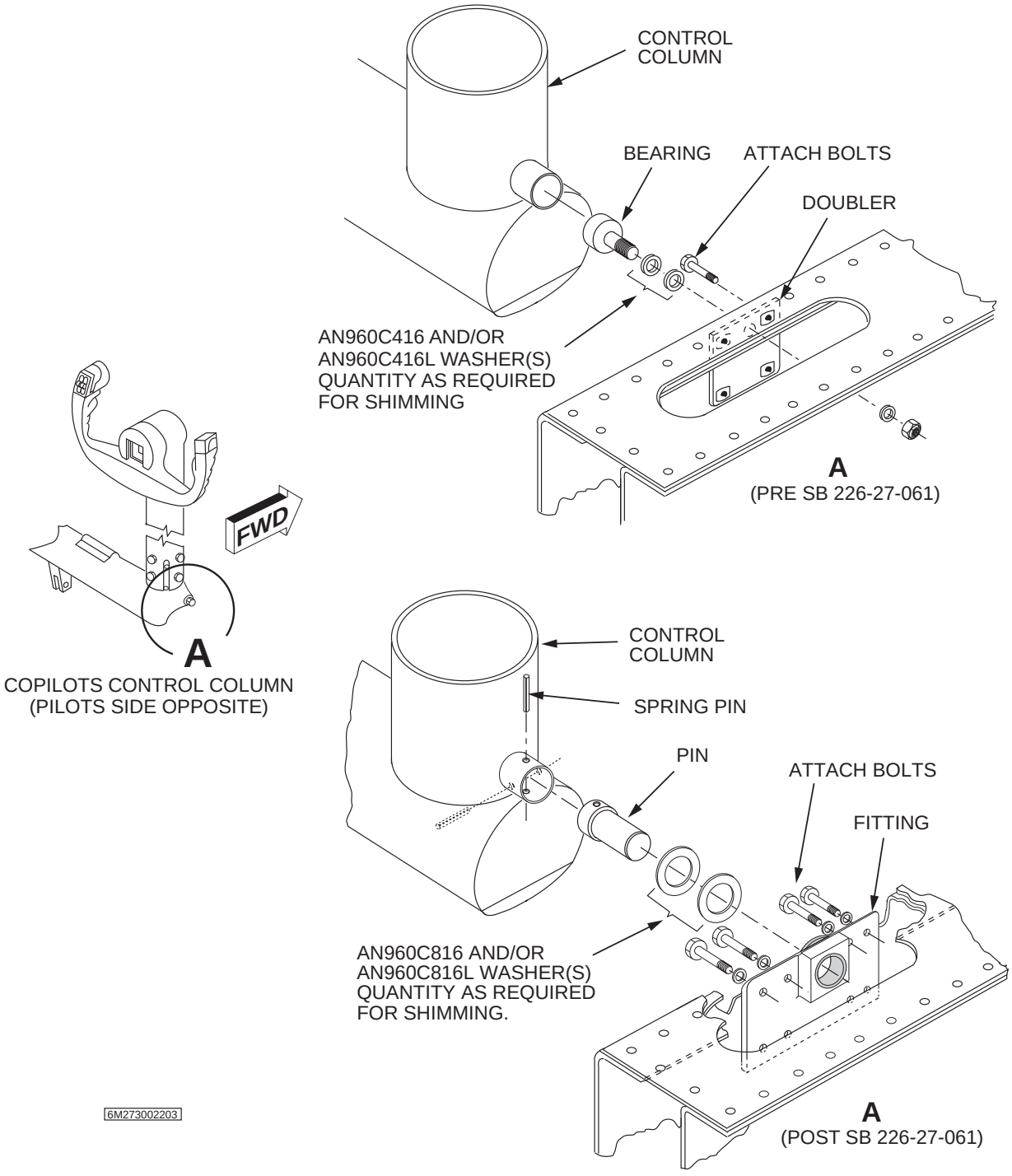
6 Position pin into control column, align bearing cup/pin holes and install spring pin.

7 Secure fitting to frame with attach bolts and washers.

(g) Functional check column for smoothness of operation.

(h) Install floorboard panel around control column IAW Section 53-30-20.

(i) Apply industrial tape and install boot and carpet.

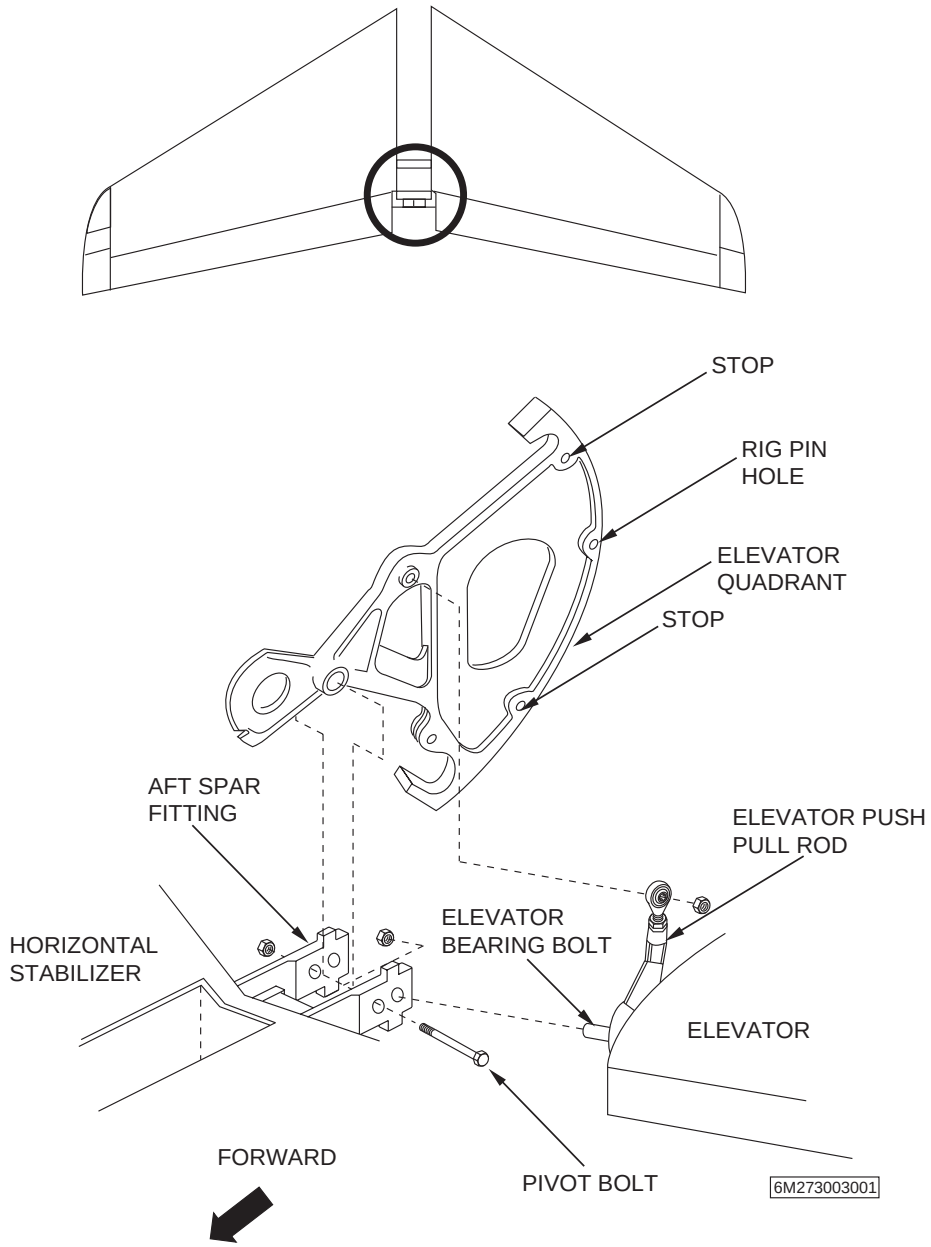


Control Column Shimming
Figure 203

ELEVATOR QUADRANT – DESCRIPTION AND OPERATION

1. General

A. The elevator quadrant is located within the vertical stabilizer of the aircraft. The quadrant is a cable controlled bellcrank. The two elevators are attached to the quadrant. The cables that control the movement of the quadrant run the entire length of the aircraft and attach to a walking beam assembly located under the cockpit floorboards. The control column activates the walking beam. When the control column is pulled back, the control cables raise the quadrant which subsequently raises the elevators. The opposite effect is realized when the control column is pushed forward.



Elevator Quadrant
Figure 1

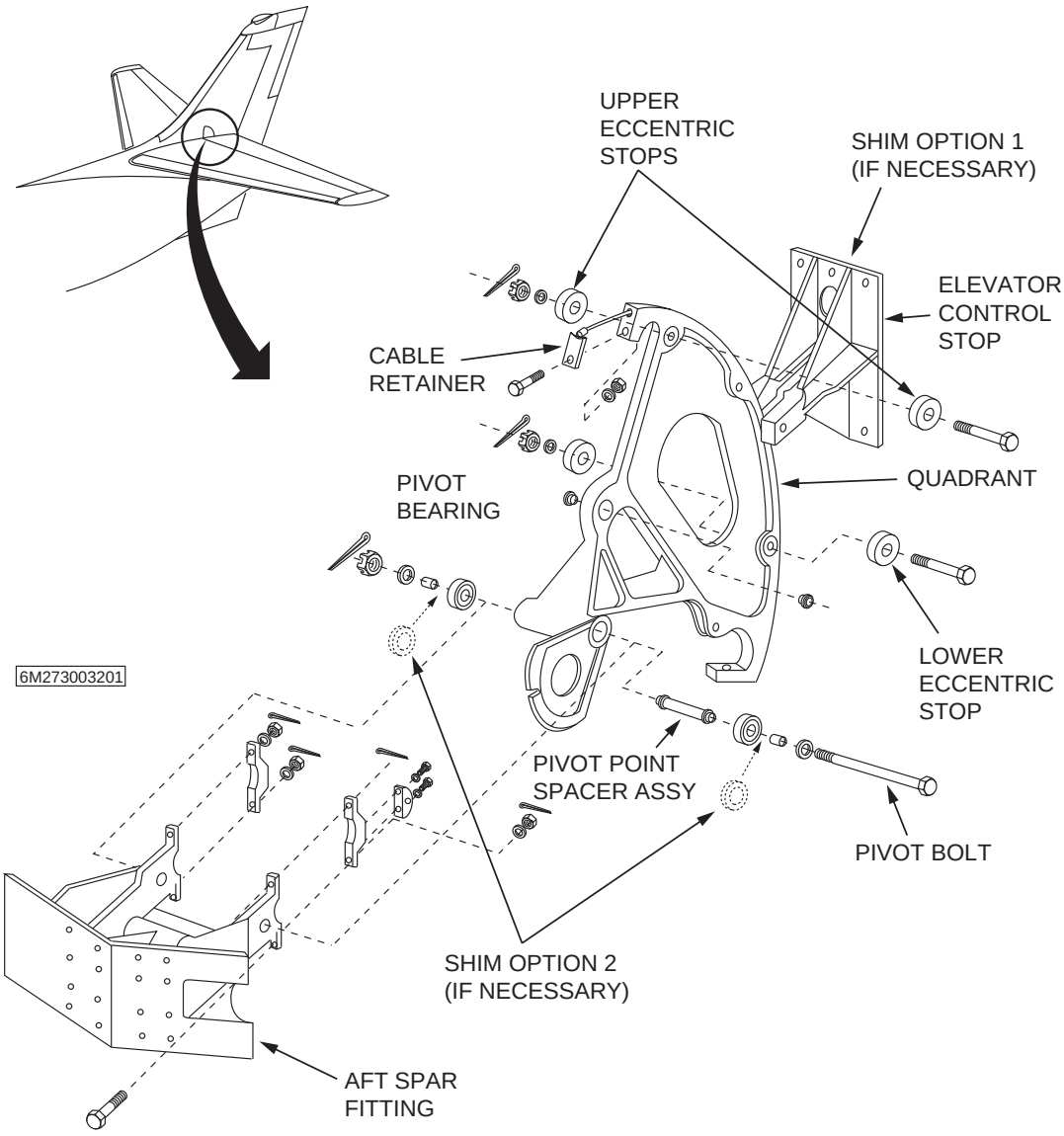
1. Removal / Installation – Elevator Quadrant

A. Removal – Elevator Quadrant (Figure 201)

NOTE: Although the quadrant can be removed with the elevators attached, time and effort will be saved by first removing the elevators.

(1) Engage gust lock and pull control column back until elevators lock.

(2) Remove aft interior bulkhead (T Model) or aft baggage compartment, aft bulkhead (AT, TC Models).



Elevator Control Quadrant Installation
Figure 201

(3) Block elevator cables forward of turnbuckles.

(4) Adjust turnbuckles to release tension on cables.

(5) Remove elevator quadrant access door on right side of vertical stabilizer at horizontal stabilizer pivot trunnion.

(6) Remove both elevators IAW Section 27-30-00.

(7) Remove screws securing cable retainers to top and bottom of quadrant. Remove retainers and cables.

(8) Tie or tape cables to inside of vertical stabilizer, to keep cables in position and handy for reinstallation.

(9) Support control column and disengage gust lock.

(10) SA226 – AT and TC aircraft only (see Figures 202 and 203):

(a) Allow quadrant to pivot down at rear (cable grooves) to release some of downspring tension.

(b) Before disconnecting quadrant, disconnect forward end of downspring to relieve tension on quadrant mounting bolts. Disconnect downspring as follows:

1 For aircraft on which SB27-032, has not been accomplished, gain access to forward end of downspring by removing forward most access door on right side of vertical stabilizer.

2 For aircraft complying with SB27-032, gain access to forward end of downspring by removing dorsal fin.

(c) Using a spring stretcher or similar tool, (make from a length of 3/8 inch cable with handle at one end and clevis, bolt, and nut arrangement at other end) attach clevis end of tool to forward end of spring.

(d) While pulling on spring to relieve tension, remove hardware securing spring to bracket.

(e) Allow spring to retract slowly. Leave stretching tool attached to spring for reinstallation.

(f) Remove hardware securing downspring / quadrant link to quadrant.

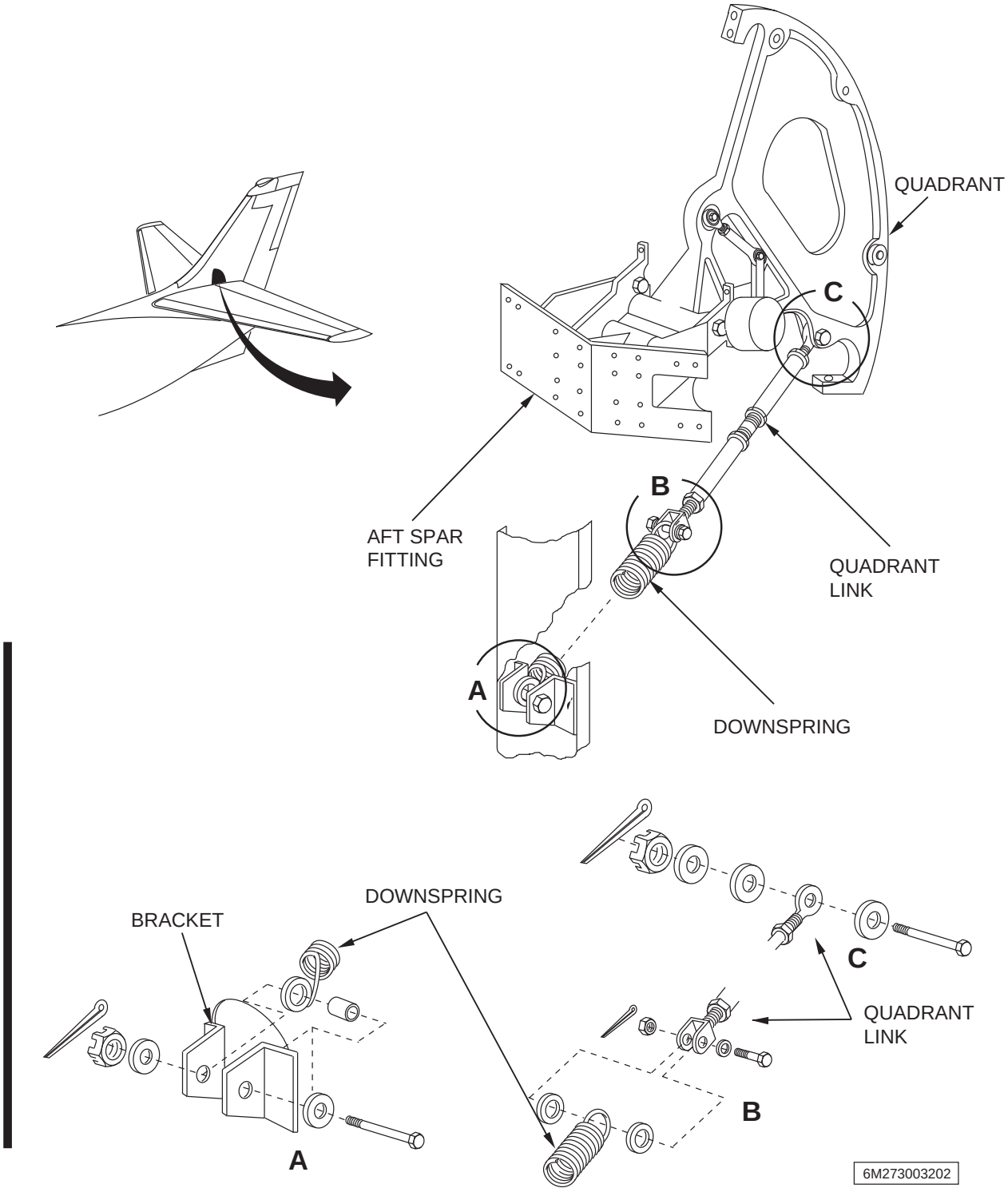
1 For aircraft on which SB27-032 has not been accomplished, remove bolt, two washers, bushing, nut, and cotter pin.

2 On aircraft complying with SB27-032, remove bolt, bushing, nut, cotter pin, and stop (bracket).

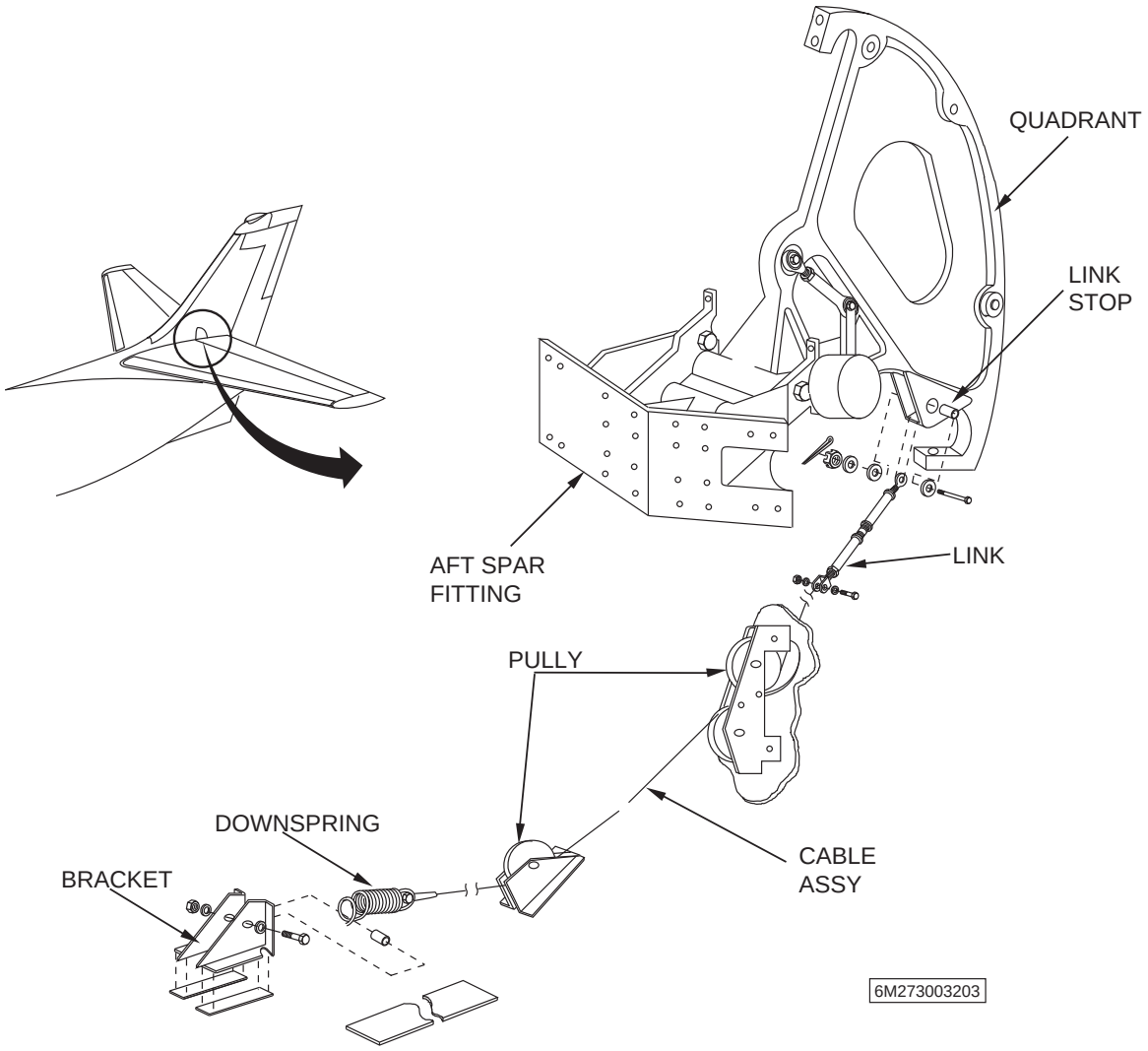
(11) Remove hardware securing upper eccentric stops to quadrant and remove stops (Refer to SA226 IPC for Pre and Post SB226-27-058 attaching hardware configurations). Allow quadrant to pivot down and out of groove in elevator control stop.

(12) Remove cotter pin, nut, washers, bushings, bearings, and spacer assembly securing quadrant to aft spar fitting at quadrant pivot point.

(13) Remove quadrant through access door.



Elevator Downspring Installation (Pre SB27-032)
Figure 202



Elevator Downspring Installation (Post SB27-032)
Figure 203

B. Installation – Elevator Quadrant (Figure 201)

(1) Position elevator quadrant and secure to pivot point using bolt, washers, bushings, bearings, spacer assembly, nut, and cotter pin.

(2) Pivot quadrant up and into groove in control stop.

(3) Verify 3/16 inch (4.76 mm) rig pin can be inserted into quadrant and control stop, and proper alignment of quadrant in control stop with 0.030 inch (0.762 mm) minimum clearance between quadrant and control stop. If necessary, two shim options are provided:

(a) Shim Option 1 - 0.080 x 3.25 x 5.10 inch (2.032 x 82.55 x 129.54 mm) laminated 0.003 inch (0.0762 mm) peel shim (1145-H19 alloy/MIL-S-22499) may be installed between control stop and spar as required.

(b) Shim Option 2 - Use AN960-416 or AN960-416L washer on pivot bolt between aft spar fitting and quadrant as required to align control stop and quadrant.

(4) Install upper eccentric stops with knurled (cross cut lines) side against quadrant (Refer to SA226 IPC for Pre and Post SB226-27-058 attaching hardware configurations).

(5) On SA226-AT and SA226-TC aircraft only, install elevator downspring. Check downspring/quadrant link for proper adjustment as follows:

(a) Connect downspring/quadrant link to quadrant.

1 For aircraft on which SB27-032 has not been accomplished, secure link to quadrant using bolt, two washers, bushing, nut, and cotter pin.

2 On aircraft complying with SB27-032, secure link to quadrant using link stop (bracket), bolt, bushing, nut, and cotter pin.

NOTE: Ensure heel of link stop contacts quadrant when bolt is tightened.

(b) Adjust link assembly to obtain the following dimensions:

1 For AT and TC aircraft not complying with SB27-032, set link to mid-point of adjustment:

2 For AT and TC aircraft complying with SB27-032, measure overall length of link. Center of bolt to center of bolt should measure 4.50 inches.

(c) Using spring stretching tool, stretch spring forward and attach spring using bolt, washers, spacer, nut, and cotter pin.

(6) Insert 3/16 inch (4.76 mm) rig pin through control stop and quadrant to hold quadrant secure.

(7) Position cables onto quadrant and secure cable retainers to quadrant with screws.

NOTE: Looking forward, cable from below goes in left groove and attaches to top of quadrant. Cable from above goes in right groove and attaches to bottom of quadrant (see Figure 201).

(8) Remove rig pin.

(9) Remove cable blocks and adjust tension of cables at turnbuckles.

(10) Replace elevators (see Removal / Installation – Elevator, this chapter).

(11) Check elevators for smooth operation and full operational travel.

(12) On SA226-AT and TC aircraft only:

(a) Check adjustment of control column loads as follows:

1 Place a hand held force gauge with a range between 0 and 100 lbs. (45.36 kg), directly behind one of the handgrips on the pilot or copilot yoke.

2 Place the palms of both hands around the gauge to pull the yoke aft, so that the force required to move the column is transmitted evenly from hands to gauge. Do not grip front of yoke with thumbs or a squeezing force may also register on the gauge.

3 Pull control column smoothly aft. Record force at neutral (vertical) position.

4 Pull control column all the way aft, then let spring pull column smoothly forward. Record force at neutral (vertical) position.

5 Add forces recorded in steps 3 and 4 divide by two. Result shall be 41 lbs. (18.60 kg), plus or minus 4 lbs. (1.81 kg), for aircraft not complying with SB27-032; and 37 lbs. (16.78 kg), plus 1 (.45) / minus 0 lbs., for aircraft complying with SB27-032.

6 If force is below required range, adjust tension at downspring/quadrant link adjustment.

(13) Replace aft interior bulkhead (T Model) or aft baggage compartment, aft bulkhead (AT, TC Models).

(14) Replace access door(s), and dorsal fin.

2. Approved Repairs – Elevator Quadrant

A. Repair – Elevator Quadrant Stops

NOTE: With each lowering or raising of the elevator, force is applied to the elevator up and elevator down stops which may, in time cause elongation of the stop bolt holes in the quadrant. Should this occur the following corrective steps may be taken:

(1) Gain access to elevator quadrant by removing quadrant access door on right side of vertical stabilizer.

(2) Run elevator up or down to gain access to quadrant stops.

(3) Remove bolt, washer, and nut attaching stop and remove stop.

(4) Enlarge stop hole in quadrant to 0.375 inch (9.525 mm) diameter.

(5) Install NAS76A4-103 bushing using Loctite 601.

(6) Install stop.

(7) Repeat steps (2) through (6) for remaining stop.

(8) Install quadrant access door.

B. Shimming Aft Spar Fitting/Elevator Torque Tube Bearing (Figure 204)

(1) Remove elevator (see Section 27-30-01).

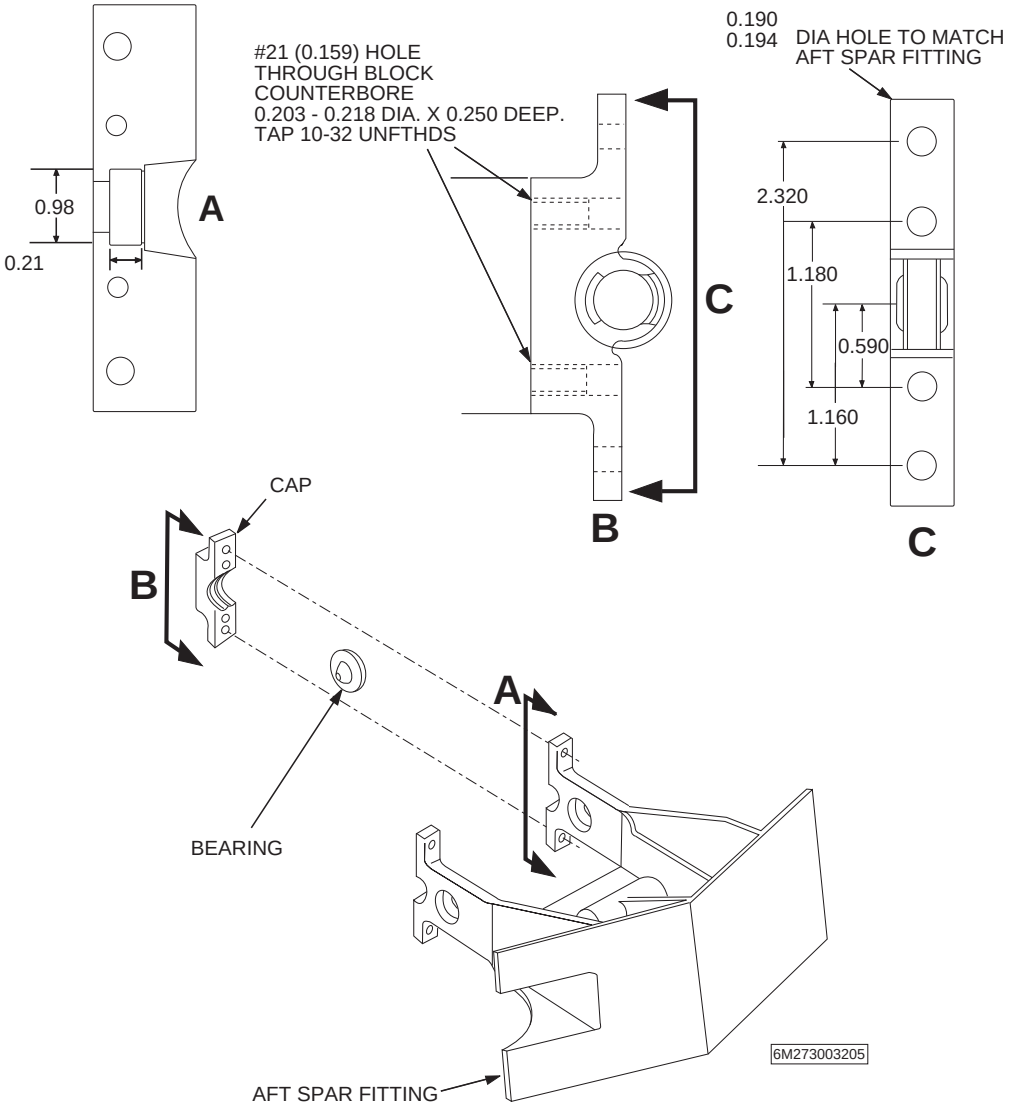
(2) Remove nut and washer securing elevator torque tube bearing and remove bearing. Retain spacer.

(3) Cut a piece of Plastigage (type APR-1) width of bearing.

(4) Place Plastigage diagonally into bearing groove in aft spar fitting.

(5) Replace bearing into groove and secure tightly with cap.

(6) Remove cap, bearing and Plastigage.



Aft Spar Fitting Bearing, and Bearing Cap
Figure 204

(7) Measure width of Plastigage. If width is less than .003 inch (.076 mm) bearing need not be shimmed.

(8) Peel laminated shim (Brass, 260-1/2 hard, MIL-22499) to half thickness of measurement obtained in step (7). Example: .003 Divided by 2 equals .0015.

(9) Cut two shim strips 0.0015 inch (0.0381 mm) thick (variable) by 0.21 inch (5.334 mm) wide, by 0.98 inch (24.892 mm) long. One strip for cap and one strip for aft spar fitting.

(10) Clean outside edge of bearing and inside grooves in cap and spar fitting with Scotch Grip Solvent #2, MPK, or equivalent.

(11) Thoroughly blend equal parts of Structural Adhesive, 1838A and 1838B.

(12) Apply a thin, even coat of adhesive to one side of each shim and to grooves in cap and spar fitting. Insert bearing into groove in spar fitting. Position cap over bearing and secure to spar fitting with two bolts and nuts. Curing time for adhesive is 12 to 18 hours at 73°F. If necessary, clean unhardened adhesive with cloth dampened with Scotch Grip Solvent #1.

(13) Remove cap and bearing.

(14) Install bearing onto elevator torque tube (against spacer) and secure with washer and nut.

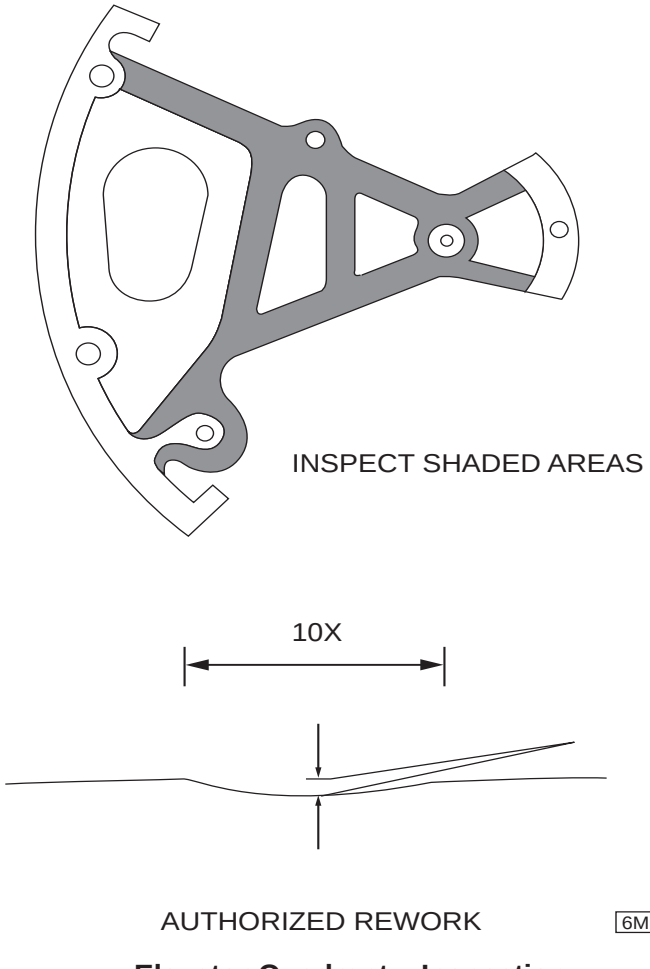
(15) Install elevator (see Section 27-30-01).

C. Replacement of Bearing Cap

NOTE: Bearing caps are received from vendor undrilled and must be drilled to match holes in aft spar fitting.

(1) Refer to Figure 204 for drill hole locations and dimensions.

(2) Use discarded cap or make a new template to ensure proper location, dimensions, and drilling angle of holes.



Elevator Quadrant – Inspection
Figure 205

3. Inspection / Check – Elevator Quadrant

A. Inspection – Quadrant

NOTE: It is recommended that the elevator quadrant installed in aircraft SA226-T201 through T287, SA226-AT001 through AT065, and SA226-TC201 through TC247 be visually inspected. Aircraft manufactured after these effectivities have a newly designed quadrant and do not require inspection. Installation of a new quadrant removes the inspection requirement. Inspection is accomplished as follows:

(1) Remove quadrant access door.

(2) Inspect all surfaces of the outer web of the elevator quadrant for nicks, tool marks, and cast chippings where stress concentration might occur, see Figure 205.

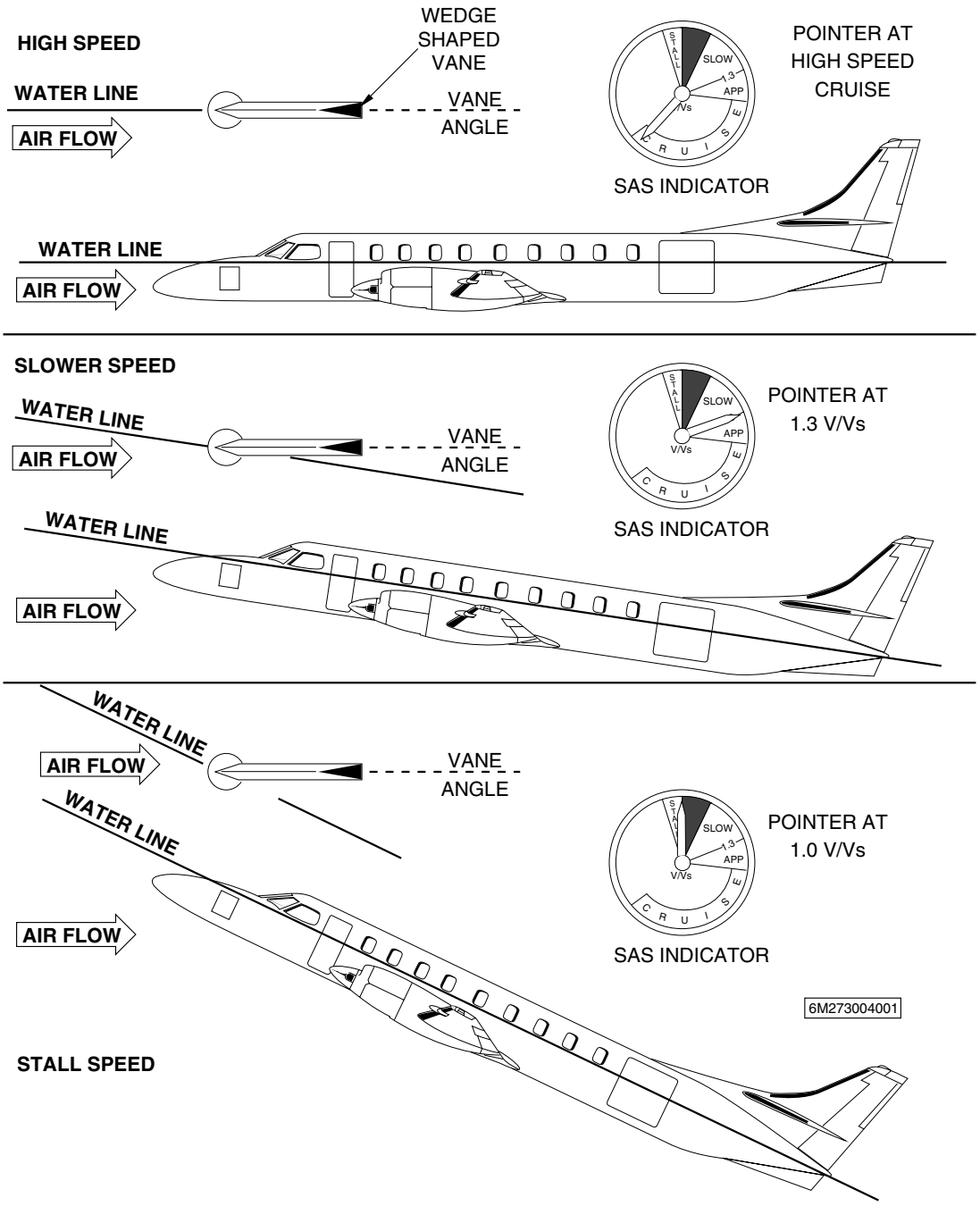
(3) Presence of any of the above described sharp marks requires a spot dye penetrant inspection for crack detection. This penetrant inspection may necessitate removal of the quadrant from the aircraft. Sharp marks with a depth greater than 0.10 inch (2.54 mm) and cracks of any size are cause for rejection. Sharp marks with a depth of 0.10 inch (2.54 mm) or less may be reworked by smoothing the surface with a small file and fine sandpaper. The length of the smoothed surface must be ten times the depth of the depression, see Figure 205. Treat the area with Dow No. 7 Dichromate and then apply epoxy primer (DuPont 825 and 826S or equivalent).

(4) Install quadrant access door.

SAS GENERAL – DESCRIPTION AND OPERATION

1. General

A. A stall avoidance system (SAS) is incorporated to warn the pilot both aurally and visually of an impending stall by use of a horn and an indicator. The system also provides for actual stall avoidance by means of a stick pusher which applies a forward force of approximately 60 pounds to the elevator control.



Stall Avoidance
Figure 1

B. A stability augmentation system is also installed on SA226-T aircraft. This is a low and gradually increasing forward force on the elevator control until the force reaches approximately 20 pounds. The combination of the stability augmentation and the stall avoidance system is known as SAS^2.

C. A SAS computer is used to, process information from an angle of attack (AOA) transmitter and a flap position compensator. This information is used to control the indicator, the warning horn, and the stick pusher. The AOA transmitter (see Figure 1) has a wedge-shaped vane mounted on a pivot in the free airstream. The vane responds much like a weather vane by aligning itself to airflow. The vane angle from the waterline of the aircraft correlates to angle of attack and thus to airspeed. When the aircraft is traveling at a high rate of speed, the airflow over the aircraft is near parallel to the waterline. As the aircraft slows, the airflow changes to a higher angle relative to the waterline. if the airspeed is allowed to go low enough the aircraft cannot sustain flight. This is known as stall speed or angle.

D. The vane is linked mechanically to a potentiometer (pot) within the AOA transmitter assembly. An electrical signal from this pot is used by the computer as angle of attack information. The position of the flaps also affects the stall speed of the aircraft. A pot connected by a mechanical linkage to the flaps provides an electrical signal used by the computer to compensate the angle of attack information for flap position.

E. The SAS computer contains small trimmer pots for adjustment and calibration of the system. Ground calibration of the system is accomplished by using the information supplied bin the SAS DATA in the Aircraft Log. The trimmer pots are then adjusted to make the computers output correspond to certain points on the face of the SAS indicator and to make horn and stick pusher actuate. Flight calibration is accomplished by flying the aircraft at indicated airspeeds that correspond to gross weight, configuration, stall speed of the aircraft, and certain points on the indicator. The trimmer pots are then adjusted to make the needle on the indicator read at these points and to make the horn and the stick pusher actuate at the proper airspeeds.

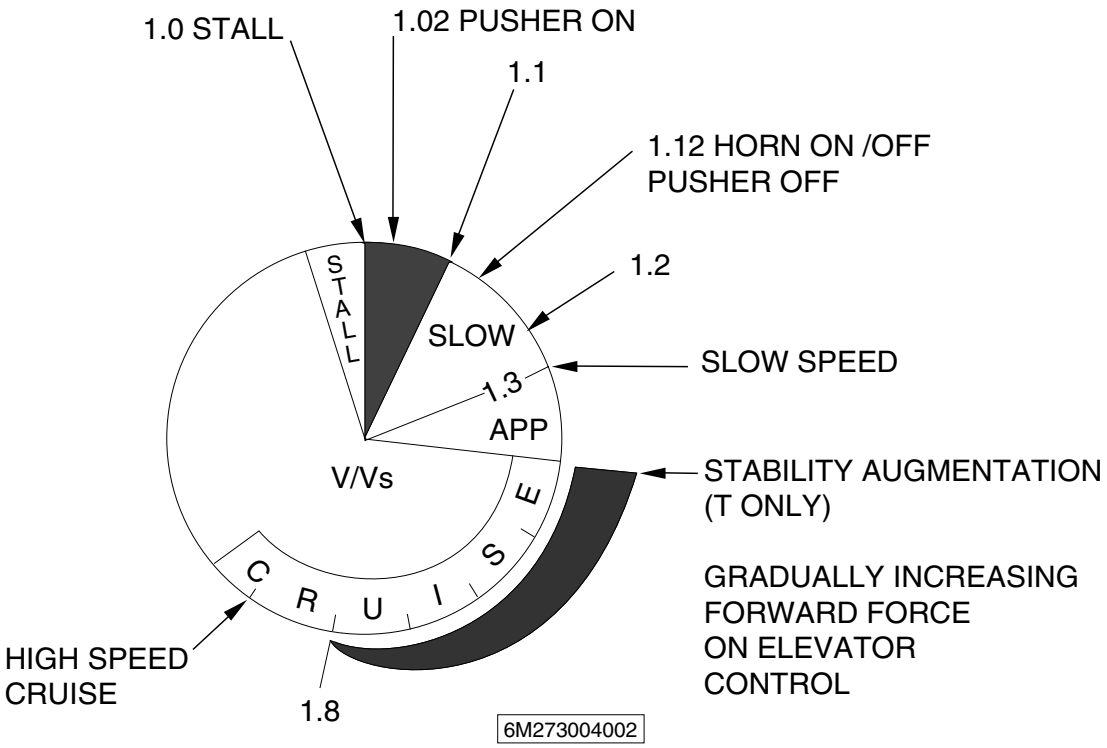
F. The stick pusher uses a servo, a capstan assembly, and an independent cable system connected to the elevator control. The servo consists of a DC powered motor and a magnetic clutch controlled by the computer. The servo drives a pulley drum within the capstan. Cables on the pulley drum are connected to the elevator control. A slip clutch within the pulley drum allows the pilot to override the stick pusher, if necessary. On the SA226-T the stick pusher is also used for stability augmentation.

G. Two lights on the annunciator panel are used to indicate system condition, a green SAS ARM light and a red SAS FAULT light. The SAS ARM light indicates the airspeed is below SAS arm speed (185 knots for the SA226-T and 140 knots for the AT, and TC). Below this speed the system is armed and the stick pusher will operate if required; above this speed the stick pusher is disabled; however, the SAS indicator will still function. A pressure sensitive switch (airspeed switch) is connected to the copilot's pitot-static system and is used for this function. The SAS FAULT light gives warning of two malfunctions. When steady it indicates a computer or angle of attack malfunction. When flashing it indicates a stick pusher malfunction.

H. A SAS CLUTCH switch on the pedestal is used to disconnect the stick pusher should a malfunction occur in flight. A strut compression switch on the left main landing gear disconnects the stick pusher while the aircraft is on the ground.

I. A SAS TEST switch provides the pilot with capability of performing a preflight check of the system. When the switch is actuated components within the computer substitutes angle of attack information and operate the warning horn and stick pusher.

J. The face of SAS indicator (see Figure 2) is marked to show a ratio of airspeed and stall speed. This is V/Vs. The V means existing velocity or airspeed and Vs is the velocity or airspeed at which the aircraft would stall at the existing gross weight and a configuration. As an example, if the pointer is at the 1.3 mark, it means that the aircraft is flying at 1.3 times its stall speed. Assume the stall speed is 100 knots for the existing gross weight and a configuration of landing gear up, flaps up, and power at flight idle. If the airspeed indicator reads 130 knots the SAS indicator should read 1.3 because 130 divided by 100 is 1.3. At 100 knots, the pointer of the indicator would be on the line between the red stall sector and the crosshatch sector or 1.0.



SAS Indicator
Figure 2

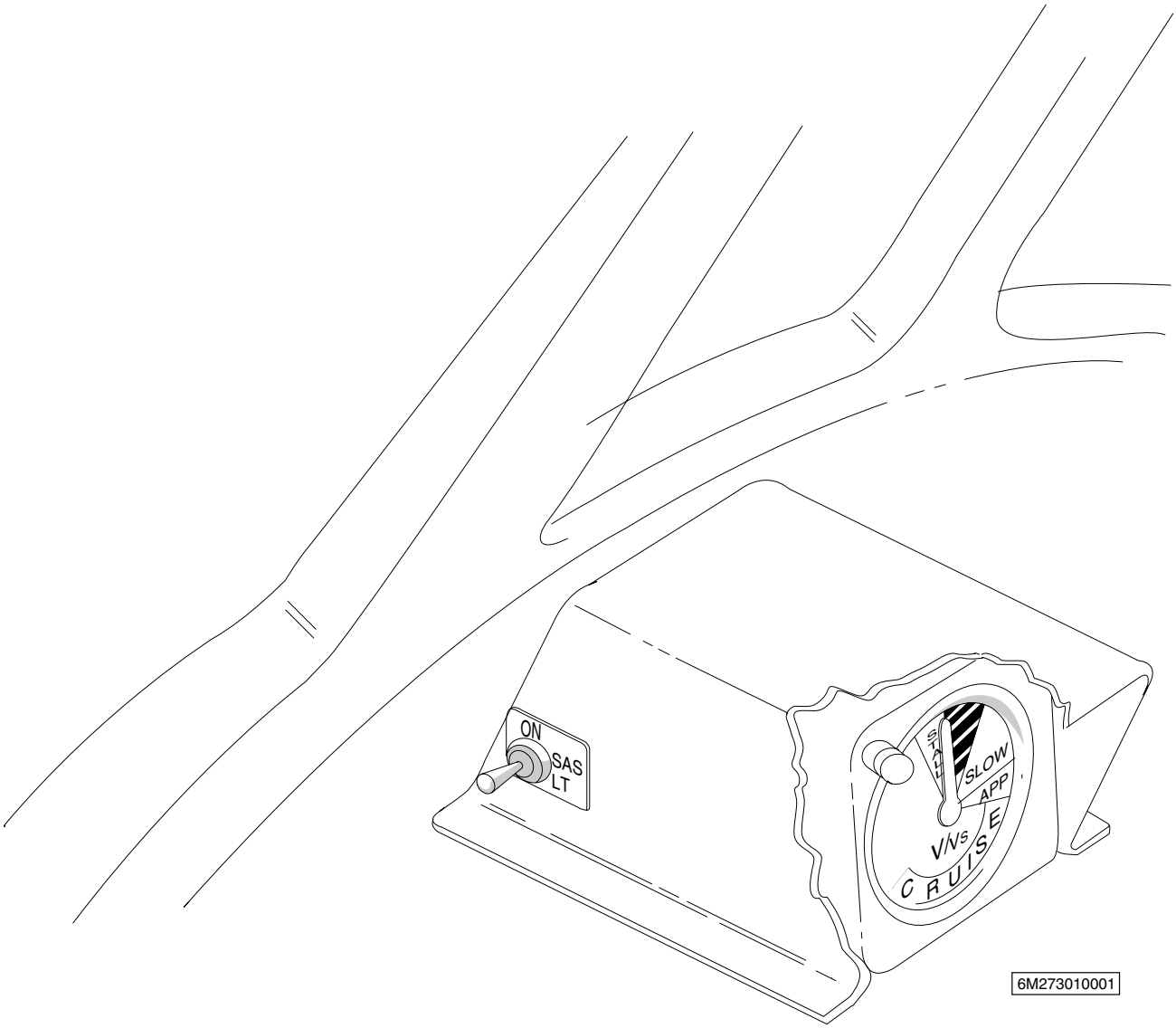
K. If the airspeed should decrease from 130 knots the pointer of the indicator would pass through the yellow SLOW sector toward the crosshatch sector. At 110 knots (1.1 on the indicator) the pointer would pass into the crosshatch sector. Slightly ahead of this point at about 1.12 the warning horn would sound and continue sounding unless airspeed is increased. If the airspeed were allowed to decease further, at 1.02 (slightly before 1.0), the stick pusher would actuate. The elevator control would be pushed forward and the aircraft would be forced toward a nose down attitude thus avoiding a stall. The force would remain until the airspeed has increased to approximately 120 knots for this example or until the pointer of the indicator moves clockwise to approximately the middle of the SLOW sector.

L. For the SA226-T only a slight forward force is applied to the elevator control by the stick pusher at approximately 1.8 V/Vs and gradually builds up as the airspeed decreases until approximately 1.4. At this point the force remains the same until pusher activation.

SAS INDICATOR – DESCRIPTION AND OPERATION

1. General

A. The SAS indicator is mounted in a shadow box on the instrument panel glareshield. It is located directly in the pilot's field of vision through the forward cockpit windows for quick reference during approach to landing. The face of the SAS indicator os divided into five sectors. A green CRUISE sector from above 1.9 V/Vs to 1.4 V/Vs. A white APP. (approach) sector from 1.4 V/Vs to 1.3 V/Vs. A yellow SLOW sector from 1.3 V/Vs to 1.1 V/Vs. A red and black crosshatch sector from 1.1 V/Vs to 1.0 V/Vs. A red STALL sector from 1.0 V/Vs to below 1.0 V/Vs.



SAS Indicator
Figure 1

SAS INDICATOR – MAINTENANCE PRACTICES

1. Removal / Installation – SAS Indicator

A. Removal – SAS Indicator – (T201-275, 277-291, AT001-999, TC-201-999)

(1) Remove four screws in back cover (forward) of shadow box and remove cover.

(2) Remove three screws in bezel mount of indicator.

(3) Push back of indicator, grasp front of indicator and pull out enough for access to the electrical connections.

(4) Mark wires and remove electrical connections.

(5) Remove post light from upper left corner of indicator.

(6) Remove indicator.

B. Installation – SAS Indicator

(1) Install post light, with ground wire, in upper left corner of indicator. Point light toward lower right corner of indicator.

(2) Install post light positive connection.

(3) Install wire 7918(A)-22 on terminal marked (+) on back of indicator. Install wire 7909(A)-2 on terminal marked (-) on back of indicator.

(4) Position indicator and push indicator into shadow box.

(5) Install three screws in bezel mount.

(6) Check that terminal studs on back of indicator do not protrude through the opening in back of shadow box.

(7) Perform SAS indicator functional check.

(8) Install cover on back of shadow box.

C. Removal – SAS Indicator – (T276, 292-999)

(1) Remove post light lens.

(2) Loosen clamp screw at upper right corner of indicator.

(3) Pull indicator out; mark and disconnect electrical connections.

(4) Remove indicator.

D. Installation – SAS Indicator

(1) Install wire 7908-22 on terminal marked (+) on back of indicator. Install wire 7909-22 on terminal marked (-) on back of indicator.

(2) Position indicator and push into shadow box.

(3) Tighten clamp screw at upper right corner of indicator.

(4) Perform SAS indicator functional check.

2. Adjustment / Test – SAS Indicator

A. SAS Indicator Functional Check (T201-275, 277-291, AT001-999, TC201-999)

(1) With power off the pointer of the SAS indicator should be at 1.0 V/Vs (dividing line between the red STALL area and red and black crosshatch area).

(2) Turn battery switch ON.

(3) Turn pilot's flight instrument light control full clockwise. On aircraft T249-275, 277-291; AT025, 034-999; TC213-999, turn SAS LT switch to ON., (left side of shadow box). Post light at upper left corner of indicator should illuminate.

(4) Make sure flaps are full up.

(5) Push and hold SAS TEST switch to CRUISE. Pointer of SAS indicator should go up to 1.3 V/Vs (dividing line between the white APP. area and yellow SLOW area).

(6) Push and hold SAS TEST switch to STALL. Pointer of SAS indicator should go to 1.0 V/Vs.

B. SAS Indicator Functional Check (T276, 292-999)

(1) With power off pointer of SAS indicator should be at 1.0 V/Vs (dividing line between the red STALL area and the red and black crosshatch area).

(2) Turn battery switches on.

(3) Turn pilot flight instrument light control full clockwise. Turn SAS LT switch to ON (left side of shadow box). Post light at upper left corner of indicator should illuminate.

(4) Make sure flaps are up.

(5) Push and hold SAS TEST switch to CRUISE. Pointer of SAS indicator should go up to 1.3 V/Vs (dividing line between the white APP. area and yellow SLOW area).

(6) Push and hold SAS TEST switch to STALL. Pointer of SAS indicator should go to left of the green CRUISE area. After a delay of one to two seconds the pointer should start traveling counterclockwise. As pointer leaves the yellow SLOW area and enters the red and black crosshatch area the stall warning horn should sound. As the pointer leaves the crosshatch area and enters the red STALL area (if the SAS CLUTCH switch is ON) the STICK PUSHER should KICK in and the SAS FAULT should illuminate. The pointer should delay to the left of the STALL area for one to two seconds and sweep clockwise. The pusher and horn should cease and the pointer should go to the APP. area just below 1.3 V/Vs.

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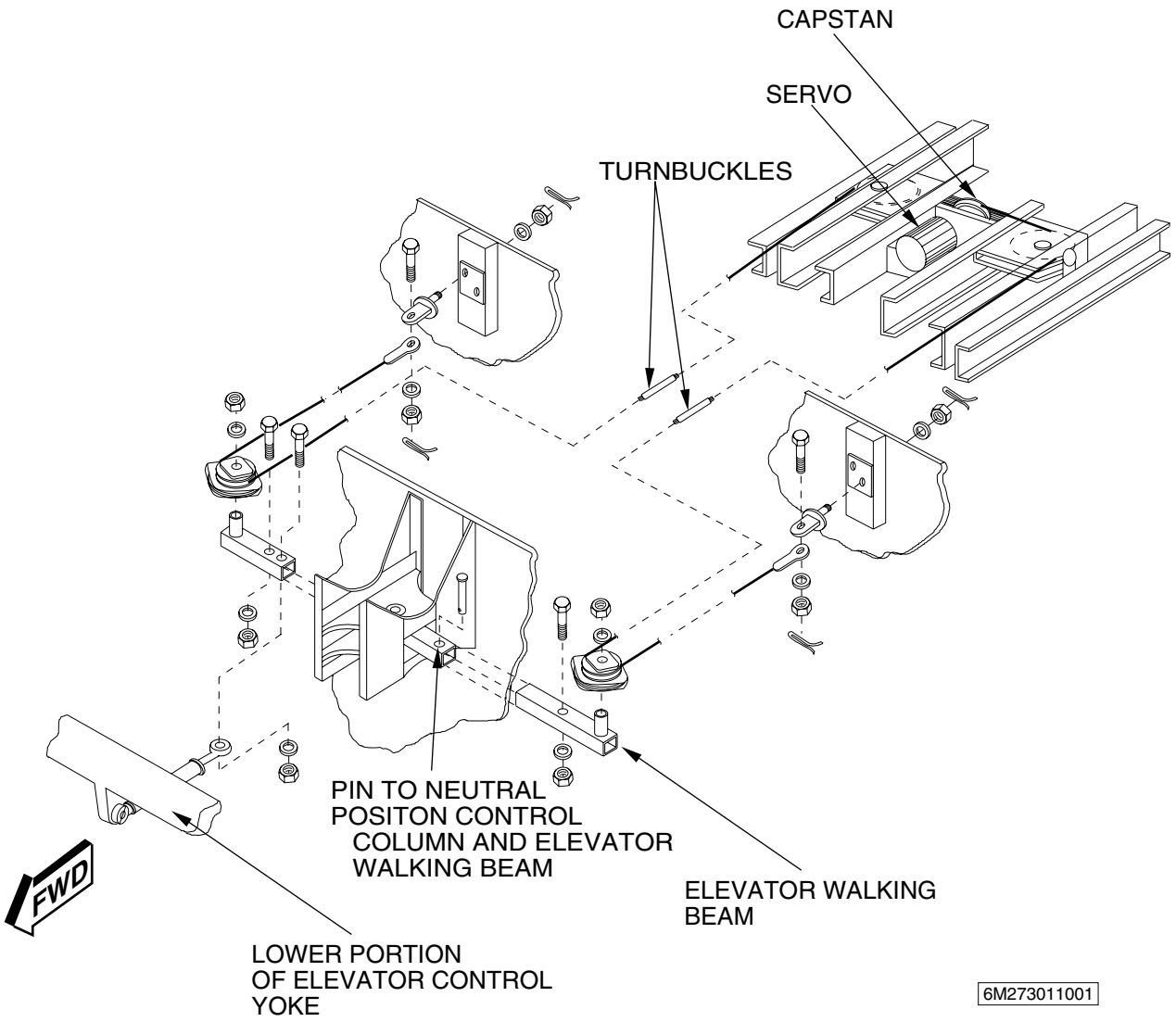
STICK PUSHER ASSEMBLY – DESCRIPTION AND OPERATION

1. General

A. The stick pusher uses a servo, capstan assembly, and an independent cable system connected to the elevator control. The servo and capstan assembly is located under the center aisle floorboards directly across from the leading edge of the wing. The cables run around a pulley drum in the capstan assembly, then outboard to two pulleys. At this point they turn forward and run under the center aisle floorboards, to another set of pulleys on each end of the elevator control walking beam. The elevator cables run up the center aisle to a larger set of pulleys on each end of the walking beam. The stick pusher cables and the elevator cables turn aft and attach to a belt frame aft of the walking beam.



Pusher Assembly
Figure 1

B. The servo consists of a continuous DC powered motor and a magnetic clutch. The clutch contains two discs with powdered iron between them. One disc is connected to the motor, the other is connected to an output shaft. When a current is allowed to pass through these particles, they grab or stick to each other, and torque from the motor is coupled to the output shaft. The amount of torque is directly related to the amount of current. The SAS computer controls the amount of current allowed to flow through the clutch.

C. The capstan assembly consists of a gear (driven by the output shaft of the servo) on the inner surface of a pulley drum. The pulley drum contains an adjustable slip clutch to allow the stick pusher to be overridden if necessary.

D. When a current is applied to the clutch by the computer the output torque of the motor is coupled to the pulley drum. The cables on the pulley drum pull the right side of the elevator walking beam aft. This applies a forward force to the elevator control and at the same time pulls the elevators down.

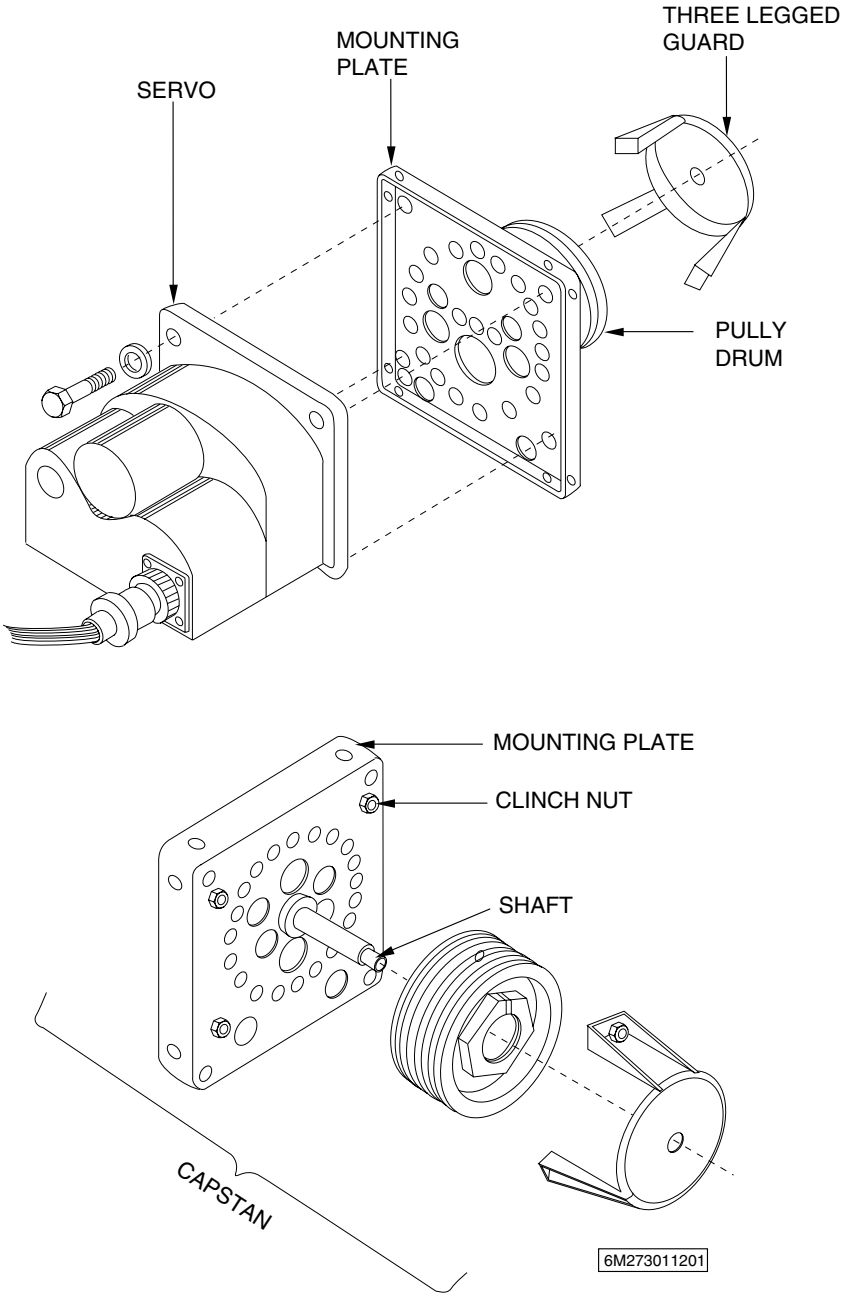
STICK PUSHER ASSEMBLY – MAINTENANCE PRACTICES

1. Removal/Installation – Stick Pusher Assembly

A. Removal – Servo

(1) Remove carpet as necessary and remove center aisle floorboard directly across from wing leading edge.

(2) Remove safety wire from electrical connector and remove connector.



Stick Pusher Servo
Figure 201

(3) Remove bolts attaching servo to mounting plate.

NOTE: Remove bottom bolts first. Do not turn clinch nuts; they are permanently attached to the mounting plate. Access to bottom bolts is through lightning holes in channels at sides of servo.

(4) Remove servo.

B. Installation – Servo

(1) Place bottom bolts in bottom mounting holes of servo.

NOTE: A rubber band or a piece of string may be used to hold bottom bolts in place until started. Remove rubber band.

(2) Position servo on mounting plate.

(3) Start top bolts.

(4) Start bottom bolts.

(5) Tighten top bolts.

(6) Tighten bottom bolts.

(7) Install electrical connector and safety wire.

(8) Perform functional check of stick pusher.

(9) Install floorboards and carpet.

C. Removal – Capstan Assembly

(1) Remove servo.

(2) Remove screws attaching three legged guard over pulley drum; remove guard.

(3) Remove bolts and nuts attaching mounting plate to side channels.

(4) Separate drum and plate.

(5) Notice how cable is wound on drum; turn drum to unwind cable.

D. Installation – Capstan Assembly

(1) Place swaged ball on cable in depression (socket) on drum.

(2) Turn drum to rewind cable. Provide two turns of cable on drum as follows:

(a) Clockwise from ball and socket, add one complete turn of cable onto drum.

(b) On other side of ball and socket, and counterclockwise, add one complete turn of cable onto drum.

(3) Position shaft on plate in pulley drum. Ball on cable and socket in drum must be in vertical or twelve o'clock position, with one complete cable turn each side.

(4) Install bolts and nuts attaching plate to side channels.

(5) Position guard and secure with screws.

(6) Work control column through a few times to check for binding or improper cable installation.

(7) Install servo.

2. Adjustment/Test – Stick Pusher

A. Servo Capstan Assembly/Disassembly/Adjustment

NOTE: Under normal operating conditions the capstan assembly torque will decrease between 10% and 25% from its preset torque value when slipping. No increase in torque will be encountered when the slippage occurs. The clutch can be slipped indefinitely without damage, and it will return to the preset value when the slippage is stopped.

It is recommended anytime the capstan torque is adjusted the capstan be disassembled, cleaned, and relubricated.

(1) Special Tools Required

(a) 1/4 inch Dowel Pin (P/N AN122721)

(b) Clutch Adjusting Block (P/N 95628)

(c) Clutch Adjusting Gear (P/N 95629)

NOTE: These three items are part of P/N 96366 assembly and can be purchased from Bendix Avionics Division, 2100 N.W. 62nd St., Ft. Lauderdale, Fla. 33310, ATTN: Service Dept.

(2) Disassembly:

(a) Separate capstan assembly from servo (see Section 27-20-11 or 27-30-11, whichever applies).

(b) Mark position of capstan guard in relation to capstan plate.

(c) Remove screws securing capstan guard to capstan plate.

(d) Remove capstan guard from capstan plate.

(e) Remove capstan gear/clutch assembly from shaft of capstan plate.

NOTE: It is recommended that anytime the capstan torque is adjusted the capstan be disassembled, cleaned, and relubricated.

(f) Loosen Allen head screw on locknut and remove locknut.

(g) Remove spring washer and clutch plate from pulley.

(h) Remove the three (3) ball bearings inside cable drive pulley.

(i) Remove all graphite grease and clean all parts with solvent. Remove graphite grease from grooves in output gear bushing.

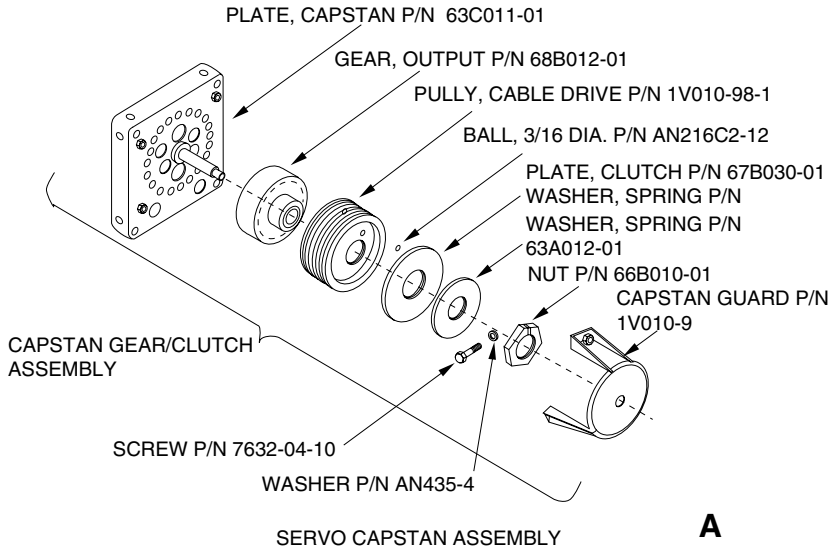
NOTE: Do not attempt to remove the clutch from inside cable drive pulley. If clutch lining is damaged, replace cable drive pulley.

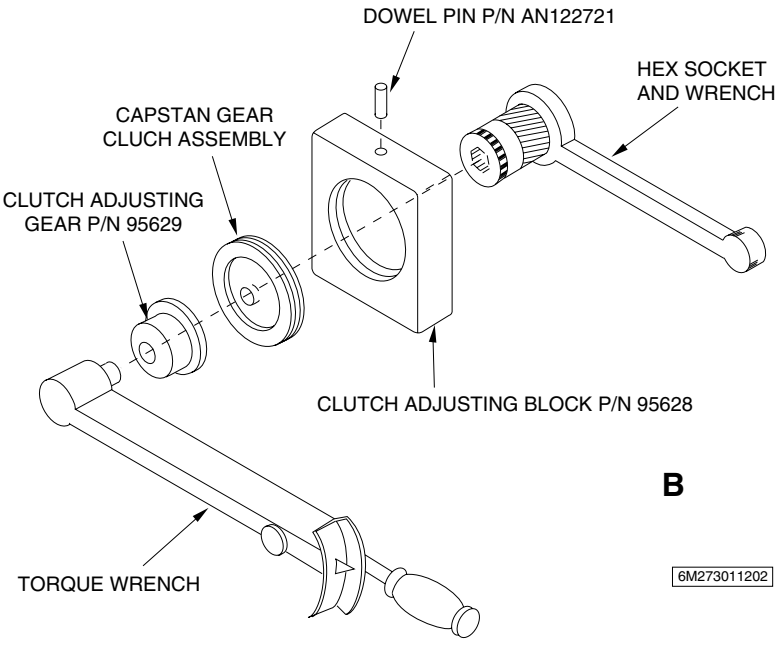
(3) Assembly:

(a) Pack grooves in output gear bushing with Molykote Type G lubricant.

(b) Apply a bead of same lubricant 0.06 inch wide by 0.38 inch long to three equally spaced places around clutch linings on both sides.

(c) Apply a thin coating of lubricant to inside diameter of cable drive pulley (output gear side only).





Servo Capstan Adjustment
Figure 202

(d) Insert all bearings.

(e) Insert output gear into cable drive pulley. Gear will only fit one end.

(f) Position clutch plate onto output gear shaft.

(g) Position spring washer onto shaft (see Figure 202).

(h) Hand-tighten locking nut.

(4) Adjustment:

NOTE: Do not install cable drive pulley into capstan plate until clutch torque has been adjusted IAW **2.**, B., (4), (e).

(a) Clamp clutch adjusting block in vice with pin hole up.

(b) Position capstan gear/clutch assembly into adjusting block and install 1/4 inch dowel pin through block into hole in cable drive pulley.

(c) Insert clutch adjusting gear into grooves in output gear.

(d) Attach torque wrench to clutch adjusting gear.

(e) Adjust torque by tightening locknut with socket wrench (stick pusher, 150 + 5 - 0 inch-pounds (16.95 +0.56 -0 N·m); yaw damper/rudder bias, 180 ± 10 inch-pounds (20.34 ± 1.13 N·m).

(f) Remove capstan gear/clutch assembly from adjusting block.

(g) Insert assembly onto capstan plate.

(h) Attach capstan guard.

B. Functional Test – Stick Pusher

(1) Depress inboard scissor switch (six wire switch) on left main gear with squat switch cheater tool.

(2) Make sure gust lock is disengaged.

(3) Pull control column back approximately vertical.

(4) Turn battery switches on.

(5) Turn SAS CLUTCH switch ON. Notice that blinking SAS FAULT light now extinguished.

(6) Hold force gauge to top center of pilots control column. Pull control column to vertical position. Set force gauge dial to zero.

(7) On SA226-T only, move AOA vane slowly upward until 1.75 V/Vs is indicated on SAS indicator. Continue to move vane upward, forward force on control column should increase to 18 plus two pounds at 1.4 V/Vs. Forward force should remain at 18 plus two pounds until 1.02 V/Vs is reached. At 1.02 V/Vs forward force should increase to 65 pounds plus or minus five pounds. Check for stall warning horn at 1.12 V/Vs. Move vane to full down position. Check that SAS fault light does not come on. (If necessary, adjust ramp current to obtain 18 plus two pounds and adjust pusher current to obtain 65 plus or minus five pounds).

(8) On SA226-AT and TC, move AOA vane slowly upward until 1.12 V/Vs is indicated. Check for stall warning horn at 1.12 V/Vs. Continue to move vane slowly upward until 1.01 V/Vs is indicated. At 1.02 V/Vs forward force should be 65 pounds plus or minus five pounds. Move vane to full down position. Check that SAS fault light does not come on. (If necessary, adjust pusher current to obtain 65 plus or minus five pounds).

(9) Pull control column back sharply. A loud ratcheting noise should be heard, accompanied by a jerking movement of the control column. This is the mechanical slip clutch being overridden.

(10) Verify that the total force required to induce ratcheting is less thab 86 pounds. If ratcheting does not occur, or if force required to induce ratcheting is greater than 86 pounds, disassemble capstan, clean, lubricate, and perform Adjustment / Test – Stick Pusher Assembly.

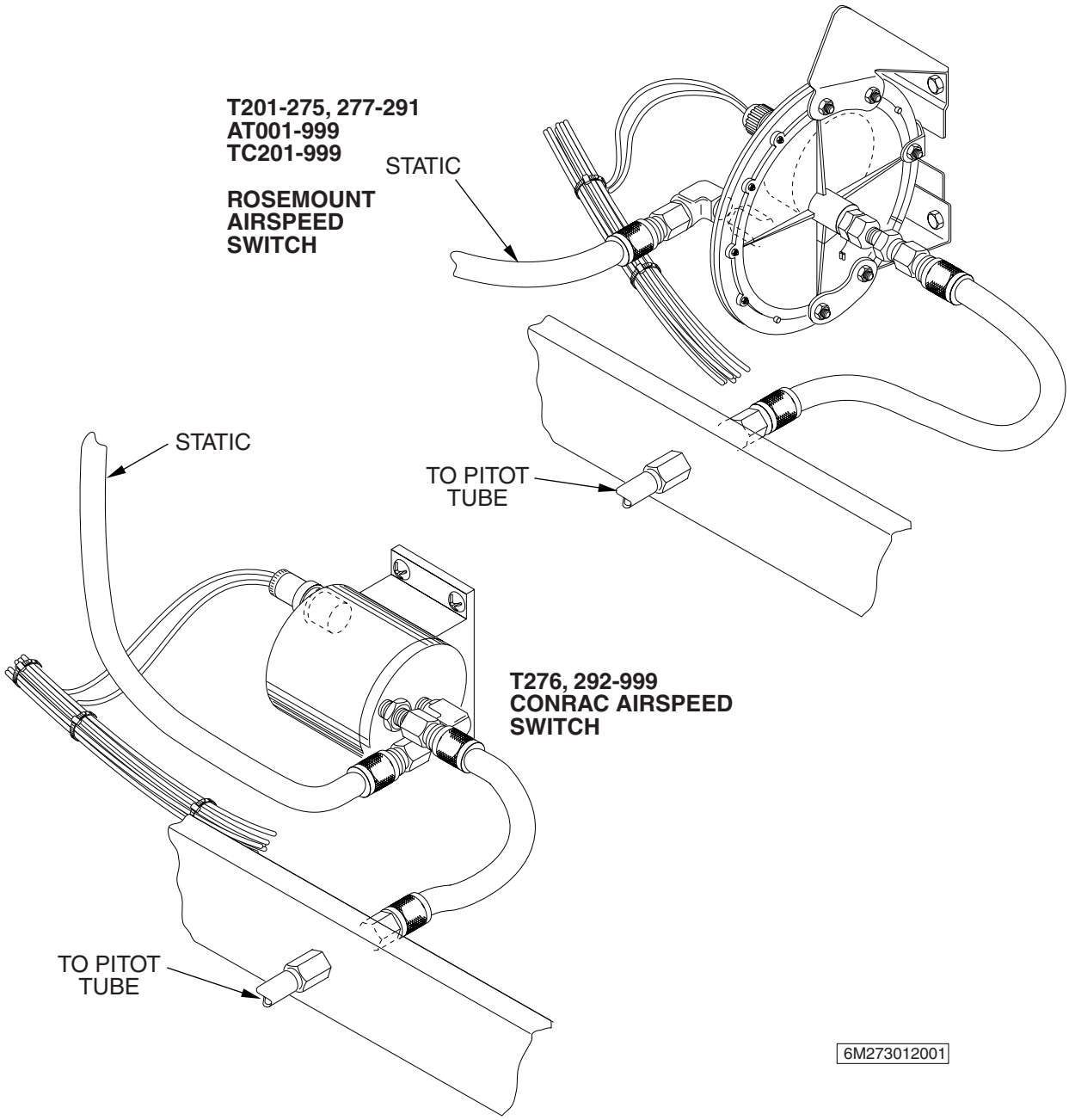
(11) Turn SAS CLUTCH switch OFF. No stick pusher force should be felt on the control column.

(12) Turn battery switches off and remove squat switch cheater tool from left main gear scissor switch.

AIRSPEED SWITCH – DESCRIPTION AND OPERATION

1. General

A. A pressure sensitive switch is used to disconnect the stick pusher above SAS arm speed. The switch is located under the right aft floorboard in the nose baggage compartment. The switch uses the copilot's pitot-static system to sense airspeed. The switch actuates at 140 knots for the SA226-AT and TC, 185 knots for the SA226-T, ± 5 knots. Other circuitry within the SAS system extinguishes the SAS ARM light and inhibits the stick pusher above this speed.



Airspeed Switch
Figure 1

AIRSPEED SWITCH – MAINTENANCE PRACTICES

1. Servicing – Airspeed Switch

A. Check airspeed switch and plumbing for condition and safety.

2. Removal/Installation – Airspeed Switch

A. Removal – Airspeed Switch

(1) Remove right aft nose baggage compartment floorboard.

(2) Mark and remove pitot-static lines.

(3) Disconnect electrical connector.

(4) Remove bolts or screws attaching airspeed switch to nose wheel well keelson.

(5) Remove fittings from pitot-static ports on switch.

NOTE: Note position of elbow for installation on new airspeed switch.

B. Installation – Airspeed Switch

(1) Install fittings in pitot-static ports on switch.

(2) Position switch and secure with screws or bolts.

(3) Connect electrical connectors.

(4) Install pitot-static lines.

(5) Perform pitot-static leak check (see Section 32-10-00 this manual.

(6) Perform Airspeed Switch Check.

NOTE: Airspeed switch check can be performed in conjunction with pitot-static check.

3. Adjustment/Test – Airspeed Switch

A. Adjustment and Test – airspeed switch check.

(1) Connect pitot-static tester to copilot's pitot tube.

(2) Turn battery switches ON.

(3) Observe SAS ARM light illuminates.

(4) Increase pitot pressure slowly.

(5) Ensure SAS ARM light extinguishes by 145 knots (SA226 – AT and TC, 190 knots (SA226 – T) on copilot's airspeed indicator.

(6) Decrease pitot pressure slowly.

(7) Ensure SAS ARM light extinguishes by 135 knots (SA226 – AT and TC, 180 knots (SA226 – T) on copilot's airspeed indicator.

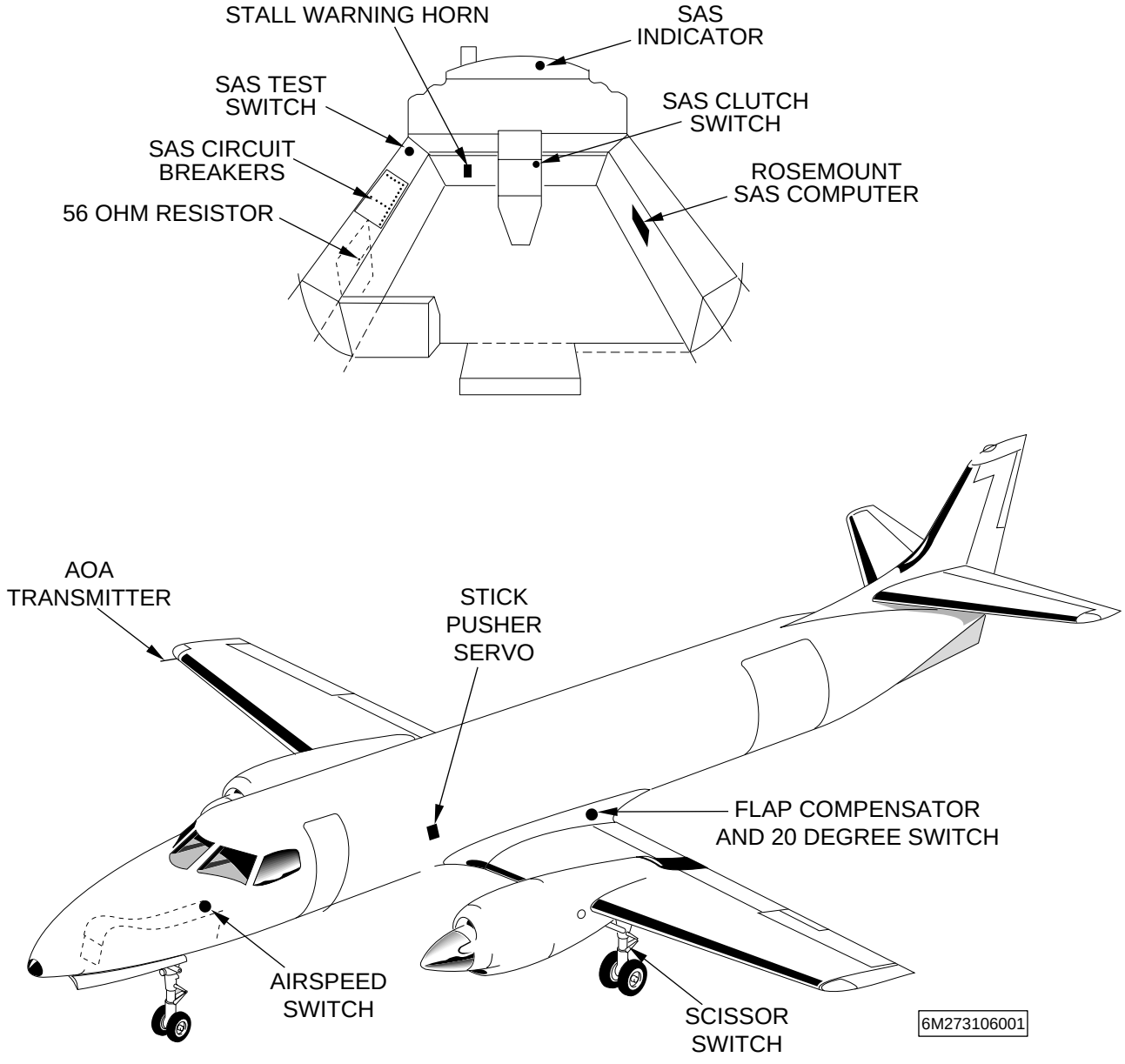
(8) Decrease pitot pressure to zero, and remove pitot-static tester.

ROSEMOUNT SAS – DESCRIPTION AND OPERATION

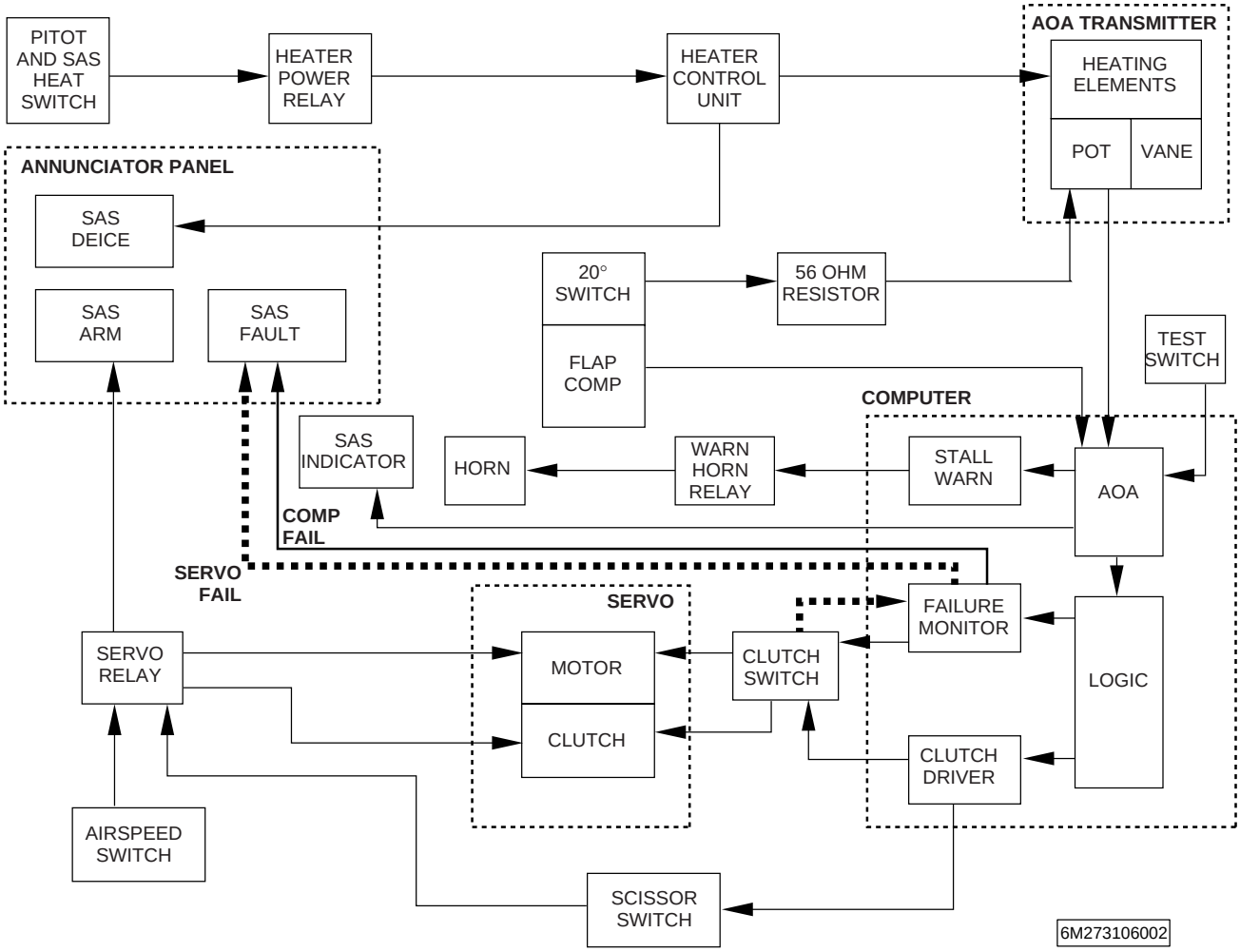
1. General

A. All 226 series aircraft except for T(B) models were factory equipped with a Rosemount SAS System. T(B) aircraft S/N 276, and 292-999 were factory equipped with a Conrac SAS System. All other T, AT, and TC aircraft may be so equipped per SB27-037, as a customer option.

B. The Rosemount SAS system consists of an angle of attack (A0A) transmitter, flap compensator, computer, airspeed switch, stick pusher servo, cockpit indicator, and controls. These components function together to warn the pilot of impending stall and, if necessary, apply a forward push to the elevator control if a stall is eminent.



Rosemount SAS Equipment Locator
Figure 1



Rosemount SAS Diagram
Figure 2

2. SAS Computer (27-30-07)

A. The computer is located in the right console, receives and processes information from the other various components of the SAS system. The information is processed by the computer and used to control a SAS indicator, stall warning horn, and a stick pusher servo. Trimmer pots installed in the computer are used for calibration of the system. Calibration of the system is necessary to compensate for installation and aerodynamic differences between individual aircraft. The computer also contains fault circuitry and monitors itself and the stick pusher servo.

3. Stall Warning Horn (27-30-07)

A. The stall warning horn is located forward of the pilot rudder pedals on the forward pressure bulkhead. The horn is controlled by a relay behind the middle door of the left console. The relay is in turn controlled by the computer.

4. Angle of Attack (AOA) Transmitter (27-30-08)

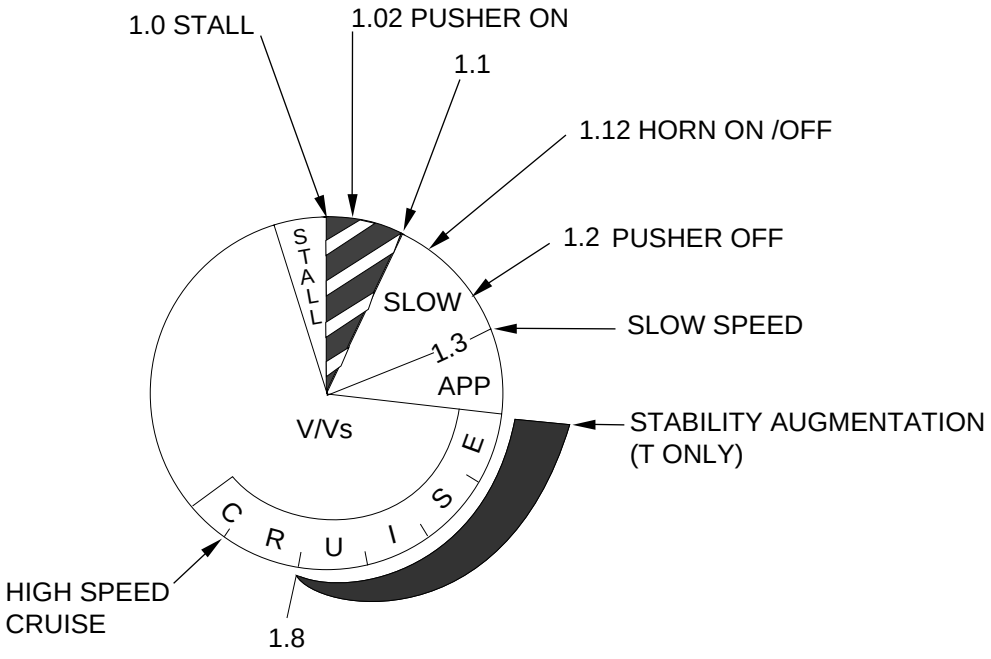
A. The AOA transmitter is attached to the right wing leading edge at the wing tip. The transmitter consists of a mast, base, vane, and a potentiometer (pot). The tip and body of the mast, the vane, plus the wing leading edge just behind the mast are electrically deiced (see appropriate section of Chapter 30 of the maintenance manual for more information on SAS deice). The vane pivots on the body of the mast and aligns itself with airflow over the wing. The vane angle from the waterline of the aircraft correlates to angle of attack. The vane is linked mechanically to the pot. Each aircraft is provided with a vane protractor, marked in degrees, and is used for ground calibration of the system.

5. Flap Compensation (27-30-09)

A. Flap compensation consists of a potentiometer and switch mechanically linked to the flaps. The changing resistance of the potentiometer (as the flaps are raised and lowered) provides a means for the SAS computer to sense flap position. A cam installed on the shaft of the flap position indicator transmitter is oriented to actuate a switch when the flaps are extended beyond 20°. Actuating this switch places an additional resistance (56 ohms) in series with the output of the AOA transmitter.

6. SAS Indicator (27-30-10)

A. The SAS Indicator is located on the glareshield. The face of the SAS indicator is divided into five sectors (see Figure 2). Points at the dividing lines of these sectors are V/Vs and are used for calibration of the system.



Rosemount SAS Indicator
Figure 3

7. Stick Pusher Servo (27-30-11)

A. The stick pusher servo is located under the center isle floorboards directly across from the wing leading edge The servo contains a continuous DC powered motor coupled by a magnetic clutch to a capstan assembly. Cables are connected from the capstan to the elevator control. When the motor is running and the clutch is engaged the elevator control is pushed forward. The capstan contains a mechanical slip clutch should it become necessary to override the stick pusher. The magnetic clutch is engaged by current controlled by the computer. The SAS CLUTCH switch on the cockpit pedestal is used to turn off the motor and disconnect the clutch.

8. Scissor Switch

A. A scissor switch on the left main gear disconnects the clutch when the aircraft is on the ground.

9. Airspeed Switch (27-30-12)

A. An airspeed switch located in the right aft section of the nose baggage compartment also controls the servo. The switch is actuated by air pressure from the copilot pitot system. The switch is connected to the servo control relay located just forward of the servo. The airspeed switch opens at the airspeed at which the SAS system is no longer needed.

10. System Operation

A. The system operates with DC power from four circuit breakers on the left essential bus. The circuit breakers power the computer, test and fault functions of the computer, stall warning horn, the servo motor, and annunciator panel lights.

NOTE: The vane angles mentioned below are averaged vane angles taken from a large number of Merlin and Metro aircraft and are for purposes of this discussion only. These vane angles should not be used for calibration of the SAS system on individual aircraft. Vane angles taken from individual aircraft can be found in the SAS DATA in the aircraft log.

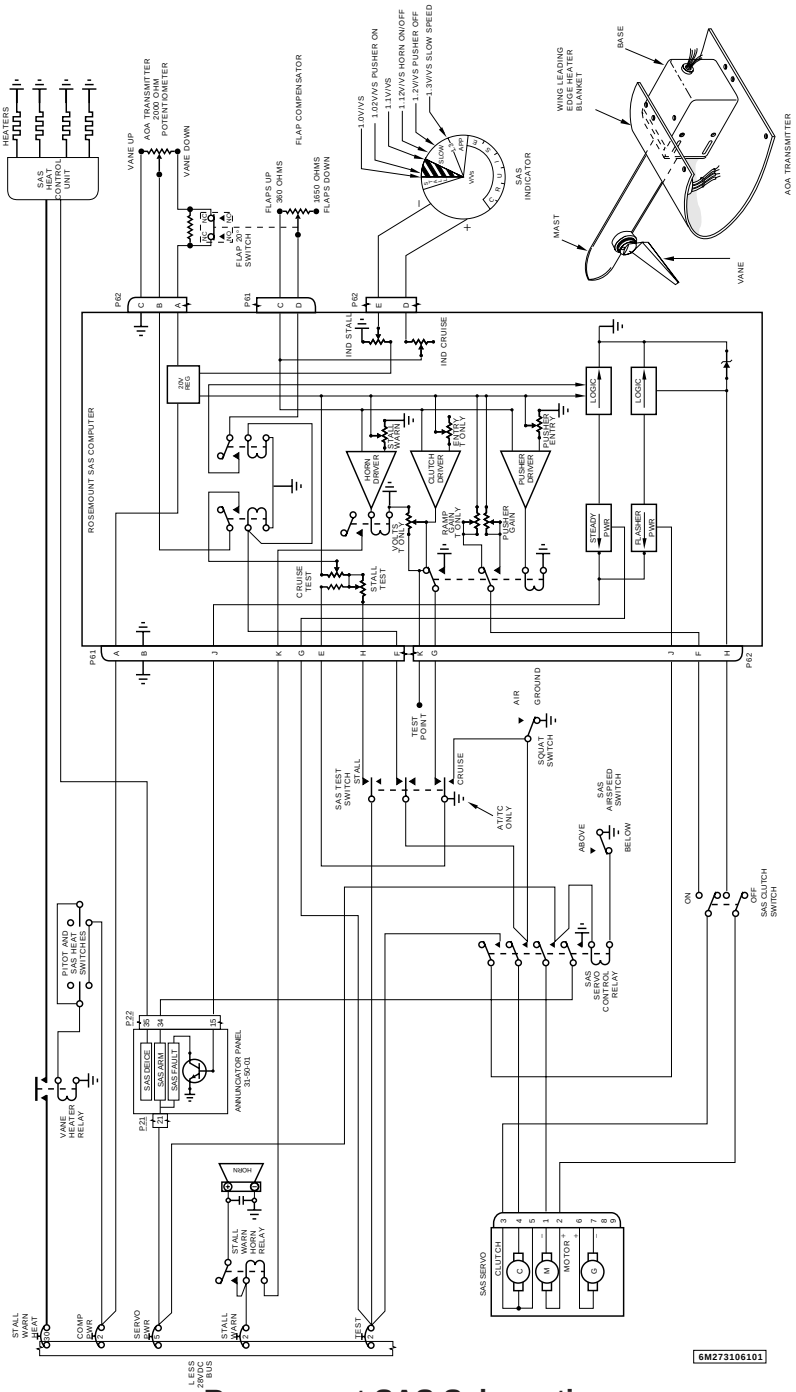
B. With the vane at approximately 0° as the flaps are lowered, the pointer of the SAS indicator starts traveling counterclockwise through the CRUISE sector.

(1) As the flaps pass 20° (slightly past 1/2 on the flap indicator) the pointer will jump about 1/8 of an inch counterclockwise. When the flaps are full down the pointer should stop near the I in the CRUISE sector (see Figure 1). If the vane is moved up slowly from its bottom stop the pointer of the SAS indicator pointer will start traveling counterclockwise.

(2) For the SA226-T only, a slight and gradually increasing forward force will be exerted on the elevator control. At 12° on the vane, 1.4 V/Vs on the indicator, the force will increase to 20 pounds and remain the same until pusher activation. This is known as stability augmentation.

(3) At 20° on the vane, the pointer of the indicator will be at 1.3 V/Vs. At 33° the indicator will be 1.12 V/Vs, the horn will sound and remain sounding as the vane moves up. At 35° the indicator will be 1.1 V/Vs. At 40° the pusher will kick in and the indicator will be 1.02 V/Vs, and a forward force of 60 pounds will be exerted on the elevator control. The pusher remains as the vane moves up. At 42° the indicator will be 1.0 V/Vs. When the vane reaches its upper stop the pointer of the indicator will be to the left of the STALL sector. If the vane is moved down slowly the horn will cease at 33° (1.12 V/Vs) and the pusher will cease at 28° (1.2V/Vs).

C. The SAS TEST switch in effect substitutes vane angles to the computer for 1.0 V/Vs and 1.3 V/Vs. With the flaps up, when held to the STALL position, the pointer of the indicator goes to 1.0 V/Vs, the horn sounds, and the pusher kicks in. When held to the CRUISE position, the pointer of the indicator goes to 1.3, and on the SA226–T the stability augmentation force can be felt on the elevator control.



Rosemount SAS Schematic
Figure 4

ROSEMOUNT SAS – FAULT ISOLATION

1. Troubleshooting – Rosemount SAS

A. When checking for 28 VDC a voltmeter or a test light may be used. When wiring, wire terminals, or relay contacts are a suspected cause a test light should be used since it places a slight load on the circuit. The slight load will cause the circuit being checked to open at the trouble point, and the light will extinguish. A voltmeter will place almost no load on the circuit, and the defective item will be harder to find.

ROSEMOUNT SAS – FAULT ISOLATION

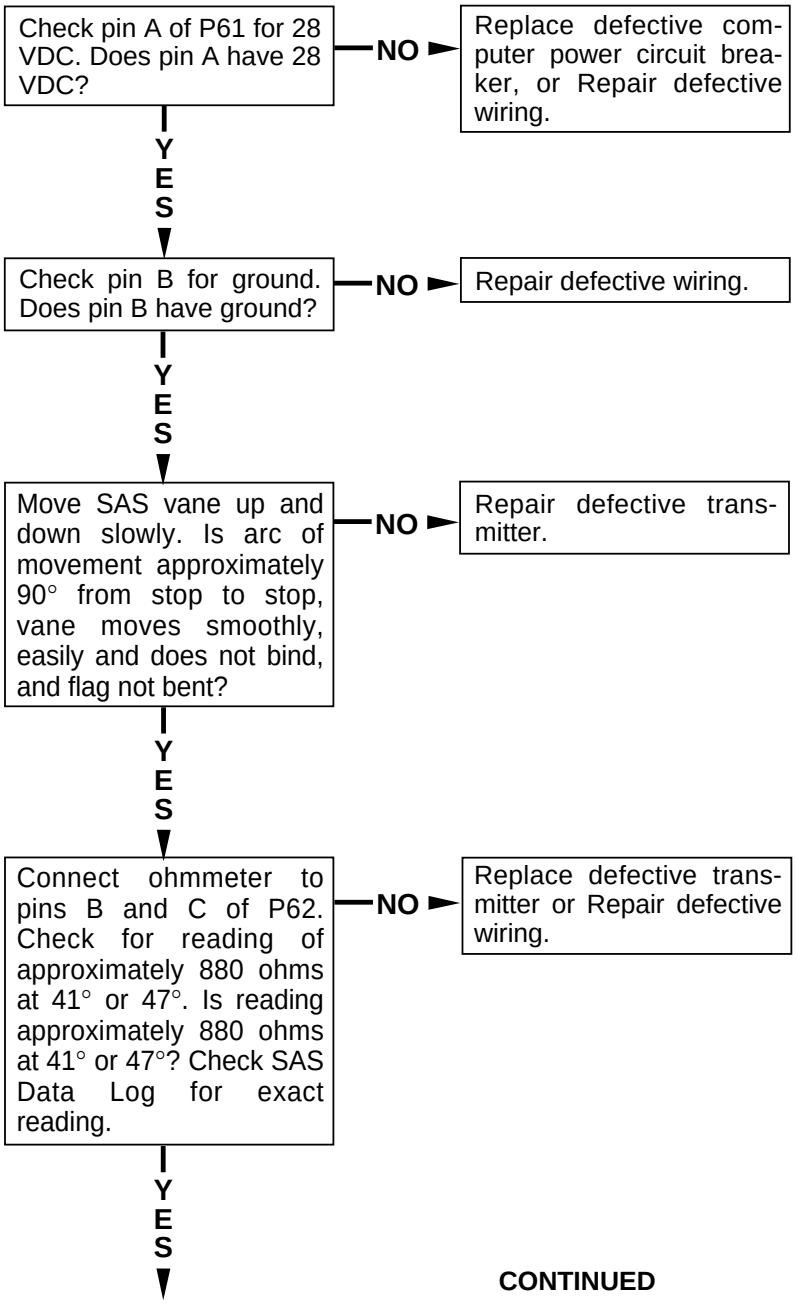
PROBLEM:

SAS SYSTEM INOPERATIVE, SAS INDICATOR ERRATIC, SAS FAULT LIGHT ILLUMINATES STEADILY, OR PUSHER OR HORN ERRATIC.

PREREQUISITES:

Electrical Power ON, Both Connectors Removed from SAS Computer.

START:



6M273106101A

CONTINUED

CONTINUE

Connect ohmmeter to pin A and C of P62. With flaps up check for reading of approximately 2,000 ohms. Is reading approximately 2,000 ohms?

NO → Replace defective transmitter, or Repair defective wiring.

YES

Check pins A, B, and C of P62 for infinite resistance to ground. Is there infinite resistance to ground?

NO → Repair defective transmitter or repair defective wiring. See Section 27-31-08 for adjustment or replacement procedures.

YES

With flaps extended, check ohm reading across pins A and C of P62 for increase of approximately 56 ohms. Is reading approximately 56 ohms?

NO → Replace defective 56 ohm resistor, or defective 20° switch, or repair defective wiring. See Section 27-31-09 for adjustment

YES

Connect ohmmeter to pins C and D of P61. With flaps up check for approximate reading of 360 ohms. Is reading approximately 360 ohms?

NO → Replace defective flap compensator or repair defective wiring. See Section 27-31-09 for adjustment or replacement procedures.

YES

CONTINUED

6M273106101B

CONTINUE

With ohmmeter connected to pins C and D of P61, extend flaps and check for smooth and even increase of approximately 1,650 ohms. Does reading increase smoothly and evenly to approximately 1,650 ohms? See SAS Data in Aircraft Log for exact ohm reading.

NO → Replace defective flap compensator, or Repair defective wiring. See Section 27-31-09 for adjustment or replacement procedures.

YES

With ohmmeter connected to pins C and D of P61, check for reading of infinite resistance to ground. Is reading infinite resistance to ground?

NO → Replace defective flap compensator, or Repair defective wiring. See Section 27-31-09 for adjustment or replacement procedures.

YES

Check glareshield connection to indicator for security. Is connection secure ?

NO → Tighten connector.

YES

Check terminals on back of indicator for security. Are terminals secure?

NO → Tighten terminals.

YES

6M273106101C

CONTINUED

CONTINUE

Connect ohmmeter to pin D of P62 and positive terminal on back of indicator Check for reading of 0 ohm continuity. Is reading 0 ohms continuity?

NO ► Replace defective SAS indicator, or Repair defective wiring.

YES ▼

Connect ohmmeter to pin E of P62 and negative terminal on back of indicator. Check for reading of 0 ohms continuity. Is reading 0 ohms continuity?

NO ► Replace defective SAS indicator, or Repair defective wiring.

YES ▼

Check that both leads to indicator read infinite resistance to ground. Rosemount SAS indicator is a 0 to 1 milliammeter; 0 mA at 1.0 V/Vs and 1 mA at left edge of green cruise band. Is reading infinite resistance to ground?

NO ► Replace defective SAS indicator, or Repair defective wiring. See Section 27-30-10 for adjustment procedures.

YES ▼

6M273106101D

CONTINUED

CONTINUE

With SAS computer disconnected, SAS Fault light should not be illuminated. Jumper pin G to pin J of P61. Does SAS Fault light illuminate?

NO → Replace defective SAS Fault light (annunciator).

YES ↓

With battery switches ON and Stall Warn circuit breaker engaged, ground pin K of P61. Check that Stall Warning Horn sounds steadily and does not warble. Does Stall Warn horn sound steadily and does not warble?

NO → Replace defective Stall Warning Horn, or Repair defective wiring.

YES ↓

If all the preceding checks are ok replace defective SAS computer.

Check positive terminal on horn for 28 VDC. Does terminal have 28 VDC?

NO → Remove middle door on left console and check Stall Warn Horn, Replace if defective, or repair defective relay wiring.

YES ↓

If connection is secure and 28 VDC is present Replace defective Stall Warning Horn. See Section 27-31-07 for replacement procedures.

6M273106101E

END

ROSEMOUNT SAS – FAULT ISOLATION

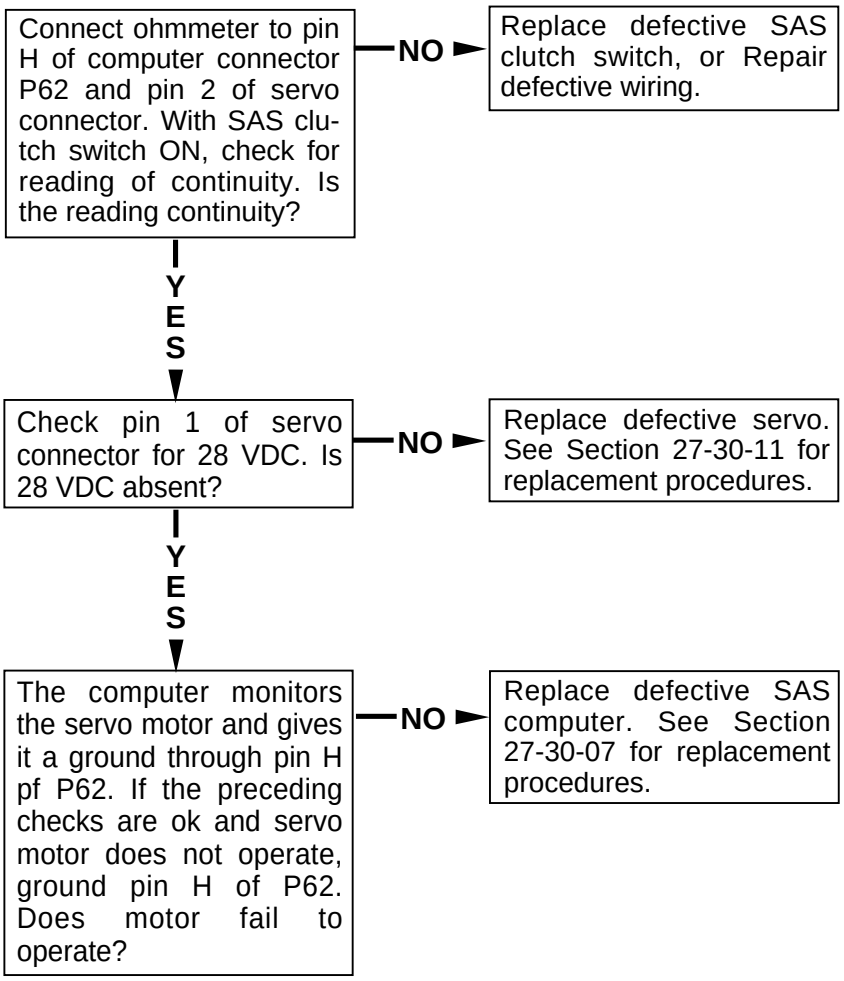
PROBLEM:

STICK PUSHER INOPERATIVE, SAS CLUTCH SWITCH ON, SAS FAULT LIGHT BLINKS.

PREREQUISITES:

Electrical Power ON, Both Connectors Removed from SAS Computer.

START:



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END

ROSEMOUNT SAS – FAULT ISOLATION

PROBLEM:

SAS FAULT LIGHT DOES NOT BLINK, OR BLINKS ERRATICALLY WHEN SAS CLUTCH SWITCH IS OFF.

PREREQUISITES:

Electrical Power ON, Both Connectors Removed from SAS Computer.

START:

Pull computer power circut breaker. Check if SAS Fault light illuminates steadily. Does SAS Fault light illuminate steadily?

NO → Replace defective SAS Fault light (annunciator).

YES ↓

With SAS clutch switch OFF, check if SAS Fault light does not blink or blinks erratically. Does SAS Fault light blink or blink erratically?

NO → Replace defective SAS computer. See Section 27-30-11 for replacement procedures.

6M273106103

END

ROSEMOUNT SAS – FAULT ISOLATION

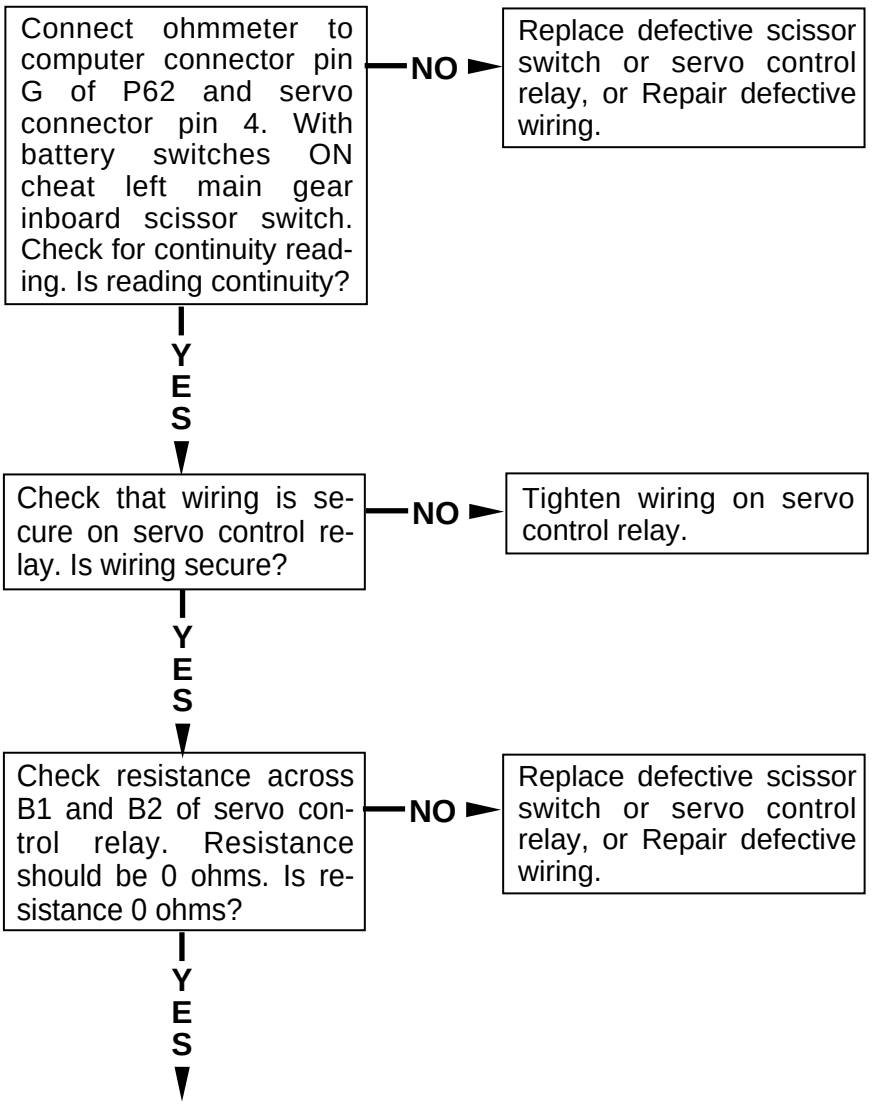
PROBLEM:

STICK PUSHER INOPERATIVE, SAS FAULT LIGHT IS NOT ILLUMINATED. SAS ARM LIGHT IS ILLUMINATED.

PREREQUISITES:

Electrical Power ON, Both Connectors Removed from SAS Computer.

START:



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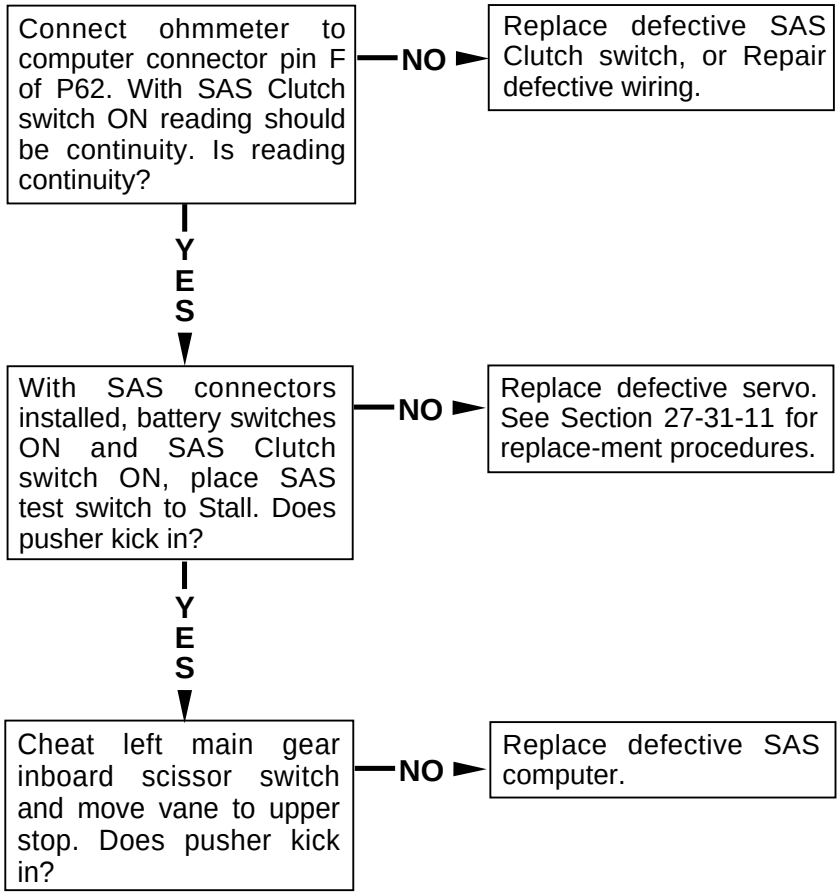
CONTINUE



END

6M273106104B

ROSEMOUNT SAS – FAULT ISOLATION

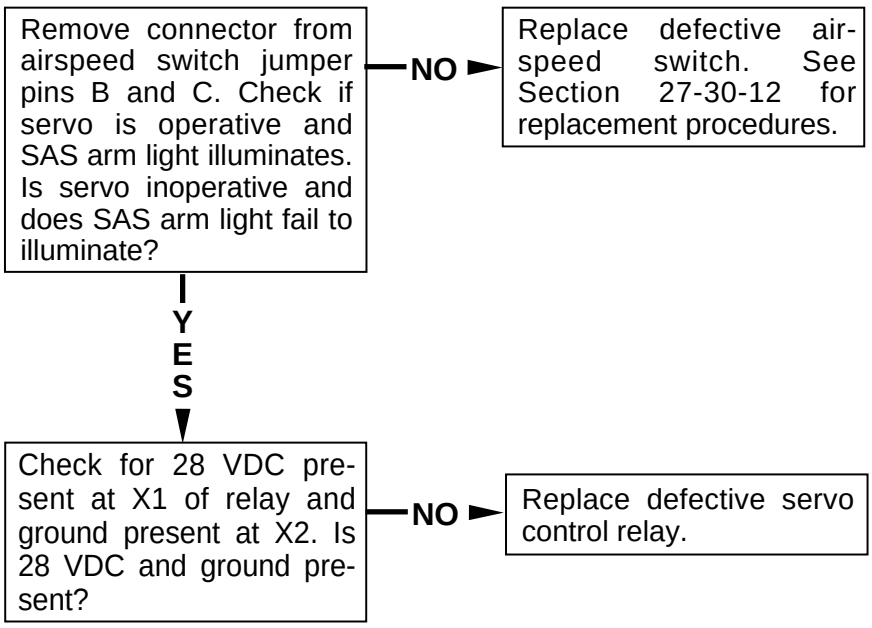
PROBLEM:

STICK PUSHER INOPERATIVE, SAS ARM LIGHT IS NOT ILLUMINATED.

PREREQUISITES:

Electrical Power ON.

START:



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END

PROBLEM:

ONE OR MORE COMPUTER TRIMMER POTS DO NOT HAVE ENOUGH ADJUSTMENT.

PREREQUISITES:

Electrical Power ON.

START:

Check SAS computer and AOA transmitter for proper adjustment. Are SAS computer and AOA trasnsmitter properly adjusted?

NO ►

Read second paragraph under B. Maintenance Practices in Section 27-30-07 and correct accordingly.

6M273106106

END

ROSEMOUNT SAS – FAULT ISOLATION

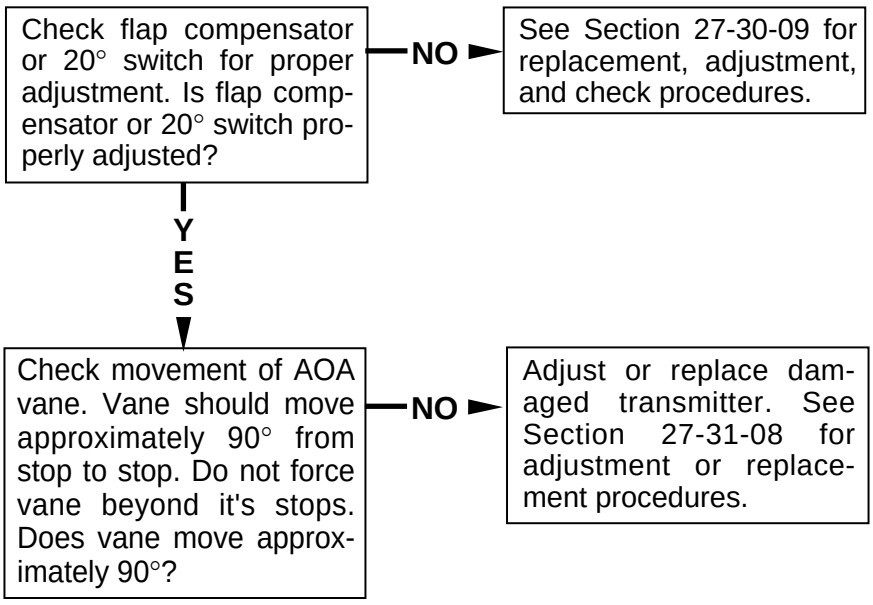
PROBLEM:

SAS SUDDENLY OUT OF ADJUSTMENT; PREVIOUSLY OK.

PREREQUISITES:

Electrical Power ON.

START:



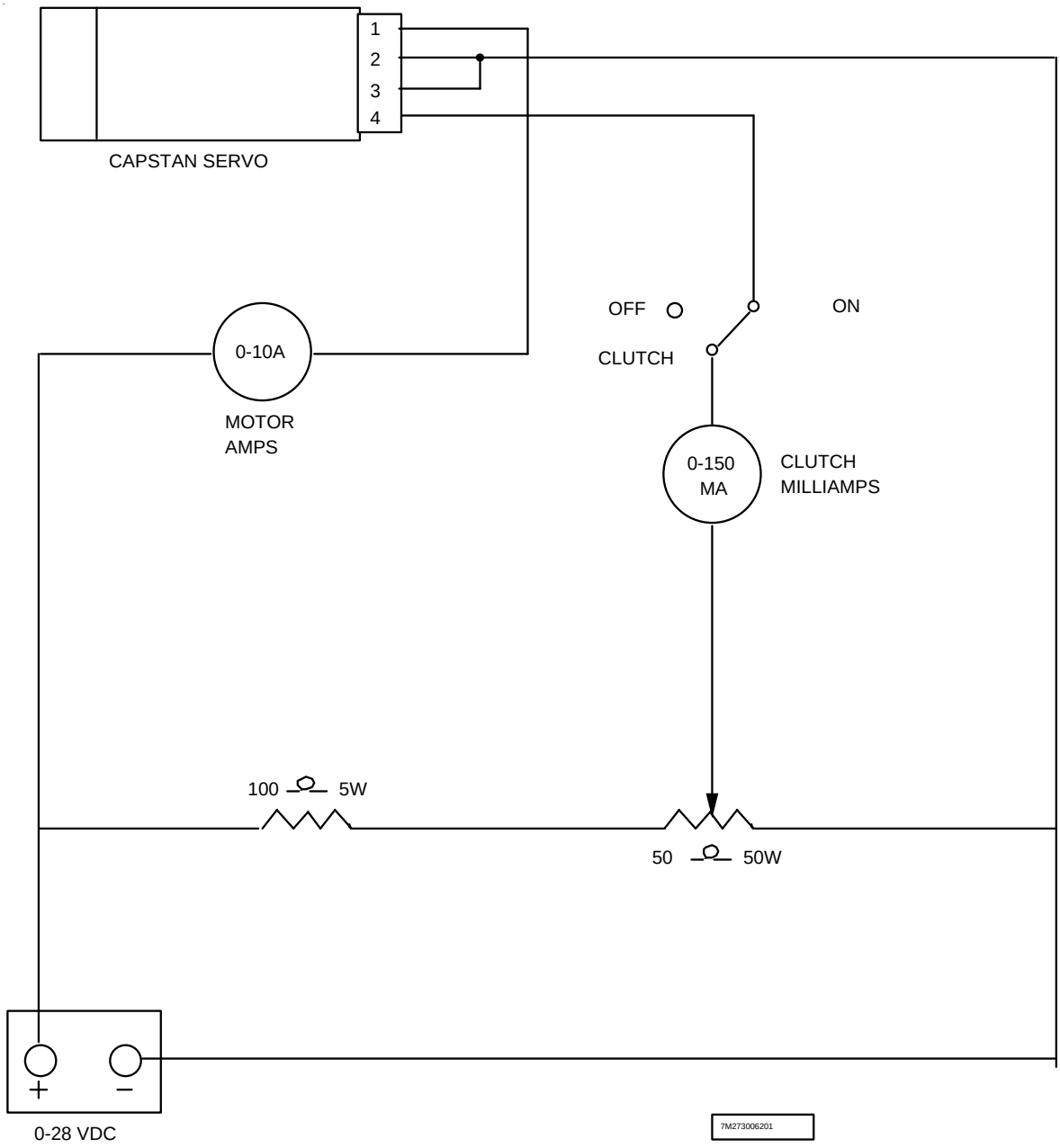
6M273106107

END

1. Inspection/Check – Rosemount SAS

NOTE: The following 200, 500, and 800 hour adjustment/tests apply to all SA226 series aircraft except SA226 – T(B) aircraft equipped with a Conrac SAS System, and T/AT/TC aircraft modified at customer's option IAW SB27-037 to use the Conrac SAS/SAS^2 system.

For Conrac SAS adjustment procedures, refer to 27-32-06, this chapter.



SAS Servo Functional Check Schematic
Figure 201

A. 200 Hour Inspection – Component Condition and Security

(1) Inspect the following components of the SAS system for condition and security.

(a) SAS Computer

(b) SAS Horn Assembly

(c) SAS/AOA Transmitter

(d) SAS Heater Blanket

(e) SAS Indicator

(f) Pusher Capstan

(g) Pusher Servo

B. 500 Hour Inspection – SAS Pusher Servo, and Capstan

NOTE: If servo fails the following inspection, it must be replaced by a serviceable unit which is either new, or overhauled by an FAA Approved Repair Station.

(1) Remove carpet as necessary, and center isle floorboards to gain access to the stick pusher servo and capstan, (approximately in line with the wing leading edge).

(2) Disconnect electrical connector from servo.

(3) Connect an instrumented variable D.C. power supply to servo as shown in Figure 201.

(4) To determine servo start voltage, turn clutch switch OFF, and starting at a minimum DC voltage, increase DC voltage until you hear motor begin to turn, note the voltage reading.

(5) To determine unloaded motor current, turn switch OFF, increase DC voltage to 28 VDC and note ampere reading.

(6) To determine loaded motor current, hold control wheel, turn clutch switch ON, with DC voltage at 28 VDC, increase clutch current to 60 milliamps and note motor ampere reading. Turn clutch switch OFF.

(7) To determine capstan condition, hold control wheel, ensure unrestricted travel, turn clutch switch ON, with voltage at 28 VDC, increase clutch current to 60 milliamps and increase force pulling aft on control wheel until capstan rachets or control wheel moves aft. (Minimum force should be approximately 70 pounds, maximum 86 pounds).

(8) If servo start voltage is in excess of 8.0 volts, DC unloaded motor current is in excess of 1.0 amps DC, or loaded motor current is in excess of 2.7 amps DC, replace SAS servo and recalibrate system.

(9) If capstan condition does not allow the control wheel and SAS servo to be overidden, the capstan must be disassembled, cleaned, recalibrated, and the torque reset IAW STICK PUSHER ASSEMBLY – ADJUSTMENT/TEST, this chapter.

(10) Disconnect variable DC power supply and reconnect aircraft harness.

(11) Turn battery switches ON, turn SAS CLUTCH switch ON, hold control column approximately vertical, turn SAS TEST switch to STALL, check that pusher comes on, release SAS TEST switch, turn battery switches OFF.

(12) Reinstall center isle floorboards and carpet.

C. 500 Hour Inspection – AOA Transmitter, SAS Computer

(1) Materials and equipment:

(a) Digital Ohmmeter – accuracy should be ± one ohm in the 2000 ohm range.

(b) Squat switch cheater tool, (see 27-SERVC).

(c) Protractor (P/N 11C0107) pointer (P/N 6023) for 12J or 9-12D1006 series transmitters, or protractor P/N 00092-2125-0001 for 12M series transmitters.

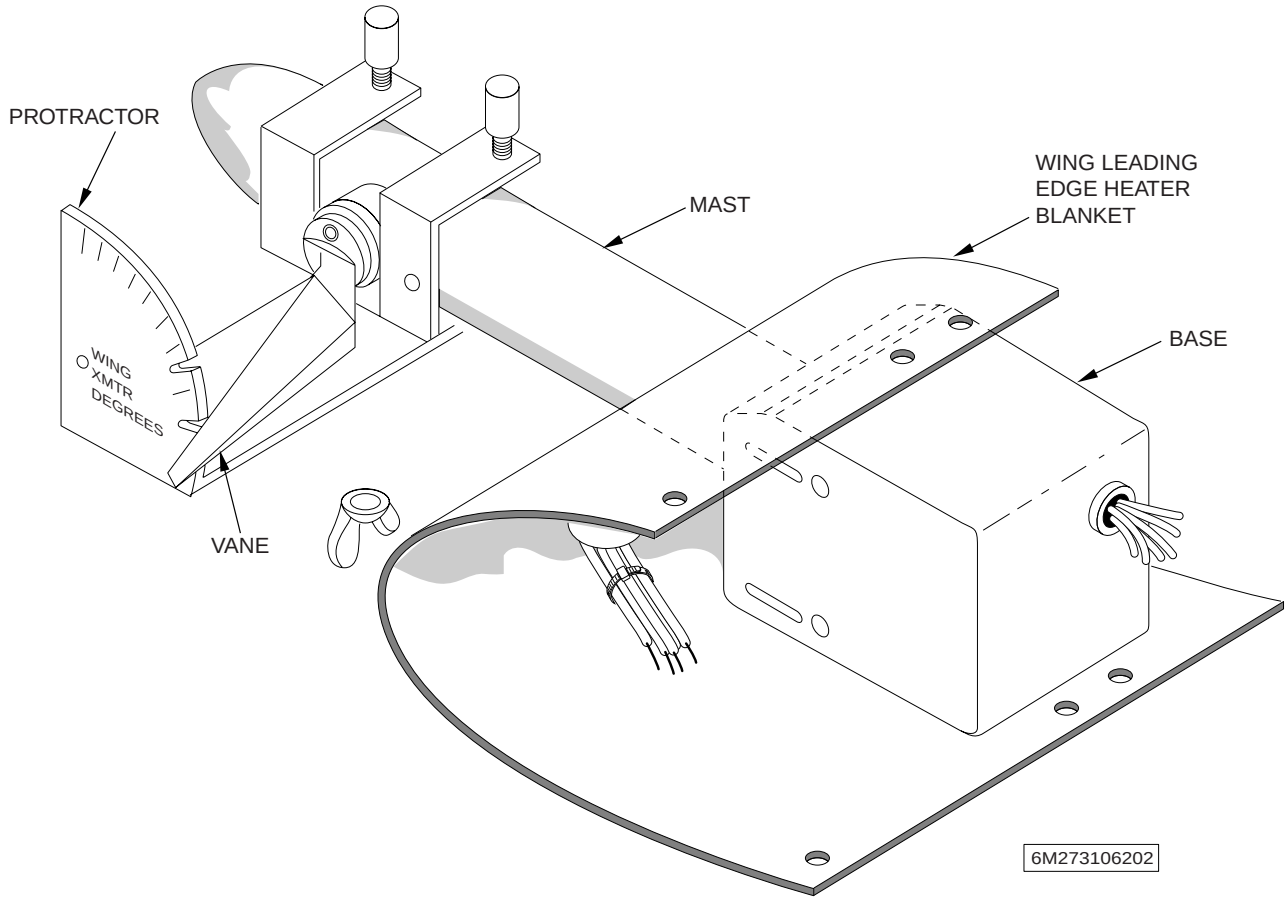
(d) Auxiliary ground unit (GPU).

(2) Remove RH wing tip to gain access to AOA transmitter.

(3) Remove screws attaching AOA transmitter and leading edge heater blanket.

(4) Clean old sealer from wing tip and leading edge.

(5) Remove cover from AOA transmitter. Inspect potentiometer, and drive belt, or drive linkage for condition and security.



AOA Transmitter
Heater Blanket, and Protractor
Figure 202

(6) Check AOA transmitter ohm reading:

(a) Remove orange and green wires from terminals 9 and 6.

(b) Connect ohmmeter leads to orange and green wires from AOA transmitter.

(c) Remove and retain screw in barrel of AOA transmitter directly behind vane. (Screw has the words REMOVE FOR CALIBRATION near it).

(d) Loosen wingnut on protractor.

(e) Position protractor base with thumb screw heads pointing toward the ground. Slide protractor base over tip of transmitter until vane is positioned at slot in base between thumb screws (see Figure 202).

(f) Rotate protractor until hole in base aligns with hole in barrel where screw was previously removed.

(g) Install screw through base into barrel and tighten.

(h) Lift vane to approximately 40° and align scale to slot in base. Push scale firmly but not tightly against vane and tighten wingnut.

(i) Vane should move in a smooth arc along scale and not move when released.

(j) For 12J, 9-12D1006, and 12M, ohm reading is taken at 47°.

(k) For 12J1 and 12m1, ohm reading is taken at 41°.

(l) Ohm reading should be ohm reading for SAS Wing Vane in Aircraft Log SAS DATA. Reading must be within 860 to 900 ohms. If vane is replaced, a new calibration and flight test are required.

(7) To adjust 12M or 12M1 transmitter:

(a) Loosen three screws in wedge clamps around base of potentiometer.

(b) If 12M transmitter is installed, place vane at 47°. If 12M1 transmitter is installed place vane at 41°. Turn potentiometer until ohmmeter reads ohm reading in Aircraft Log SAS DATA.

(c) Tighten screws in wedge clamps.

(d) Hold wires and slide cover over base of transmitter. Take care not to bunch wires up inside of case. Ensure grommet is installed in cover.

(e) Install screw attaching cover.

(8) To adjust 12J, 9-12D1006, or 12J1 transmitters:

(a) Loosen locknut at rod end.

(b) Remove C clip on pin through rod end and arm on potentiometer. Take care not to lose C clip. No other type of attachment is authorized.

(c) Adjust rod end in or out as required. Transmitter is adjusted to ohm reading in Aircraft Log SAS DATA at 47° for 12J or 9-12D1006, or at 41° for 12J1.

(d) Install pin through rod end and arm of potentiometer.

(e) Install C clip and tighten locknut to rod end.

(f) Recheck ohm reading.

(g) Install cover and attaching screw. Care should be taken not to bunch wires up inside of case. Ensure grommet is installed.

(9) Position transmitter on mounting pad and install four screws.

(10) Check terminals on wires for good connection. Replace or solder if necessary. Do not cut wires shorter, leave original length. Open-ended or forked terminals should be replaced.

(11) Reinstall leading edge heater blanket. Do not seal yet.

(12) Connect and energize GPU.

(13) Check AOA transmitter operation:

(a) Turn battery switches to ON.

(b) Move vane up; check pointer of SAS indicator moves counterclockwise toward STALL. Move vane down; check indicator needle moves clockwise.

(c) Turn pitot heat switch to PITOT and SAS. After a delay of 10 to 15 seconds check SAS DEICE light illuminates. Turn switch to OFF.

(d) Check that tip, vane, and barrel of transmitter are warm. Check that heater blanket is warm.

(e) Turn battery switches OFF.

(14) Reinstall wingtip.

(15) Check NAV and STROBE light operation.

(a) Turn battery switches ON.

(b) Turn NAV light switch to NAV&STROBE. Check NAV AND STROBE lights illuminate. Turn switch to OFF.

(c) Turn battery switches OFF.

(16) Seal edges of heater blanket per Maintenance Practices 27-31-08.

(17) Computer Functional Check:

(a) Remove forward door on right console and remove connectors on computer.

NOTE: Power should always be off when connecting or disconnecting computer.

(b) Connect ohmmeter to pins B and C of computer connector P62 (big key over pin A).

(c) With vane protractor installed, set vane angle to 47° (12J, 9-12D1006, and 12M), or 41° (12J1 and 12M1).

(d) Reading should be ohm reading for SAS Wing Vane in Aircraft Log SAS DATA.

(e) Connect SAS computer to aircraft wiring.

(f) Make sure SAS vane is at bottom stop.

(g) Using squat switch cheater tool depress inboard scissor switch (six wire switch) on left main gear.

(h) Make sure control gust lock is disengaged.

(i) Turn SAS CLUTCH switch OFF.

(j) Turn battery switches ON.

(k) Check SAS ARM light illuminated and SAS FAULT light blinking.

(l) Hold control column back slightly whenever stick pusher is expected to actuate.

(m) Turn SAS CLUTCH switch ON.

(n) Check SAS FAULT light extinguished.

(o) Lower flaps.

(p) Have assistant move SAS vane up slowly.

(q) Check that needle of SAS indicator moves counterclockwise.

(r) Check that the stall warning horn sounds, and the stick pusher actuates at the angles indicated on previous SAS DATA Sheet in Aircraft Log.

(s) Remove squat switch cheater tool from scissor switch.

(t) Remove protractor from SAS transmitter.

(u) Check pusher off.

(v) Turn SAS CLUTCH switch OFF.

(w) Turn battery switches OFF.

D. 800 Hour Inspection and Recalibration

NOTE: Accomplish 200 flight hour inspection, and 500 flight hour servo functional check in conjunction with this procedure.

This procedure provides recalibration procedures for the Rosemount SAS P/N 9-20C1006, 9-20C1008, or 20AA1 computers in AT and TC aircraft and the Rosemount SAS P/N 9-20C1007, 9-20C1009, or 20AA1 computers in T aircraft.

(1) Materials and Equipment:

(a) Digital Ohmmeter - accuracy should be one ohm in the 2000 ohm range.

(b) Squat switch cheater tool, (see 27-SERVC).

(c) Protractor (P/N 11C0107) pointer (P/N 6023) for 12J or 9-12D1006 series transmitters or protractor P/N 00092-2125-0001 for 12M series transmitters.

(d) Control column force gauge.

(e) Auxiliary ground power unit (GPU).

(f) Auxiliary hydraulic power unit.

(g) SAS Calibrator, TS27-03 Model 7106 or equivalent.

(2) Remove forward door on RH console and remove SAS computer.

(3) Remove cover from computer.

NOTE: Be sure power is OFF when connecting or disconnecting computer.

(4) Connect GPU and hydraulic unit.

(4) Connect ohmmeter to pins C and D of computer connector P61. Connect computer calibrator if applicable.

(5) Flaps UP reading should be reading for Flap Compensator in Aircraft log SAS DATA, 360 ± 10 ohms. Flaps DOWN reading should be SAS DATA reading, 1650, ± 10 ohms.

(6) Connect ohmmeter to pins B and C of computer connector P62.

(7) Install protractor on SAS transmitter. Reading should be reading for SAS Wing Vane in Aircraft Log SAS DATA or 880, ± 10 ohms at 47° for 12J, 9-12D1006, or 12M transmitter, or 41° for 12J1 or 12M1 transmitter.

(8) If a new vane is installed, adjust to match ohm reading from Aircraft.

(9) Install connectors on computer.

(10) Turn SAS CLUTCH switch ON.

(11) Turn Battery switches ON. Observe SAS ARM light ON, and SAS FAULT light OFF.

(12) Turn SAS CLUTCH switch OFF.

(13) Observe SAS FAULT light now blinking.

(14) Move vane to approximately 0°.

(15) Move flaps down.

(16) As flaps extend, needle on indicator should move counterclockwise. T205E, T218, & up, AT006 & up, TC203 & up only, needle should jump about a needle width in direction of movement as flaps pass between 1/2 and 3/4.

(17) Pull STALL WARN circuit breaker.

(18) Depress inboard scissor switch on left main gear (six wire switch) with squat switch cheater tool.

(19) Merlin III (SA226-T) only – Substitute appropriate vane angles from Aircraft Log. Refer to Table 201 for vane angles and expected SAS indicator readings on your aircraft.

(a) Move vane to ___°, adjust IND STALL so indicator reads 1.0.

(b) Move vane to ___°, adjust IND CRUISE so indicator reads 1.3.

(c) Reset STALL WARN circuit breaker.

(d) Move vane up to ___°, adjust S/W HORN so horn comes on a needle width before 1.1 on indicator (1.12 V/Vs).

(e) Pull STALL WARN circuit breaker.

(f) Turn SAS CLUTCH switch ON.

(g) Move vane to bottom step (should be between 0° and 10°), adjust CLUTCH ENTRY to obtain a reading of 0 to 3 volts.

NOTE: If vane only goes to 0°, adjust to 3 volts; if vane goes to -5°, adjust to 1.5 volts, etc. The numbers in the Aircraft Log are taken with the flaps down and these numbers should be used. This adjustment can be made with flaps up, move vane until indicator needle is between the R and U in CRUISE and set CLUTCH ENTRY to 0 volts.

(h) Move vane to ___°, or until needle on indicator is on 1.4; adjust CLUTCH VOLTS to 8.25 volts and adjust CLUTCH GAIN to 30mA.

(i) Move vane up slowly to ___°, adjust PUSHER so pusher kicks in as needle reaches 1.02 on indicator. Adjust GAIN to 58 mA. On the 20AA1 computer the pusher gain pot (R49) is located in the middle of the top PC board.

(j) Hold force gauge to top center of pilot control column. Pull control column to vertical position. SET FORCE GAUGE DIAL TO ZERO. Activate the pusher and check for 60 to 68 pound force. If necessary, adjust gain control to obtain 60 to 68 pound force.

(k) With stick pusher force present, increase force pulling aft on control wheel until capstan ratchets. The total force required must be less than 86 pounds. Capstan ratcheting can be felt and heard as a jerking aft movement of the control column.

(l) If this jerking movement is not present, or the force required is 86 pounds or more, the capstan must be disassembled , cleaned, relubricated, and the torque reset per STICK PUSHER ASSEMBLY – Adjustment/Test, this chapter.

COMPUTER ADJUSTMENT	SAS INDICATOR V/Vs	SA226AT and TC AOA TRANSMITTER		SA226T AOA TRANSMITTER	
		12J/12J1	12M/12M1	12J/12J1	12M/12M1
IND STALL	1.0 V/Vs	42°	45°	43°	46°
IND CRUISE	1.3 V/Vs	19°	22°	20°	23°
PUSHER (ENTRY)	1.02 V/Vs	41°	44°	42°	45°
HORN	1.12V/Vs	32°	35°	33°	36°
CLUTCH ENTRY 0 VOLTS	1.8 V/Vs	N/A	N/A	-10°	-7°
CLUTCH VOLTS 8.25	1.4 V/Vs	N/A	N/A	12°	15°

Rosemount SAS
Vane Angles with Flaps Down
Table 201

(20) Merlin IV and Metro (SA226-AT and TC) only – substitute appropriate vane angles from Aircraft Log. Refer to Table 203 for vane angles and expected SAS indicator readings on your aircraft.

(a) Move vane to ___°, adjust IND STALL so indicator reads 1.0.

(b) Move vane to ___°, adjust IND CRUISE so indicator reads 1.3.

(c) Reset STALL WARN circuit breaker.

(d) Move vane up to ___°, adjust S/W HORN so horn comes on a needle width before 1.1 on indicator (1.12 V/Vs).

(e) Pull STALL WARN circuit breaker.

(f) Turn SAS CLUTCH switch ON.

(g) Move vane up slowly to ___°, adjust ENTRY (PUSHER 20AA1 computer) so pusher kicks in at 1.02 on indicator, adjust GAIN to 58 mA. Gain should not required adjustment, but can be checked by connecting SAS calibrator. 20AA1 computer trimmer pot R49 is located in middle of top PC board.

(h) Hold force gauge to top center of pilot control column. Pull control column to vertical position. SET FORCE GAUGE TO ZERO Activate the pusher and check for 60 to 68 pound force. If necessary, adjust gain control to obtain 60 to 68 pound force.

(i) With stick pusher force present, increase force pulling aft on control wheel until capstan ratchets. The total force required must be less than 86 pounds. Capstan ratcheting can be felt and heard as a jerking aft movement of the control column.

(j) If this jerking movement is not present, or the force required is 86 pounds or more, the capstan must be disassembled , cleaned, relubricated, and the torque reset per STICK PUSHER ASSEMBLY – Adjustment/Test, this chapter.

(21) Remove squat switch cheater tool from left scissor switch.

(22) Move flaps up.

(23) Turn SAS TEST switch to CRUISE, adjust CRUISE TEST so needle on indicator is on 1.3.

(24) Turn SAS TEST switch to STALL, adjust STALL TEST so needle on indicator is in 1.0.

(25) Reset STALL WARN circuit breaker.

(26) Turn SAS TEST switch to STALL, horn should come on and pusher should kick in.

(27) Turn SAS CLUTCH switch OFF.

(28) Turn battery switches OFF.

(29) Disconnect GPU and hydraulic power unit.

(30) Remove vane protractor and pointer.

(31) Preflight Set-Up Check

(a) Confirm SAS vane is free from drag.

(b) Obtain Airspeed Calibration card and have available for use during flight testing.

(c) Obtain gross weight of aircraft and make note of Table 202. It is suggested that aircraft be at maximum takeoff weight.

GROSS WEIGHT:

Basic_____pounds

Crew_____pounds

Fuel_____pounds

Total_____pounds (at engine start)

Gross Weight
Table 202

(32) Preflight System Check

(a) With at least one engine running and generator on the line, and with all SAS circuit breakers engaged, turn the SAS CLUTCH switch OFF.

(b) Confirm SAS ARM light is illuminated and SAS FAULT light is blinking.

(c) Turn the SAS CLUTCH switch ON, observe the SAS FAULT light extinguished.

(d) Confirm flaps up.

(e) Hold SAS TEST switch to CRUISE. Confirm pointer of SAS indicator reads 1.3 V/Vs precisely.

(f) While holding SAS TEST switch to CRUISE:

1 Move flaps down.

2 Observe pointer of SAS indicator travels counterclockwise. T205E, T218 & up, AT006 & up, TC203 & up only, needle should jump about a needle width in direction of movement as flaps pass between 1/2 and 3/4.

3 While flaps are full down, confirm pointer of SAS indicator reads approximately 1.2 V/Vs (middle of yellow SLOW area).

4 Move flaps up, confirm pointer of SAS indicator returns to 1.3 V/Vs when flaps are up.

5 Merlin III only – Approximately 20 pounds of forward stick force should be felt.

(g) Hold the SAS TEST switch to STALL and observe the SAS indicator and elevator control. Pointer of the SAS indicator should travel counterclockwise to 1.0 V/Vs, the pusher should kick in, and the horn should sound. Release switch.

(33) In-Flight Adjustment/Test

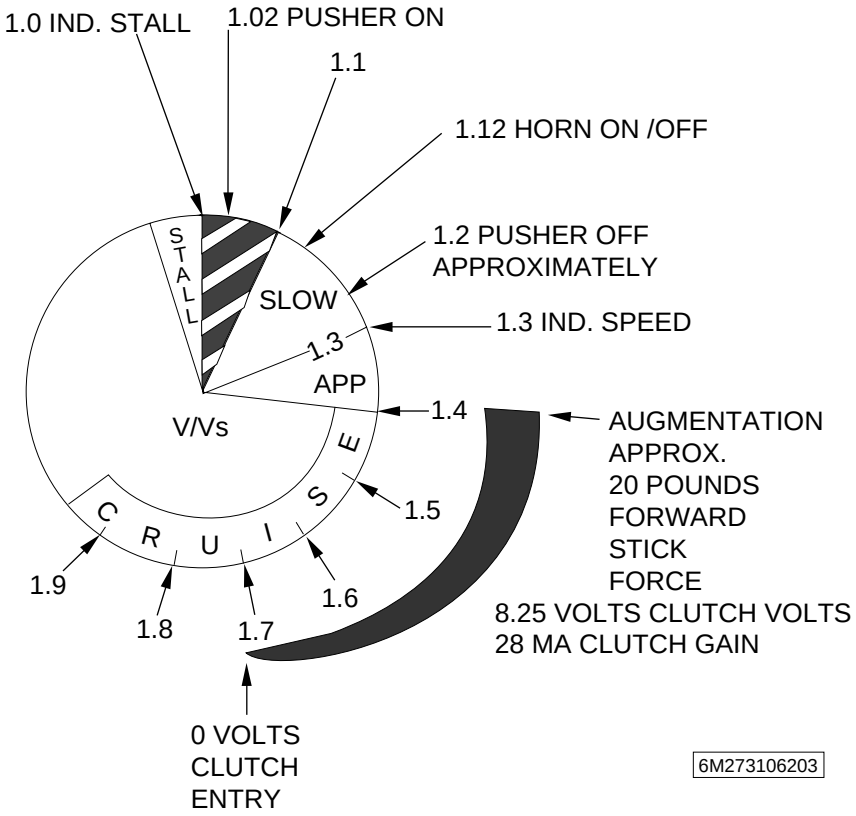
WARNING: DO NOT PERMIT THE AIRCRAFT TO ENTER FULL STALL. TERMINATE THE TESTS AT THE LISTED STALL SPEEDS OR AT THE FIRST INDICATION OF EXCESSIVE WING HEAVINESS OR BREAK.

ALL PERTINENT FAA REGULATIONS SHOULD BE OBSERVED WHILE PERFORMING THE MANEUVERS REQUIRED IN SAS SETUP. IN ADDITION, THE FOLLOWING MANEUVERS PLACE THE AIRCRAFT IN A DESCENDING CONFIGURATION AND AMPLE ALTITUDE SHOULD BE ALLOWED FOR PROPER RECOVERY.

PRIOR TO THE FIRST CALIBRATION FLIGHT, PLACE THE SAS CLUTCH IN THE OFF POSITION DURING TAKEOFF AND INITIAL CLIMB OUT.

(a) For aircraft S/N T 201-275, and T 277-291, refer to Figures 203 and 204 for In-Flight test procedures. For aircraft S/N AT 001-999, and TC 201-999, refer to Figures 205 and 206 for In-Flight test procedures.

(b) SAS calibrator TS27-03 contains a number of test jacks, a voltmeter, and a milliammeter. The jacks are primarily for ground use. The voltmeter and milliammeter are used for steps 4 and 5 of the In-Flight Set-Up, Merlin III only. The milliammeter can also be used for setting pusher current (PUSHER GAIN). The PUSHER GAIN pot (R49) for the 20AA1 computer is located in the middle of the top PC board.



SAS INDICATOR ADJUSTMENT POINTS
T 201-275, 277-291
Figure 203

ROSEMOUNT SAS
MERLIN III SA226T SAS^2 IN-FLIGHT SET-UP

STEP	PURPOSE OF MANEUVER	SAS CLUTCH SWITCH	GEAR/ FLAPS	POWER LEVERS	WEIGHT POUNDS	AIRSPEED KIAS		PROCEDURE/REMARKS
						TRIM TO	TEST SPEED	
1	To adjust IND. STALL pot, aircraft clean	OFF	UP	F/I	12,500	142	98	Decrease 1 knot/sec. to test speed. Adjust IND. STALL pot to position SAS indicator pointer at 1.0V/Vs when at the appropriate test speed.
					12,000	140	96	
					11,500	137	94	
					11,000	134	92	
					10,500	131	90	
					10,000	128	88	
2	To adjust IND. STALL pot, aircraft dirty	OFF	FULL DOWN	F/I	12,500	127	83	Decrease 1 knot/sec. to test speed. Confirm that SAS indicator pointer is positioned at 1.0 V/Vs. If not, adjust IND. STALL pot to split the difference with the test speed step 1. Repeat steps 1 and 2 as required.
					12,000	125	82	
					11,500	122	80	
					11,000	120	79	
					10,500	118	77	
					10,000	115	76	
3	To adjust IND. CRUISE pot, aircraft dirty	OFF	FULL DOWN	F/I	12,500	110	110	Trim to speed, adjust IND. CRUISE pot to position the SAS indicator pointer 1.3 V/Vs.
					12,000	108	108	
					11,500	106	106	
					11,000	103	103	
					10,500	101	101	
					10,000	98	98	
4	To adjust CLUTCH ENTRY pot. Start of stability augmentation ramp.	OFF	UP	As Required	N/A	180	180	Verify that SAS ARM light on annunciator is illuminated. Adjust CLUTCH ENTRY pot to obtain 0 volts on SAS calibrator (TS27-03).

6M273106204A

In-Flight Set-Up Charts
T 201-275, 277-291
Figure 204

ROSEMOUNT SAS
MERLIN III SA226T SAS^2 IN-FLIGHT SET-UP (Continued)

STEP	PURPOSE OF MANEUVER	SAS CLUTCH SWITCH	GEAR/ FLAPS	POWER LEVERS	WEIGHT POUNDS	AIRSPEED KIAS		PROCEDURE/REMARKS
						TRIM TO	TEST SPEED	
5	To adjust CLUTCH VOLTS and CLUTCH GAIN pots. Top of stability augmentation ramp.	ON	UP	As Required	N/A	See Procedures	See Procedures	Trim to maintain SAS indicator pointer on 1.4 V/Vs. Adjust CLUTCH VOLTS pot to obtain 8.25 volts plus or minus 25 volts on SAS calibration (TS27-03). Adjust CLUTCH GAIN pot to obtain 28 MA plus or minus 1 MA on SAS calibration (TS27-03).
6	Short cut to adjusting CLUTCH PUSHER pot. This step is not mandatory. If you do not care to do it, go directly to step 7.	OFF/ ON	UP	As Required	N/A	See Procedures	See Procedures	Hold SAS TEST switch to STALL. Adjust TEST STALL pot to position SAS indicator pointer to 1.02 V/Vs. Then turn SAS CLUTCH switch ON. Hold SAS TEST switch to STALL and adjust CLUTCH PUSHER pot so pusher is off then just on. When adjusting pusher off, release SAS TEST switch, make adjustment, then reset SAS TEST switch to STALL.
7	To adjust CLUTCH PUSHER pot with aircraft clean.	ON	UP	F/I	12,500	142	100	Decrease 1 knot/sec. to test speed. Adjust CLUTCH PUSHER pot to actuate pusher at test speed. Because pusher remains actuated until speed is increased above 1.2 Vs, several attempts may be required.
					12,000	140	98	
					11,500	137	96	
					11,000	134	94	
					10,500	131	92	
					10,000	128	90	
8	To adjust PUSHER with aircraft dirty.	ON	FULL DOWN	F/I	12,500	127	83	Decrease 1 knot/sec. to test speed. Verify that pusher activates within 4 knots above test speed. If not, adjust CLUTCH PUSHER pot and repeat steps 7 and 8 to achieve optimum intermediate adjustment.
					12,000	125	82	
					11,500	122	80	
					11,000	120	79	
					10,500	118	77	
					10,000	115	76	

6M273106204B

In-Flight Set-Up Charts
T 201-275, 277-291 (Continued)
Figure 204

ROSEMOUNT SAS
MERLIN III SA226T SAS^2 IN-FLIGHT SET-UP (Continued)

STEP	PURPOSE OF MANEUVER	SAS CLUTCH SWITCH	GEAR/ FLAPS	POWER LEVERS	WEIGHT POUNDS	AIRSPEED KIAS		PROCEDURE/REMARKS
						TRIM TO	TEST SPEED	
9	Short cut to adjusting S/W HORN pot. This step is not mandatory. If you do not care to do it, go directly to step 10.	ON	FULL DOWN	As Required	N/A	See Procedures	See Procedures	Fly aircraft so SAS indicator pointer is on 1.12 V/Vs. Adjust S/W HORN pot so horn is off, then just on.
10	To adjust S/W HORN pot.	ON	FULL DOWN	F/I	12,500	127	88 - 93	Decrease 1 knot/sec. to test speed. Adjust S/W HORN pot to actuate horn within the specified test speed range.
					12,000	125	87 - 92	
					11,500	122	85 - 90	
					11,000	120	84 - 89	
					10,500	117	82 - 87	
					10,000	115	80 - 85	
11	To adjust TEST CRUISE pot.	ON	UP	N/A	N/A	N/A	N/A	Hold SAS TEST switch to CRUISE. Adjust TEST CRUISE pot to position SAS indicator pointer at 1.3 V/Vs.
12	To adjust TEST STALL pot.	OFF/ ON	UP	N/A	N/A	N/A	N/A	Hold SAS TEST switch to STALL. Adjust TEST STALL pot to position SAS indicator pointer at 1.0 V/Vs. Then turn SAS CLUTCH switch ON. Hold SAS TEST switch to STALL and verify pusher activates.

6M273106204C

In-Flight Set-Up Charts
T 201-275, 277-291 (Continued)
Figure 204

(c) Do not test during turbulent conditions.

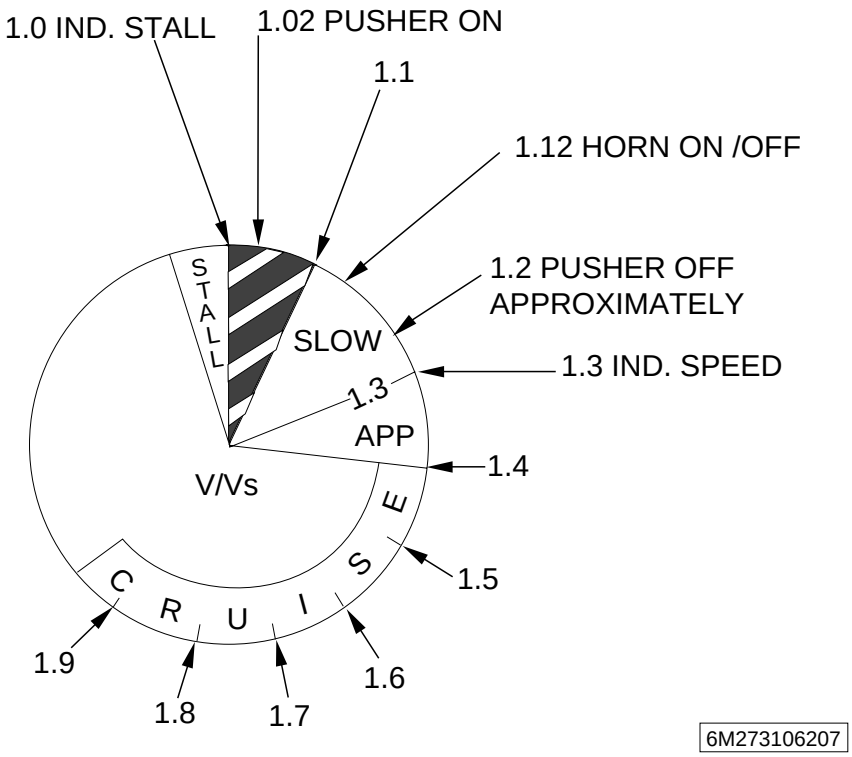
(d) Adjustments to the computer must be accomplished in the order they appear on the In-Flight Set-Up sheet as they are interacting, and if not done in order will affect the previous adjustment.

(e) RPM levers should remain at 100% at all times.

(f) Trim to zero stick force at the trim speed corresponding to airplane weight as listed in each test when applicable. Do not retrim within a test. Achieve test speeds by elevator control.

(g) Approach each test speed from the appropriate trim speed with a speed bleed-off rate of one knot per second. Otherwise, wrong values of stall, stick pusher, and stall warning will result.

(h) Required speeds are tabulated as functions of airplane weight requiring that the weight be calculated for each test. This requires fuel accountability throughout the flight program.



SAS Indicator Adjustment Points
AT001-000, TC201-999
Figure 205

ROSEMOUNT SAS
MERLIN IV AND METRO SA226AT AND TC SAS IN-FLIGHT SET-UP

STEP	PURPOSE OF MANEUVER	SAS CLUTCH SWITCH	GEAR/ FLAPS	POWER LEVERS	WEIGHT POUNDS	AIRSPEED KIAS		PROCEDURE/REMARKS
						TRIM TO	TEST SPEED	
1	To adjust IND. STALL pot, aircraft clean	OFF	UP	F/I	12,500	146	98	Decrease 1 knot/sec. to test speed. Adjust IND. STALL pot to position SAS indicator pointer at 1.0V/Vs when at the appropriate test speed.
					12,000	143	96	
					11,500	140	94	
					11,000	137	92	
					10,500	134	90	
					10,000	131	88	
2	To adjust IND. STALL pot, aircraft dirty	OFF	FULL DOWN	F/I	12,500	132	87	Decrease 1 knot/sec. to test speed. Confirm that SAS indicator pointer is positioned at 1.0 V/Vs. If not, adjust IND. STALL pot to split the difference with the test speed step 1. Repeat steps 1 and 2 as required.
					12,000	128	85	
					11,500	126	84	
					11,000	123	82	
					10,500	120	80	
					10,000	118	78	
3	To adjust IND. CRUISE pot, aircraft dirty	OFF	FULL DOWN	F/I	12,500	113	113	Trim to speed, adjust IND. CRUISE pot to position the SAS indicator pointer 1.3 V/Vs.
					12,000	110	110	
					11,500	108	108	
					11,000	106	106	
					10,500	103	103	
					10,000	101	101	
4	Short cut to adjusting CLUTCH ENTRY/ PUSHER pot. This step is not mandatory. If you do not care to do it, go directly to step 5.	OFF/ ON	UP	As Required	N/A	See Procedures	See Procedures	Hold SAS TEST switch to STALL. Adjust TEST STALL pot to position SAS indicator pointer to 1.02 V/Vs. Then turn SAS CLUTCH switch ON. Hold SAS TEST switch to STALL and adjust CLUTCH ENTRY/PUSHER pot so pusher is off, then just on. When adjusting pusher off, release SAS TEST switch, make adjustment, then reset SAS TEST switch to STALL.

6M273106206A

In-Flight Set-Up Charts
AT001-999, TC201-999
Figure 206

ROSEMOUNT SAS
MERLIN IV AND METRO SA226AT AND TC SAS IN-FLIGHT SET-UP (Continued)

STEP	PURPOSE OF MANEUVER	SAS CLUTCH SWITCH	GEAR/ FLAPS	POWER LEVERS	WEIGHT POUNDS	AIRSPEED KIAS		PROCEDURE/REMARKS
						TRIM TO	TEST SPEED	
5	To adjust CLUTCH ENTRY/PUSHER pot with aircraft clean.	ON	UP	F/I	12,500	146	99	Decrease 1 knot/sec. to test speed. Adjust CLUTCH ENTRY/PUSHER pot to actuate pusher at test speed. Because pusher remains actuated until speed is increased above 1.2 Vs, several attempts may be required.
					12,000	143	97	
					11,500	140	95	
					11,000	137	93	
					10,500	134	91	
					10,000	131	89	
6	To check PUSHER with aircraft dirty.	ON	FULL DOWN	F/I	12,500	132	87	Decrease 1 knot/sec. to test speed. Verify that pusher activates within 4 knots above test speed. If not, adjust CLUTCH ENTRY/PUSHER pot and repeat steps 5 and 6 to achieve optimum intermediate adjustment.
					12,000	128	85	
					11,500	126	84	
					11,000	123	82	
					10,500	120	80	
					10,000	118	78	
7	Short cut to adjusting S/W HORN pot. This step is not mandatory. If you do not care to do it, go directly to step 8.	ON	FULL DOWN	As Required	N/A	See Procedures	See Procedures	Fly aircraft so SAS indicator pointer is on 1.12 V/Vs. Adjust S/W HORN pot so horn is off, then just on.
8	To adjust S/W HORN pot.	ON	FULL DOWN	F/I	12,500	132	92 - 97	Decrease 1 knot/sec. to test speed. Adjust S/W HORN pot to actuate horn within the specified test speed range.
					12,000	128	90 - 95	
					11,500	126	89 - 94	
					11,000	123	87 - 92	
					10,500	120	85 - 90	
					10,000	118	83 - 88	

6M273106206B

In-Flight Set-Up Charts
AT001-999, TC201-999 (Continued)
Figure 206

ROSEMOUNT SAS
MERLIN IV AND METRO SA226AT AND TC SAS IN-FLIGHT SET-UP (Continued)

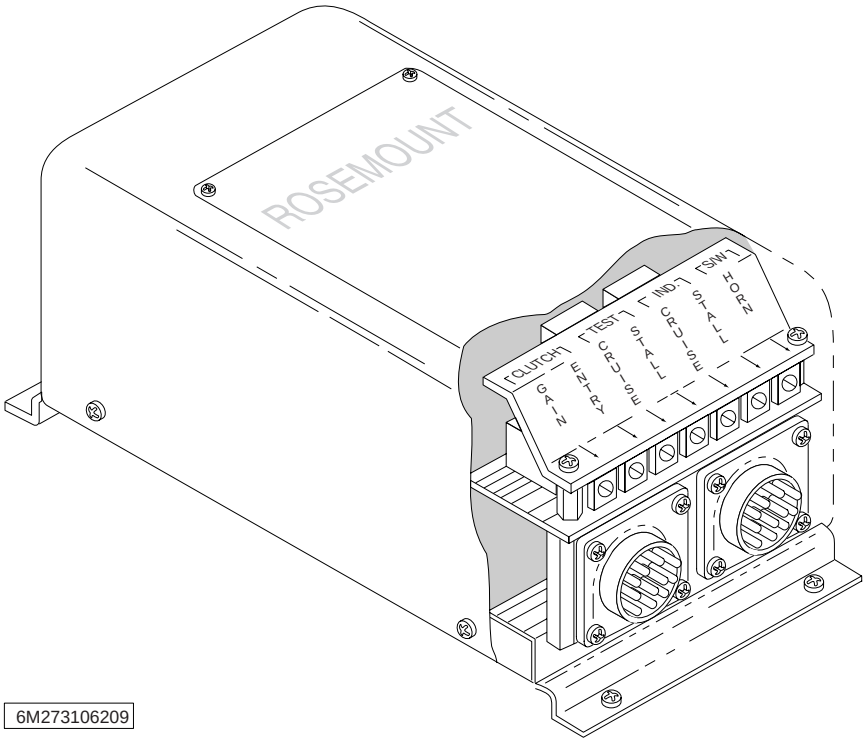
STEP	PURPOSE OF MANEUVER	SAS CLUTCH SWITCH	GEAR/ FLAPS	POWER LEVERS	WEIGHT POUNDS	AIRSPEED KIAS		PROCEDURE/REMARKS
						TRIM TO	TEST SPEED	
9	To adjust TEST CRUISE pot.	ON	UP	N/A	N/A	N/A	N/A	Hold SAS TEST switch to CRUISE. Adjust TEST CRUISE pot to position SAS indicator pointer at 1.3 V/Vs.
10	To adjust TEST STALL pot.	OFF/ ON	UP	N/A	N/A	N/A	N/A	Hold SAS TEST switch to STALL. Adjust TEST STALL pot to position SAS indicator pointer at 1.0 V/Vs. Then turn SAS CLUTCH switch ON and hold SAS TEST switch to STALL verify activates.

6M273106206C

In-Flight Set-Up Charts
AT001-999, TC201-999 (Continued)
Figure 206

(i) If the aircraft stalls at a higher speed than the listed test speed, the airspeed indication system is faulty, or the wrong bleedoff rate was used.

(j) Computer adjustment pots are shown in Figure 207.



Location
Adjustment Pots
Figure 207

(k) To move SAS indicator pointer clockwise or to increase airspeed to test speed, turn pot clockwise. To move SAS indicator pointer counterclockwise or to decrease airspeed to test speed, turn pot counterclockwise. CLUTCH PUSHER pot – To increase pusher speed turn pot clockwise; to decrease pusher speed turn pot counterclockwise.

(34) Record SAS data after In-Flight Set-Up.

NOTE: If any adjustments were made during In-Flight Adjustment/Test, record the new numbers on a new SAS DATA sheet (see Figure 209), using the following procedures, and replace old SAS DATA sheet in Aircraft Log book.

Read all vane angles with vane traveling up the scale.

(a) Connect and energize GPU and hydraulic power unit.

(b) Install protractor on AOA transmitter.

(c) Set vane at 41° for 12J1 or 12M1 transmitter, 47° for 12J, 9-12D1006, or 12M transmitter.

(d) Remove connectors from computer.

(e) Connect ohmmeter leads to pins B and C of computer connector P62.

(f) Record ohm reading for SAS Wing Vane.

(g) Connect ohmmeter leads to C and D of computer connector P61.

(h) With flaps up record ohm reading for flap Compensator.

(i) Move flaps down. Record reading for flap Compensator.

(j) With battery switches OFF, install connectors on computer.

(k) Depress inboard scissor switch on left main gear (six wire switch) with squat switch cheater tool.

(l) Move vane to bottom stop.

(m) Turn battery switches ON. Turn SAS CLUTCH switch ON.

(n) Merlin III only: Move vane up until voltmeter reads volts. Record vane degrees for 0 Volts on SAS Calibrator. If vane is at bottom stop and voltmeter is reading above 0 volts change 0 to read that voltage, and record minus vane degrees.

(o) Merlin III only: Move vane up until voltmeter reads 8.25 volts. Record vane degrees for 8.25 volts on SAS Calibrator.

(p) Move vane up until needle on SAS indicator reads 1.3. Record vane degrees for SAS indicator pointer 1.3.

(q) Move vane up slowly until stall warning horn sounds. Record vane degrees for stall warning horn.

(r) Pull STALL WARN circuit breaker.

(s) Move vane up slowly until pusher kicks in. Record vane degrees for Stick Pusher.

(t) Remove squat switch cheater tool from left main gear scissor switch.

(u) Move vane up until SAS indicator needle reads 1.0. Record vane degrees for SAS Indicator Pointer 1.0.

(v) Remove protractor from transmitter and move vane to bottom stop.

(w) Move flaps up.

(x) Recheck TEST STALL and TEST CRUISE.

(y) Turn battery switches OFF.

(z) Replace all components and disconnect support equipment:

1 Remove connectors from computer.

2 Remove test box if applicable.

3 Install cover on computer.

4 Install computer in right console.

5 Install connectors on computer.

6 Disconnect GPU and hydraulic power unit from aircraft.

SAS DATA AIRPLANE SERIAL NO. _____

Flaps Down:

SAS Indicator Pointer

1.0 _____°

1.3 _____°

Stick Pusher _____°

Stall Warning Horn _____°

0 Volts on SAS Calibrator * _____°

8.25 Volts on SAS Calibrator * _____°

Flap Compensator (Ohms) _____

Flaps Up:

Flap compensator (Ohms) _____

SAS Wing Vane at 47° or 41° (Ohms) _____

*** Merlin III, SA226-T only**

SAS Data Sheet
Table 202

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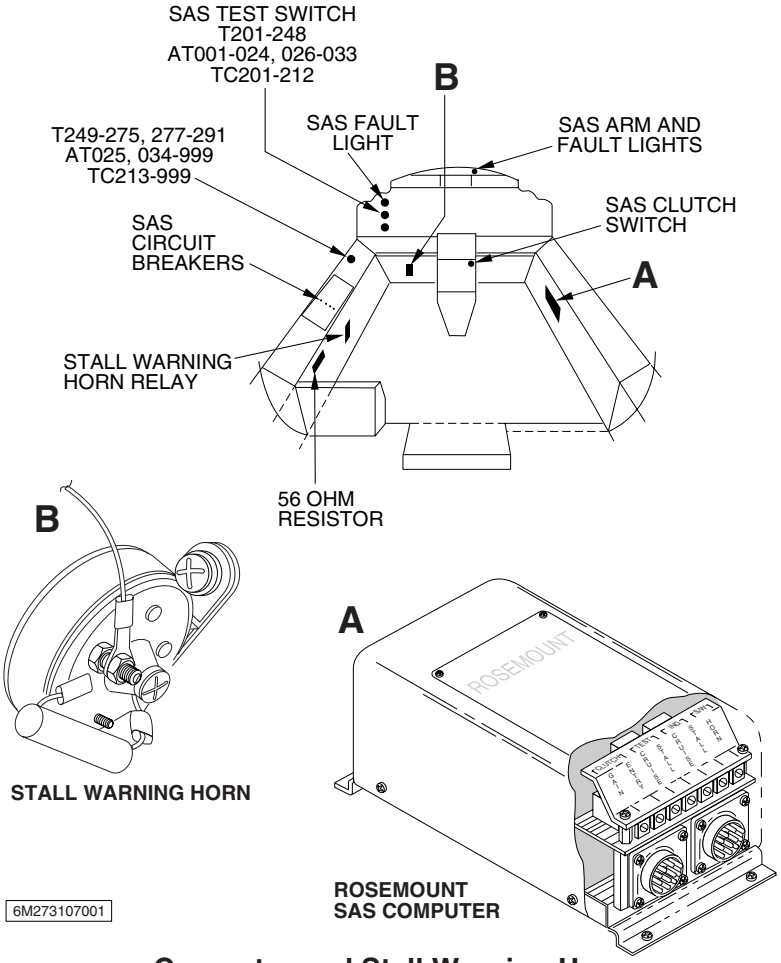
SAS COMPUTER AND STALL WARNING HORN – DESCRIPTION AND OPERATION

1. General

A. The SAS computer and stall warning horn are two cockpit installed components of the SAS system. The SAS computer receives inputs from the AOA transmitter, flap compensator, SAS TEST switch, and a servo fail input from the servo motor via the SAS CLUTCH switch. These inputs are processed to usable outputs for the servo clutch stall warning horn, SAS indicator, and SAS FAULT light. Trimmer pots within the computer adjust these outputs to specifications required by the flight characteristics of the aircraft.

B. The computer on aircraft T201-248; AT001-024, 026-033; TC201-212 is located behind the forward access door of the right console on the forward electrical equipment panel. On aircraft T249-275, 277-291; AT025, 034-999; TC213-999 it is mounted to the inner surface of the right console between the access doors. During calibration of the system the cover of the computer is removed to expose the trimmer pots. During this time a temporary cover with a cutout for the trimmer pots and the nomenclature should be installed to protect the components on the printed circuit boards from any damage that might occur. For convenience during calibration the computer can be attached to the outside of the console by one of the forward access door attaching screws.

NOTE: A placard under the cover of the 20AA1 computer reads "DO NOT USE ON METRO". This means that the three pots the placard is covering are not used on the Metro and the Merlin IV. It does not mean that the computer is not used on the metro.



Computer and Stall Warning Horn
Figure 1

C. The trimmer pots, their nomenclature, and their purpose in the order of their adjustment are as follows:

IND STALL – Adjusts the indicator to read correctly for stall regimes between 1.3 V/Vs and 1.0 V/Vs.

IND CRUISE – Adjusts the indicator to read correctly for cruise regimes between 1.4 V/Vs and 1.2 V/Vs.

S/W HORN – Adjusts stall warning horn trip point at 1.12 V/Vs.

(PUSHER) ENTRY CONT – Adjusts stick pusher trip point at 1.02 V/Vs.

PUSHER GAIN – Adjusts amount of stick pusher force, Does not normally require adjustment. On 20AA1 computer trimmer pot is mounted in middle of top PC board (R49).

CLUTCH ENTRY – T only. Adjusts the clutch voltage to zero (start of stability augmentation ramp) at 1.85 V/Vs.

CLUTCH VOLT CONT – T only. Adjusts the clutch voltage to 8.25 (top of stability augmentation ramp) at 1.4 V/Vs.

CLUTCH GAIN – T only. Adjusts the maximum amount of stability augmentation stick force (top of ramp) 1.4 V/Vs.

TEST CRUISE – Adjusts pointer of indicator to 1.3 V/Vs when SAS TEST switch is held to CRUISE.

TEST STALL – Adjusts pointer of indicator to 1.0 V/Vs when SAS TEST switch is held to STALL.

SAS COMPUTER AND STALL WARNING HORN – MAINTENANCE PRACTICES

1. Servicing – SAS Computer and Stall Warning Horn

A. No regularly scheduled maintenance is required on the SAS computer or the stall warning horn. During regularly scheduled inpections, check computer, horn, and wiring for condition and security of attachment. Replacement of the computer requires ground and flight calibration of the system. Replacement of the horn requires only a functional check of the horn. Certain computers can only be used with certain type numbers of AOA transmitters, Computer P/N 9-20C1007 for Merlin III or P/N 9-20C1006 for Merlin IV and Metro require 12J or 12M AOA transmitter. Computer P/N 9-20C1009 for merlin III or P/N 9-20C1008 for Merlin IV and Metro require 12J1 or 12M1 AOA transmitters. The new computer P/N 20AA1 for Merlin IV and Metro and 20AA for Merlin III can be used with any type AOA transmitter.

2. Removal / Installation – SAS Computer and Stall Warning Horn

A. Removal – Computer

(1) Ensure battery switches are off.

(2) Remove right forward console door.

(3) Remove four screws attaching computer to console.

(4) Remove electrical connectors.

(5) Remove computer from console.

B. Installation – Computer

(1) For reinstallation of same computer (if AOA transmitter was not changed at this time) proceed to step (2). For installation of new computer or if vane was changed remove cover from computer and install temporary cover that exposes trimer pots. Connect computer to aircraft wiring to SAS test box if required. Perform functional check, ground calibration flight calibration, and extract SAS DATA for Aircraft Log (see Section 27-31-06).

(2) Before installing computer in console move flaps up and check STALL TEST and CRUISE TEST for proper adjustment. They must be 1.0 V/Vs and 1.3 V/Vs precisely. If system was just flight calibrated these two adjustments may have to be made (see Section 27-31-06). If same computer is to be reinstalled, no calibration was done to the system and these two adjustments are off check flap compensator or 20° switch adjustment (see Section 27-31-09).

(3) Install cover on computer.

(4) Position computer in console and install four screws.

(5) Install electrical connector.

(6) Recheck CRUISE TEST and STALL TEST.

(7) Close console door.

C. Removal – Stall Warning Horn

(1) If horn is the type with a positive terminal stud protruding from back of case remove nut on terminal stud and remove wire number 7945-22. If horn is type with three place terminal strips mounted to ground attach point at center of horn; remove screw at bottom of horn and loosen screw in clamp at top of horn. Rotate horn to bring strip around for easy access. De-solder wire number 7945-22 from terminal strip.

(2) Remove grounding screw at center of horn. If orange capacitor is installed retain for reinstallation.

(3) Loosen screw in clamp at top of horn.

(4) Remove horn from clamp.

D. Installation – Stall Warning Horn

(1) Position horn in clamp and install screw at center with capacitor or terminal strip.

(2) If horn replacement is type with terminal strip cut terminal off wire number 7945-22.

(3) Solder wire number 7945-22 to same terminal that small black wire is soldered to, or install on positive terminal stud.

(4) Install screw at bottom of horn, and tighten screw in clamp at top of horn.

(5) Perform functional check.

3. Adjustment / Test – SAS Computer and Stall Warning Horn

A. Adjustment of computer is ground and flight set-up. Functional check of computer is system functional check (see Section 27-31-06).

B. Functional Test – Horn

(1) Turn battery switches on.

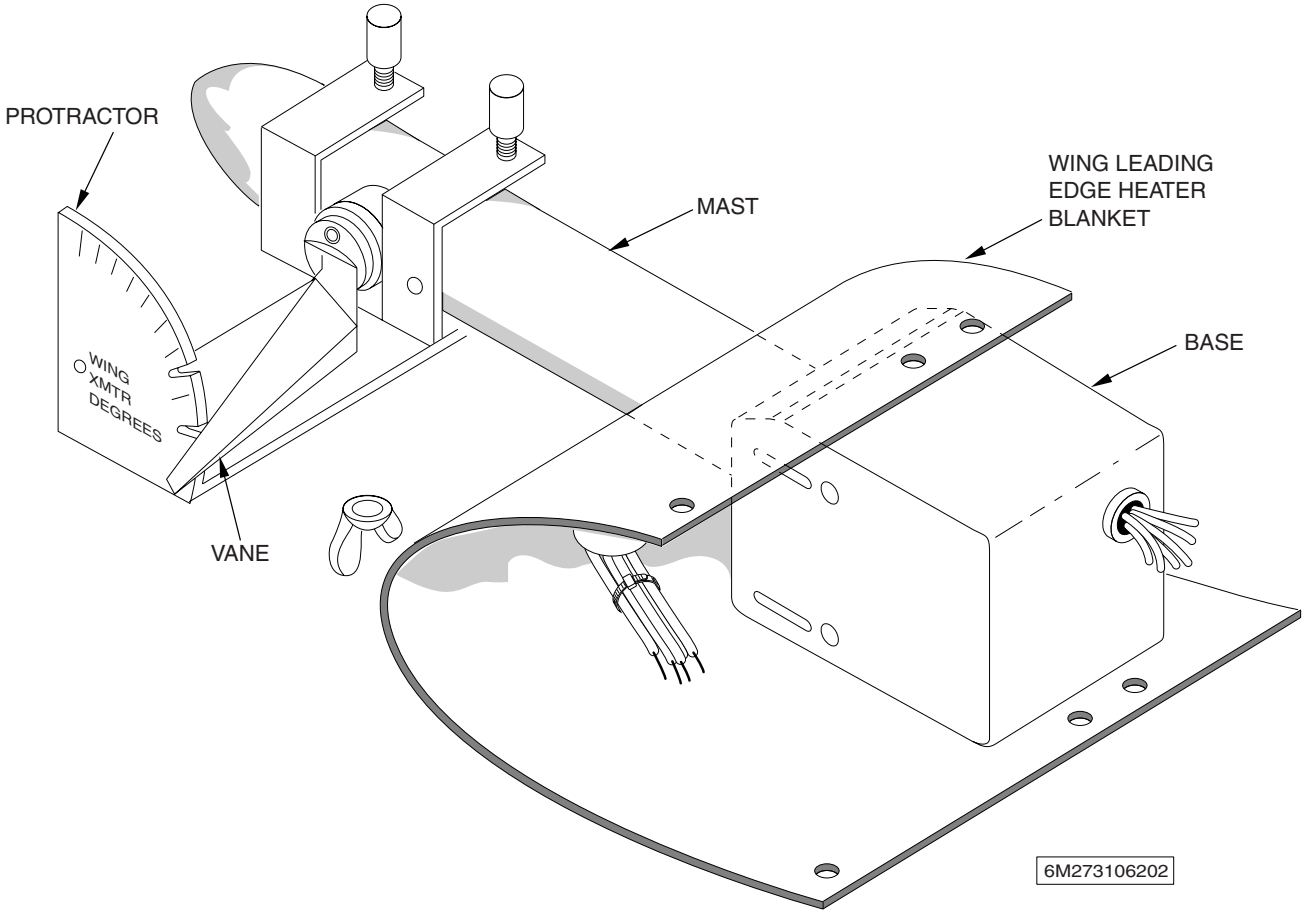
(2) Push SAS TEST switch to STALL. Horn should sound in an even and steady tone and not warble.

(3) Release SAS TEST switch and turn battery switches off.

ANGLE OF ATTACK TRANSMITTER – ROSEMOUNT – DESCRIPTION AND OPERATION

1. General

A. The angle of attack transmitter for the Rosemount SAS system is located on the right wing leading edge at the wing tip. The transmitter consists of a mast, base, vane, and a 2,000 ohm potentiometer (pot). The tip and body of the mast, the vane, plus the wing leading edge just behind the mast are electrically deiced (see Maintenance Manual, Chapter 30 for more information on SAS deice). The vane pivots on a shaft that extends into the body of the mast. The vane aligns itself with the airflow over the wing. The pivot shaft is linked mechanically to the pot. The electrical signal from this pot is used by the SAS computer for angle of attack information. The vane has a range of 90° (approximately 80 above zero degrees and approximately 10 below). Each aircraft is provided with a vane protractor, marked in degrees, to be used for calibration of the system.



Rosemount AOA Transmitter
Figure 1

ANGLE OF ATTACK TRANSMITTER – ROSEMOUNT – MAINTENANCE PRACTICES

1. Servicing – Angle of Attack Transmitter – Rosemount

A. Check AOA transmitter vane for condition and freedom of movement; check tip and barrel of AOA transmitter for security and condition; check heater blanket for security and condition (rain erosion); coat with EC801 or equivalent, if necessary. Check heater control box, terminal strip, and wiring beneath wing tip for condition and security.

B. Replacement of AOA transmitter requires ground calibration and flight calibration of the SAS system. SAS computer P/N 9-20C1007 for Merlin III or P/N 9-20C1006 for Merlin IV and Metro require 12J or 12M AOA transmitter. SAS computer P/N 9-20C1009 for Merlin III or P/N 9-20C1008 for Merlin IV and Metro require 12J1 or 12M1 AOA transmitter. The new computer P/N 20AA1 for Merlin IV and Metro and 20AA for Merlin III can be used with any type AOA transmitter, Record SAS DATA after in-flight set-up.

1. Removal/Installation – AOA Transmitter – Rosemount

A. Removal – AOA Transmitter (12J, 12M, 12J1, or 12M1) and Leading Edge Heater

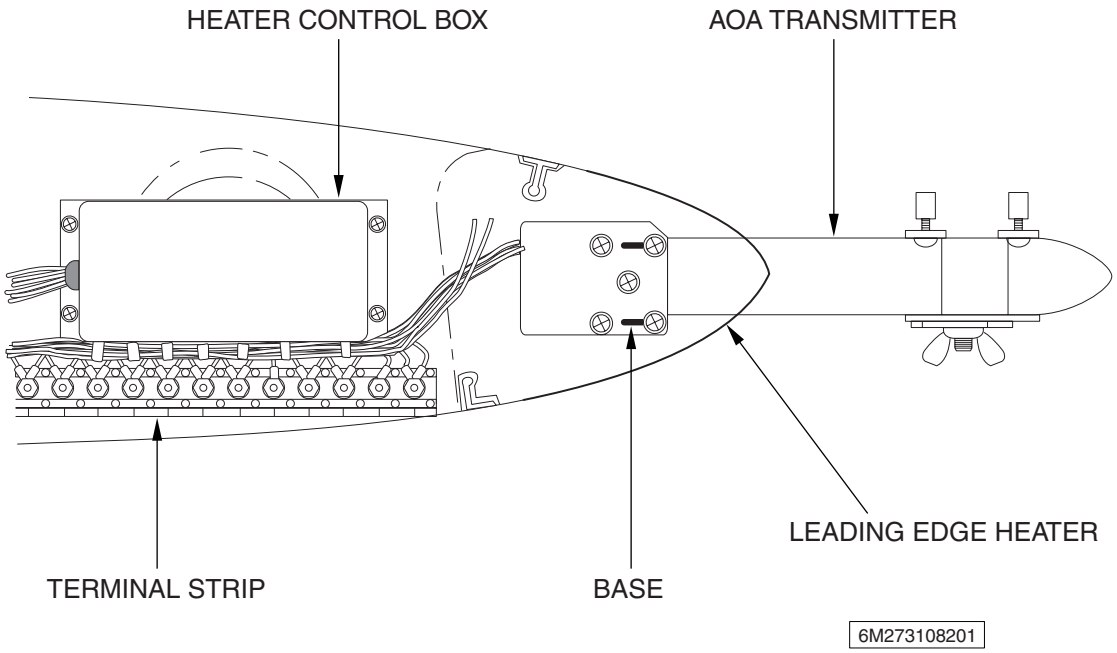
(1) Remove screws attaching right wing tip and remove wing tip. Wing tip can be placed on top of wing and secured with two screws.

(2) Remove screws attaching heater control unit. Heater control unit can be stowed by installing one screw through lower aft hole in unit and upper aft nutplate.

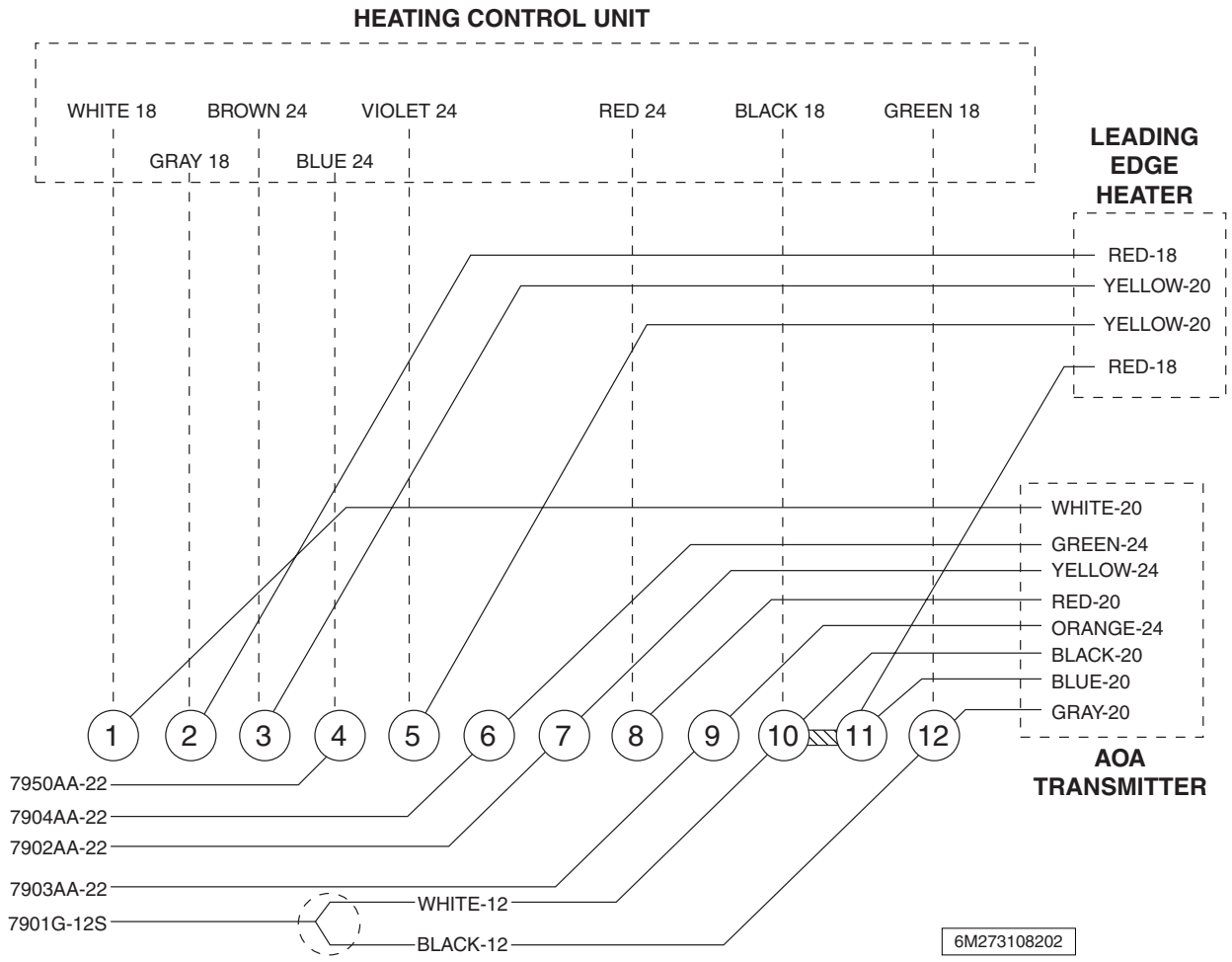
(3) Remove ties securing wires near terminal strip.

(4) Remove nuts and washers securing terminals on terminal strip.

(5) Remove only those wires from terminal strip attached to AOA transmitter and leading edge heater.



AOA Transmitter
Wing Tip Installation
Figure 201



SAS Wing Tip Wiring
Figure 202

(6) Remove screws attaching AOA transmitter and leading edge heater.

(7) Remove AOA transmitter and leading edge heater.

(8) Clean old sealer from wing tip and leading edge.

B. Installation – AOA Transmitter

(1) Check ohm reading (see Adjustment and Test 12J, 12M, 12J1, or 12M1 AOA transmitters.

(2) Position transmitter on mounting pad and install four screws.

(3) Check terminals on wires for good connection. Replace or solder if necessary. Do not cut wires shorter, leave original length. Open ended or forked terminals should be replaced.

(4) Install wires on terminal strip. Refer to Wiring Diagram Manual Section 27-30-00 and 30-10-02 for wiring. The only difference in wiring of the 12m series transmitter is one white wire from the transmitter to terminal one. The 12J series has two white wires to terminal one.

(5) The new leading edge heater (Rosemount P/N 00012-2101-0001) is split on its outboard edge and can be removed and replaced without removing the transmitter. The red wires on the new leading edge heater are for the heating element and connect to terminals two and eleven. The yellow wires are for the thermister and connect to terminals five and three.

(6) Slip heater over barrel of transmitter. The old type leading edge heater is part of the AOA transmitter and should be replaced with AOA transmitter.

(7) Feed wires through hole in leading edge and connect to terminal strip (see Figure 202).

(8) Check AOA transmitter operation:

(a) Turn battery switches on.

(b) Move vane up; check pointer of SAS indicator moves counterclockwise toward STALL. Move vane down; check indicator needle moves clockwise.

(c) Turn pitot heat switch to PITOT and SAS. After a delay of 10 to 15 seconds check SAS DEICE light illuminates. Turn switch OFF.

(d) Check that tip, vane, and barrel of transmitter warm. Check that leading edge heater is warm.

(9) Install wing tip.

(10) Check NAV and STROBE light operation.

(a) Turn NAV light switch to NAV & STROBE. Check NAV and STROBE lights illuminated. Turn switch to OFF.

(b) Turn battery switches OFF.

(11) Push leading edge heater against edge of wing. Align heater to deice boot and NAV light cover. Locate and drill holes for attaching screws. A hole finder may be used, or draw three lines on wing converging across the center of each hole. Align heater, and continue lines onto heater. Trim top and bottom of heater to match wing if necessary. Do not trim sides of heater. Countersink holes and install screws and washers.

(12) Using EC 801 or equivalent seal edges and split in heater. Seal around barrel of transmitter. Do not seal over vent hole in barrel of transmitter.

(13) Ground set SAS (see Section 27-31-06).

3. Adjustment / Test – AOA Transmitter – Rosemount

A. Adjustment and Test of 12J, 12M, 12J1, and 12M1 AOA transmitters

(1) Remove right wing tip.

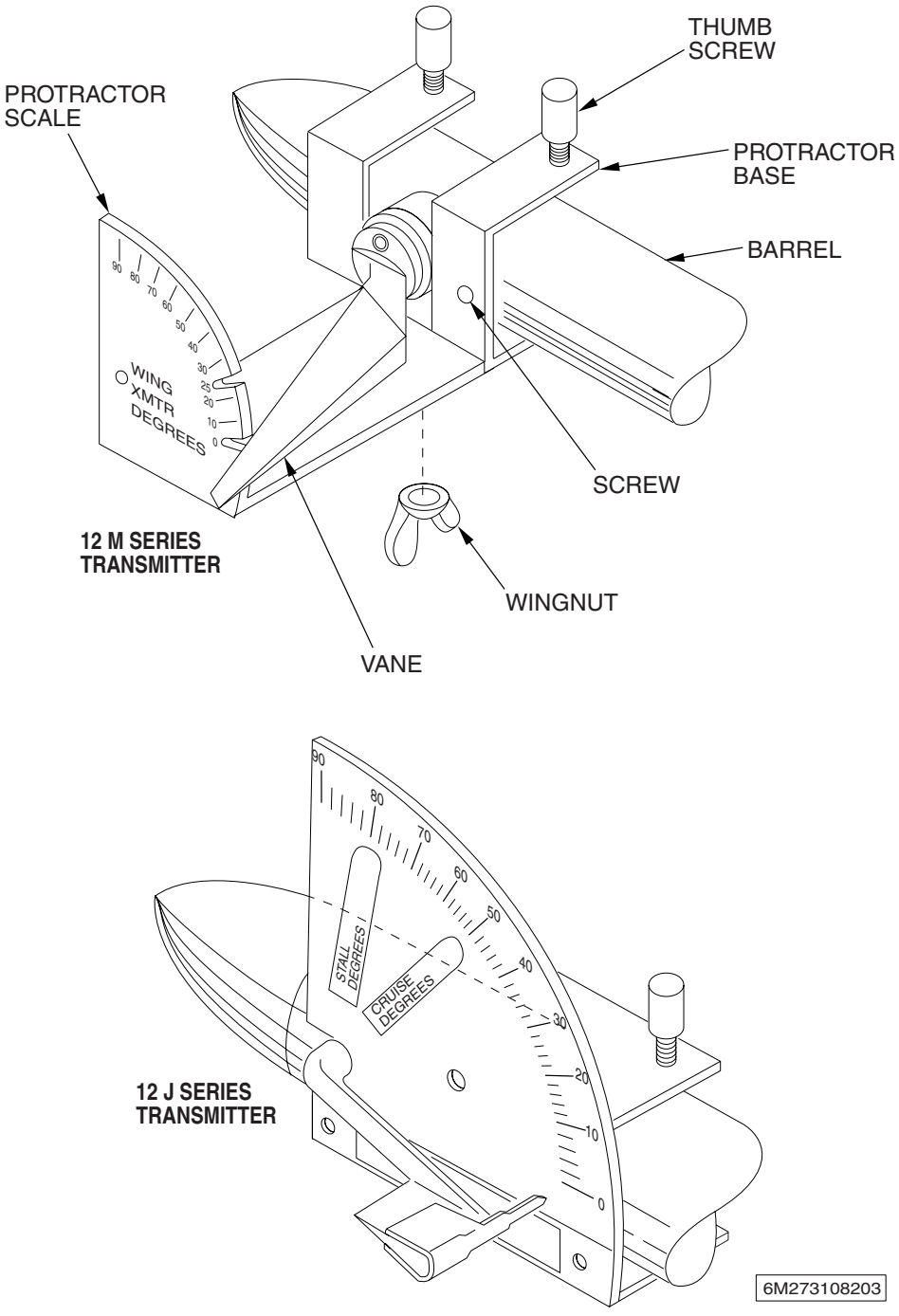
(2) Remove orange and green wires from terminals nine and six.

(3) Remove four screws attaching transmitter to mounting pad.

(4) Obtain an accurate ohmmeter. Connect ohmmeter leads to orange and green wires from AOA transmitter.

(5) Install protractor on AOA transmitter.

(6) For 12J and 12M ohm reading is taken at 47°. For 12J1 and 12M1 ohm reading is taken at 41°. Ohm reading should be 880 plus or minus 10 ohms. Reading must not be below 860 ohms or above 900 ohms. If reading is 880 plus or minus 10 ohms do not adjust AOA transmitter to agree with aircraft log ohm reading for vane.



12M / 12J Series
Transmitters / Protractors
Figure 203

(7) To adjust 12M or 12M1 transmitter:

(a) Remove screw attaching cover to base of transmitter.

(b) Slide cover back just far enough to clear potentiometer.

(c) Loosen three screws in wedge clamps around base of potentiometer.

(d) If 12M transmitter is required place vane at 47°. If 12M1 transmitter is required place vane at 41°. Turn potentiometer until ohmmeter reads 880 plus or minus 10 ohms.

(e) Tighten screws in wedge clamps.

(f) Hold wires and slide cover over base of transmitter. Take care not to bunch wires up inside of case. Ensure grommet is installed in cover.

(g) Install screw attaching cover.

(h) All 12M series transmitters are manufactured as 12M1 transmitters. If transmitter was adjusted to 12M configuration (880 ohms at 47°) place a dot of white paint over the 1 in 12M1 on the data plate of the transmitter.

(8) To adjust 12J or 12J1 transmitters:

NOTE: Do not adjust 12J transmitter to 12J1 configuration. The rod in the 12J transmitter is shorter than in the 12J1 and may cone loose from the rod end.

(a) Remove screw attaching cover to base of transmitter.

(b) Slide cover back just far enough to clear potentiometer.

(c) Loosen locknut at rod end.

(d) Remove C clip on pin through rod end and arm on potentiometer. Care should be taken not to lose C clip. No other type of attachment is authorized.

(e) Adjust rod end in or out as desired. 12J transmitter is adjusted to 880 ohms at 47° and 12J1 880 ohms at 41°.

(f) Install pin through rod end and arm on potentiometer.

(g) Install C clip and tighten locknut to rod end.

(h) Recheck ohm reading.

(i) Install cover and attaching screw. Care should be taken not to bunch wires up inside of case. Ensure grommet is installed.

(j) If 12J1 transmitter was changed to 12J configuration place a dot of white paint over the 1 in 12J1 on data plate.

(9) Install AOA transmitter and wing tip.

(10) Ground set SAS (see Section 27-31-06).

B. Procedure for installing 12M / 12M1 vane protractor for calibration of SAS.

(1) Remove and retain screw in barrel of AOA transmitter directly behind vane. (Screw has the words REMOVE FOR CALIBRATION near it).

(2) Loosen wing nut on protractor.

(3) Position protractor base with thumb screw heads pointing toward the ground. Slide protractor base over tip of transmitter until vane is positioned at slot in base between thumb screws. Rotate protractor until hole in base aligns with hole in barrel where screw was previously removed.

(4) Install screw through base into barrel and tighten.

(5) Lift vane to approximately 40° and align scale to slot in base. Push scale firmly but not tightly against vane and tighten wingnut.

(6) Vane should move in a smooth arc along scale and not move when released.

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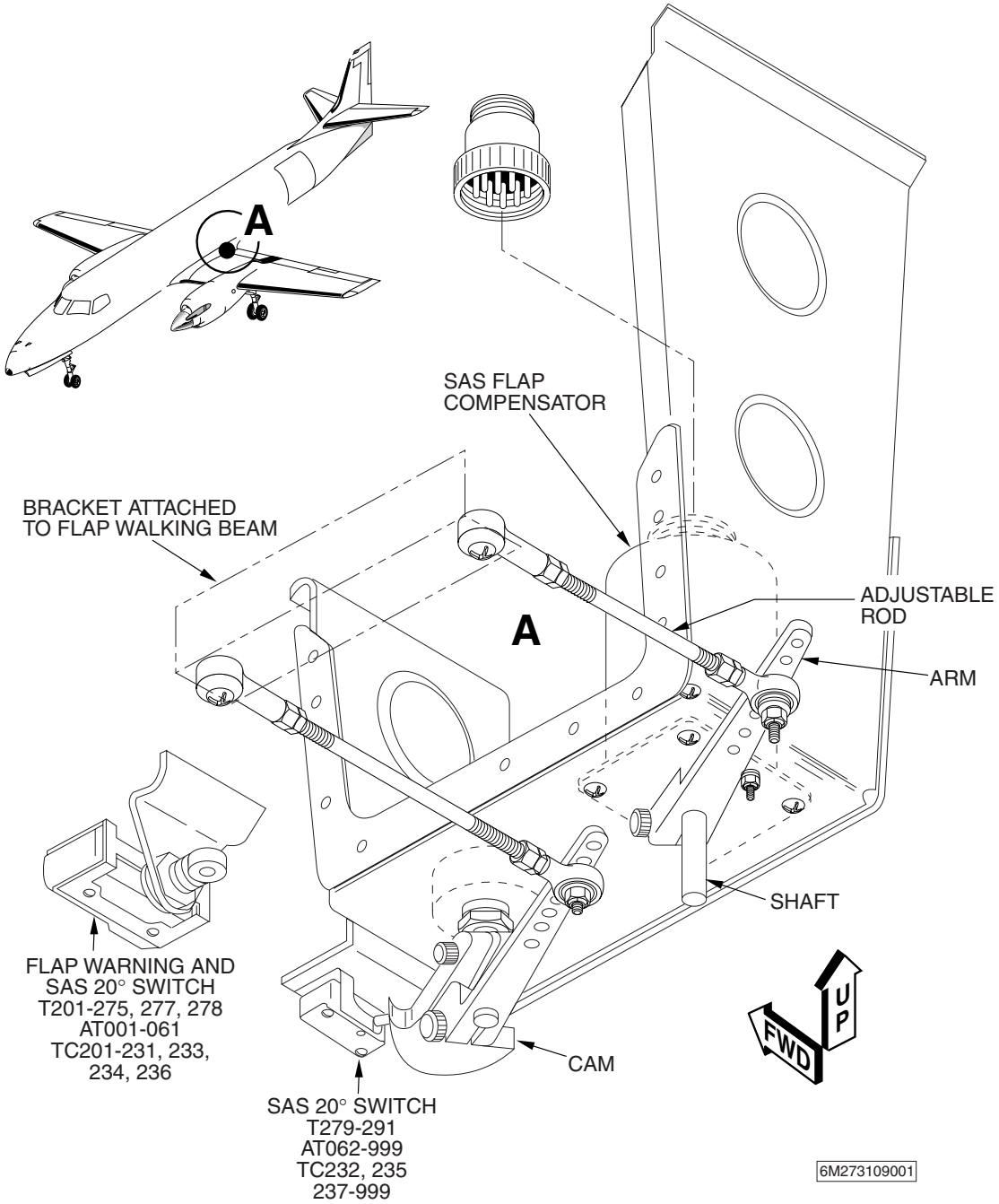
FLAP COMPENSATION – ROSEMOUNT – DESCRIPTION AND OPERATION

1. General

A. Flap position compensation for the rosemount SAS system is accomplished by a 2000 ohm potentiometer (pot) and a micro-switch linked mechanically to the flaps. The pot and the transmitter for the flap position indication system are mounted on a bracket just to the right of the left flap.



Flap Compensation – Rosemount
Figure 1

B. An arm on the shaft of the pot is connected by an adjustable rod to the flap walking beam. When the flaps are extended the shaft turns. The resistance increases the SAS computer uses this to compensate angle of attack information from the AOA transmitter for flap position.

C. The micro-switch is attached to the right side of the bracket. A cam on the shaft of the flap position transmitter is orientated such that it actuates the switch when the flaps are extended beyond 20°. When the switch is actuated it places a 56 ohm resistor in the AOA transmitter circuit. The 56 ohm resistor is located on a terminal strip in the rear of the cockpit console.

D. As the flaps extend the pointer of the SAS indicator will travel counterclockwise. As the flaps pass 20° (just past 1/2 on the flap indicator) the pointer of the SAS indicator will jump approximately 1/3 of an inch counterclockwise. The pointer will continue to travel counterclockwise until the flaps are fully extended.

FLAP COMPENSATION – ROSEMOUNT – MAINTENANCE PRACTICES

1. Servicing – Flap Compensation – Rosemount

A. Check the 20° switch and mechanical linkage for condition and security. Replacement of flap compensator pot requires ground set-up, in-flight set-up, and record data for Aircraft Log.

2. Removal/Installation – Flap Compensation – Rosemount

A. Removal – Flap Compensator Pot

(1) Remove left walking beam panel. Flap compensator is the left outboard transmitter.

(2) Loosen allenhead screw in arm on shaft of pot.

(3) Slide arm down and off shaft.

(4) Disconnect electrical connector.

(5) Remove screws, nuts, and washers attaching case to bracket.

(6) Remove two smaller nuts and washers from studs attaching pot to bracket.

(7) Remove pot and case.

B. Installation – Flap Compensator Pot

(1) Position pot and case on bracket with studs on pot through holes in bracket.

(2) Install two washers and nuts on studs. Cut off and smooth ends of studs if they protrude more than 0.125 inch (3.175 mm) through nuts.

(3) Install screws, nuts, and washers attaching case to bracket. Screws may be installed with head down to clear mechanical linkage.

(4) Install arm on shaft and adjust pot (see Adjustment Flap Compensator pot).

(5) Connect connector and close walking beam access panel.

3. Adjustment/Test – Flap Compensation – Rosemount

A. Adjustment – Flap Compensator Pot

(1) Adjust rod attached to arm on shaft of flap compensator pot so a piece of 0.032 safetywire will not pass through check holes in ends.

(2) Install rod end on arm in forth adjustment hole from center. If arm or rod were not replaced, leave rod end in same hole on arm as previously installed.

(3) Connect ohmmeter to pins A and B at flap compensator pot connector or D and C of SAS computer connector P61 (big key over H).

(4) With flaps up, turn shaft of pot until ohmmeter reads 360 ohms. If AOA transmitter or SAS computer were not changed at this time set to ohm reading for flap compensator in Aircraft Log SAS DATA.

(5) Install arm on shaft and tighten allenhead screw. Recheck ohmmeter reading.

(6) Move flaps down observing linkage for binding And obstructions.

(7) Check ohmmeter reads 1650 ohms or reading for flap compensator in Aircraft Log SAS DATA.

(8) If reading is less than desired move rod end in one hole on arm. If reading is more than desired move rod end out one hole on arm.

(9) Move flaps up and reset 360 ohms of flap compensator reading in Aircraft Log.

(10) Move flaps down and check ohm reading.

(11) Repeat steps **3.**, A., (8), (9), and (10) until reading of 360 ± 10 ohms flaps up, and 1650 ± 10 flaps down; or readings for flap compensator in Aircraft Log SAS DATA is obtained.

B. Adjustment – 20° Switch

(1) The 20° switch is also part of the landing gear warning system on aircraft T201-275, 277 and 278, AT001-062, TC201-231, 233, 234, and 236. If any adjustment is made to the switch the landing gear warning system must be checked. For adjustment and checkout procedures see Section 32-61-03.

(2) For aircraft T279-291, AT062-999, and TC232, 235, 237-999 the 20° switch adjustment procedures are:

(a) Remove left walking beam panel.

(b) Move flaps to 1/2 (18°).

(c) Loosen two screws attaching switch and allen head screw attaching cam to shaft of pot.

(d) Adjust switch so actuator roller barely makes contact with cam. Tighten screws in switch.

(e) Turn cam until it starts to put pressure on actuator roller. Tighten allen head screw in cam.

(f) Lower flaps to 20° (just past 1/2 on flap indicator). Switch should actuate. If not, adjust switch in until it does.

C. Functional Test – Flap Compensation

(1) With power on the aircraft and SAS system hold SAS TEST switch to CRUISE.

(2) Pointer of SAS indicator should be on 1.3 V/Vs with flaps up.

(3) Move flaps down.

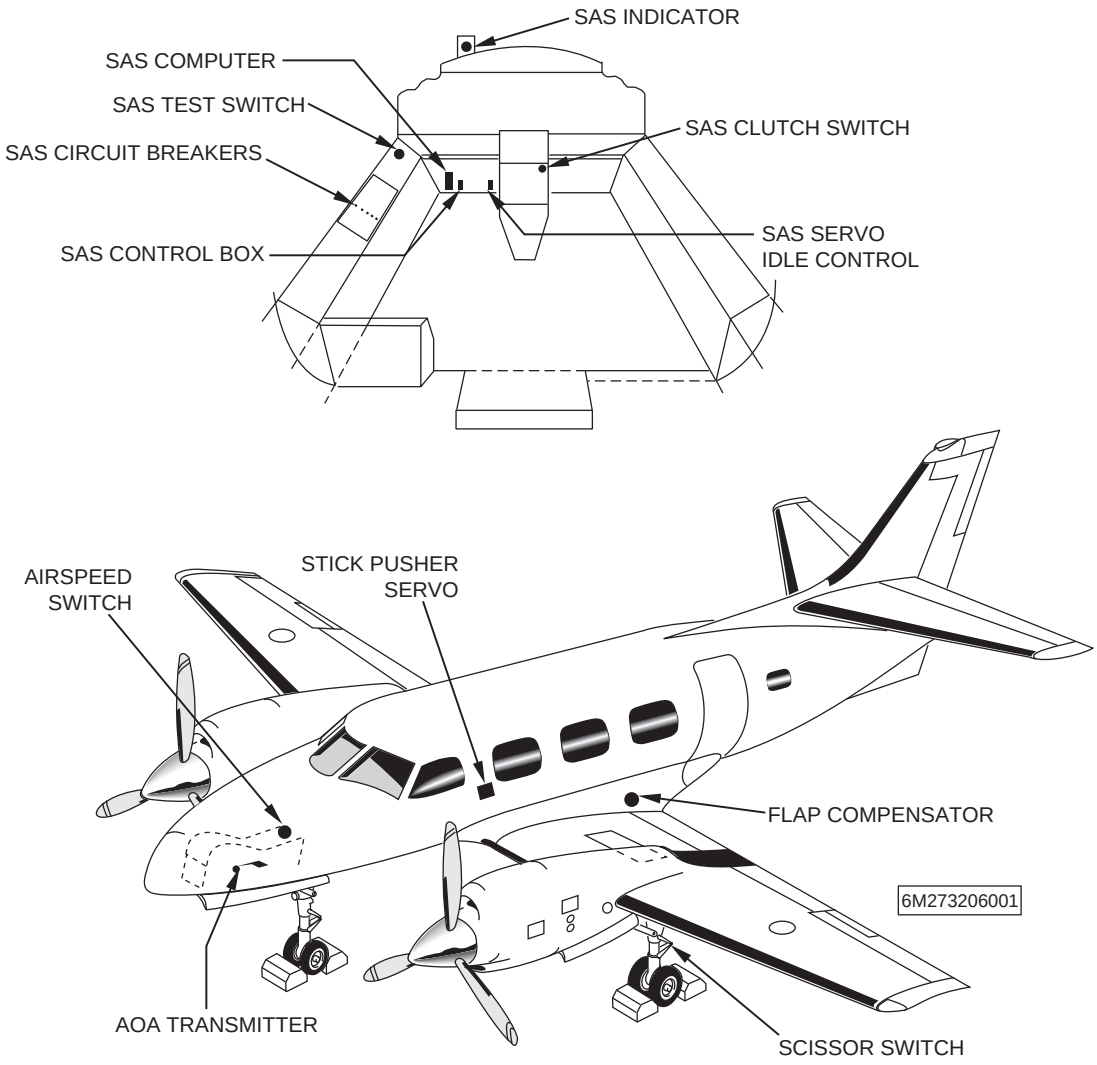
(4) As flaps travel down the pointer of the SAS indicator should travel counterclockwise into the SLOW area. As flaps pass 20° (just past 1/2 on the flap indicator) the pointer of the SAS indicator should jump approximately a pointers width. When flaps are full down the pointer of SAS indicator should be approximately in the middle of the yellow SLOW area.

CONRAC SAS – DESCRIPTION AND OPERATION

1. General

A. A Conrac SAS system is standard equipment on T(B) aircraft S/N 276, 292-999. It may also be installed as a replacement to the Rosemount SAS system on SA226 Series T/AT/TC aircraft modified in accordance with STC SA4725SW.

B. The SAS system consists of an angle of attack transmitter, computer, airspeed switch, stick pusher servo, cockpit indicator, and control box. These components function together to warn the pilot of an impending stall and, if necessary, apply a forward push to the elevator control if the stall is imminent.



Conrac SAS Equipment Locator
Figure 1

C. SAS Computer

(1) For T(B) aircraft, the computer (see Section 27-32-07) is located forward of the pilot's left rudder pedal on the forward pressure bulkhead. For aircraft modified by STC SA4725SW, the computer is located in the RH console, mounted on a removable panel. The computer receives information from other various components of the SAS system. The information is processed by the computer and used to control the SAS indicator, stall warning horn, and a stick pusher servo. Trimmer pots installed in the computer are used for calibration of the system. Calibration of the system is necessary to compensate for installation and aerodynamic differences between individual aircraft. The computer also contains fault circuitry and monitors itself and the stick pusher servo.

D. Control Box (SAS Interface Assembly).

(1) The control box (see Section 27-32-07) located just to the right of the computer, contains a printed circuit board (PCB), stall warning horn, and the servo, power compensation, vane heater, and warning horn relays. The PCB contains circuitry to make the SAS FAULT light blink or illuminate steadily when it receives a servo fail or a computer fail signal from the computer. Other components on the PCB use a signal from the airspeed switch to control the servo relay.

(2) In aircraft modified according to STC SA4725SW, another part of the PCB keeps the servo motor turning at slow or idle speed above SAS arm speed. This keeps the particles in the servo clutch from settling during prolonged operation at high cruise speeds.

E. Angle of Attack (AOA) Transmitter

(1) The AOA transmitter (see Section 27-32-08) is located on the nose of the aircraft just aft of the radome. The transmitter consists of a vane, a base, and a potentiometer (pot). The vane is electrically deiced (for more information on SAS Deice, see Maintenance Manual, Section 30-30-10). The vane pivots on a shaft that extends into, the base. The vane aligns itself with airflow over the fuselage; the vane angle from the waterline of the aircraft correlates to angle of attack. The vane is linked by the pivot shaft to the pot in the vane base. The pot sends angle of attack information to the computer.

F. Flap Compensator

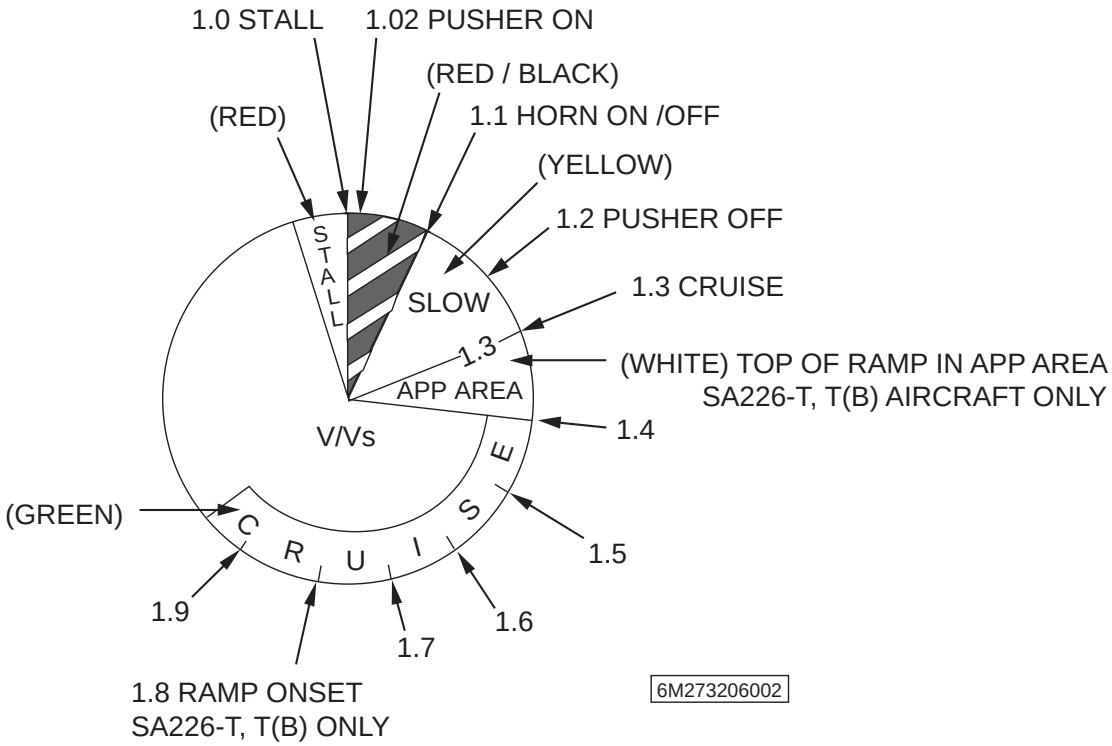
(1) The flap compensator (see Section 27-32-09) consists of a pot mechanically linked to the flaps. When the flaps are extended or retracted the pot rotates. This provides a means for the SAS computer to sense flap position.

G. SAS Indicator

(1) The SAS indicator (see Section 27-30-10) is located on the glareshield. The face of the indicator is divided into five sectors (see Figure 3). Points at the dividing lines of these sectors are V/Vs and are used for calibration of the system.

H. Stick Pusher Servo

(1) The stick pusher servo (see Section 27-30-11) is located under the center isle floorboards directly across from the wing leading edge. The servo contains a continuous DC powered motor coupled by a magnetic clutch to a capstan assembly. Cables are connected from the capstan to the elevator control. When the motor is running and the clutch is engaged, the elevator control is pushed forward. The capstan contains a mechanical slip clutch, should it become necessary to override the stick pusher. The magnetic clutch is engaged by current controlled by the computer. The SAS CLUTCH switch on the cockpit pedestal is used to turn off the motor and disconnect the clutch.



Conrac SAS Indicator
Figure 2

I. Airspeed Switch

(1) An airspeed switch (see Section 27-30-12) located in the right aft section of the nose baggage compartment also controls the servo. The switch is actuated by air pressure from the copilot system. The switch is connected to the servo relay located within the control box (SA226-T(B), or aft of the servo motor (STC SA4725SW). The airspeed switch actuates at the airspeed at which the SAS system is no longer needed. When the switch actuates, it relaxes the servo control relay which in turn disconnects the servo, clutch, extinguishes the SAS ARM light on the annunciator panel, and switches the servo motor from normal speed to idle speed.

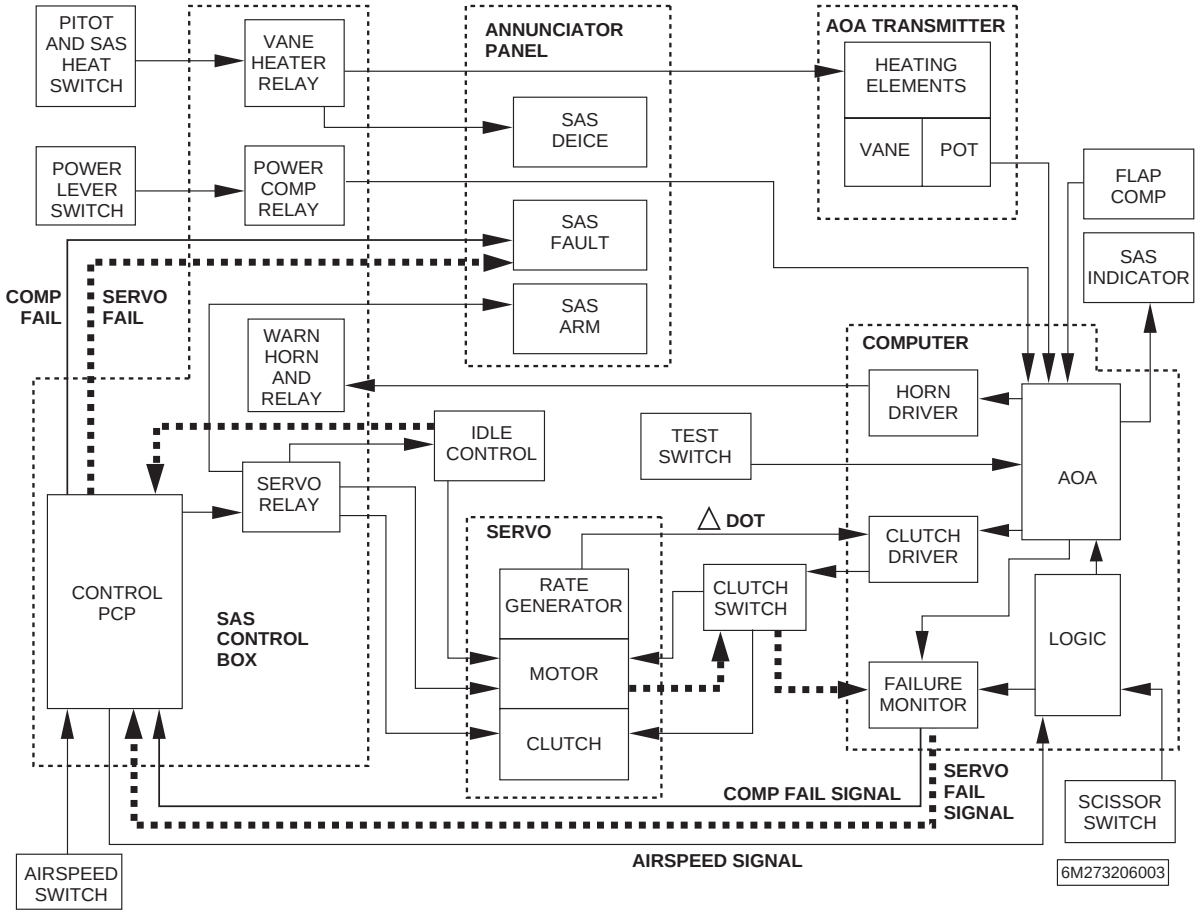
J. Servo Idle Control

(1) In SA226 T(B) aircraft, the servo idle control (see Section 27-32-13) is located behind the left side of the pedestal on the forward pressure bulkhead. The idle control keeps the servo motor turning at slow or idle speed above SAS ARM speed. This keeps the particles in the servo clutch from settling at prolonged operation at high cruise speeds.

K. Scissor Switch

(1) A scissor switch on the left main gear disconnects the servo clutch and inhibits the stall warning horn when the aircraft is on the ground.

CONRAC SAS – DESCRIPTION AND OPERATION



Conrac SAS Diagram
Figure 3

L. System Operation

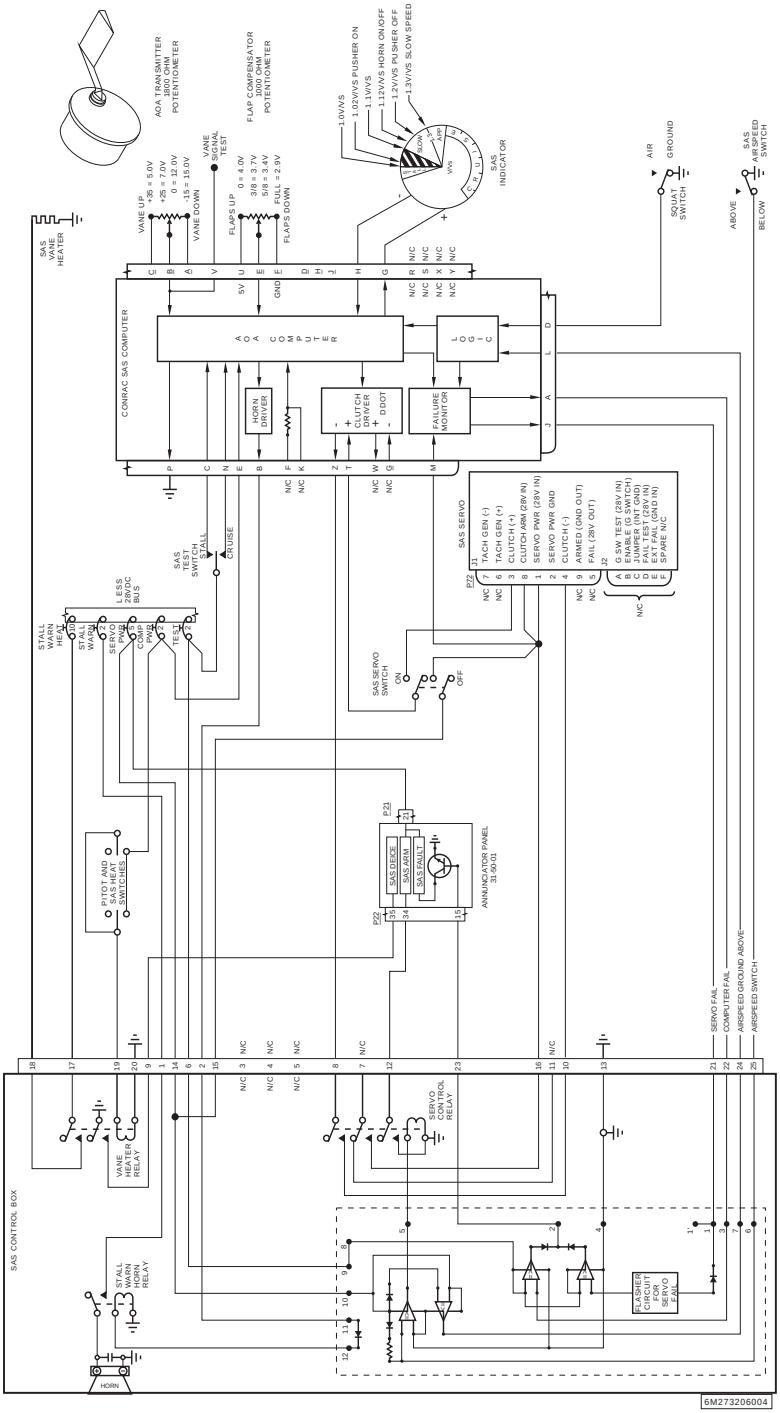
(1) The system operates on DC power from four circuit breakers on the left essential bus. The circuit breakers power the stall warning horn and relay, the SAS computer, control box, and idle control, the test function of the computer and the servo motor and annunciator panel lights.

(2) With the vane at approximately zero degrees as the flaps are lowered, the pointer of the SAS indicator will jump three times counterclockwise. Angle of attack is compensated in steps for flap positions of UP to 1/8, 1/8 to 1/2, just past 1/2 to 7/8, and from 7/8 to FULL DOWN. When the flaps are full down and the vane is at its bottom stop the pointer of the SAS indicator will be to the left of the CRUISE sector. If the vane is moved slowly up from its bottom stop the pointer of the SAS indicator will start traveling counterclockwise.

(a) On the SA226-T(B) and SA226-T models, a slight and gradually increasing forward force will be exerted on the elevator control. At 1.3 V/Vs on the indicator the force will have increased to 20 pounds (9.07 kg). At this point the force will remain the same until pusher actuation. This is known as stability augmentation.

CONRAC SAS – DESCRIPTION AND OPERATION

(b) On all installations, as the vane continues traveling up, the pointer of the indicator will travel through the SLOW sector. At just before 1.1 V/Vs (1.12) the horn will sound. As the pointer travels through the crosshatch sector, the horn will remain sounding.



Conrac SAS Schematic
Figure 4

(c) Just before 1.0 V/Vs (1.02) the pusher will engage and remain engaged. As the pointer travels into the STALL sector the SAS FAULT light on the annunciator panel will illuminate steadily.

(d) When the vane is at its upper stop the pointer of the indicator will be to the left of the STALL sector.

(e) If the vane is moved down the pusher will not disengage until the pointer of the SAS indicator travels below the middle of the SLOW sector (1.2 V/Vs).

(3) The computer contains circuitry to test itself. When the SAS TEST switch is held to STALL the computer substitutes angle of attack information and tests itself as described above. When the switch is held to the CRUISE position the pointer of the indicator will go to 1.3 V/s.

(4) On the SA226-T(B) and SA226-T models, a switch actuated by the left power lever controls the power compensation relay located within the control box. When the left power lever is pulled back to flight idle the relay opens. This is sensed by the computer which in turn moves the pointer of the SAS indicator clockwise approximately 1/8 of an inch. Because of the proximity of the propeller to the AOA vane this action is necessary to compensate for decreased airflow over the vane when the engine is operating at decreased power.

2. Troubleshooting – Conrac SAS

A. If a defect exists that complies with the items in the DEFECT column comply with the PRELIMINARY and follow the steps in the CORRECTIVE ACTION column. If you suspect an item in the PROBABLE CAUSE column comply with the PRELIMINARY and follow the steps in the CORRECTIVE ACTION column next to the item.

B. When checking for 28 VDC a voltmeter or a test light may be used. When wiring, wire terminals, or relay contacts are a suspected cause a test light should be used since it places a slight load on the circuit. The slight load will cause the circuit being checked to open at the trouble point, and the light will extinguish. A voltmeter will place almost no load on the circuit, and the defective item will be harder to find.

NOTE: SAS computer connector has 32 pins. Twenty-three of the pins are marked with upper case (capital) letters. Nine are marked with lower case (small) letters. On some connectors these nine pins are marked with upper case letters with a line beneath them.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
(SAS ARM light is illuminated). SAS system inoperative, OR SAS indicator erratic, OR SAS FAULT light illuminates steadily, OR Pusher or horn erratic.		PRELIMINARY Refer to Wiring Diagram Manual Series (B) 27-30-06. Remove connector from computer. Because of accessibility to computer, connector, and difficulty of reading voltages it is recommended that TS27-02 SAS test box and extender harness be used.
	1. COMP PWR circuit breaker or wiring defective.	1. Check pin E of computer connector for 28 VDC. If 28 VDC is not present, circuit breaker or wiring is defective. Check pin P for ground. If ground is not present, wiring is defective. If the above checks okay, see step 6. a.
	2. AOA transmitter or wiring defective	2. Move vane up and down slowly. Arc of movement should be approximately 50°, 35° above zero and 15° below. Vane should move smoothly, easily, and not bind. Check flag not damaged or bent. Connect ohmmeter to pins a and c of computer connector. Reading should be approximately 1,800 ohms. Connect ohmmeter to pins b and c of computer connector. Move vane up. reading should decrease smoothly and evenly. Check pins a, b, and c, read infinite resistance to ground. If the above checks okay, see step 6. a. or 6. b. If not, AOA transmitter or wiring defective, see Section 27-32-08 for replacement procedures.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
(SAS ARM light is illuminated). SAS system inoperative, OR SAS indicator erratic, OR SAS FAULT light illuminates steadily, OR Pusher or horn erratic.	3. Flap compensator or wiring defective.	3. Connect ohmmeter to pins U and F wiring defective. of computer connector. Reading should be approximately 1,000 ohms. Connect ohmmeter to pins e and f of computer connector. As flaps extend reading should decrease smoothly and evenly. Check pins U, e and f, read infinite resistance to ground. If the above checks okay see step 6. a. or 6. c. If not, flap compensator or wiring is defective, see Section 27-32-09 for replacement procedures.
	4. SAS indicator or wiring defective.	4. Check that glareshield connection to indicator is secure. Check that terminals on back of indicator are secure. Connect ohmmeter to computer connector pin G and positive terminal on back of indicator. Reading should be continuity (0 ohms). Connect ohmmeter to computer connector pin H and negative terminal on back of indicator. Reading should be continuity (0 ohms). Check both leads to indicator, read infinite resistance to ground. If not, wiring to indicator is defective. To determine whether indicator or computer is defective, see step 6. d.
	5. SAS control box or wiring defective.	5. This step details procedures to determine whether the wiring outside the control box or components within the box are defective. If a component within the control box is found to be defective, the entire box should be replaced. For replacement procedures, see Section 27-32-07. For troubleshooting procedures for AOA transmitter deice system, see Chapter 30-12-02
	a. Stall warn horn relay, STALL WARN circuit breaker, or wiring defective.	a. Jumper pins E and B of computer connector. Turn battery switches on; horn should sound steadily and not warble. If horn sounds, proceed to step 6. e. If horn does not sound, remove connector from control box. Check pins 1 and 2 for 28 VDC and pin 13 for ground. If not, wiring or STALL WARN circuit breaker is defective. If wiring checks okay,horn or relay is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
(SAS ARM light is illuminated). SAS system inoperative, OR SAS indicator erratic, OR SAS FAULT light illuminates steadily, OR Pusher or horn erratic.	b. Power compensation relay; TEST circuit breaker, wiring or left power lever switch out of adjustment or defective.	b. Connect ohmmeter to pins F and K of computer connector. Turn battery switches on. With left power at flight idle, reading across pins F and K should be inifnite. Push left power lever forward, reading should be continuity. If reading is okay, proceed to step 6. f. If not, remove connector from control box. With ohmmeter connected to pins F and K jumper pins 3 and 4 of control box connector. Reading should be continuity. If not, wiring is defective. Connect ohmmeter to pin 5 of control box connector and ground. Reading should be continuity with left power forward and infinite with left power lever at flight idle. If not, switch or wiring is out of adjustment or defective, see Chapter 32-60-00 for power lever switch adjustment procedures. Check for 28 VDC at pin b. If 28 VDC is not present, TEST circuit breaker or wiring is defective.
	c. SAS FAULT light (annunciator), control PCB, or wiring defective.	c. Ground pin A of computer connector. Turn battery switches on. SAS FAULT light should illuminate steadily. If light illuminates, proceed to step 6. a. If not, remove connector from control box; remove jumper from pin A. Connect ohmmeter to pin 22 of A of computer connector. Reading should be continuity. Remove lead from pin A and connect to ground, reading should be infinite If not, wiring is defective. With SA control box disconnected, SAS FAULT light should not be illuminated. Jumper pin 6 to pin 23, SAS FAULT light should illuminate. If not, SAS FAULT light (annunciator) is defective. If wiring and light checks okay, PCB is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
(SAS ARM light is illuminated). SAS system inoperative, OR SAS indicator erratic, OR SAS FAULT light illuminates steadily, OR Pusher or horn erratic.	6. SAS computer AOA transmitter. Flap compensator, SAS indicator, or Squat switch defective or out of adjustment.	6. This step details procedures to determine whether the computer is defective or another component of the system is defective or merely out of adjustment. These steps are designed for use with SAS test box and extender harness TS27-02 although an alternate method of measuring voltages is given, If the computer is found to be defective, see Section 27-32-07 for replacement procedures. Connect all components and engage all circuit breakers of SAS system. Turn battery switch on.
	a. SAS FAULT light illuminated, SAS computer defective.	a. A steady SAS FAULT light indicates a possible seven problems. (1) Computer power see step 1. (2) AOA transmitter, see step 2. (3) Flap compensator, see step 3. (4) SAS FAULT light, see step 5. c. (5) SAS control box printed circuit board (PCB), see step 5. c. (6) AOA transmitter vane is stuck at it's upper stop. (7) SAS computer. Check (1) through (6) before replacing computer.
	b. Vane voltage, SAS computer defective.	b. Connect voltmeter to VANE VOLTS jacks on SAS test box TS27-02 or remove right rear console door and connect voltmeter to terminal 5 of TS16 (bottom terminal strip) and ground. Move vane to upper stop, voltmeter should read 5 VDC, move vane to lower stop voltmeter should read 15 VDC. If not, check step 2. If step 2 checks okay, computer is defective, see Section 27-32-08 to check vane voltage adjustment.
	c. Flap voltage, SAS computer defective.	c. Connect voltmeter to FLAP VOLTS jacks on SAS test box TS27-02. There is no other convenient way of measuring flap voltage at pins e and f of computer connector. With flaps up, voltmeter should read 4.00 VDC plus or minus .05. With flaps down, voltmeter should read 2.90 VDC plus or minus .05. If voltage reading is 0 volts, check step 3. If step 3 checks okay, computer is defective. If voltage reading is not 0, see Section 27-32-09 for flap compensator adjustment procedures.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
(SAS ARM light is illuminated). SAS system inoperative, OR SAS indicator erratic, OR SAS FAULT light illuminates steadily, OR Pusher or horn erratic.	d. SAS indicator, SAS computer defective.	d. Connect voltmeter to IND VOLTS jacks on SAS test box TS27-02 or pull indicator out and connect voltmeter to terminals on back of indicator. With flaps up and vane at it's upper stop, voltmeter should read approximately – 1.6 VDC. With vane at it's lower stop, voltmeter should read approximately +18 VDC. If not, check step 4, If step 4 checks okay and voltage not okay, computer is defective. If voltage is okay and indicator does not read 0.00 volts when pointer is on 1.0 V/Vs, 4.59 volts when pointer is on 1.3 V/Vs, and 11,30 volts when pointer is on 1.8 V/Vs, indicator is defective. Check indicator does not take more than a light tap to make it read correctly.
	e. STALL warning, SAS indicator defective.	e. If SAS test box TS27-02 is connected place horn/mute switch to HORN. Place squat switch/override switch to OVRD. If not, place squat switch cheater tool TF27-03 in left main gear inboard scissor switch. Have an assistant move vane up slowly while watching SAS indicator. When pointer of indicator passes to within one needle width of the red and black crosshatch sector, the horn should sound. If horn does not sound, check step 5.a. If step 5. a. checks okay, see next defect in DEFECT column. If horn sounds, but at the wrong place, see step 6. d.
	f. Power compensation, computer defective.	f. Have an assistant move vane up until the pointer of the SAS indicator is in the middle yellow SLOW sector. When the left power lever is pushed forward from flight idle the pointer of the SAS indicator should jump counterclockwise approximately 1/8 of an inch. If not, check step 5. b. If step 5. b. checks okay, computer is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
(SAS ARM light is illuminated). SAS system inoperative, OR SAS indicator erratic, OR SAS FAULT light illuminates steadily, OR Pusher or horn erratic.	g. Pusher, SAS indicator defective.	g. If SAS test box TS27-02 is connected place squat switch/overide switch to OVRD. If not, place squat switch cheater toll TP27-03 in left main gear inboard scissor switch. Have assistant move vane up slowly while watching SAS indicator. Pusher should kick in when pointer at SAS indicator passes to within one needle width of the redSTALL sector. Horn should also sound at place stated in step 6. e. If pusher kicks in but the wrong place, see step 6.d. If pusher does not kick in, see next defect in DEFECT column.
	h. SAS test, computer defective.	h. Hold SAS TEST switch to CRUISE. With flaps up pointer of SAS indicator should go to 1.3 V/Vs. Hold switch to STALL. Pointer of SAS indicator should go clockwise to the left of the green CRUISE band. Then sweep counterclockwise. The horn should sound, the pusher should kick in, and the SAS FAULT light should illuminate. Then the pointer should jump to just below 1.3 V/Vs and the above action should cease. If not, remove connector from SAS computer. Check for 28 VDC on pin N when SAS TEST switch is held to CRUISE. Check for 28 VDC on pin C when switch is held to STALL. If not, switch or wiring is defective. If 28 VDC is present, computer is defective. If pointer of SAS indicator does move but does not go to 1.3 when switch is held to CRUISE. Check adjustment of flap compensator, see Section 27-32-09.
Pusher and horn inoperative. SAS ARM light is illuminated. SAS FAULT light is not illuminated.		PRELIMINARY Refer to Wiring Diagram Manual 27-32-06. If the aircraft is on the ground and the left main gear scissor switch is not depressed, the pusher will not kick in and the horn will not sound when the vane is moved up.
	1. Left main gear inboard scissor switch or wiring defective.	1. Remove connector from computer. Connect ohmmeter to pin D and ground. When scissor switch is not depressed, reading should be continuity (0 ohms). When switch is depressed, reading should be infinite.
	2. SAS computer defective.	2. If step 1 checks okay, computer is is defective. See 27-32-07 for replacement procedures.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pusher and horn inoperative. SAS ARM light not illuminated.	1. Airspeed switch or wiring defective.	1. Remove connector from airspeed switch, Jumper pin A and B. If pusher and horn are operative and SAS ARM light illuminates, airspeed switch is defective, see 27-30-12 for replacement procedures. If horn is operative and pusher and SAS ARM light are inoperative, proceed to step 2. If SAS ARM light illuminates and pusher and horn are inoperative, proceed to step 3.
	2. SAS control box, servo relay, PCB, SAS ARM light or wiring defective.	2. Replace connector to airspeed switch. Remove connector from control box. Check pin 25 for ground. If not, wiring to airspeed switch is defective. Ground pin 12, SAS ARM light should illuminate. if not, SAS ARM light (annunciator) is defective. If okay, control box is defective. see Section 27-32-07 for replacement procedures.
	3. SAS computer or control box defective.	3. Replace connector to airspeed switch. Remove connector from computer. Check pin L for ground. If not grounded, computer is defective. If grounded, SAS control or wiring is defective. See Section 27-32-07 for replacement procedures.
Pusher inoperative. SAS CLUTCH switch ON. SAS FAULT light blinks. SAS ARM light is illuminated.		PRELIMINARY Refer to Wiring Diagram Manual 27-32-06. A blinking SAS FAULT light indicates the servo motor is not operating.
	1. SAS CLUTCH switch or wiring defective.	1. Remove connector from computer. Check pin M for 28 VDC. If not present, SAS CLUTCH or wiring is defective.
	2. Servo defective.	2. Remove connector from servo. Check pin 1 for 28 VDC and pin 2 for ground. If 28 VDC and ground are present, servo is defective, see Section 27-30-11 for replacement procedures.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pusher inoperative. SAS CLUTCH switch ON. SAS FAULT light is not illuminated. SAS ARM light is illuminated.	1. SAS CLUTCH switch or wiring defective.	1. Remove connector from computer and servo. Connect ohmmeter to pin T of computer connector and pin 3 of servo connector. Reading should be continuity with SAS CLUTCH switch ON and infinite with switch OFF. If not, switch or wiring is defective.
	2. SAS control box or wiring defective.	2. Connect ohmmeter to pin Z of computer connector and pin 4 of servo connector. With power on reading should be continuity. If not, SAS control box or wiring is defective, see Section 27-32-07 for replacement procedures.
	3. Servo defective.	3. Install computer connector. Connect milliammeter to pins 3 (positive) and 4 (negative) of servo connector. Depress left main gear inboard scissor switch with squat switch cheater tool TF27-03. Turn SAS CLUTCH switch ON. Move vane to upper stop. Milliammeter reading should be approximately 50 mA. If 50 mA, servo is defective, see Section 27-30-11 for replacement procedures.
	4. Computer defective.	4. If steps 1 and 2 check okay and step 3 check 0 mA, computer is defective, see Section 27-32-07 for replacement procedures.
SAS FAULT light does not blink or blinks erratically when SAS CLUTCH is OFF.	1. SAS control box (PCB) defective.	1. Pull COMP PWR circuit breaker. SAS FAULT light should illuminate steadily. If SAS FAULT light illuminates, SAS control box (PCB) is defective, see Section 27-32-07 for replacement procedures. If light does not illuminate, remove connector from SAS control box. Jumper pin 6 to pin 23. If light illuminates, SAS control box is defective.
	2. SAS FAULT light (annunciator) defective.	2. If light does not illuminate, SAS FAULT light (annunciator) is defective.
	3. SAS computer defective.	3. Install connector on control box and remove connector from computer. Ground pin A, SAS FAULT light should illuminate steadily. Ground pin J, light should blink. If so, computer is defective. If not, control box is defective, see Section 27-32-07 for replacement procedures.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
SAS FAULT light blinks above SAS ARM speed.	1. SAS idle control defective.	1. Remove connector from SAS idle control. Check pin 5 for 28 VDC. Check pin 1 and 4 for ground. Turn SAS CLUTCH switch ON. Check pin 2 does not read any voltage. Ground pin 3 SAS FAULT light should blink. Turn power off. Remove connector from computer. Connect ohmmeter from pin 2 of idle control and pin M of computer connector, Reading should be approximately 20 ohms. If all the above checks okay, SAS idle control is defective, see Section 27-32-13 for further check procedures and replacement procedures.
SAS suddenly out of adjustment; previously okay.	1. Flap compensator out of adjustment.	1. See Section 27-32-09 for replacement, adjustment, and check procedures.
	2. AOA transmitter damaged or out of adjustment.	2. Check condition of vane. Vane should move smoothly and easily in a 50 degree arc from stop tp stop, see Section 27-32-08 for adjustment, replacement, and check procedures.
	3 Power compensation/ left power lever switch out of adjustment.	3. See steps 6. f. and 5. b. of first defect.
	4. SAS indicator defective.	4. See Section 27-30-10 for replacement and check procedures.

1. Inspection/Check – SAS General

A. 200 Hour Inspection:

(1) The following components of the SAS system will be inspected for condition and security:

(a) SAS Indicator

(b) SAS Interface Control Assembly

(c) SAS Computer

(d) SAS Servo Idle Control (When installed)

(e) SAS Flap Compensator

(f) AOA Transmitter

(g) Airspeed Switch

(h) Pusher Servo

(i) Pusher Capstan

B. 500 Hour Inspection:

NOTE: Steps **2.**, B., (1) through **2.**, B., (6) of the following functional check procedures apply to SAS Servo, Part Number 27-44049 only. Omit steps **2.**, B., (1) through **2.**, B., (6) for aircraft with the 32-19023 SAS Servo installed.

If the SAS Servo fails the following inspection, it must be replaced using a serviceable unit which is either new, or overhauled by an FAA Approved Repair Station.

Because the SAS Servo must remain in motion during flight (slower speed to keep the clutch magnetic particles from settling), it is recommended that the Servo be allowed to run at least 5 minutes prior to the accomplishment of any Functional Check of the SAS System (operation in 32°F (0°C) or lower temperatures may require longer Servo run time).

(1) Remove carpet as necessary, and center aisle floorboards to gain access to stick pusher servo and capstan. Servo capstan is located in the center aisle, approximately in line with the wing leading edge.

(2) Disconnect servo electrical connector.

(3) Connect an instrumented variable DC power supply (0-28 VDC, minimum 5 ampere output) as shown in Figure 201.

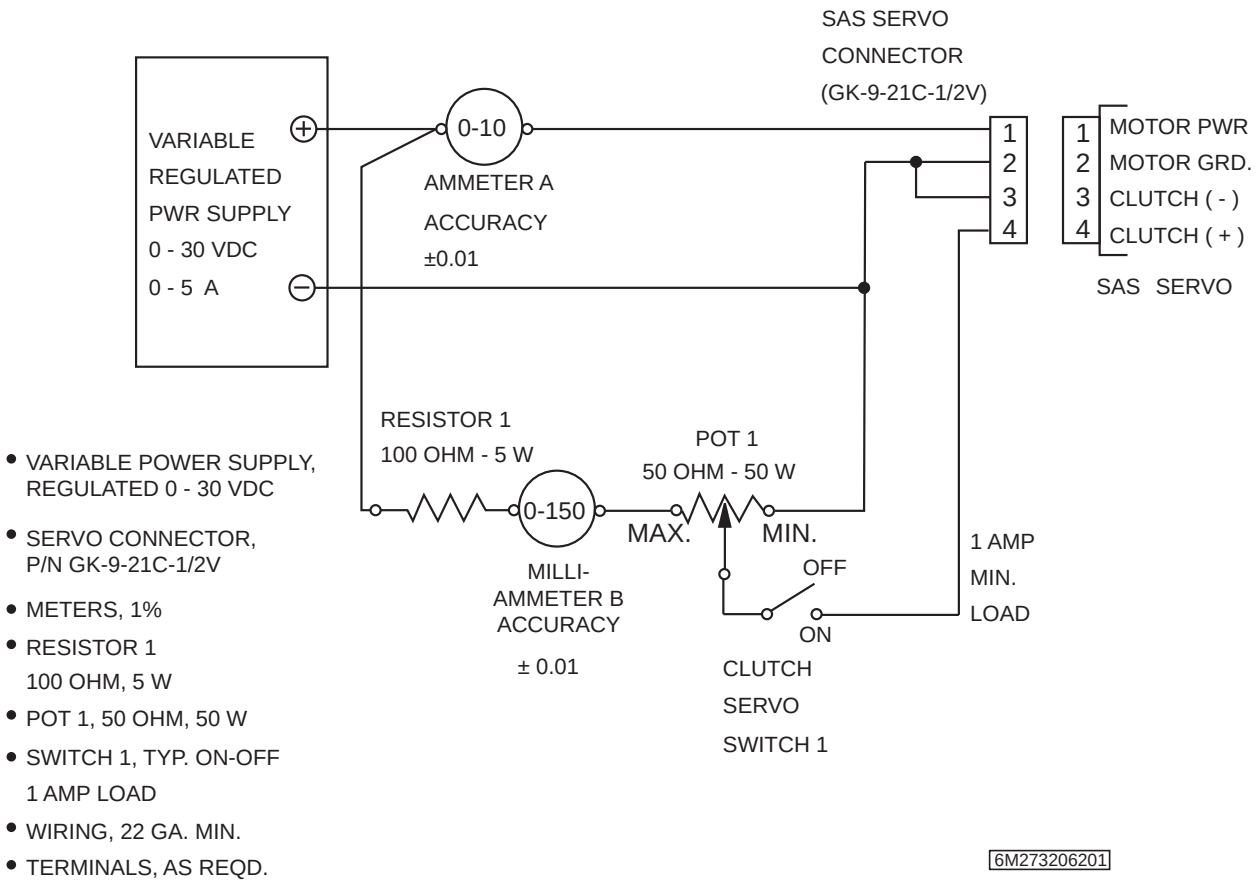
(4) Determine servo start voltage:

(a) Select CLUTCH/SERVO SWITCH 1 to OFF.

(b) Starting at a minimum DC voltage, increase POWER SUPPLY voltage until the motor begins to turn.

(c) Note voltage on POWER SUPPLY VOLTMETER where motor begins to turn.

(d) If start voltage is more than 8 VDC, replace SAS servo and recalibrate system.



SAS Servo Functional Check Schematic
Figure 201

(5) Determine unloaded motor current:

(a) Select CLUTCH/SERVO SWITCH 1 to OFF

(b) Increase DC voltage to 28 VDC and note ampere reading on AMMETER A.

(c) If unloaded motor current is more than 1.0 ampere, replace SAS servo and recalibrate system.

(6) Determine loaded motor current:

(a) Ensure power supply voltage is set at 28 VDC.

(b) Hold control column firmly at mid travel point and select CLUTCH/SERVO SWITCH 1 to ON.

(c) Rotate POT 1 until MILLIAMMETER B reads 60 mA.

(d) Note motor ampere reading on AMMETER A. Turn CLUTCH/SERVO SWITCH 1 to OFF. Release SAS TEST switch.

(e) If loaded motor current is more than 2.7 amperes, replace SAS servo and recalibrate system.

(7) Determine capstan condition:

(a) Position and hold force gauge to forward top center of pilots control column (see Figure 202).

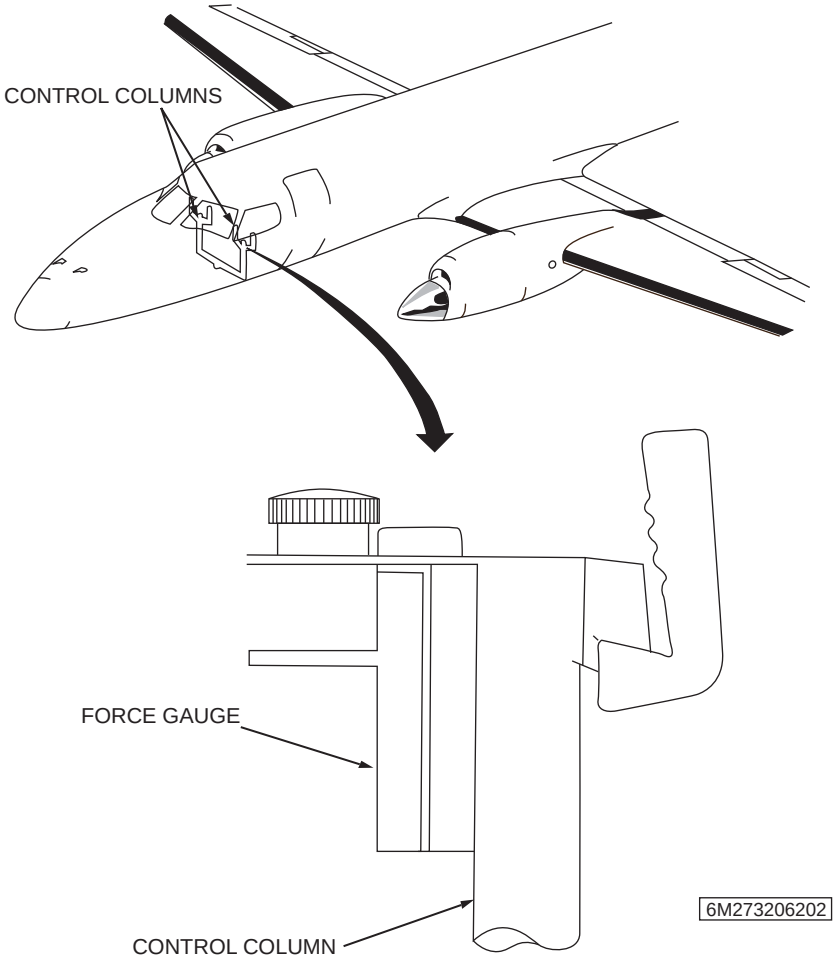
(b) Hold pilots control column securely in neutral position. Set force gauge to read zero (0).

(c) Ensure unrestricted travel of control column.

(d) For airplanes with P/N 32-19023 SAS Servo, turn CLUTCH/SERVO switch ON. Hold SAS TEST switch to STALL.

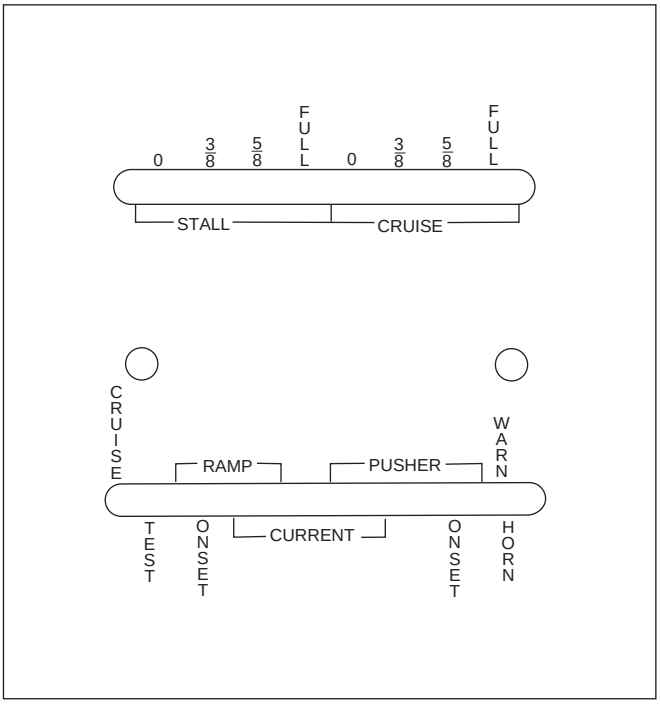
(e) For airplanes with P/N 27-44049 SAS Servo, increase clutch current to 60 mA.

(f) Increase force pulling aft on control column until capstan ratchets or control column moves aft. Minimum force should be 60 pounds (27.22 kg), maximum 86 pounds (39.01 kg).

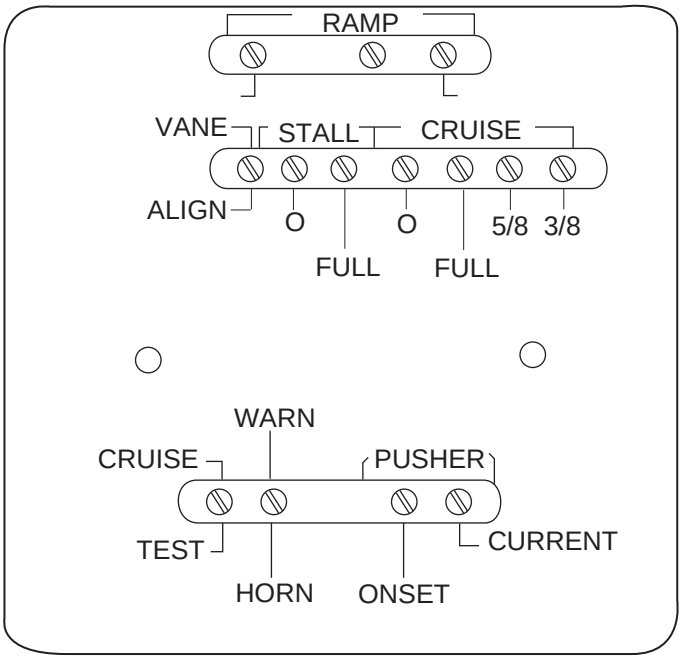


Control Column
Force Gauge Installation
Figure 202

SAS COMPUTER ADJUSTMENT
POTENTIOMETER
LOCATION PLATE
TF 27-01
(For -2 Computers)



SAS COMPUTER ADJUSTMENT
POTENTIOMETER
LOCATION PLATE
TF 27-01A
(For -3 and -4 Computers)
(Can be used for -5 Computer
but no Ramp Adjustment)



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Computer Adjustment
Location Plate TF 27-01
Figure 203

(g) Turn CLUTCH/SERVO switch OFF. Release SAS TEST switch.

(h) If capstan condition does not allow the control column and SAS servo to be overridden, the capstan must be disassembled, cleaned, calibrated and the torque reset IAW STICK PUSHER ASSEMBLY – Adjustment/Test, in section 27-30-11.

(8) Disconnect variable DC power supply and connect aircraft harness.

(9) Turn BATT switches ON.

(10) Turn SAS CLUTCH/SERVO switch ON.

(11) Hold control column approximately vertical.

(12) Turn SAS TEST switch to STALL, check that pusher comes ON.

(13) Release SAS TEST switch and turn SAS CLUTCH/SERVO and BATT switches OFF.

(14) Remove force gauge from control column.

(15) Install center aisle floorboards and carpet.

C. 2000 Hour Recalibration Procedures – On Ground Recalibration, Using 32-82032-001 or AEC2000-1 Calibrator. A flight check is not required when one of these calibration units is used to re-calibrate the SAS computer.

NOTE: Accomplish 200 and 500 flight hour servo functional checks, in conjunction with these procedures.

If S. T. C. SA4725SW is installed, follow procedures for Computer Ground Set-Up Procedure, Vane Voltage Adjustment, and Flap Compensator Voltage Adjustment in Consolidated Engineering Services operating procedures manual, and step **1.**, C., (2). Do not use procedure defined in steps **1.**, C., (3), (4), or (5).

The following procedures apply to recalibration of the Conrac SAS P/N 54302-1 , -2 and -3 computers in SA226-T and T(B) Model airplanes using AEC 2000-1 or 32-82032-001 calibrators. A flight check IS NOT REQUIRED when one of these calibration units is used to recalibrate the SAS Computer.

SA226-T and T(B) Model airplanes with P/N 54302-1 or -2 Computer must have P/N 27-82079-003 Potentiometer/Relay Assembly installed and adjusted as described in SB 226-27-037.

The following procedures also apply to SA226-AT and TC Model airplanes equipped with Conrac SAS P/N 54302-4 and -5 Computers installed per S.T.C. SA4725SW, Rosemount to Conrac SAS conversion.

If the TS27-01 Computer Calibrator is used, In-Flight Recalibration is required. Comply with **1.**, D. 2000 Hour Recalibration Procedures; using TS27-01 Calibrator and **1.** E. 2000 Hour Recalibration Procedures – Flight Recalibration, this section.

Service Bulletin 226-27-026 should be accomplished prior to these procedures if not already incorporated to simplify calibration.

(1) Materials and Equipment Required:

(a) One of the following Conrac SAS Computer Calibrators:

1 Conrac SAS Computer Calibrator, AEC-2000-1 (Consolidated Engineering Services, San Antonio, TX). Refer to Figure 204.

2 Conrac SAS Computer Calibrator 32-82032-001. (M7 Aerospace, San Antonio, TX). Refer to Figure 204.

(b) SAS Computer Calibration Data Sheet and SAS Computer Calibration Data Graph from aircraft log or previous calibration setup, (see Figures 206 and 208).

(c) SAS Computer Adjustment Potentiometer Location Plate, (see Figure 203).

(d) SAS AOA Vane Adjustment Tool, (see Figure 210).

(e) Squat switch cheater tool, (see Figure 212).

(f) Control column force gauge, 0-200 lbs (90.72 kg) range.

(g) Conrac SAS Indicator (P/N 3065847, or 562-892) removed from aircraft.

(h) Ground Power Unit (GPU).

(i) Small common screwdriver (for potentiometer adjustments).

(2) Computer Functional Check (Figure 204)

NOTE: SAS Computer Set-Up Procedure, **1.**, C., (3), this section must be accomplished before the computer functional check. If set-up procedures are not accomplished FIRST, the values called for in this procedure may be erroneous.

(a) Connect AEC 2000 -1 or 32-82032 -001 Conrac SAS Computer Calibrator to the aircraft SAS system as shown in Figure 204.

(b) Position CALIBRATE/NORMAL switch on back of calibrator, Figure 204, to NORMAL.

(c) Position PWR/OFF switch to PWR.

(d) Position A/C-BENCH switch to BENCH.

(e) Position VANE L- R switch to R.

NOTE: L- R-VANE switch is installed on some early model AEC2000-1 calibrators only. There is no L- R-VANE switch on 32-82032-001 calibrators.

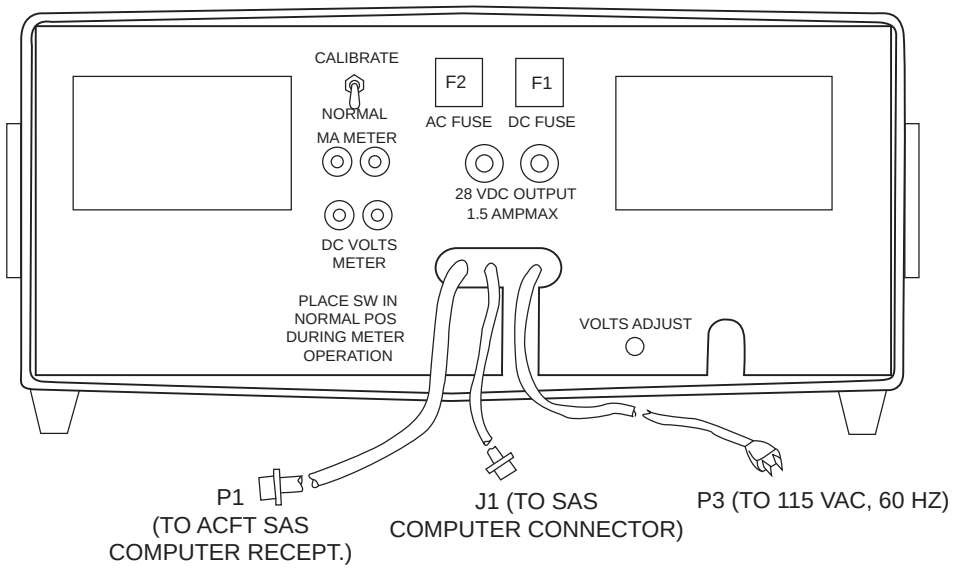
(f) Position FLAP-VANE-SIG SWITCH (S1) to FLAP.

(g) Position A/S switch to BELOW.

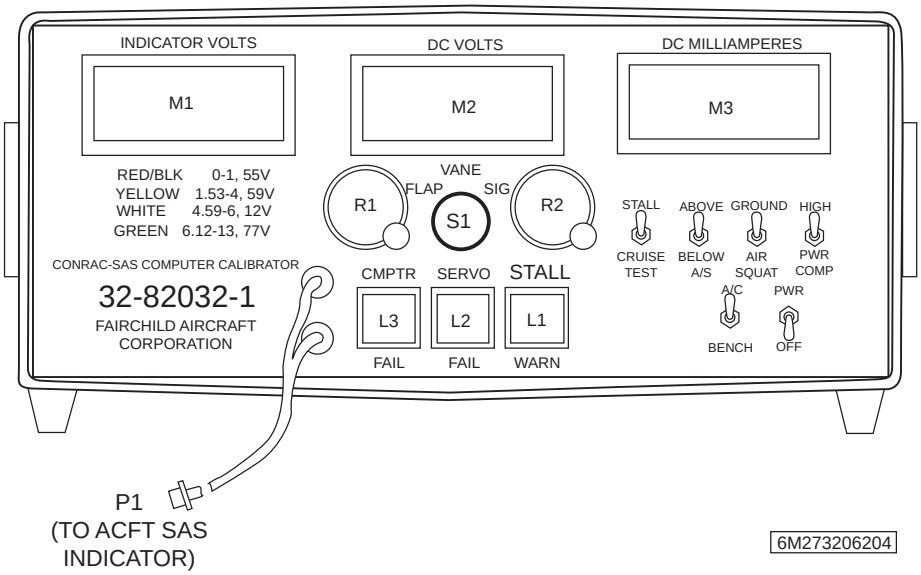
(h) Position SQUAT switch to AIR.

(i) Move FLAP knob (R1) until DC VOLTS meter (M2) indicates approximately 2.9 volts.

NOTE: All values in column A and B are ± 0.01 VDC. All values in column D are ± 0.40 VDC. All values in column E are ± 1.0 mA for P/N 27-44049 servo and + 5.0, - 0.0 mA for P/N 32-19023 servo.



REAR VIEW



FRONT VIEW

AEC2000-1 and 32-82032-001
SAS Calibrator and Connections
Figure 204

(j) Position FLAP-VANE-SIG switch (S1) to VANE.

(k) Move VANE knob (R2) counterclockwise (DC VOLTS must be at least 12 VDC), until DC VOLTS meter (M2) indicates voltage called for in column B, Step 10, SAS data sheet. Observe STALL WARN light (L1) illuminates when INDICATOR VOLTS meter (M1) reads 1.75 VDC.

(l) Continue to turn VANE knob (R2) in the same direction until INDICATOR VOLTS meter (M1) reads approximately 0.20 volts and DC MILLIAMPERES meter (M3) indicates approximately 55 mA.

(m) Continue to turn VANE knob (R2) in the same direction until INDICATOR VOLTS meter (M1) reads approximately -1.50 volts and the CMPTR FAIL light (L3) illuminates. Observe DC MILLIAMPERES meter (M3) indicates:

1 0.00 mA on airplanes with -4 and -5 computers.

2 ramp current (25 mA) on airplanes with -2 (with mod) and -3 computers.

(n) Turn VANE knob (R2) in opposite direction until INDICATOR VOLTS meter (M1) reads 0.00 volts. Observe DC MILLIAMPERES meter (M3) indicates 55 mA and CMPTR FAIL light (L3) is extinguished.

(o) Position A/S switch to ABOVE and hold. Observe STALL WARN light (L1) extinguishes and DC MILLIAMPERES meter (M3) indicates approximately 0 mA.

(p) Move A/S switch to BELOW. Observe STALL WARN light (L1) illuminates and DC MILLIAMPERES meter (M3) returns to approximately 55 mA.

(q) Position SQUAT switch to GROUND and hold. Observe STALL WARN light (L1) is extinguished and DC MILLIAMPERES meter (M3) indicates approximately 0 mA.

(r) Continue to hold SQUAT switch in GROUND and rotate VANE knob (R2) counterclockwise until COMPTR FAIL light (L3) illuminates. Observe DC MILLIMAPERES meter (m3) indicates approximately 0 mA.

(s) Release SQUAT switch and allow it to return to AIR position. Observe STALL WARN light (L1) illuminates. Turn the VANE knob (R2) clockwise until the CMPTR FAIL light (L3) extinguishes, and DC MILLIAMPERES meter (M3) indicates approximately 55 mA.

(t) Press and hold CMPTR FAIL light (L3). Observe CMPTR FAIL light (L3) illuminates, STALL WARN light (L1) extinguishes and all meter indications (M1), (M2), and (M3) drop to 0.

NOTE: The INDICATOR VOLTS meter (M1) may indicate a negative voltage.

(u) Release CMPTR FAIL light (L3) and observe CMPTR FAIL light extinguishes, STALL WARN light (L1) illuminates and meters (M1), (M2), and (M3) return to indications prior to step (t).

(v) Press and hold SERVO FAIL light (L2). Observe SERVO FAIL light illuminates. Release SERVO FAIL light and observe light extinguishes.

(w) Rotate VANE knob (R2) to obtain approximately 9.90 volts as indicated by the DC VOLTS meter (M2).

(x) Position PWR COMP switch to HIGH. Observe INDICATOR VOLTMETER (M1) decreases approximately 0.35 volts.

(y) Release PWR COMP switch. INDICATOR VOLTS meter (M1) should return to original indication.

(z) If computer will not perform functional check, replace computer.

(3) Computer Set-Up Procedure (Figures 204, 206, and 208), Using 32-82032-001 or AEC2000-1 Calibrator:

NOTE: The following steps apply to and correlate with the steps in SAS Data Sheet, Figure 206. Accomplish steps 1, 2, 7, and 8, then steps 3, 4, 5, and 6. Repeat steps 1 through 8 a second time to verify settings.

Use SAS DATA sheet and SAS COMPUTER CALIBRATION graph from aircraft log, or from previous set-up.

(a) Connect aircraft SAS Computer and Indicator to AEC2000-1 or 32-82032-001 Conrac SAS Computer Calibrator.

(b) Remove end plate from computer, and install adjustment location plate. Position CALIBRATE/NORMAL switch on back of calibrator, to NORMAL.

(c) Position PWR/OFF switch to PWR (front of calibrator).

(d) Position A/C-BENCH switch to BENCH.

(e) Position L R-VANE switch to R.

NOTE: L R-VANE switch is installed on some early model AEC2000-1 calibrators only. There is no L R-VANE switch on 32-82032-001 calibrators.

(f) STEP 1 – SAS Data Sheet:

1 Set FLAP-VANE-SIG switch (S1) to FLAP. Adjust FLAP knob (R1) to the voltage listed in SAS Data Sheet, column A, as indicated on DC VOLTS meter (M2).

NOTE: All values in column A and B are ± 0.01 VDC. All values in column D are ± 0.40 VDC. All values in column E are ± 1.0 mA for P/N 27-44049 servo and + 5.0, - 0.0 mA for P/N 32-19023 servo.

Voltages in SAS Data Sheet, column B are taken from the original SAS Data Sheet supplied with the airplane at delivery. If the original data sheet numbers are lost or otherwise not available, perform a SAS In-Flight Calibration. Extract voltages after In-Flight Set-Up and enter those voltages in SAS Data Sheet, column B.

2 Set FLAP-VANE-SIG switch (S1) to VANE. Adjust VANE knob (R2) to the voltage listed in SAS Data Sheet, column B, as indicated on DC VOLTS meter (M2).

3 Locate the SAS Computer adjustment trimmer potentiometer called out in SAS Data Sheet column C (see Figure 203).

4 Adjust SAS Computer potentiometer as necessary to obtain voltage called for in column D, SAS Data Sheet as indicated on the INDICATOR VOLTS meter (M1). Record actual voltage in column D, SAS Data Sheet.

5 Milliampere readings, column E, SAS Data Sheet, does not apply to steps 1 through 8.

6 Aircraft SAS Indicator should read V/V_s in column F, SAS Data Sheet. Lightly tap indicator to settle needle.

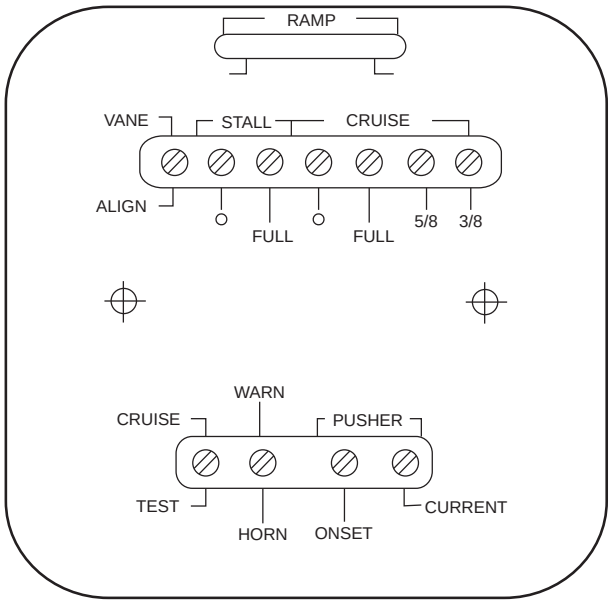
(g) STEP 2 – SAS Data Sheet:

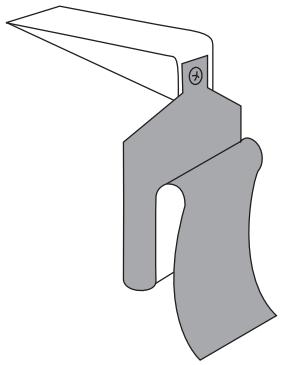
1 Step 2 procedures are identical to Step 1. Comply with Step 1, sub-steps 1 through 6.

(h) STEP 3 – SAS Data Sheet:

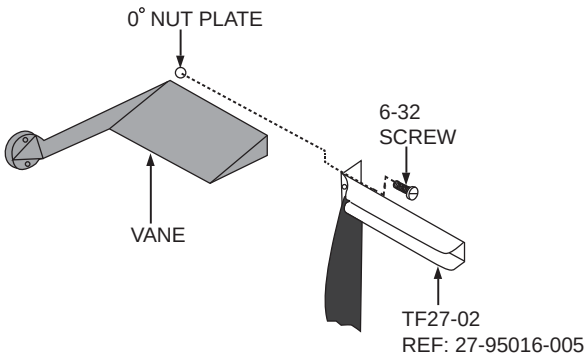
NOTE: Step 3 procedures do not apply to 54302-4/5 computers.

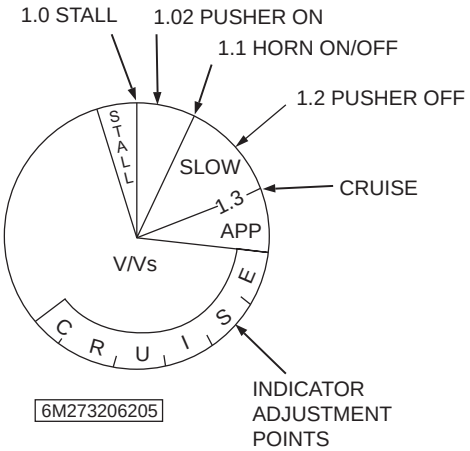
1 Step 3 procedures are identical to Step 1. Comply with Step 1, sub-steps 1 through 6.





SQUAT SWITCH
CHEATER TOOL
TF27-03





Ground Set-Up Equipment
Figure 205

SAS COMPUTER CALIBRATION DATA SHEET

FOR: { AIRCRAFT MODEL_____ AIRCRAFT S/N_____
AIRCRAFT TAIL NO._____ COMPUTER PART NO. _____
COMPUTER S/N _____ DATE: _____

STEP	COLUMN A (M-2 CALIBRATOR) SET FLAP VOLTS TO	COLUMN B (M-2 CALIBRATOR) SET VANE VOLTS TO	COLUMN C ADJUST COMPUTER POT	COLUMN D TO OBTAIN IND. VOLTS PREFERRED (+) OR (-) 0.10	COLUMN D TO OBTAIN IND. VOLTS ACTUAL (RECORD)	COLUMN E TO OBTAIN MILLAMPS	COLUMN F SAS INDICATOR V/V_S
1	4.00		0 STALL	0.00			1.0
2	4.00		0 CRUISE	4.59			1.3
3	3.70		N/A(3/8 STALL)	0.00			1.0
4	3.70		3/8 CRUISE	4.59			1.3
5	3.40		N/A(5/8 STALL)	0.00			1.0
6	3.40		5/8 CRUISE	4.59			1.3
7	2.90		FULL STALL	0.00			1.0
8	2.90		FULL CRUISE	4.59			1.3
9	TURN SLOWLY CW	DO NOT TOUCH		JUMP -1			
				JUMP -2			
				JUMP -3			
10	2.90	TURN SLOWLY CCW	WARN HORN	STALL WARN LIGHT ON			1.12
				1.75			
11	2.90	TURN SLOWLY CCW	PUSHER ONSET	mA INCREASE (+) OR (-) 0.05		INCREASE	1.02
			PUSHER CURRENT	AT 0.20	AT	55 mA	
12	2.90		RAMP CURRENT	AT 6.04	AT	25 mA	1.40
			TOP OF RAMP	mA STOPS AT		TOP 25 mA	
				6.04			
13	2.90		RAMP ONSET	11.30		8 mA	1.8
14	4.00		CRUISE TEST	4.59			1.3
15	4.00		CHECK ONLY	STALL WARN LIGHT ON	CHECK		1.12
16	4.00		CHECK ONLY	mA INCREASE	CHECK	55 mA	1.02
17	4.00	12.00	CHECK ONLY			0 mA	

6M273206206

SAS Computer
Calibration Data Sheet
Figure 206

(i) STEP 4 – SAS Data Sheet:

1 Step 4 procedures are identical to Step 1. Comply with Step 1, sub-steps 1 through 6.

(j) STEP 5 – SAS Data Sheet

NOTE: Step 5 procedures do not apply to 54302-4/5 computers.

1 Step 5 procedures are identical to Step 1. Comply with Step 1, sub-steps 1 through 6.

(k) STEP 6 – SAS Data Sheet:

1 Step 6 procedures are identical to Step 1. Comply with Step 1, sub-steps 1 through 6.

(l) STEP 7 – SAS Data Sheet:

1 Step 7 procedures are identical to Step 1. Comply with Step 1, sub-steps 1 through 6.

(m) STEP 8 – SAS Data Sheet:

1 Step 8 procedures are identical to Step 1. Comply with Step 1, sub-steps 1 through 6.

(n) STEP 9 – SAS Data Sheet:

NOTE: DO NOT adjust VANE knob (R2) during this step. VANE voltage should remain as set in step 8, column B, SAS Data Sheet.

DO NOT adjust SAS Computer potentiometers during this step.

1 Set FLAP-VANE-SIG switch (S1) to FLAP.

2 Adjust FLAP knob (R1) slowly clockwise while watching INDICATOR VOLTS meter (M1). INDICATOR VOLTS meter should increase in three jumps from the actual voltage reading called out in step 8, column D, SAS Data Sheet. Each increase (jump) should be at least 1 VDC from previous reading.

3 If voltages do not increase as called out, repeat SAS Data Sheet, steps 1 through 8.

4 Milliampere reading, Column E, SAS Data Sheet, does not apply to step 9.

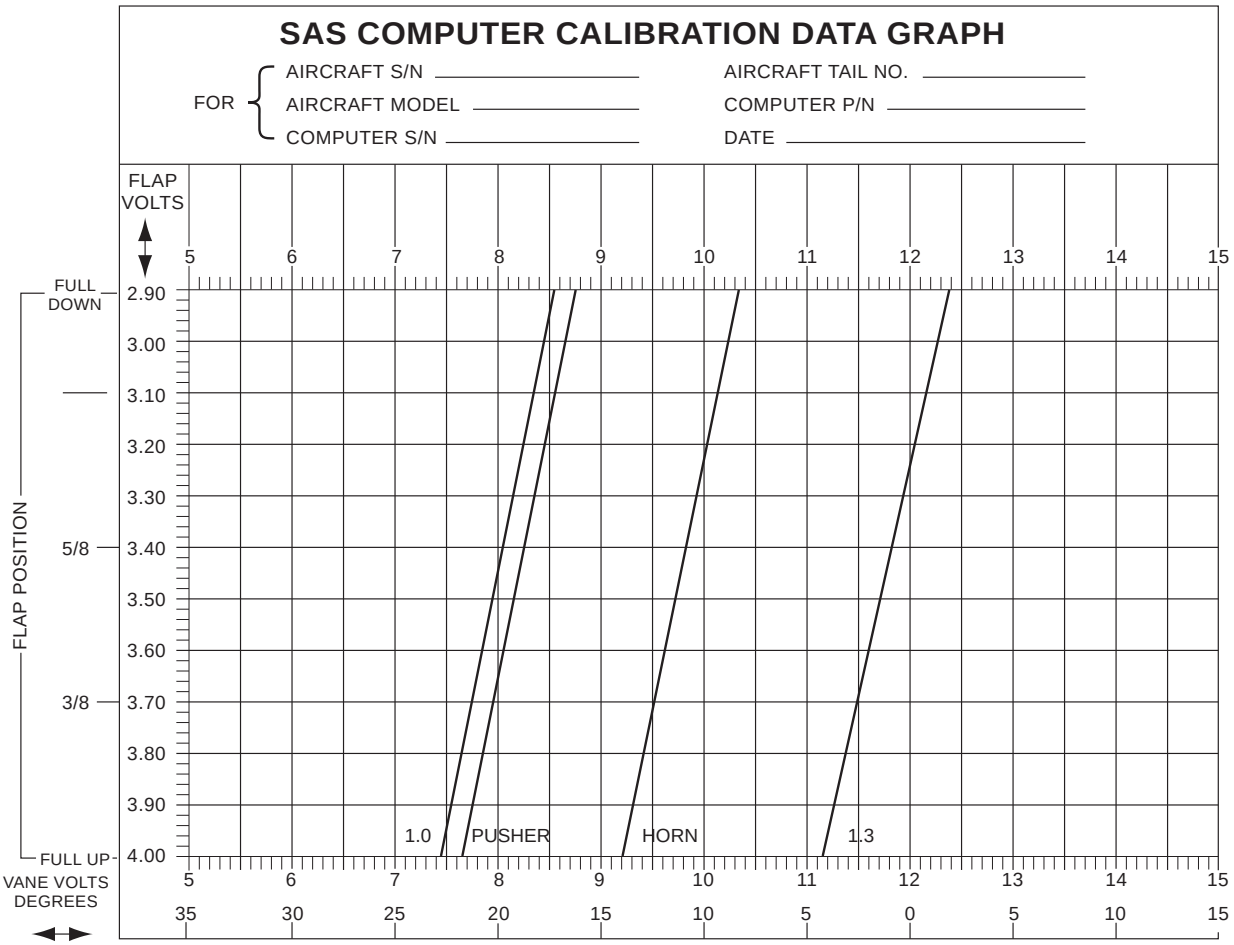
5 Aircraft SAS Indicator should read V/V_s in Column F, SAS Data Sheet for Indicator VDC value in column D, SAS Data Sheet. Lightly tap indicator to settle needle.

(o) STEP 10 – SAS Data Sheet:

1 Set FLAP-VANE-SIG switch (S1) to FLAP.

2 Adjust FLAP knob (R1) to obtain the voltage in column A, SAS Data Sheet, as indicated on the DC VOLTS meter (M2).

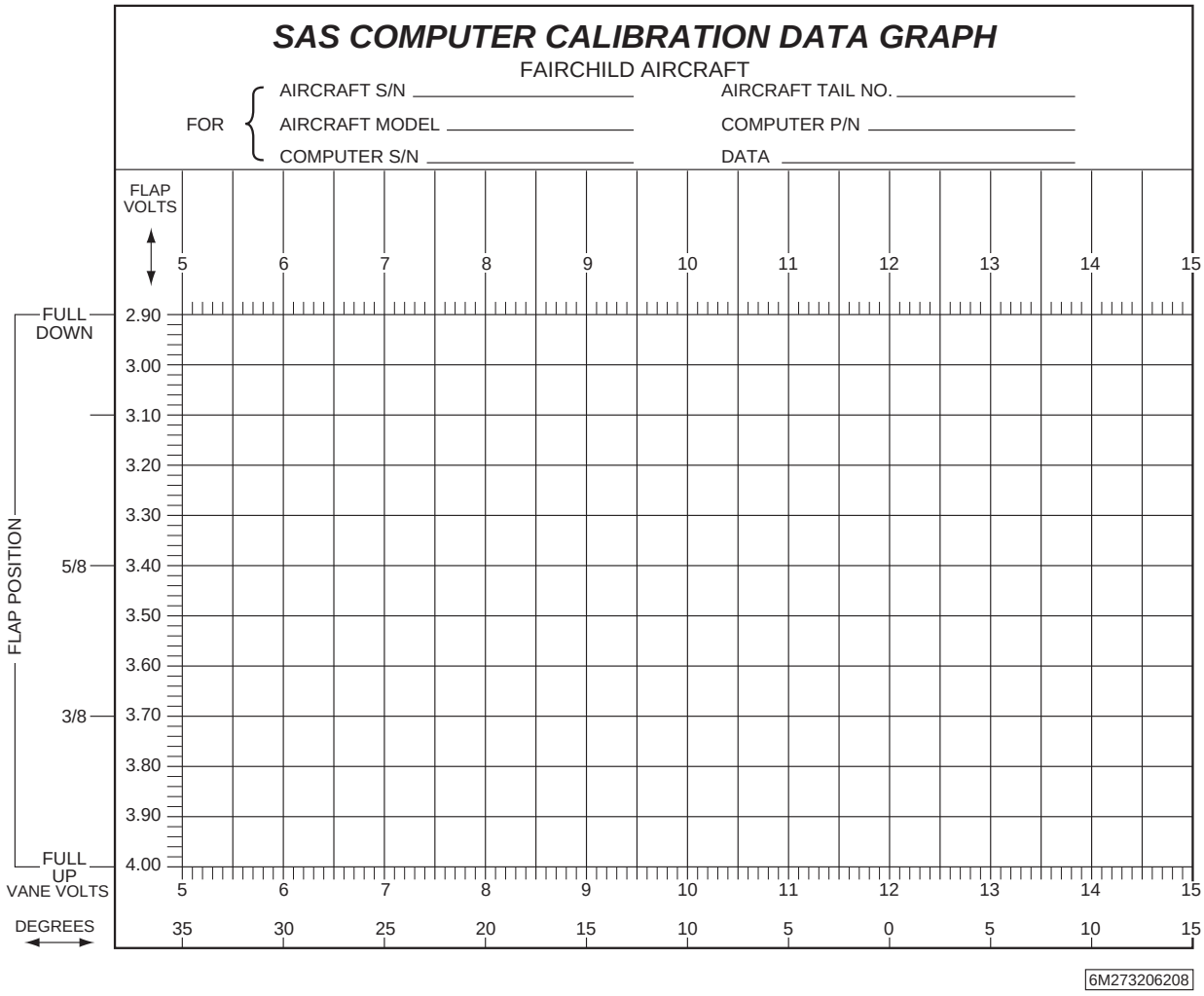
3 Set FLAP-VANE-SIG switch (S1) to VANE.



EXAMPLE ONLY

6M273206207

SAS Computer Calibration
Data Graph – Example Only
Figure 207



SAS Computer Calibration
Data Graph
Figure 208

4 Slowly adjust VANE knob (R2) counterclockwise while watching DC VOLTS meter (M2) and STALL WARN light (L1). STALL WARN light should illuminate as voltage decreases to the vane voltage called out in column B, SAS Data Sheet. If voltage fails to decrease, see step 10, sub-step 5.

5 Locate Computer WARN HORN potentiometer, Figure 203. Adjust WARN HORN potentiometer to make STALL WARN light (L1) illuminate as vane voltage decreases to value called out in column B SAS Data Sheet.

6 Check INDICATOR VOLTS meter (M1) reads voltage in column D, SAS Data Sheet. Record actual voltage in column D, SAS Data Sheet.

7 Milliamperes reading, column E, SAS Data Sheet, does not apply to step 10.

8 Aircraft SAS Indicator should read V/V_s, in column F, SAS Data Sheet, when STALL WARN light (L1) illuminates. Tap indicator lightly to settle needle.

(p) STEP 11 – SAS Data Sheet:

1 Set FLAP-VANE-SIG switch (S1) to FLAP.

2 Adjust FLAP knob (R1) to obtain the voltage in column A, SAS Data Sheet, as indicated on the DC VOLTS meter (M2).

3 Set FLAP-VANE-SIG switch (S1) to VANE.

4 Slowly adjust VANE knob (R2) counterclockwise while watching DC VOLTS meter (M2) and DC MILLIAMPERES meter (M3). MILLIAMPERES meter (M3) should jump (increase) to 55 mA as the voltage on DC VOLTS meter (M2) deceases to the value called out in column B, SAS Data Sheet. If voltage does not jump, adjust IAW step 11, sub-step 5 below.

NOTE: Indicator voltage must be increased by approximately 2.0 volts before DC MILLIAMPERES meter will reset to 0.00 mA.

5 Locate Computer PUSHER ONSET potentiometer (Figure 203). Adjust PUSHER ONSET potentiometer to make DC MILLIAMPERES meter (M3) jump (increase) to 55 mA when voltage in column B, SAS Data Sheet, is reached, as indicated on INDICATOR VOLTS meter (M2).

6 Check that INDICATOR VOLTS meter (M1) reads voltage in column D, SAS Data Sheet. Record actual voltage in column D, SAS Data Sheet.

7 DC MILLIAMPERES meter (M3) should indicate 55 mA, after jump (increase). If M3 does not read 55 mA, adjust PUSHER CURRENT potentiometer (Figure 203) to value in column E, SAS Data Sheet.

8 Check that aircraft SAS Indicator reads V/V_s,in column F, SAS Data Sheet, when DC MILLIAMPERES meter (M3) increases to value in column E, SAS Data Sheet. Lightly tap indicator to settle needle.

(q) STEP 12 – SAS Data Sheet:

NOTE: STEPS 12 and 13 do not apply to 54302-4 or 54302-5 computers. For 54302-4 and -5 computers, proceed to STEP 14.

Substeps 7 through 11 are applicable to 54302-2 and -3 computers only.

1 Move FLAP-VANE-SIG switch (S1) to FLAP.

2 Adjust FLAP knob (R1) to obtain voltage listed in column A, SAS Data Sheet, as indicated on DC VOLTS meter (M2).

3 Move FLAP-VANE-SIG switch (S1) to VANE.

4 Adjust VANE knob (R2) to obtain 12 volts, as indicated on DC VOLTS meter (M2).

5 Locate computer RAMP/CURRENT potentiometer, Figure 203.

6 Adjust RAMP/CURRENT potentiometer until DC MILLIAMPERES meter (M3) indicates, column E value, when INDICATOR VOLTS meter (M1) reads 6.04 VDC. Record actual voltage in column D, SAS Data Sheet.

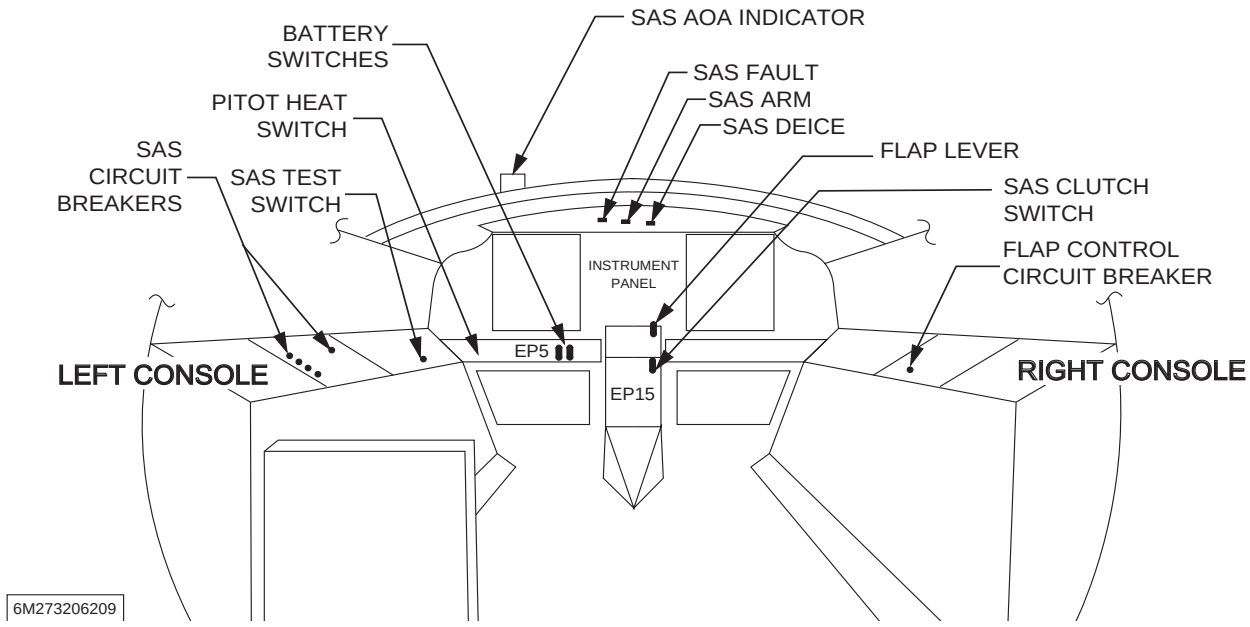
7 Locate computer TOP OF RAMP potentiometer, Figure 203.

8 Adjust TOP OF RAMP potentiometer so that DC MILLIAMPERES meter (M3) stops increasing at 6.04 volts as indicated on INDICATOR VOLTS meter (M1).

9 Adjust VANE knob (R2) to obtain 6.04 volts, as indicated on INDICATOR VOLTS meter (M1) while watching DC MILLIAMPERES meter (M3).

10 DC MILLIAMPERES meter (M3) should increase to approximately 25 mA and stop increasing when INDICATOR VOLTS meter (M1) reaches 6.04 volts. If this condition does not exist, repeat sub-steps 5 through 9.

11 Observe SAS Indicator reads V/Vs, in column F, SAS Data Sheet, when INDICATOR VOLTS meter (M1) displays 6.04 volts. Lightly tap in indicator to settle needle.



SAS Component Locator – Cockpit
Figure 209

(r) STEP 13 – SAS Data Sheet, (54302-1 Computers Only):

1 Move FLAP-VANE-SIG switch (S1) to FLAP position.

2 Adjust FLAP knob (R1) to obtain voltage listed in column A, SAS Data Sheet, as indicated on DC VOLTS meter (M2).

3 Locate computer RAMP/ONSET potentiometer, Figure 203.

4 Adjust RAMP/CURRENT potentiometer until DC MILLIAMPERES meter (M3) indicate column E value when INDICATOR VOLTS meter (M1) reads 11.30 VDC.

5 Adjust VANE knob (R2) to obtain voltage in column D, SAS Data Sheet, as indicated on INDICATOR VOLTS meter (M1).

6 DC MILLIAMPERES meter (M3) should read column E, SAS Data Sheet. If this condition does not exist, repeat sub-steps 3 and 4.

7 Observe aircraft SAS Indicator reads V/Vs in column F, SAS Data Sheet.

(s) STEP 14 – SAS Data Sheet:

1 Set FLAP-VANE-SIG switch (S1) to FLAP.

2 Adjust FLAP knob (R1) to obtain voltage called out in column A, SAS Data Sheet, as indicated on DC VOLTS meter (M2).

3 Vane voltage, column B, SAS Data Sheet, does not apply to step 14.

4 Locate computer CRUISE/TEST trimmer potentiometer (Figure 203).

5 Position TEST switch in CRUISE and hold. Adjust computer CRUISE/TEST potentiometer so that aircraft SAS Indicator reads V/Vs in column F, SAS Data Sheet. Lightly tap indicator to settle needle. INDICATOR VOLTS meter (M1) should read voltage called out in column D, SAS Data Sheet.

6 Milliamperes reading, column E, SAS Data Sheet, does not apply to step 14.

7 Release TEST switch.

(t) STEP 15 – SAS Data Sheet:

NOTE: DO NOT adjust any SAS Computer potentiometers during Step 15.

1 Set FLAP-VANE-SIG switch (S1) to FLAP.

2 Adjust FLAP knob (R1) to obtain voltage called out in column A, SAS Data Sheet, as indicated on DC VOLTS meter (M2).

3 Set FLAP-VANE-SIG switch (S1) to VANE. Adjust VANE knob (R2) so that STALL WARN light (L1) just illuminates. Check voltage on DC VOLTS meter (M2) is same as voltage called out in column B, SAS Data Sheet.

4 Check that INDICATOR VOLTS meter (M1) reads voltage called out in column D, SAS Data Sheet.

5 Milliamperes, column E, SAS Data Sheet, does not apply to Step 15 procedure.

6 Check aircraft SAS Indicator reads V/Vs in column F, SAS Data Sheet. Lightly tap indicator to settle needle.

(u) STEP 16 – SAS Data Sheet:

NOTE: DO NOT adjust any SAS Computer potentiometers during Step 16.

1 Set FLAP-VANE-SIG switch (S1) to FLAP.

2 Adjust FLAP knob (R1) to obtain voltage called out in column A, SAS Data Sheet, as indicated on DC VOLTS meter (M2).

3 Set FLAP-VANE-SIG switch (S1) to VANE. Adjust VANE knob (R2) counterclockwise slowly until DC MILLIAMPERES meter (M3) jumps (increases) to value called out in column E, SAS Data Sheet.

4 Check that INDICATOR VOLTS meter (M1) reads voltage called out in column D, SAS Data Sheet.

5 DC MILLIAMPERES meter (M3) should read 50 - 55 mA.

6 Check aircraft SAS Indicator reads V/Vs in column F, SAS Data Sheet. Lightly tap indicator to settle needle.

(v) STEP 17 – SAS Data Sheet:

NOTE: DO NOT adjust any SAS Computer potentiometers during Step 17.

1 Set FLAP-VANE-SIG switch (S1) to FLAP.

2 Adjust FLAP knob (R1) to obtain voltage called out in column A, SAS Data Sheet, as indicated on DC VOLTS meter (M2).

3 Set FLAP-VANE-SIG switch (S1) to VANE. Adjust VANE knob (R2) to obtain voltage called out in column B, SAS Data Sheet, as indicated on the DC VOLTS meter (M2).

4 Check that INDICATOR VOLTS meter (M1) reads voltage called out in column D, SAS Data Sheet.

5 DC MILLIAMPERES meter (M3) should read value called out in column E, SAS Data Sheet.

6 Check aircraft SAS Indicator reads V/Vs in column F, SAS Data Sheet. Lightly tap indicator to settle needle. Read exact V/Vs, and write zero degrees = 12 VDC = (this) V/Vs on computer. This V/Vs reading is to be used to check the zero (0) degree setting of the AOA transmitter vane when the computer is installed in the aircraft.

(4) Angle of Attack (AOA) Vane Voltage Set Up (Figures 204, 209, and 210)

CAUTION: TO PREVENT ELECTRICAL SHOCK AND/OR SPARK DAMAGE, MAKE SURE ALL AIRCRAFT ELECTRICAL POWER IS OFF BEFORE DISCONNECTING OR CONNECTING ELECTRICAL EQUIPMENT/ COMPONENTS TO AIRCRAFT.

(a) Disconnect SAS Computer and connect SAS Computer Calibrator to aircraft SAS system as shown in Figure 204.

(b) Apply 115 volts, 60 Hz electrical power to SAS Calibrator.

(c) Connect GPU to aircraft and set voltage to 29.0, plus 0.0 or minus 0.5 volts, as indicated by the Essential Bus Voltmeter.

(d) Attach TF27-02 (27-95016-005) Vane Adjustment Tool to zero degree nutplate as shown in Figure 210.

(e) Remove radome and access cover from F.S. 0.00 to gain access to back of zero degree nutplate.

(f) Loosen screws on vane adjustment bracket.

(g) Move LH and RH BATT switch to ON (see Figure 209).

(h) Set SAS Calibrator PWR-OFF switch to PWR.

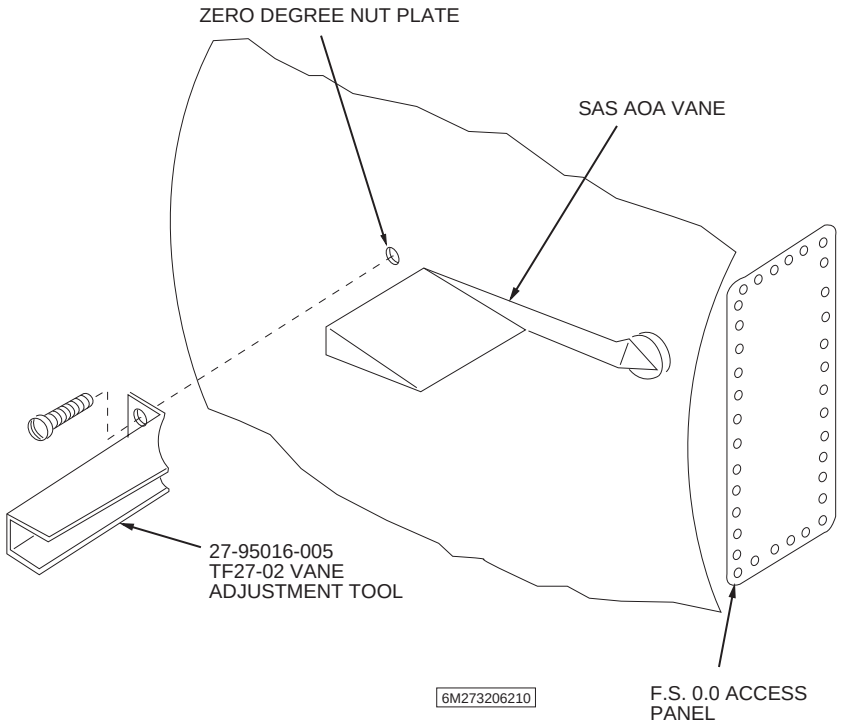
(i) Set SAS Calibrator A/C-BENCH switch to A/C.

(j) Set SAS Calibrator L R-VANE switch to R.

NOTE: L R-VANE switch is installed on some early model AEC 2000-1 calibrators only. There is no L R-VANE switch on 32-82032-001 calibrators.

(k) Set FLAP-VANE-SIG switch (S1) to VANE.

(l) Engage all SAS circuit breakers on LH Essential Bus (see Figure 209).



SAS Vane Adjustment Tool and Set-Up
Figure 210

(m) Rotate Vane Adjustment Bracket to achieve 12.0 volts, ± 0.10 volts, as indicated on DC VOLTS meter (M2).

(n) Tighten screws to secure adjustment bracket.

NOTE: Verify voltage remains as set after tightening adjustment bracket screws. If voltage has changed, readjust to correct voltage IAW **1.**, C., (4), (m).

(5) SAS Flap Compensator Voltage Set Up (Figures 204, 209, and 211)

CAUTION: TO PREVENT ELECTRICAL SHOCK AND/OR SPARK DAMAGE, MAKE SURE ALL AIRCRAFT ELECTRICAL POWER IS OFF BEFORE DISCONNECTING OR CONNECTING ELECTRICAL EQUIPMENT/ COMPONENTS TO AIRCRAFT.

(a) Connect Hydraulic Power Cart.

(b) Move LH and RH BATT switches to ON (see Figure 209).

(c) Set SAS Calibrator PWR/OFF switch to PWR.

(d) Set SAS Calibrator A/C-BENCH switch to A/C.

(e) Set SAS Calibrator L R-VANE switch to R.

NOTE: L R-VANE switch is installed on some early model AEC 2000-1 calibrators only. There is no L R-VANE switch on 32-82032-001 calibrators.

(f) Set FLAP-VANE-SIG switch (S1) to FLAP.

(g) Engage all SAS circuit breakers on LH Essential Bus (see Figure 209).

(h) Engage Flap Control circuit breaker on RH console Non-Essential Bus (see Figure 209).

(i) Position FLAP CONT switch to FULL UP (see Figure 209).

(j) Loosen Allen head screw securing flap compensator arm to flap compensator potentiometer shaft (see Figure 211).

(k) Using a small common screwdriver, rotate potentiometer shaft slowly to achieve 4.00, ± 0.05 volt, as indicated on the DC VOLTS meter (M2).

(l) Tighten Allen head screw and recheck voltage. If voltage has changed from correct setting, readjust potentiometer IAW **1.**, C., (5), (k).

(m) Position FLAP CONT switch (Figure 209) to FULL DOWN. DC VOLTS meter (M2) should decrease to 2.90 volts, ± 0.05 volt.

(n) If voltage fails to decrease to 2.90 volts, readjust flap compensator potentiometer linkage as follows:

1 Perform flap compensator rod adjustment per step 6 and repeat 1., C., (5), (j) through (m).

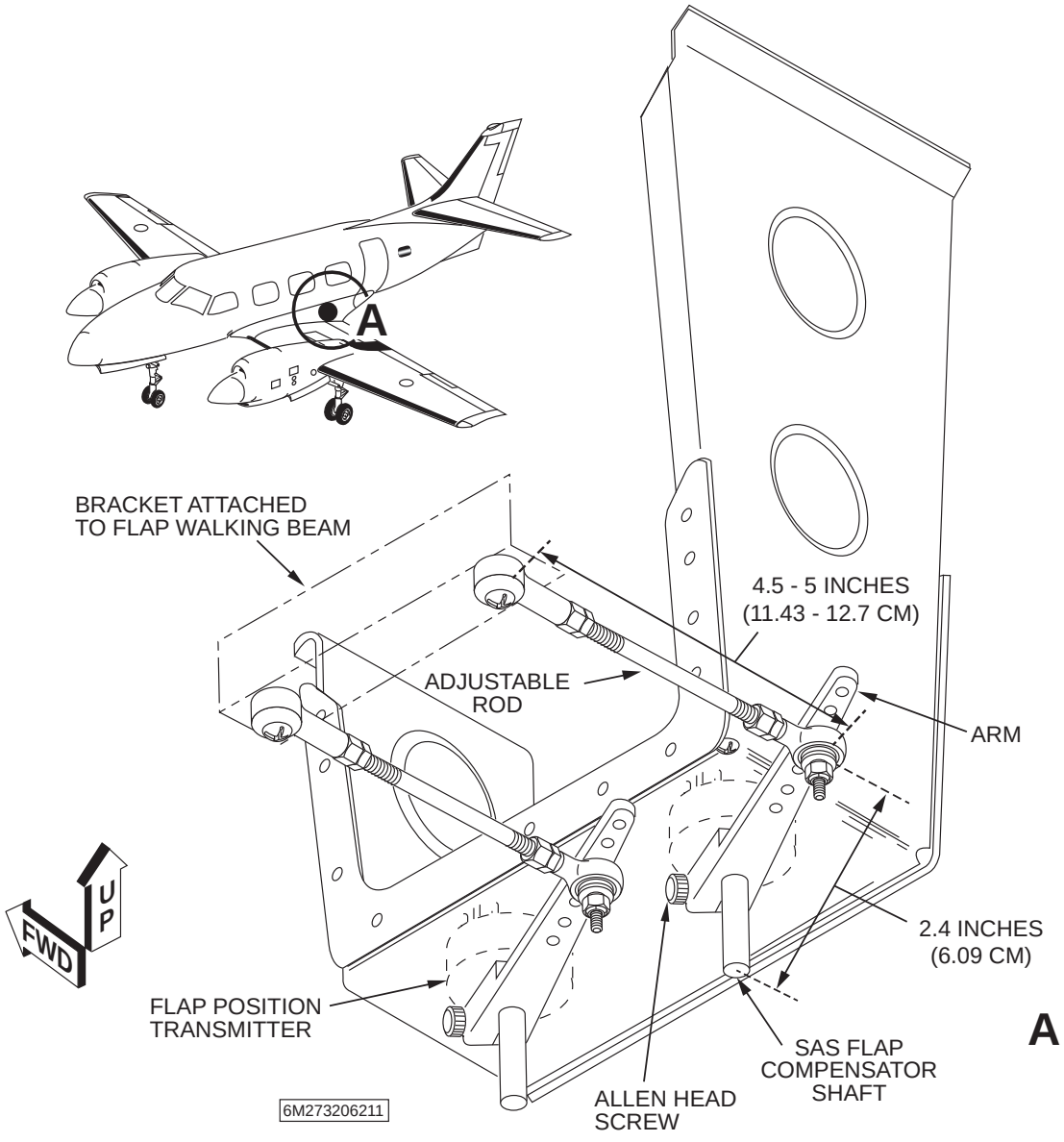
2 If DOWN voltage is higher than 2.90 ± 0.05 volts, move rod end to next bolt hole in arm closer to flap compensator shaft.

3 If DOWN voltage is lower than 2.90 ± 0.05 volts, move rod end to next bolt hole in arm farther from flap compensator shaft.

(o) Repeat **1.**, C., (5), (j) through (n) as many times as necessary to achieve correct voltage settings of 4.00, ± 0.05 volt, in FULL UP, and 2.90, ± 0.05 volt, in FULL DOWN.

(p) Position LH and RH BATT switches OFF.

(q) Disconnect Hydraulic Power Cart and GPU.



SAS Flap Compensator Transmitter Assembly
Figure 211

(r) Remove AOA Vane Adjustment tool; install radome and F.S. 0.00 access panel.

(s) Disconnect SAS Computer Calibrator and reconnect SAS Computer.

(6) Flap Compensator Rod Adjustment:

(a) Adjust rod 4.5 to 5.0 inches long. Adjust arm so rod end center is 2.4 inches from center of potentiometer shaft. Tighten screws.

(b) Check rod end engagement to rod by inserting a piece of 0.032 inch safety wire into check hole in rod ends.

(c) Repeat **1.**, D., (5), (j) through (n) to ensure correct voltage settings of 4, ± 0.05 VDC, in FULL UP, and 2.9, ± 0.05 VDC, in FULL DOWN.

NOTE: If necessary to increase or decrease the range of voltage change between Flaps UP and DOWN, move rod end to arm connection one bolt hole closer to or farther away from potentiometer respectively.

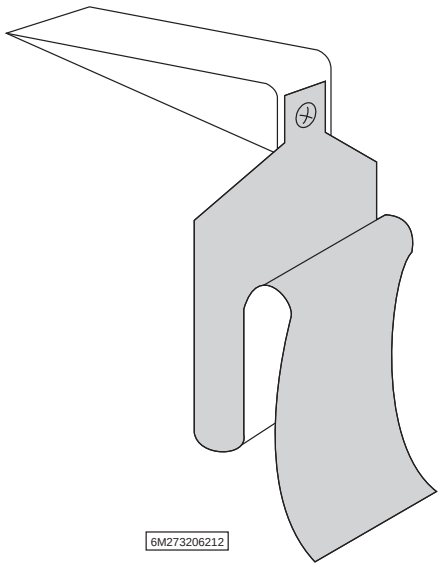
(7) SAS System Functional Check (Figures 202, 203, 209, 212, and 213)

NOTE: Before attempting to perform a system functional check, verify computer is connected to aircraft wiring, SAS angle of Attack (AOA) Vane is against lower stop and control gust lock is disengaged.

(a) Install 32-82020-021 (TF27-03) Squat Switch Cheater Tool, Figure 212, on LH inboard main landing gear squat switch so that squat switch plunger is depressed.

(b) Using an external Hydraulic Power Cart, apply hydraulic power to the aircraft hydraulic system.

(c) Connect a Ground Power Unit (GPU) and apply 29.0, + 0.0 or - 0.5 volts as measured at Essential Bus voltmeter.



TF27-03 (32-82032-021) Squat Switch Cheater Tool
Figure 212

(d) Position LH and RH BATT switches (Figure 209) ON.

(e) Observe aircraft annunciator panel SAS ARM light (Green) illuminates and SAS FAULT light (Red) blinks on and off.

(f) Position SAS CLUTCH switch (Figure 209) ON.

(g) Observe aircraft annunciator SAS FAULT light (Red) extinguishes.

(h) Observe SAS Indicator pointer, Figure 213, moves clockwise completely out of green CRUISE band.

(i) Position and hold force gauge to forward top center of pilots control column (Figure 202).

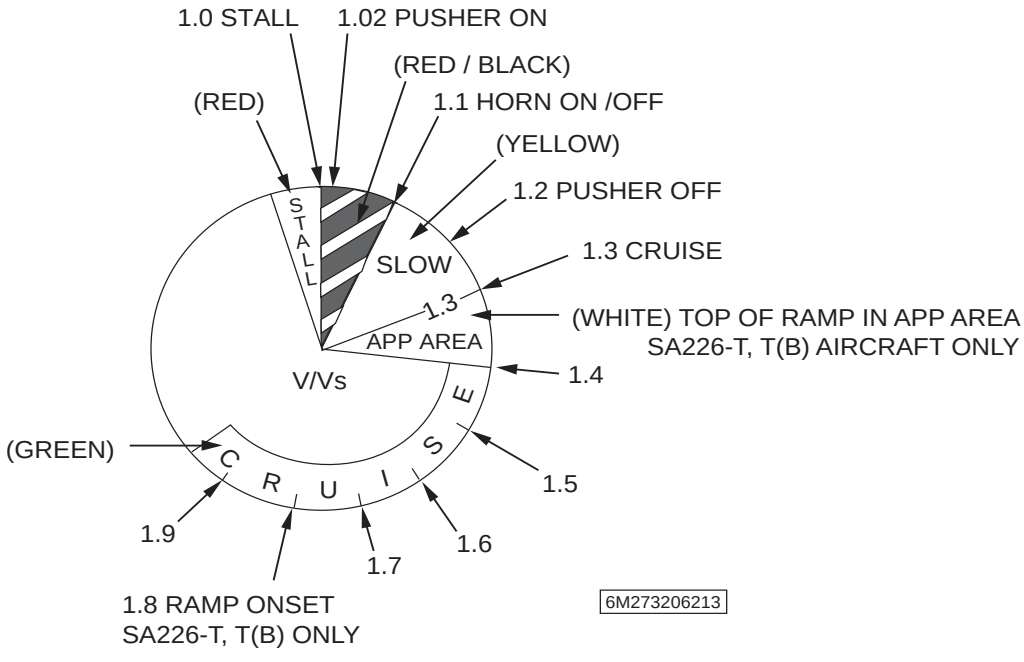
(j) Hold pilots control column securely in neutral position. Set force gauge to read zero (o).

(k) Vane Adjustment Check – SA226 T/T(B)

1 Have an assistant slowly move SAS Vane UP until SAS Indicator indicates 1.75 V/Vs at upper end of slow band.

2 Continue to move vane slowly upward; stick force should increase to 20, ± 3 pounds (7.71 - 10.43 kg) at 1.4 V/Vs. Stick force should remain at 20, ± 3 pounds (7.71 - 10.43 kg), until 1.02 V/Vs is reached in step below.

3 Check for stall warning horn at 1.12 V/Vs.



SAS Indicator Adjustment Points
Figure 213

4 At 1.02 V/Vs, stick force should increase to 65, ± 5 pounds (27.22 - 31.75 kg).

5 Continue upward movement until SAS FAIL light illuminates. Verify as SAS FAIL light illuminates, stick force drops to approximately 20 lbs (9.07 kg).

6 Have assistant move vane to FULL DOWN. Check that SAS FAULT light extinguishes.

(l) Vane Adjustment Check – SA226 AT/TC

1 Have an assistant slowly move SAS Vane UP until SAS Indicator indicates 1.12 V/Vs at upper end of slow band. Stall warning horn should sound.

2 Continue to move vane slowly upward until SAS Indicator indicates 1.02 V/Vs at upper end of Red and Black band.

3 Observe force gauge indicates 65, ± 5 pounds (27.22 - 31.76 kg).

4 Continue upward movement until SAS FAIL light illuminates. Verify as SAS FAIL light illuminates, stick pusher disengages.

(m) Remove force gauge from pilots control column.

(n) Move SAS Vane to FULL DOWN position.

(o) Observe annunciator SAS FAULT light (Red) is extinguished.

(8) Torque Motor Output Adjustment (32-19023 SAS Servo):

NOTE: Aircraft with S.T.C. SA4725SW and P/N 32-19023 SAS Servo installed must be equipped with a 5 amp SAS SERVO PWR circuit breaker (CB197).

(a) Remove SAS Computer end cover and locate Pusher Current Pot. Adjust until 65 pounds ± 5 pounds (27.22 - 31.76 kg) is achieved.

CAUTION: DO NOT EXCEED COMPUTER LIMITS OF 55 mA, + 5.0, - 0.0 mA.

(b) If 65 ± 5 pounds (27.22 - 31.76 kg) can not be accomplished, set Computer Output to 55 mA, + 5.0, - 0.0 mA.

(c) Install ammeter in line of SAS SERVO PW circuit breaker (CB197). At SAS Servo, remove connector from SAS Servo.

(d) Remove end cover from Servo, (see Figure 206). Connect Servo connector.

(e) Move SAS Vane upward until SAS Indicator indicates 1.02 at upper end of Red and Black band.

(f) Locate R8 on SAS Servo internal PC board, (see Figure 206). Adjust R8 in Servo until 60 ± 5 pounds (27.22 - 31.75 kg) is achieved.

CAUTION: DO NOT EXCEED 4.5 AMPS AT 28.5 VDC APPLIED THROUGH SAS SERVO 5 AMP CIRCUIT BREAKER.

(g) Remove ammeter in line of circuit breaker (CB197). Remove connector from Servo. Install cover on Servo end plate. Connect Servo connector.

(h) Remove force gauge from pilot's control column.

(i) Move SAS Vane to FULL DOWN position.

(j) Observe annunciator SAS FAULT light (Red) is extinguished.

(k) On SA226 AT/AC aircraft, move SAS TEST switch to STALL and hold. Check for the following conditions:

1 SAS Indicator pointer moves counterclockwise to 1.00 V/Vs.

2 Stick pusher activates and column pushes forward.

3 Stall Warning Horn sounds.

(l) On SA226 T/T(B) aircraft, Move SAS TEST switch to STALL and hold. Observe the following on SAS indicator and elevator control:

1 The pointer of the SAS indicator should, depending on vane position, descend to, or start from, the left side of the green CRUISE band and sweep counterclockwise.

2 As the pointer sweeps through the CRUISE band, a low and gradually increasing forward force should be felt on the elevator control.

3 This force should reach its peak. approximately 20 pounds (9.07 kg), when the pointer of the indicator reaches 1.40 V/Vs, and remain the same until pusher activation.

4 As the pointer of the SAS indicator leaves the yellow SLOW area and enters the red and black crosshatch area, the stall warning horn should sound.

5 As the pointer of the SAS indicator leaves the red and black crosshatch area and enters the red STALL area, the pusher should engage.

6 The SAS FAULT light should illuminate as the pointer passes the middle of the STALL area.

7 The pointer should travel to the left side of the STALL area and delay for a few seconds.

8 The pointer should return to the APP area, just below 1.30 V/Vs.

9 As the pointer returns, the SAS FAULT light should extinguish and the pusher and horn should cease.

10 Release SAS TEST switch and allow it to return to OFF. Check that:

a Stick pusher OFF.

b SAS Warning Horn stops sounding.

c SAS Indicator pointer returns to original position.

11 Move SAS CLUTCH/SERVO switch to OFF.

12 Remove squat switch cheater tool.

13 Move SAS CLUTCH/SERVO switch to ON.

14 Place SAS TEST switch to CRUISE and hold. Check that SAS Indicator moves to 1.30 V/Vs.

15 Move FLAP CONTROL switch to FULL DOWN and look for three (3) distinct jumps of SAS Indicator needle in counterclockwise direction.

CAUTION: REMOVE SAS VANE AND PITOT COVERS BEFORE PROCEEDING. KEEP PERSONNEL CLEAR OF SAS VANE AND PITOT HEADS TO AVOID INJURY. VERY HIGH SAS VANE AND PITOT HEAD TEMPERATURES ARE REACHED WHEN HEATERS ARE SWITCHED ON.

16 Momentarily position LH PITOT HEAT switch to PITOT/SAS and check for the following conditions:

a Annunciator panel SAS DEICE light (Green) illuminates.

b LH Pitot mast and SAS Vane start to heat up.

17 Momentarily position RH PITOT HEAT switch to PITOT/SAS and check for the following conditions:

a Annunciator panel SAS DEICE light (Green) illuminates.

b RH Pitot mast and SAS Vane start to heat up.

18 Position SAS CLUTCH/SERVO switch (Figure 209) to OFF.

19 Move BATT switches (Figure 209) to OFF.

20 Switch GPU OFF and disconnect from aircraft.

21 Switch Hydraulic Power Cart OFF and disconnect from aircraft.

D. 2000 Hour Recalibration Procedures; using TS27-01 Calibrator:

NOTE: Accomplish 200 hour and 500 flight hour checks, in conjunction with this procedure.

The following procedures apply to recalibration of the Conrac SAS P/N 54302-1 and -2 computers in T(B) aircraft or -2, -3, -4, and -5 computers in T, AT, and TC aircraft with STC4725SW installed, using TS27-01 calibrator. A flight check IS REQUIRED when this calibrator unit is used to recalibrate the SAS computer.

Aircraft which do not have Service Bulletin 226-27-026 accomplished should have that bulletin accomplished at this time to simplify calibration procedures.

(1) Materials and Equipment Required

(a) CONRAC SAS Computer Calibrator TS 27-01 (see Figure 214).

(b) SAS Calibration Data Sheet (see Figure 206).

(c) SAS Computer Calibration Graph (see Figure 208).

(d) Digital Voltmeter - accuracy 0.01 VDC within the 20 volt range.

(e) Milliammeter - accuracy 1.0 mA within the 100 mA range.

(f) Power supply of 28.5 VDC, capable of supplying 2 amps.

(g) Adjustment Location Plate (see Figure 203).

(h) CONRAC SAS Indicator (remove from aircraft).

(i) SAS AOA Vane Adjustment Tool (see Figure 210).

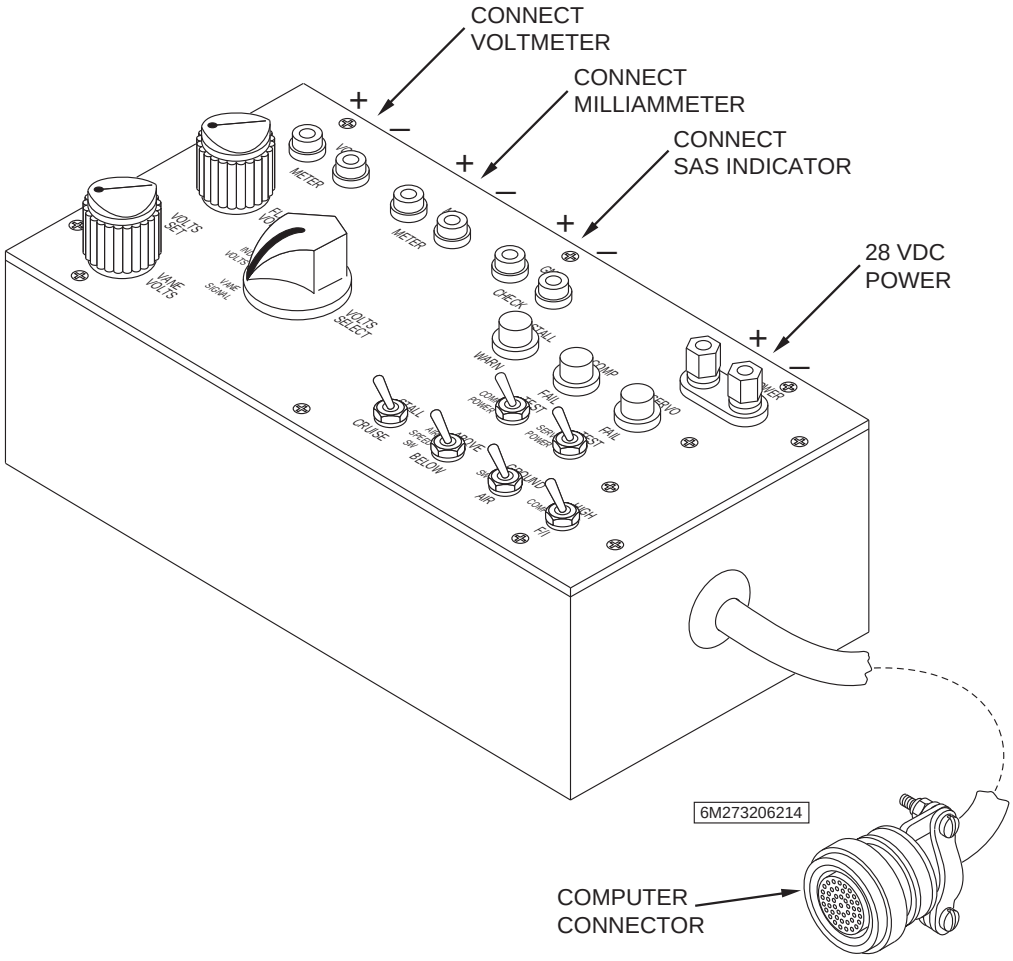
(j) Squat Switch Cheater Tool (see Figure 212).

(k) Control column force gauge, 0-200 lbs (90.72 kg) range.

(l) Auxiliary Ground Power Unit.

(m) Auxiliary Hydraulic Power Unit.

(n) Small Common Screwdriver.



TS27-01 Computer Calibrator
Figure 214

(2) Computer Functional Check

NOTE: SAS Computer Ground Set-Up procedures, section **2.**, D., (3), must be accomplished before the computer functional check. If set-up procedures are not accomplished FIRST, the values called for in this procedure may be erroneous.

(a) Remove SAS computer from aircraft.

(b) Connect computer to TS27-01 SAS COMPUTER CALIBRATOR.

(c) Connect voltmeter to VOLT METER terminals.

(d) Connect milliammeter to MA METER terminals.

(e) Connect a SAS Indicator to IND CHECK terminals.

(f) Connect power supply to 28 VDC POWER terminals and energize.

(g) Turn VANE VOLTS SET slowly counterclockwise until STALL WARN light illuminates. Continue turning counterclockwise until mA jumps up to approximately 55 mA.

(h) Push and hold AIRSPEED switch to ABOVE. Milliammeter should drop to 0 mA, and STALL WARN light should extinguish. Release switch.

(i) Push and hold SQUAT switch to GROUND. Milliammeter should drop to 0 mA, and STALL WARN light should extinguish. Release switch.

(j) Turn VANE VOLTS SET counterclockwise until COMP FAIL light illuminates, Milliammeter should drop to a lower indication.

(k) Turn VOLTS SELECT switch to IND VOLTS.

(l) Turn VANE VOLTS SET clockwise until voltmeter reads approximately 2.00 volts. Pointer of SAS Indicator should be within the yellow SLOW area.

(m) On SA226-T and SA226-T(B) airplane models, push and hold POWER COMP switch to HIGH. Voltmeter should decrease, and pointer of SAS Indicator should move counterclockwise. Release switch.

(n) Push and hold COMP PWR switch to TEST. COMP FAIL light should illuminate, milliammeter should drop to 0 mA, voltmeter should drop to 0 volts, and pointer of SAS Indicator should go counterclockwise to 1.0 V/Vs. Release switch.

(o) Push SERVO PWR switch to TEST, SERVO FAIL light should illuminate. Release switch.

(3) Computer Ground Set-Up using TS27-01 Calibrator:

(a) Use SAS DATA sheet and SAS Computer Calibration graph from Aircraft Log, or from previous set-up.

(b) Remove end plate from computer, and install adjustment location plate.

(c) Computers with VANE ALIGN pot only:

1 Turn VOLTS SELECT switch to VANE VOLTS.

2 Turn VANE VOLTS SET knob so voltmeter reads 12 ± 0.01 VDC.

3 Turn VOLTS SELECT switch to VANE SIGNAL.

4 Adjust computer VANE ALIGN pot so voltmeter reads 12 ± 0.01 VDC.

NOTE: The following steps apply to and correlate with the steps in SAS Data Sheet, see Figure 206. Accomplish steps 1, 2, 7, and 8, then steps 3, 4, 5, and 6. Repeat steps 1 through 8 a second time to verify settings.

All values in column A and B are ±0.01 VDC. All values in column D are ± 0.40 VDC. All values in column E are ± 1.0 mA for P/N 27-44049 servo and + 5.0 - 0.0 mA for P/N 27-19023 servo.

Voltages in SAS Data Sheet, column B are taken from the original SAS Data Sheet supplied with the airplane at delivery. If the original data sheet numbers are lost or otherwise not available, perform a SAS In-Flight Calibration, see Figure 215. Extract voltages after In-Flight Set-Up and enter those voltages in SAS Data Sheet, column B.

(d) Step 1 on SAS Data Sheet:

1 Turn VOLTS SELECT switch to FLAP VOLTS. Turn FLAP VOLTS SET so voltmeter reads column A, SAS Data Sheet.

2 Turn VOLTS SELECT switch to VANE VOLTS. Turn VANE VOLTS SET so voltmeter reads column B, SAS Data Sheet.

3 Turn VOLTS SELECT switch to IND VOLTS.

4 Adjust computer potentiometer called out in column C to obtain IND VOLTS column D, SAS Data Sheet.

5 Milliammeter reading, column E, of SAS Data Sheet, does not apply for steps 1 through 8.

6 Aircraft SAS Indicator should read within a 1/2 needle width of V/Vs in column F, SAS Data Sheet. Lightly tap indicator to settle needle.

(e) STEP 2 on SAS Data Sheet:

1 STEP 2 procedures are identical to STEP 1. Comply with STEP 1, sub-steps 1 through 6.

(f) STEP 3 on SAS Data Sheet:

NOTE: STEP 3 procedures do not apply to 54302-4 or -5 computers.

1 STEP 3 procedures are identical to STEP 1. Comply with STEP 1, sub-steps 1 through 6.

(g) STEP 4 on SAS Data Sheet:

1 STEP 4 procedures are identical to STEP 1. Comply with STEP 1, sub-steps 1 through 6.

(h) STEP 5 on SAS Data Sheet:

NOTE: STEP 5 procedures do not apply to 54302-4 or -5 computers.

1 STEP 5 procedures are identical to STEP 1. Comply with STEP 1, sub-steps 1 through 6.

(i) STEP 6 on SAS Data Sheet:

1 STEP 6 procedures are identical to STEP 1. Comply with STEP 1, sub-steps 1 through 6.

(j) STEP 7 on SAS Data Sheet:

1 STEP 7 procedures are identical to STEP 1. Comply with STEP 1, sub-steps 1 through 6.

(k) STEP 8 on SAS Data Sheet:

1 STEP 8 procedures are identical to STEP 1. Comply with STEP 1, sub-steps 1 through 6.

(l) STEP 9 on SAS Data Sheet:

1 With VOLTS SELECT switch at IND VOLTS, turn FLAP VOLTS SET slowly clockwise while watching voltmeter and SAS indicator. See columns D and F.

2 Do not touch VANE VOLTS SET during this step. VANE VOLTS SET should still be set from previous steps.

3 Do not adjust any computer potentiometers during this step.

4 When step 1 is performed, IND VOLTS should increase in three (3) jumps from actual voltage reading called out in STEP 8 column D, SAS Data Sheet. Each increase (jump) should be at least 1.0 VDC from previous reading.

5 Milliammeter reading do not apply to STEP 9.

6 SAS indicator should read V/Vs in column F, SAS Data Sheet. Tap indicator lightly if necessary.

(m) STEP 10 on SAS Data Sheet:

1 Turn VOLTS SELECT switch to FLAPS VOLTS. Turn FLAP VOLTS SET so voltmeter reads column A, SAS Data Sheet.

2 Turn VOLTS SELECT switch to VANE VOLTS. Turn VANE VOLTS SET knob slowly counterclockwise while watching voltmeter and STALL WARN light. STALL WARN light should illuminate as voltage decreases to VANE VOLTS called out in column B, SAS Data Sheet. If not, adjust IAW next step.

3 Locate computer WARN HORN adjustment potentiometer. Adjust to make STALL WARN light illuminate as VANE VOLTS decrease to column B, SAS Data Sheet.

4 Milliammeter readings do not apply to STEP 10.

5 SAS indicator should read V/Vs in column F, SAS Data Sheet, when STALL WARN light illuminates. Tap indicator lightly to settle needle.

(n) STEP 11 on SAS Data Sheet:

1 Turn VOLTS SELECT switch to FLAP VOLTS. Turn FLAP VOLTS SET so voltmeter reads column A, SAS Data Sheet.

2 Turn VOLTS SELECT switch to VANE VOLTS. Turn VANE VOLTS SET slowly counterclockwise while watching voltmeter and milliammeter. Milliammeter should jump (increase) as voltage decreases to VANE VOLTS called out in column B, SAS Data Sheet. If not, adjust IAW next step.

NOTE: Indicator voltage must be increased by approximately 2.00 volts before DC MILLIAMPERES meter will reset.

3 Locate computer PUSHER ONSET adjustment potentiometer. Adjust to make milliammeter jump (increase) to approximately 55 mA when VANE VOLTS decrease to column B, SAS Data Sheet. Locate computer PUSHER CURRENT adjustment potentiometer. Adjust to make milliammeter indicate 55 mA.

4 SAS indicator should read V/Vs in column F, SAS Data Sheet, when milliammeter increases to value called out in column E, SAS Data Sheet. Lightly tap indicator to settle needle.

(o) STEP 12 on SAS Data Sheet (54302-2 and -3 Red Dot Computer):

NOTE: Steps 12 and 13 do not apply to 54302-4 or 54302-5 computers. For 54302-4 and -5 computers, proceed to STEP 14.

1 Turn VOLTS SELECT switch to FLAP VOLTS. Turn FLAP VOLTS SET so voltmeter indicates column A, ± 0.05 volts.

2 Turn VOLTS SELECT switch to VANE VOLTS. Turn VANE VOLTS SET knob so voltmeter reads 12 VDC.

3 Turn VOLTS SELECT switch to IND VOLTS. Turn VANE VOLTS SET slowly until voltmeter reads 6.04 VDC while watching milliammeter.

4 Milliammeter should increase to approximately 25 mA and stop increasing when voltmeter reaches 6.04 VDC. Milliammeter should read column E, SAS Data Sheet. If not, adjust IAW next step.

5 Locate computer RAMP/CURRENT adjustment potentiometer.

6 SAS indicator should read V/Vs in column F, SAS Data Sheet, when voltmeter reads 6.04 VDC. Lightly tap indicator to settle needle.

(p) STEP 13 on SAS Data Sheet (54302-2 and -3 Red Dot Computer:

1 Turn VOLTS SELECT switch to FLAP VOLTS. Turn FLAP VOLTS SET so voltmeter reads column A, SAS Data Sheet.

2 Turn VOLTS SELECT switch to IND VOLTS. Turn VANE VOLTS SET so voltmeter indicates 11.30 VDC.

3 Milliammeter should read column E, SAS Data Sheet. If not, adjust IAW next step.

4 Locate computer RAMP ONSET adjustment potentiometer. Adjust to make milliammeter read column E, SAS Data Sheet, when IND VOLTS are 11.30 VDC.

5 SAS Indicator should read V/Vs in column F, SAS Data Sheet. Lightly tap on indicator to settle needle.

(q) STEP 14 on SAS Data Sheet:

1 Turn VOLTS SELECT switch to FLAP VOLTS. Turn FLAP VOLTS SET so voltmeter reads column A, SAS Data Sheet.

2 Vane volts do not apply to STEP 14.

3 Locate computer CRUISE/TEST adjustment potentiometer.

4 Turn VOLTS SELECT switch to IND VOLTS. Place TEST switch to CRUISE and hold. Adjust CRUISE TEST potentiometer so voltmeter indicates 4.59 VDC. Release switch.

5 Milliammeter reading do not apply to STEP 14.

6 SAS Indicator should read V/Vs in column F, SAS Data Sheet. Tap indicator lightly to settle needle.

(r) STEP 15 on SAS Data Sheet:

1 Turn VOLTS SELECT switch to FLAP VOLTS. Turn FLAP VOLTS SET so voltmeter reads column A, plus or minus 0.05 volts.

2 Turn VOLTS SELECT switch to VANE VOLTS. Turn VANE VOLTS SET so STALL WARN light just illuminates. Voltmeter should read column B, plus or minus 0.01 volts.

3 Do not adjust any computer potentiometers during step 15.

4 Milliammeter reading do not apply to Step 15.

5 SAS Indicator should read V/Vs in column F, SAS Data Sheet. Lightly tap on indicator to settle needle.

(s) STEP 16 on SAS Data Sheet:

1 Turn VOLTS SELECT switch to FLAP VOLTS. Turn FLAP VOLTS SET so voltmeter reads column A, SAS Data Sheet.

2 Turn VOLTS SELECT to VANE VOLTS. Turn VANE VOLTS SET so milliammeter jumps (increases) to 55 mA. Voltmeter should read column B, SAS Data Sheet.

3 Do not adjust any computer potentiometers during step 16.

4 Milliammeter should read 55 mA.

5 SAS Indicator should read V/Vs in column F, SAS Data Sheet. Lightly tap indicator to settle needle.

(t) STEP 17 on SAS Data Sheet:

1 Turn VOLTS SELECT switch to FLAP VOLTS. Turn FLAP VOLTS SET so voltmeter indicates column A, SAS Data Sheet.

2 Turn VOLTS SELECT switch to VANE SIGNAL. Turn VANE VOLTS SET so voltmeter reads VDC in column B, SAS Data Sheet.

3 Do not adjust any computer potentiometers during step 17.

4 Turn VOLTS SELECT switch to IND VOLTS. Voltmeter should read column D, SAS Data Sheet.

5 Milliammeter should read column E, SAS Data Sheet.

6 SAS Indicator should read V/Vs in column F, SAS Data Sheet. Lightly tap indicator to settle needle. Read exact V/Vs and write zero degrees = 12.00 VOLTS = (this) V/Vs on computer. This V/Vs reading is to be used to check zero degree setting of AOA transmitter vane when computer is installed in aircraft.

(u) Connect SAS test box to aircraft wire harness and computer to test box. Connect voltmeter to VANE VOLTS terminals on test box.

(v) Connect GPU and hydraulic unit to aircraft and energize.

(4) Angle of Attack (AOA) Vane Voltage Set-Up:

(a) Install TF27-02 (27-95016-005) Vane Adjustment Tool to zero nutplate as shown in Figure 210.

(b) Gain access to nose baggage compartment at transmitter location just behind radome bulkhead.

(c) Loosen screws in slotted bracket.

(d) With power on the system and vane adjustment tool installed and pinned at zero, turn base of transmitter so voltmeter reads 12.0, ± 0.10 volts. Tighten screws and recheck voltage reading.

(5) Flap Compensator Voltage Set-Up:

NOTE: The SAS test box contains test jacks for measuring voltages and for troubleshooting The box contains a horn mute switch for silencing the horn when adjusting the computer and a squat switch override to activate the stick pusher and the horn when the aircraft is on the ground.

(a) Put flaps in FULL UP position.

(b) Connect voltmeter to flap volts terminals on test box. Adjust flap compensator potentiometer for 4.00, ± 0.05 volts.

(c) Put flaps in FULL DOWN. Voltmeter should read 2.90, ± 0.05 volts. Adjust compensator potentiometer if required.

(d) Repeat above **1.**, D., (5), (a) through (c) until correct readings are obtained.

(6) Perform Flap Rod Compensator Rod Adjustment:

(a) Follow procedure IAW **1.**, C., (6).

(7) Perform SAS System Functional check:

(a) Follow procedure IAW **1.**, C., (7).

(8) Perform Torque Motor Output Adjustment for 32-19023 Servo:

(a) Follow procedure IAW **1.**, C., (8).

E. 2000 Hour Recalibration Procedures – Flight Recalibration

NOTE: In-Flight Recalibration of the SAS Computer IS NOT REQUIRED when the SAS Computer has been calibrated using AEC 2000-1 or 32-82032-001 SAS Calibrator. Comply with paragraph D, 2000 Hour Recalibration Procedures – On Ground Recalibration, this section.

If original SAS Data Sheet column B values are lost or unknown, contact M7 Aerospace for a copy of original data. If original values cannot be determined, In-Flight Recalibration is the only method available to determine voltage values for the 2000 Hour Recalibration.

(1) Preflight System Check:

(a) Comply with **1.**, D., (4), Angle of Attack (AOA) Vane Voltage Setup, this section.

(b) Comply with **1.**, D., (5), SAS Flap Compensator Voltage Set Up, this section.

(c) Verify SAS vane is free from drag (no more than 10 grams of force to move vane in either direction).

NOTE: Three (3) US nickels weigh approximately ten (10) grams.

(d) Obtain Airspeed Calibration Card and have available for use during flight testing.

(e) Obtain gross weight and instrument corrected speeds of aircraft. Make note of these in appropriate SAS calibration work sheet. Aircraft should be at maximum gross takeoff weight.

CAUTION: ENGINE STARTING, OPERATION AND SHUTDOWN PROCEDURES MUST BE PERFORMED BY QUALIFIED PILOTS OR PERSONNEL IN ACCORDANCE WITH THE AIRCRAFT FLIGHT MANUAL. OBSERVE SAFETY PRECAUTIONS CALLED OUT IN CHAPTER 71.

(f) Remove SAS computer from LH F.S. 069 bulkhead or RH console as applicable. Connect SAS calibrator to SAS computer and aircraft wiring system as shown in Figure 204.

(g) Remove SAS Computer end plate and attach Computer Adjustment Locator plate, Figure 203.

(h) Start one engine. See AFM for start procedures. Bring generator on line.

(i) Engage all SAS circuit breakers (Figure 209). Turn SAS CLUTCH/SERVO switch OFF.

(j) Verify the following indications on the annunciator panel:

1 SAS ARM light illuminated.

2 SAS FAULT light blinking.

3 Turn the SAS CLUTCH/SERVO switch ON, note SAS FAULT light extinguished.

(k) Verify flaps FULL UP.

(l) Hold SAS TEST switch to CRUISE.

(m) Note pointer of SAS indicator reads 1.3 V/Vs precisely.

(n) While holding SAS TEST switch to CRUISE:

1 Move flaps FULL DOWN.

2 Note pointer of SAS indicator travels counterclockwise.

3 Needle should jump about a needle width in direction of movement as flaps pass between 1/2 and 3/4.

4 Verify pointer of SAS indicator reads. less than 1.2 V/Vs.

5 Move flaps FULL UP. Verify pointer of SAS indicator returns to 1.3 V/Vs

(o) Release SAS TEST switch.

(p) On SA226 AT/TC aircraft, hold the SAS TEST switch to STALL. Observe the following on SAS indicator and elevator control:

1 The pointer of the SAS indicator travels counterclockwise to 1.00 V/Vs.

2 The horn sounds and pusher engages.

(q) Release SAS TEST switch.

(r) On SA226 T/T(B) aircraft, hold the SAS TEST switch to STALL. Observe the following on SAS indicator and elevator control:

1 SAS indicator pointer (depending on vane position) descends to, or starts from, left side of CRUISE band and sweeps counterclockwise.

2 As pointer sweeps through CRUISE band, a gradually increasing forward force is felt on the elevator control.

3 Force should reach its peak when indicator pointer reaches 1.4 V/Vs. Force should remain same until pusher activation. Peak force should be approximately 20 pounds (9.07 kg).

4 As SAS indicator pointer leaves SLOW area and enters red and black crosshatch area, STALL WARN HORN should sound.

5 As SAS indicator pointer leaves red and black crosshatch area and enters STALL area pusher should engage.

6 Annunciator SAS FAULT light should illuminate as SAS indicator pointer passes mid range of STALL area.

7 Pointer should travel to left side of STALL area and delay for a few seconds.

8 Pointer should return to APP area (just below 1.3 V/Vs).

9 As pointer returns, annunciator SAS FAULT light should extinguish and pusher and horn stop sounding.

(s) Move SAS vane until SAS indicator pointer is in mid range of SLOW area.

(t) Move LH power lever to flight idle.

(u) Slowly advance LH power lever above flight idle. SAS indicator pointer should jump counterclockwise approximately 1/8 inch when power lever has traveled approximately 1/8 inch forward of flight idle gate.

(v) Move power lever back to flight idle gate and SAS pointer should return clockwise to previous position.

(w) Release SAS TEST switch.

(2) In-Flight Check and Adjustments (Figures 203, 215, and 216).

WARNING: DO NOT PERMIT AIRCRAFT TO ENTER FULL STALL. TERMINATE TESTS AT THE LISTED STALL SPEEDS OR AT FIRST INDICATION OF EXCESSIVE WING HEAVINESS OR BREAK.

ALL PERTINENT FAA REGULATIONS SHOULD BE OBSERVED WHILE PERFORMING THE MANEUVERS REQUIRED IN SAS SETUP. IN ADDITION, THE FOLLOWING MANEUVERS PLACE THE AIRCRAFT IN A DESCENDING CONFIGURATION AND AMPLE ALTITUDE SHOULD BE ALLOWED FOR PROPER RECOVERY.

DO NOT TEST DURING TURBULENT CONDITIONS.

PRIOR TO THE FIRST CALIBRATION FLIGHT, PLACE THE SAS CLUTCH IN THE OFF POSITION DURING TAKEOFF AND INITIAL CLIMB OUT.

(a) Refer to Aircraft Flight Manual to determine if ballast is required to ensure mid to forward C/G.

(b) Set RPM levers at 100%.

(c) Using In-Flight Set-Up Sheet, (Figures 215 or 216), trim to zero stick force at the trim speed listed in step A through 11. Do not re-trim within a test. Achieve test speeds by elevator control.

NOTE: Use current Airspeed Calibration Card to determine airspeed indicator correction factors.

(d) Approach each test speed from appropriate trim speed with a speed bleed-off rate of one knot per second. Otherwise, incorrect stall, stick pusher, and stall warning values will result.

(e) Computer adjustments must be accomplished in the order they appear on the In-Flight Set-Up sheet, (Figures 215 or 216). If the adjustments are not accomplished in correct order, the previous adjustments will be affected.

NOTE: Required speeds are tabulated as functions of airplane weight requiring that the weight be calculated for each test. This requires fuel accountability throughout the flight test program.

STEP	SAS CLUTCH SWITCH	GEAR/ FLAPS	POWER LEVERS	AIRSPEED IND. CORR KIAS	AJDUST COMP. POT	TO OBTAIN	REMARKS
A	OFF	UP	F/1	V_{S1}	N/A	N/A	Trim to 1.5 V_{S1}. Decrease 1 knot/Sec to stall. Check aerodynamic stall speed equal to or less than V_{S1}.
B	OFF	FULL DOWN	F/1	V_{SO}	N/A	N/A	Trim to 1.5 V_{SO}. Decrease 1 knot/Sec to stall. Check aerodynamic stall speed equal to or less than V_{SO}.
1	ON	UP	AS REQ'D	APPROX. 183 KNOTS	RAMP ONSET	START OF RAMP	Airspeed to just above SAS arm speed (SAS ARM light on) Decrease to below SAS arm speed. Check no bump (kick) on elevator control. Adjust RAMP ON-SET for no bump. Gross weight must be 12,500 to 12,000 pounds. Pointer to 1.8 V/V_S.
2≥	OFF	UP	F/1	V_{S1}	0 STALL	1.0 V/V_S	Trim to approx. 1.3 V_{S1}. Decrease 1 knot/Sec to V_{S1} check that pointer is at 1.0 V/V_S. Fine adjust 0 STALL if required.
3	OFF	UP	F/1	1.3 V_{S1}	0 CRUISE	1.3 V/V_S	Trim to approx. 1.3 V_{S1} hands off. Pointer to 1.3 V/V_S precisely.
4	OFF	FULL DOWN	F/1	V_{SO}	FULL STALL	1.0 V/V_S	Trim to approx. 1.3 V_{SO}. Decrease 1 knot/Sec. to V_{SO} Check that pointer is at 1.0 V/V_S. Fine adjust FULL STALL is required.
5	OFF	FULL DOWN	F/1	1.1 V_{SO}	FULL STALL	V_{SO} 1.3 V/V_S	Trim to 1.3 V_{SO} hands off. Pointer to 1.3 V/V_S precisely.
6	ON	UP	F/1	V_{S1} PLUS 4 MINUS 1 KNOT	PUSHER ONSET	PUSHER AT APPROX. 1,02 V/V_S	Trim to 1.5 V_{S1}. Decrease 1 knot/Sec to pusher. pointer at approx. 1.02 V/V_S.
7	ON	UP	F/1	N/A	WARN HORN	HORN AT 1.1 V/V_S	Fly at 1.1 V/V_S. Adjust horn off then on. Pointer at 1.1 V/V_S.
8	ON	FULL DOWN	F/1	1.1 VSO	FULL CRUISE	HORN AT 1.1 V/V_S	Trim to 1.5 V_{SO}. Decrease 1 knot/Sec. to horn and pusher.
				V_{SO} PLUS 4 MINUS 1	PUSHER ONSET	APPROX. PUSHER AT 1.02 V/V_S	Fine adjust WARN HORN and PUSHER ONSET if re-quired.
9	ON	FULL DOWN	F/1	RECORD	N/A	N/A	Trim to 1.5 V_{SO}. Decrease 1 knot/Sec. to horn and pusher. Record airspeed for horn and pusher. Pusher must be V_{SO} plus 4 minus 1 knot. Horn must be 5 to 10 knots above pusher speed.
10	ON	UP	F/1	RECORD	N/A	N/A	Trim to 1.5 V_{SI}. Decrease 1 knot/Sec. to horn and pusher. Record airspeed for horn and pusher. Pusher must be V_{SI} plus 4 minus 0 knot. Horn must be 10 knots above pusher speed.
11	ON	UP	N/A	N/A	CRUISE TEST	1.3 V/V_S	Hold SAS TEST switch to CRUISE. Adjust pointer to 1.3 V/V_{SI} precisely.

6M273206215

In-Flight Set-Up
SA226-T/T(B)
Figure 215

STEP	SAS CLUTCH SWITCH	GEAR/ FLAPS	POWER LEVERS	AIRSPEED IND. CORR KIAS	AJDUST COMP. POT	TO OBTAIN	REMARKS
A	OFF	UP	F/1	V_{S1}	N/A	N/A	Trim to 1.5 V_{S1}. Decrease 1 knot/Sec to stall. Check aerodynamic stall speed equal to or less than V_{S1}.
B	OFF	FULL DOWN	F/1	V_{SO}	N/A	N/A	Trim to 1.5 V_{SO}. Decrease 1 knot/Sec to stall. Check aerodynamic stall speed equal to or less than V_{SO}.
1	OFF	UP	F/1	V_{S1}	0 STALL	1.0 V/V_S	Trim to approx. 1.3 V_{S1}. Decrease 1 knot/Sec to V_{S1} check that pointer is at 1.0 V/V_S. Fine adjust 0 STALL if required.
2	OFF	UP	F/1	1.3 V_{S1}	0 CRUISE	1.3 V/V_S	Trim to approx. 1.3 VS1 hands off. Pointer to 1.3 V/V_S precisely.
3	OFF	FULL DOWN	F/1	V_{SO}	FULL STALL	1.0 V/V_S	Trim to approx. 1.3 V_{SO}. Decrease 1 knot/Sec. to V_{SO} Check that pointer is at 1.0 V/V_S. Fine adjust FULL STALL is required.
4	OFF	FULL DOWN	F/1	1.1 V_{SO}	FULL STALL	V_{SO} 1.3 V/V_S	Trim to 1.3 V_{SO} hands off. Pointer to 1.3 V/V_S precisely.
5	ON	UP	F/1	VS1 PLUS 4 MINUS 1 KNOT	PUSHER ONSET	PUSHER AT APPROX. 1,02 V/V_S	Trim to 1.5 V_{S1}. Decrease 1 knot/Sec to pusher. pointer at approx. 1.02 V/V_S.
6	ON	UP	F/1	N/A	WARN HORN	HORN AT 1.1 V/V_S	Fly at 1.1 V/V_S. Adjust horn off then on. Pointer at 1.01 V/V_S.
7	ON	FULL DOWN	F/1	1.1 V_{SO}	FULL CRUISE	HORN AT 1.1 V/V_S	Trim to 1.5 V_{SO}. Decrease 1 knot/Sec. to horn and pusher.
				V_{SO} PLUS 4 MINUS 1	PUSHER ONSET	APPROX. PUSHER AT 1.02 V/V_S	Fine adjust WARN HORN and PUSHER ONSET if re-quired.
8	ON	FULL DOWN	F/1	RECORD	N/A	N/A	Trim to 1.5 V_{SO}. Decrease 1 knot/Sec. to horn and pusher. Record airspeed for horn and pusher. Pusher must be V_{SO} plus 4 minus 1 knot. Horn must be 5 to 10 knots above pusher speed.
9	ON	UP	F/1	RECORD	N/A	N/A	Trim to 1.5 V_{SI}. Decrease 1 knot/Sec. to horn and pusher. Record airspeed for horn and pusher. Pusher must be V_{SI} plus 4 minus 0 knot. Horn must be 10 knots above pusher speed.
10	ON	UP	N/A	N/A	CRUISE TEST	1.3 V/V_S	Hold SAS TEST switch to CRUISE. Adjust pointer to 1.3 V/V_{SI} precisely.

6M273206216

In-Flight Set-Up
SA226-AT/TC
Figure 216

GROSS WEIGHT:	POUNDS	
CLEAN:	STALL WARNING	_____ KIAS
FLIGHT IDLE POWER	STICK PUSHER	_____ KIAS
	AFM/Vs1	_____ KIAS
CLEAN:	STALL WARNING	_____ KIAS
55% TORQUE	STICK PUSHER	_____ KIAS
DIRTY:	STALL WARNING	_____ KIAS
FLIGHT IDLE POWER	STICK PUSHER	_____ KIAS
	AFM/Vso	_____ KIAS
DIRTY:	STALL WARNING	_____ KIAS
55% TORQUE	STICK PUSHER	_____ KIAS

NOTES: STALL WARNING HORN AT 5 TO 10 KIAS ABOVE STICK PUSHER AT FLIGHT IDLE POWER. 5 KIAS MINIMUM POWER ON. STICK PUSHER AT AFM/Vs + 4, -0 KIAS.

6M273206217

Stall Warning Horn and Stick Pusher Speeds
Figure 217

(f) If the aircraft stalls at a higher speed than the listed test speed, the airspeed indication system is faulty, the wrong bleed-off rate was used, or the aircraft is heavier (i.e., fuel load).

(g) When adjusting computer potentiometers, to move SAS indicator pointer clockwise, turn potentiometer clockwise. To move SAS indicator pointer counterclockwise, turn potentiometer counterclockwise.

(h) On SA226 T/T(B) aircraft only; Step A, In-Flight Set-Up Sheet, Figure 215, can be performed during climb to altitude and should be performed at a gross weight as close to MTOW pounds as possible.

(i) After completion of In-Flight Set-Up, record stall warning and stick pusher speeds in Stall Warning Horn and Stick Pusher Speeds chart, Figure 217.

(j) After calibration, remove adjustment location plate and install end cover plate on computer.

(k) Disconnect SAS calibrator wiring and install SAS computer in RH console.

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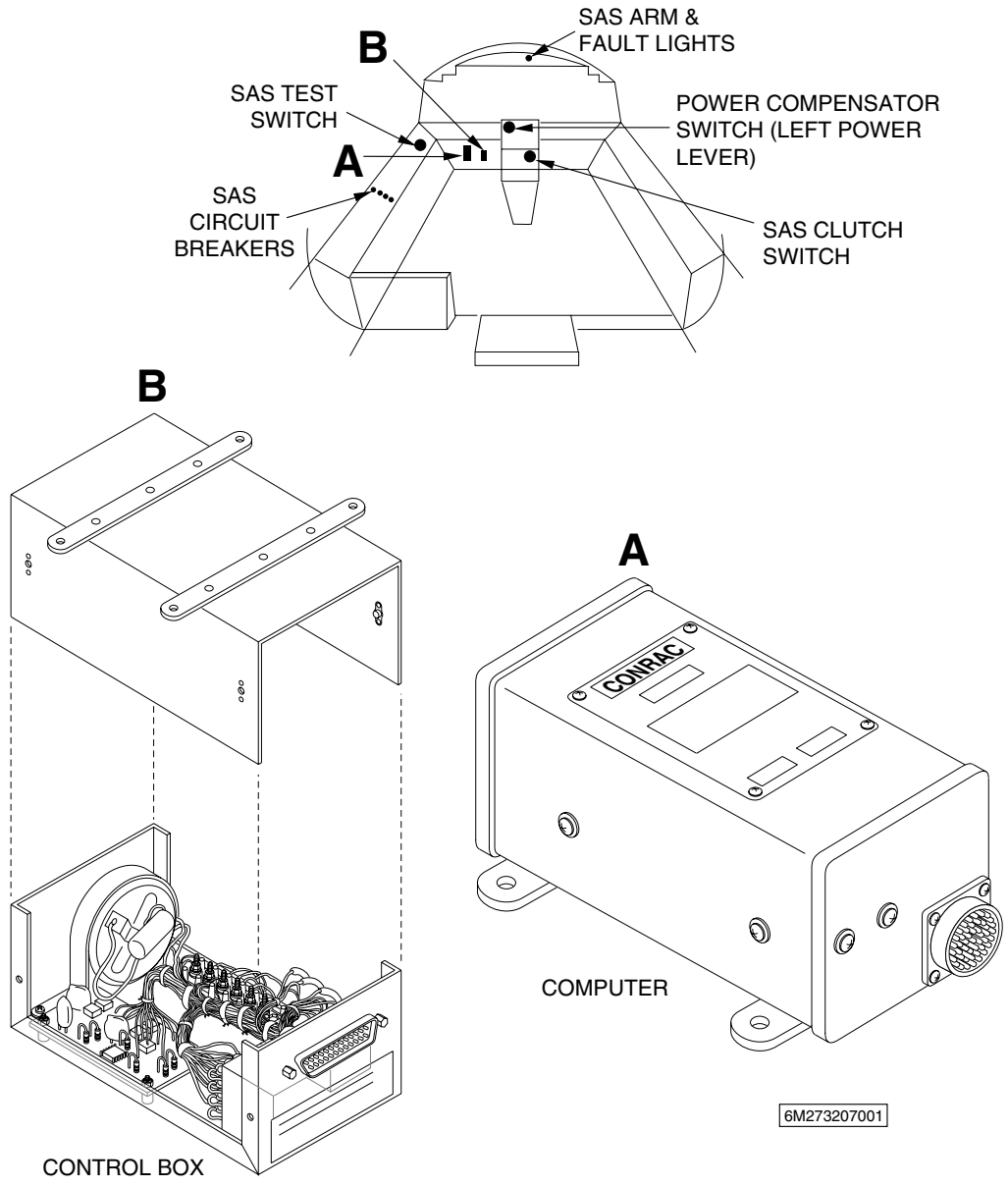
SAS COMPUTER AND CONTROL BOX – DESCRIPTION AND OPERATION

1. General

A. The conrac SAS computer is located forward of the pilot left rudder pedal on the forward pressure bulkhead. The computer receives input information from various components of the system. These inputs are processed by the computer to usable information for its various outputs. The computer contains a number of adjustable potentiometers (trimmer pots) to adjust the output information for calibration of the system.



Computer and Control Box
Figure 1

SAS COMPUTER AND CONTROL BOX – DESCRIPTION AND OPERATION

B. The SAS control box, located just to the right of the computer, interfaces some of the computers inputs and outputs. The control box also consolidates some components of the system to one location.

C. The computer inputs are: Angle of Attack (AOA) information from AOA transmitter, flap position information from the flap compensator, aircraft on the ground information from strut compression switch on the left main gear, aircraft below SAS arm speed from the airspeed switch, left engine power lever from a switch actuated by the left power lever, computer self test from the SAS TEST switch on the left console, stick pusher servo motor power monitor from the positive side of the motor, stability augmentation rate of change (known as delta dot from a rate generator within the stick pusher servo), and 28 VDC power from the COMP PWR circuit breaker on the left essential bus. The computer monitors itself for input power, regulated power, and the AOA transmitter pot for loss of continuity.

D. The computer outputs are: a signal to the SAS indicator for V/Vs, a signal to the stick pusher servo clutch for stability augmentation and stick pusher function, a computer and AOA transmitter power fail signal, a servo power fail signal, and a stall warning signal to operate the stall warning horn.

E. The control box contains a printed circuit board (PCB), the servo relay, the stall warning horn and relay, the power compensation relay, and the vane heater relay. Components on the PCB interface the signal from the airspeed switch to the computer and operate the servo relay. Other components on the PCB make the SAS FAULT light illuminate steadily when they receive a computer or vane fail signal from the computer or flash when they receive a servo fail signal from the computer or the SAS servo idle control.

SAS COMPUTER AND CONTROL BOX – MAINTENANCE PRACTICES

1. Servicing – SAS Computer and Control Box

A. No regularly scheduled maintenance is required on the computer or the control box. During regularly scheduled inspections the computer and control box should be checked for condition and security. Replacement of the computer requires a functional check, ground calibration, and flight calibration of the system (see Section 27-32-06). Replacement of control box requires system functional check (see Section 27-30-12). and AOA transmitter deice check (see Section 30-12-02).

2. Removal/Installation – SAS Computer and Control Box

A. Removal – Computer

(1) Ensure battery switches are off.

(2) Disconnect electrical connector from computer.

(3) Remove screws securing computer to equipment plate on forward pressure bulkhead.

(4) Remove computer from location.

B. Installation – Computer

(1) After ground calibration, flight calibration, and extracting number of voltages, remove SAS test box and connect computer to aircraft wiring.

(2) Move flaps UP. Turn battery switch ON and hold SAS TEST switch to CRUISE. Pointer of SAS indicator must be precisely 1.3 V/Vs. If not refer to Section 27-32-06.

(3) Remove adjustment location plate and install end plate on computer.

(4) Position computer on equipment plate on forward pressure bulkhead and secure with screws.

(5) Connect electrical connector.

C. Removal – Control Box

(1) Remove screws securing box to equipment plate on forward pressure bulkhead.

(2) Disconnect electrical connector.

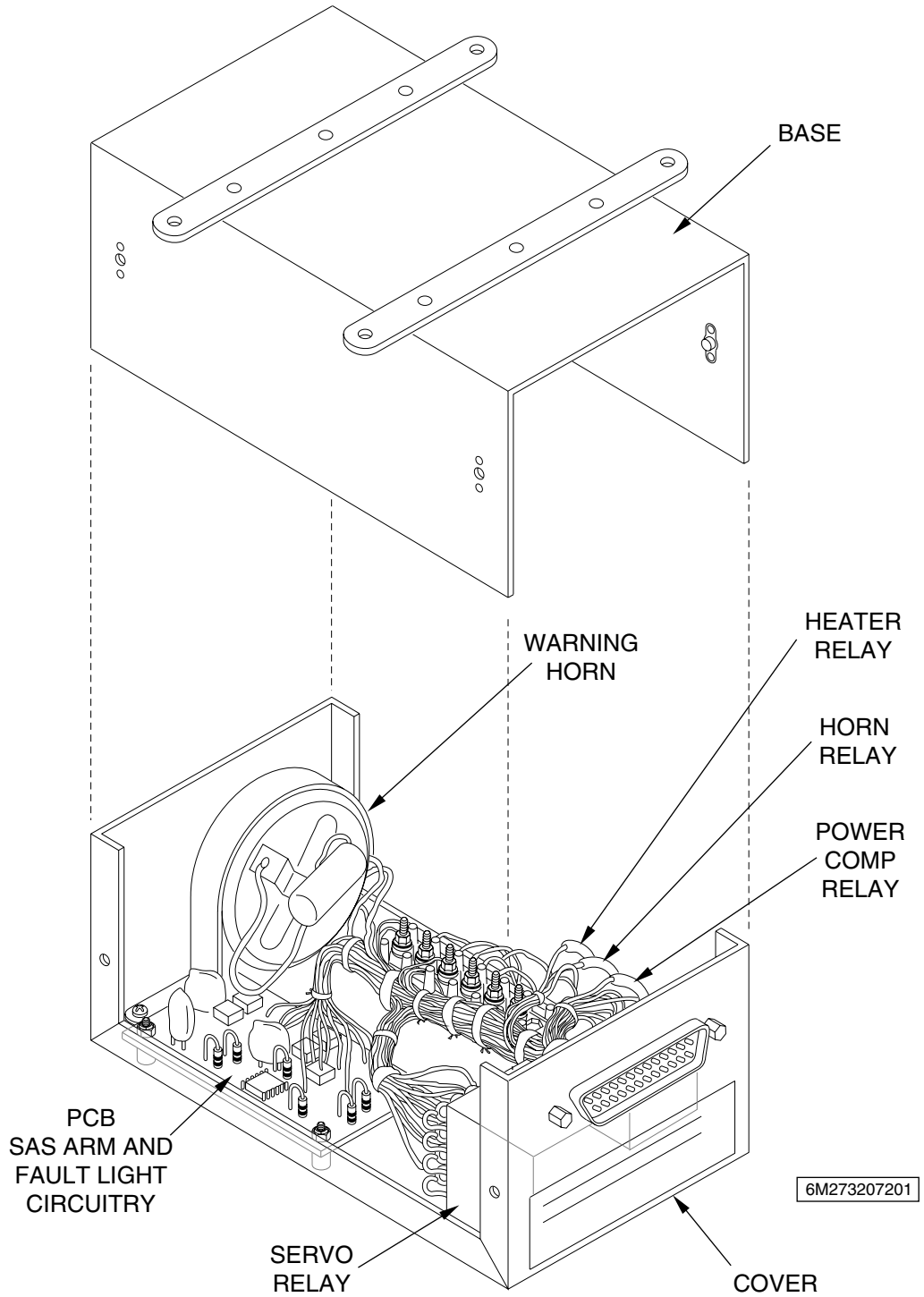
D. Installation – Control Box

(1) Connect electrical connector.

(2) Position box on equipment plate on forward end pressure bulkhead and secure with screws.

(3) Install screws.

(4) Perform SAS System Functional Check, Airspeed Switch Functional Check, and AOA Transmitter Heat Functional Check.



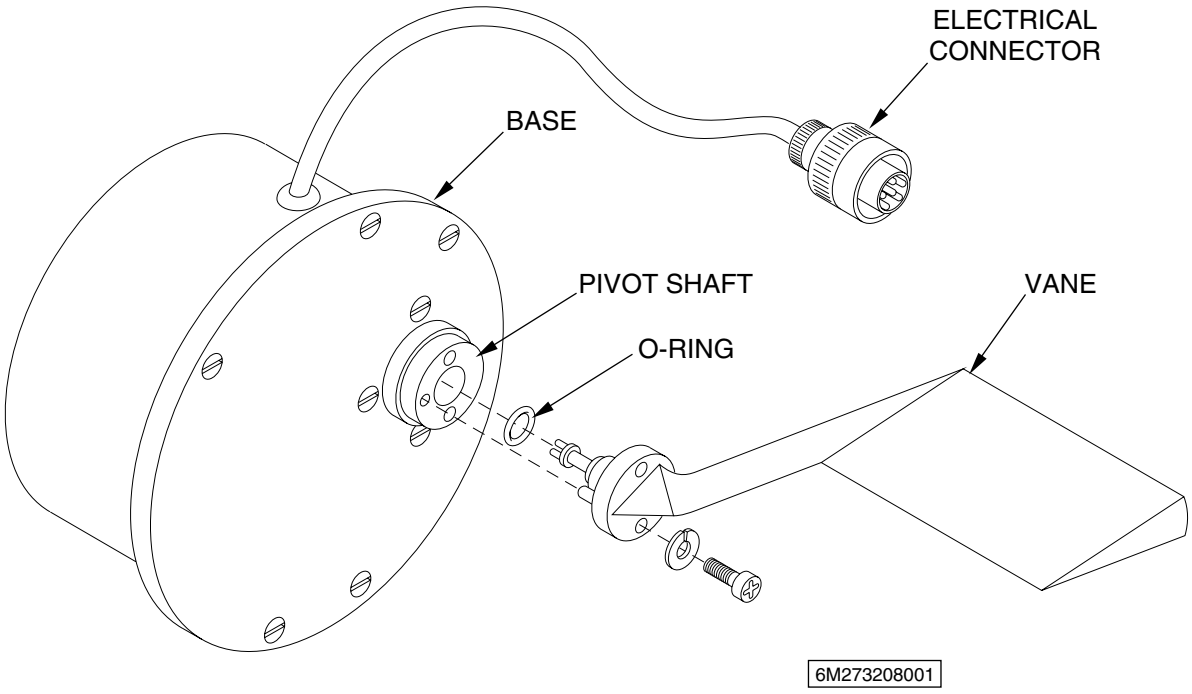
SAS Control Box
Figure 201

AOA TRANSMITTER – CONRAC – DESCRIPTION AND OPERATION

1. General

A. The Angle Of Attack (AOA) transmitter for the CONRAC SAS system is located on the nose of the aircraft just aft of the radome. The transmitter consists of a detachable vane, a base, and an 1,800 ohm potentiometer (pot). The vane is electrically deiced (see Section 30-12-09). The vane pivots on a shaft that extends into the base. The vane aligns itself to airflow over the fuselage, the vane angle from the waterline of the aircraft correlates to angle of attack. The vane is linked mechanically by the pivot shaft to a counterweight, a viscous damper, and the pot. The counterweight and the viscous damper work together to prevent overshooting and excessive response of the vane The vane has a range of 50°, 35° above the zero waterline and 15° below.

B. When the vane pivots, the wiper of the pot passes over the resistance winding. Fifteen volts is placed across the resistance winding by the computer. The low end of the resistance winding is lifted from ground five volts. When the wiper travels over the resistance winding it picks off a voltage. This voltage corresponds to angle of attack and is the basic input to the computer.



Conrac AOA Transmitter Figure 1

AOA TRANSMITTER – CONRAC – MAINTENANCE PRACTICES

1. Servicing – AOA Transmitter – CONRAC

A. Check AOA transmitter, vane, and wiring for condition and security.

2. Removal/Installation – AOA Transmitter – CONRAC

A. Removal – Vane from Transmitter Base

(1) Remove and retain screws and lockwashers at pivot shaft of vane.

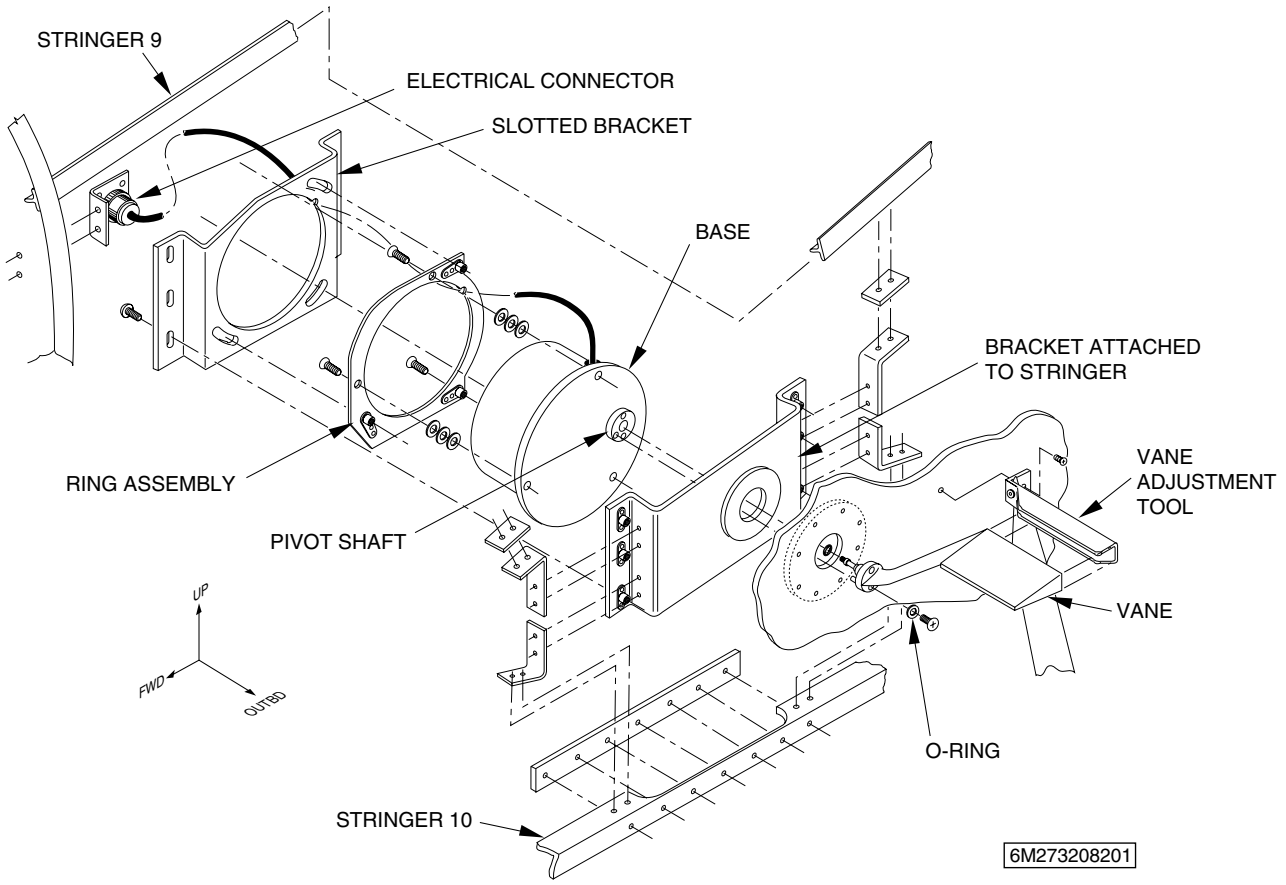
(2) Remove vane from base of transmitter by carefully separating gray vane and silver shaft of pivot point. After separation has started pull vane straight out and do not turn. Care must be taken not to bend vane deice electrical pins on the center shaft.

B. Removal – Transmitter Base from Aircraft

(1) Gain access to nose baggage compartment at transmitter location just behind radome bulkhead.

(2) Disconnect electrical connector.

(3) Remove screws attaching slotted bracket to bracket attached to stringers.



AOA Transmitter – Conrac Installation
Figure 201

(4) Remove base assembled with slotted bracket and ring assembly from location.

(5) Remove screws attaching slotted bracket to ring assembly and remove slotted bracket noting its location with respect to the ring assembly.

(6) Remove screws attaching ring assembly to base and remove ring assembly noting its location with respect to the base.

C. Installation – New AOA Transmitter

(1) Remove vane from base of transmitter.

(2) Ensure O-ring is installed (see Figure 1).

(3) Slip ring assembly over base. Position ring assembly so that cutout for wire harness is closest to the point where the wire harness exits the base, and install countersunk screws.

(4) Position slotted bracket over base and onto ring assembly. Install but do not tighten screws attaching slotted bracket to ring assembly.

(5) Position entire assembly on bracket attached to stringers with vane pivot shaft through hole in bracket and skin.

(6) Install but do not tighten screws attaching slotted bracket to other bracket.

(7) Connect electrical connector.

D. Installation – Vane to Base

(1) Position vane at pivot shaft with pin on vane aligned with hole in shaft. Push center shaft of vane straight into hole in pivot shaft. Care must be taken not to bend vane deice electrical connector pins on the center shaft.

(2) Install screws and lockwashers attaching vane to pivot shaft. Torque screws 10 to 14 inch-pounds (1.13 to 1.58 N·m).

(3) Adjust vane voltage (see AOA Transmitter Adjustment and Test).

(4) Perform SAS Functional Check.

3. Adjustment/Test – AOA Transmitter – CONRAC

A. Vane Voltage Adjustment

(1) Connect GPU to aircraft and energize.

(2) Install vane adjustment tool TF 27-02 on vane, and attach to zero nutplate using a No. 6-32 screw.

(3) Remove CONRAC SAS computer from left side of forward pressure bulkhead. Power should always be off when disconnecting or connecting electrical connector on SAS computer.

(4) Connect SAS test box TS 27-02 to aircraft wire harness and computer to test box. Connect voltmeter to VANE VOLTS terminals on test box.

NOTE: In lieu of SAS test box, voltage can be measured at vane signal test point terminal 5 to ground on TS16 right console behind aft door lower terminal strip.

(5) Gain access to nose baggage compartment at transmitter location just behind radome bulkhead.

(6) Loosen screws in slotted bracket.

(7) With power on the system and the vane adjustment tool installed and pinned at zero, turn base of transmitter so voltmeter reads 12.00 ± 0.01 volts. Tighten screws, and recheck voltage reading.

B. AOA Vane Flag Kit

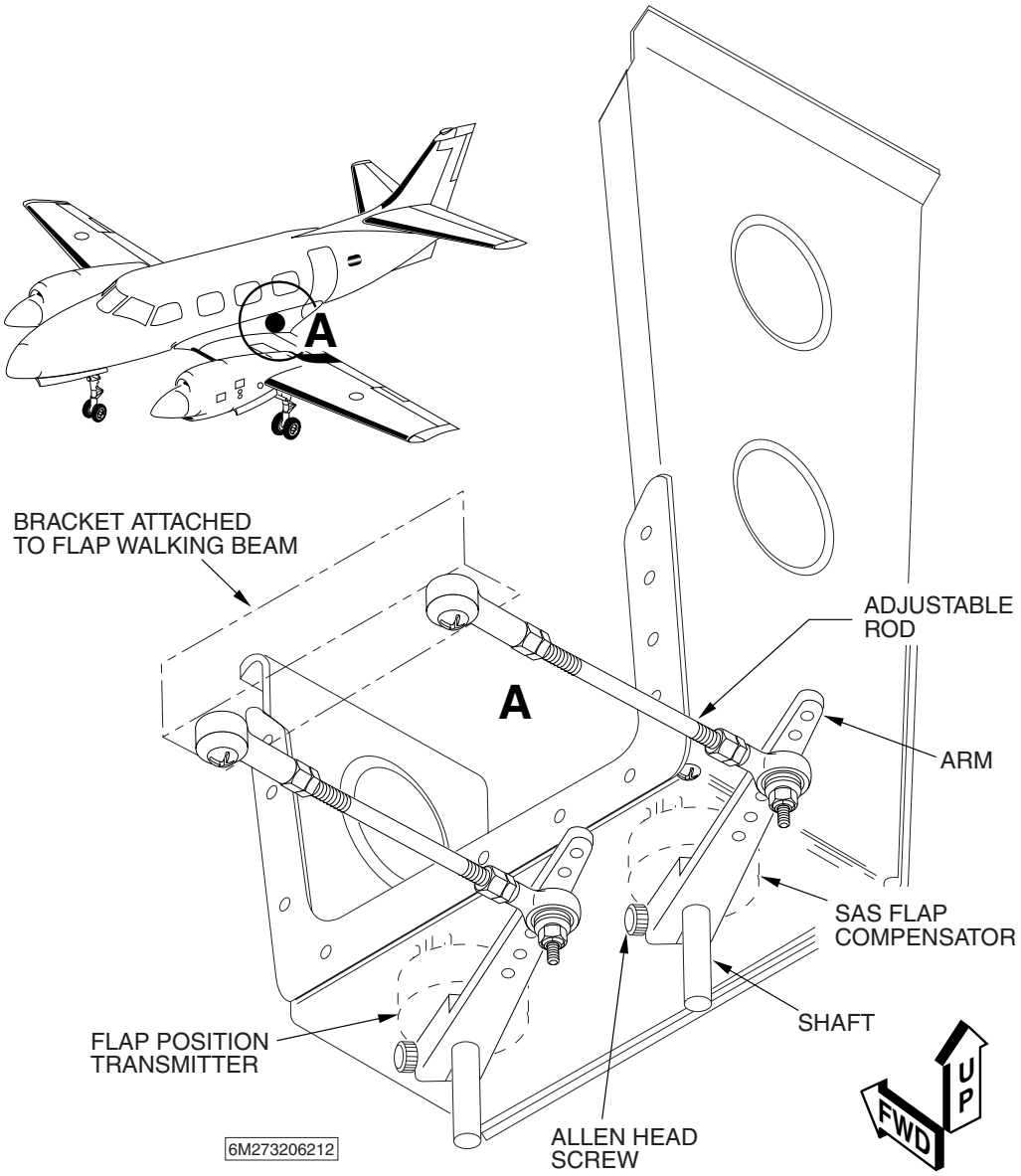
(1) When vane is replaced IAW Gulton-Statham kit number 25147A-5 kit 1, check vane for 12.00 ± 0.01 volts with vane at 0° angle. If the voltage is not 12.00 ± 0.01, perform Vane Voltage Adjustment.

MAINTENANCE MANUAL

FLAP COMPENSATOR – DESCRIPTION AND OPERATION

1. General

A. Flap position compensation for the CONRAC SAS system is accomplished by a 1,000 ohm potentiometer (pot) mechanically linked to the flaps. The pot along with another pot for the flap position indication system are mounted on a bracket just to the right of the left flap.



Flap Compensator Conrac
Figure 1

FLAP COMPENSATOR – DESCRIPTION AND OPERATION

B. An arm on the shaft of the pot is connected by an adjustable rod to the flap walking beam. When the flaps are extended the shaft turns and the wiper passes over the resistance winding of the pot. Five volts are placed across the resistance winding by the computer. When the wiper travels over the resistance winding it picks off a voltage. This voltage corresponds to flap position. The computer uses this voltage to compensate a similar voltage from the AOA transmitter pot for flap position. Circuitry within the computer compensates for four discrete flap positions: up (0), 3/8, 5/8, full down (FULL).

MAINTENANCE MANUAL

FLAP COMPENSATOR – MAINTENANCE PRACTICES

1. Servicing – Flap Compensator – CONRAC

A. Check flap compensator and mechanical linkage for condition and security.

2. Removal/Installation – Flap Compensator – CONRAC

A. Removal – Flap Compensator

(1) Remove left walking beam access panel.

(2) Flap compensator is the left outboard potentiometer.

(3) Loosen allenhead screw in arm on shaft of pot.

(4) Slide arm down and off shaft.

(5) Remove nut and lockwasher securing pot to bracket.

(6) Pull pot out of bracket and cut wires as close to pot as possible.

B. Installation – Flap Compensator

(1) Solder wires on pot IAW Figure 201.

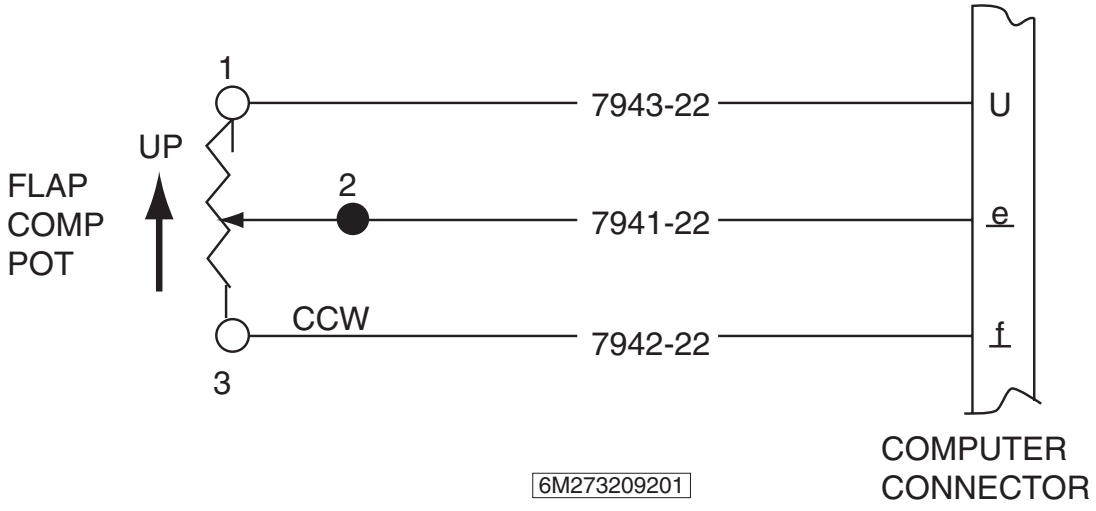
(2) Position pot on bracket with alignment pin on pot in hole in bracket.

(3) Install lockwasher and nut.

(4) Pot wire connections with 3145 Dow Corning.

(5) Slip arm on shaft and tighten allenhead screw loosely.

(6) Adjust flap voltage (see Flap Voltage Adjustment).



Flap Compensator Pot
Figure 201

FLAP COMPENSATOR – MAINTENANCE PRACTICES

C. Flap Voltage Adjustment – Flap Compensator

(1) Connect and energize GPU and hydraulic cart.

(2) Remove connector from computer and install SAS test box TS 27-02.

(3) Connect voltmeter to FLAP VOLTS terminals on test box. In lieu of SAS test box voltage can be measured at e and f at computer connector.

(4) Remove left walking beam access panel.

(5) Adjust rod attached to arm on shaft of flap compensator pot so a piece of 0.032 safetywire will not pass through check holes in rod ends.

(6) Install rod end on arm in the fourth adjustment hole from center.

(7) Move flaps down observing flap linkage for binding and obstructions.

(8) Loosen allenhead screw in arm. With power on the system, adjust shaft of pot so the voltmeter reads 2.90, ± 0.05 volts.

(9) Tighten allenhead screw and confirm voltage reading.

(10) Move flaps up and confirm flap voltage of 4.00 volts, ± 0.05 volts.

(11) Perform Flap Compensation Functional Check.

(12) Install walking beam panel.

D. Functional Check – Flap Compensation

(1) With power on the aircraft and SAS system, hold SAS TEST switch to CRUISE.

(2) Pointer of SAS indicator should be on 1.3 V/Vs with flaps up.

(3) Move flaps down.

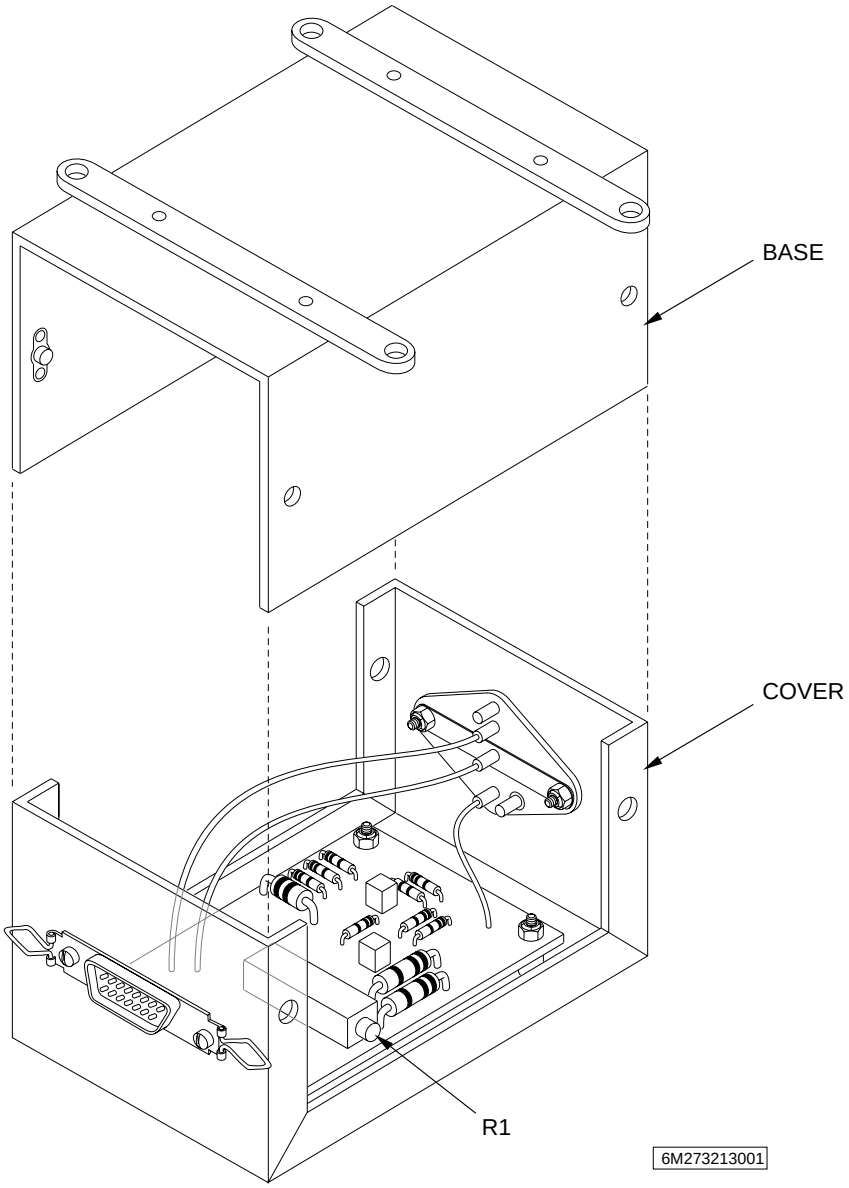
(4) As flaps travel down, the pointer of the SAS indicator should jump three times counter-clockwise. When flaps are full down, the pointer of SAS indicator should be approximately in the middle of the yellow SLOW area.

MAINTENANCE MANUAL

SAS SERVO IDLE CONTROL – DESCRIPTION AND OPERATION

1. General

A. The servo idle control regulates the stick pusher servo motor at a slower speed to keep the clutch magnetic particles from settling during prolonged flight at high speed cruise. The unit is located forward of pilots right rudder pedal on the forward pressure bulkhead. Circuitry within the box lifts the motor negative from ground which causes the motor to run at a slower rate. This rate is adjustable by a pot (R1) within the box. If the motor stops, underspeeds, or overspeeds the servo fail (blinking SAS FAULT) light illuminates.



SAS Servo Idle Control
Figure 1

SAS SERVO IDLE CONTROL – MAINTENANCE PRACTICES

1. Removal/Installation – SAS Servo Idler Control

A. Removal – Idle Control

(1) Disconnect connector.

(2) Remove screws securing box to bulkhead bracket.

(3) Remove box.

B. Installation – Idle Control

(1) Position box on bracket and secure with screws.

(2) Connect connector.

(3) Perform SAS servo idle control check.

2. Adjustment/Check – SAS Servo Idler Control

A. Adjustment/Test – SAS Servo Idle Control Check

(1) Remove screws securing cover to base.

(2) Remove cover and place in such a position as not to short-out circuitry to ground.

(3) Connect GPU to aircraft and energize.

(4) Connect pilot static tester to copilots pitot tube.

(5) Depress left main gear inboard scissor switch (six wire switch) with squat switch cheater tool.

(6 Turn battery switches ON (SAS FAULT blinking, SAS ARM on).

(7) Turn clutch switch ON (SAS FAULT out).

(8) Apply pitot pressure above 140 knots AT - TC, 180 knots T.

(9) SAS ARM light should be extinguished. Turn (R1) clockwise until the SAS FAULT light begins to flash, then turn (R1) counterclockwise one and one-half turns. The SAS FAULT light will extinguish.

(10) Decrease and increase pitot pressure above and below airspeed actuation point a few times to see that the SAS ARM light does not come on.

(11) Decrease pitot pressure and remove tester.

(12) Turn clutch switch OFF.

(13) Turn battery switches OFF.

(14) Remove squat switch cheater tool from left scissor switch.

(15) Disconnect APU.

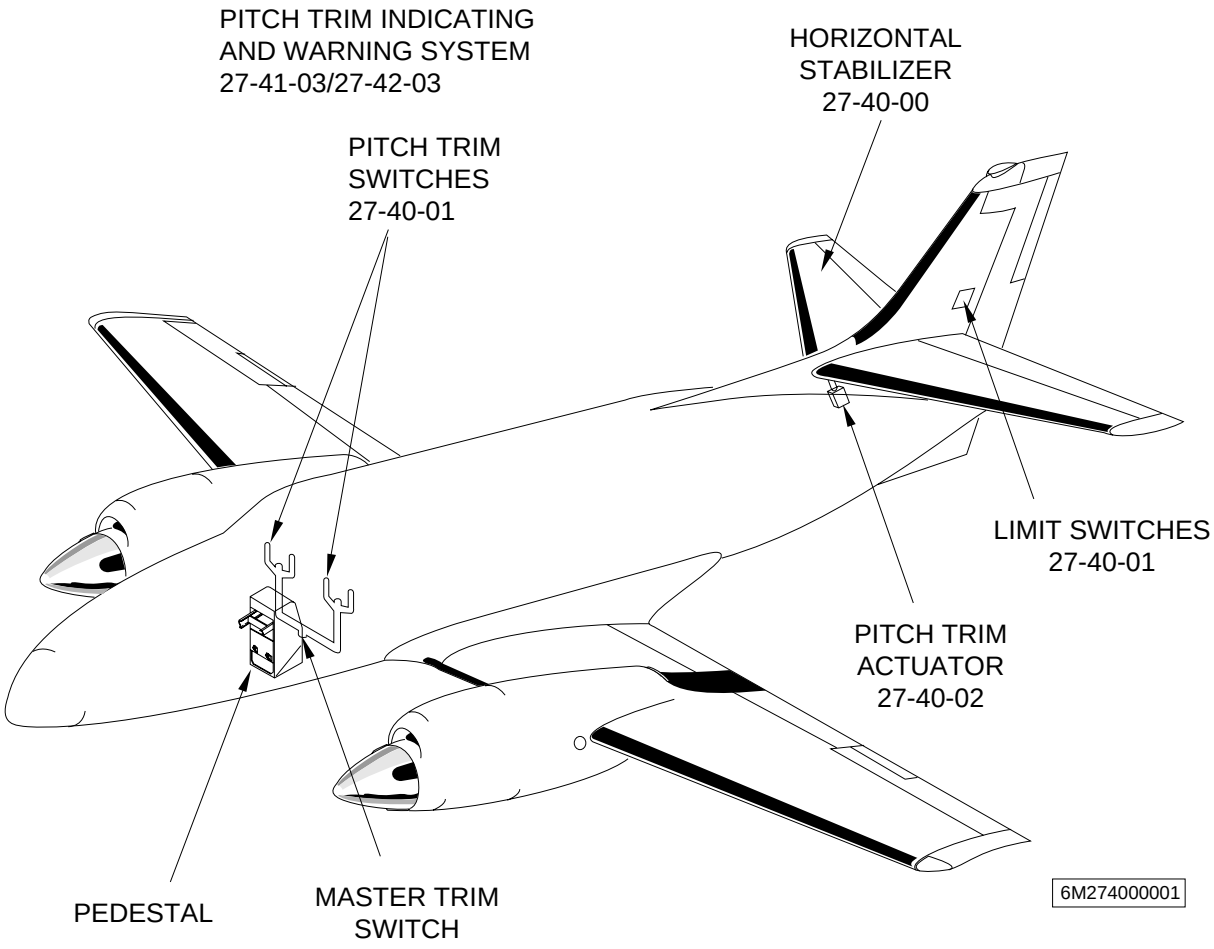
(16) Position cover and secure with screws.

MAINTENANCE MANUAL

HORIZONTAL STABILIZER – DESCRIPTION AND OPERATION

1. General

A. The horizontal stabilizer is a metal cantilever design structure consisting of left and right units which are bolted together at the fuselage centerline. The horizontal stabilizer is attached to the vertical stabilizer with a large trunnion bolt. The unit pivots around the trunnion to change the angle of incidence of the stabilizer. The angle of incidence is controlled by the pilot or copilot with a trim switch in the control wheel. The stabilizer is moved by two motor operated jackscrews mounted on the rear top of the fuselage.



Horizontal Stabilizer System
Figure 1

MAINTENANCE MANUAL

HORIZONTAL STABILIZER – MAINTENANCE PRACTICES

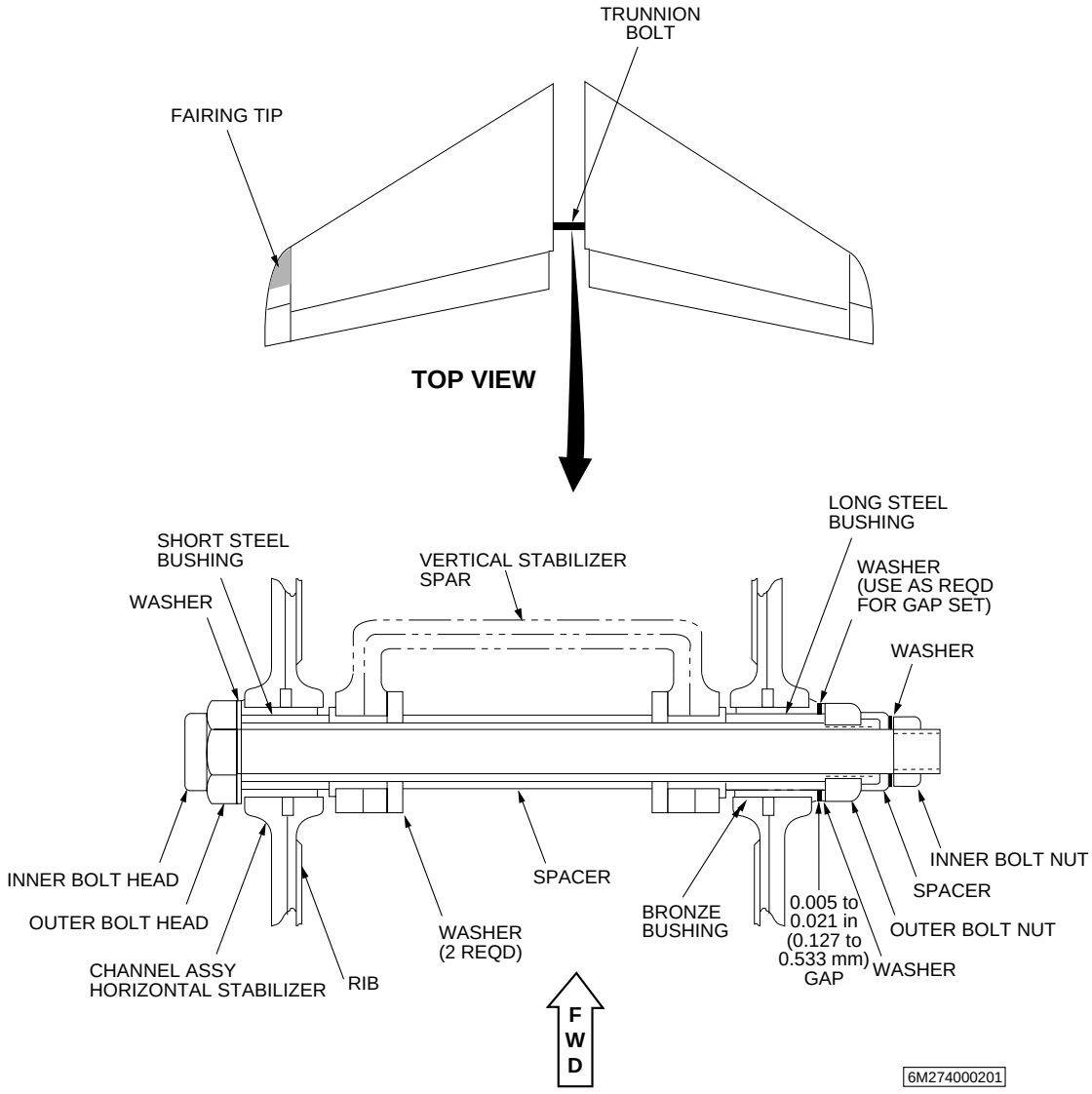
1. Servicing – Horizontal Stabilizer

A. The only maintenance required of the horizontal stabilizer is periodic lubrication of the trunnion bolt. Lubricate with general type grease (MIL-G-23827 or equivalent). The lubrication fittings are located on the aft interconnecting spar and are accessible through access panels located on the underside of the horizontal stabilizer.

2. Removal/Installation – Horizontal Stabilizer

A. Removal – Horizontal Stabilizer

(1) Provide adequate shoring for support of both horizontal stabilizers.



Trunnion Bolt
Figure 201

(2) Remove both elevators IAW Section 27-30-01.

(3) Extend pitch trim actuator (nose down position).

(4) Remove screws from bottom portion of actuator access panel (saddle).

(5) Retract actuator (nose up position).

(6) Remove remaining screws from saddle and remove saddle.

(7) Remove bolts securing trim actuator rods to actuator brackets.

(8) Remove bolts securing right trim actuator attaching bracket to interconnecting spar.

(9) Remove round access panel located on right forward lower side of vertical stabilizer.

(10) Through access panel disconnect deice boot line.

(11) Peel back approximately six inch section of deice boot from inboard leading edge of right horizontal stabilizer.

(12) Drill out top and bottom rivets for removal of short inboard butt rib (forward of trim actuator attaching bracket) from forward inboard edge of right horizontal stabilizer.

(13) Remove rib section.

(14) Remove screws from top and bottom of right stabilizer securing skin to inboard portion of forward spar.

(15) Remove round access panel located forward bottom of right stabilizer.

(16) Working through access panel remove Hi-lock bolts securing right stabilizer to forward interconnecting spar.

(17) Remove elevator quadrant IAW Section 27-30-04.

(18) Disconnect elevator quadrant lock cable from arm.

(19) Remove Hi-lock bolts securing quadrant lock bracket (red) and lay aside.

(20) Remove two square access panels located aft bottom of right and left horizontal stabilizer.

(21) Working through access panel remove Hi-lock bolts securing right stabilizer to aft interconnecting spar.

(22) Through right aft access panel remove inner trunnion bolt nut (5/8 inch) while holding inner (5/8 inch) bolt head through left access panel.

(23) Remove washer and chamfered spacer. Note placement and direction of spacer; with chamfered side towards the bolt head. Save for installation.

(24) Through right access panel remove outer trunnion bolt nut (7/8 inch) while holding outer (7/8 inch) bolt head through left access panel.

(25) Remove washers. Note order of removal for installation.

(26) Remove trunnion bolt.

NOTE: While removing bolt simultaneously insert 0.875 inch (22.22 mm) rod (or similar tool) to replace bolt. This ensures inner spacers and bushings stay fast and aids in bolt installation.

(27) Separate and remove left and right horizontal stabilizers.

B. Installation – Horizontal Stabilizer

(1) Mate and align left and right horizontal stabilizer trunnion bolt holes with bolt holes in vertical stabilizer.

NOTE: Spacers and washers must be installed on trunnion bolt in same order as removed to result in a 0.005 to 0.021 inch (0.13 to 5.11 mm) gap space between 7/8 inch nut, washer, and bushing (see Figure 201).

(2) Clean inner and outer bolt with MPK or equivalent.

(3) Apply MIL-S-8802, Type A (PR1422, A) sealant to ID of outer bolt and shank of inner bolt so that a continuous bead is extruded at each end.

(4) Insert inner bolt into outer bolt. Ensure bolts are sealed properly to keep moisture out.

(5) Insert trunnion bolt from left while simultaneously removing tool holding inner spacers.

(6) Through right access door, insert washers and 7/8 inch nut onto outer bolt and torque 1,000 to 1,200 inch-pounds (112.98 to 135.58 N·m).

(7) Measure clearance between 7/8 inch nut, washer, and bushing for 0.005 to 0.021 inch (0.13 to 5.11 mm) gap.

(8) Through right access door, insert spacer and 5/8 inch nut onto inner bolt and torque 600 to 780 inch-pounds (67.79 to 88.13 N·m).

NOTE: Washer may be installed on inner bolt between nut, and spacer as required to prevent bottoming out of retaining nut.

(9) Secure right stabilizer to aft interconnecting spar with Hi-lock bolts.

(10) Secure quadrant lock bracket to interconnecting spar.

(11) Connect quadrant lock cable to arm.

(12) Secure right stabilizer to forward interconnecting spar with Hi-lock bolts.

(13) Install screws in top and bottom of right stabilizer securing skin to inboard portion of forward spar.

(14) Install short inboard butt rib section in right stabilizer forward of trim actuator attaching bracket and replace rivets to secure rib and skin.

(15) Install deice boots.

(16) Connect deice boot line.

(17) Install trim actuator attaching bracket to interconnecting spar.

(18) Attach trim actuator rods to brackets.

(19) Install elevator quadrant IAW Removal/Installation – Elevator Control, this chapter.

(20) Install elevators IAW Removal/Installation – Elevators, this chapter.

(21) Remove support from both stabilizers.

(22) Functional/Check deice boot.

(23) Functional/Check tel-tail light.

(24) Perform operational check.

(25) Check for rubbing at saddle.

(26) Check for rubbing at trailing edge of stabilizer and elevators.

(27) Replace all access panels and doors.

C. Removal – Horizontal Stabilizer Tip Fairing

(1) Support control column and unlock gust lock.

(2) Remove bolts (two) from clip inside horizontal stabilizer tip.

(3) Remove screws top and bottom of tip and remove tip.

D. Installation – Horizontal Stabilizer Tip Fairing

(1) Position tip and replace enough screws to hold tip in place.

(2) Install bolts (two) to inner clip.

(3) Install remaining screws to tip and secure.

3. Inspection/Check – Horizontal Stabilizer

A. Wear – Stabilizer Trunnion Bolt

(1) Have a mechanic push stabilizer tip fore and aft.

NOTE: Stabilizer flexes if pushed too hard. When moving stabilizer tip, remember that purpose is to measure wear between bushing and bolt only.

(2) Record movement at tip.

(3) If movement is less than 0.250 inch (6.35 mm), bushing wear is within allowable tolerance, and no further wear check is required.

(4) If movement is more than 0.250 inch (6.35 mm), disassemble trunnion bolt assembly following procedure in Approved Repairs – Stabilizer Trunnion Bushings, this section. Examine bolt, sleeve, and bushing for wear.

(5) If wear between bolt and bushing exceeds 0.005 inch (0.127 mm), or if out-of-round is more than 0.002 inch (0.051 mm), replace bushings, and bolt.

B. Free Play Measurement – Horizontal Stabilizer

(1) Check horizontal stabilizer free play IAW Flight Controls, this chapter.

4. Approved Repairs – Horizontal Stabilizer

A. Approved Repairs – Tip Fairing and Installing

NOTE: Horizontal stabilizer tips are made of two hydraulically formed aluminum shells welded together. Basic instructions on fitting and installing a new stabilizer tip are as follows:

(1) Place horizontal stabilizer in streamline (zero) position.

(2) On aircraft equipped with plunger gust lock, engage lock and pull control column back until elevators lock. On other aircraft, secure gust lock belt.

(3) Apply 0.750 inch (19.05 mm) masking tape evenly along top edge of skin and over deice boot of horizontal stabilizer leaving rib with screw holes exposed. Apply a second layer of tape along inboard edge of first strip of tape. Apply a third layer of tape making a 2.250 inch (5.71 cm) border of masking tape along edge of stabilizer.

(4) Using rotary file, remove enough of weldment from inside of leading edge of new tip so that when trimmed, tip will fit flush with deice boot.

(5) Remove riveted holding clip from stabilizer rib.

(6) Position tip onto horizontal stabilizer, overlapping tape and deice boot on stabilizer.

(7) Align outboard edge of tip with outboard edge of elevator and secure with tape.

(8) Lower elevator and measure distance between inboard rib of tip and rib of stabilizer. Form a spacer block the thickness of the measured distance by cutting and taping together pieces of aluminum about 2.00 inch (5.08 cm) square.

(9) Remove tip and secure spacer block to rib of horizontal stabilizer with tape.

(10) Position tip so that leading edge of tip is flush against deice boot at leading edge and even with elevator at outboard edge, and inboard rib of tip is flush with spacer block on horizontal stabilizer.

(11) Secure tip in place with tape.

(12) Carefully mark tip at inboard edge (top and bottom) far enough back so that at least 0.250 inch (6.35 mm) of tape applied along edge of aluminum skin will show when trimmed.

(13) Remove tip and trim excess where marked.

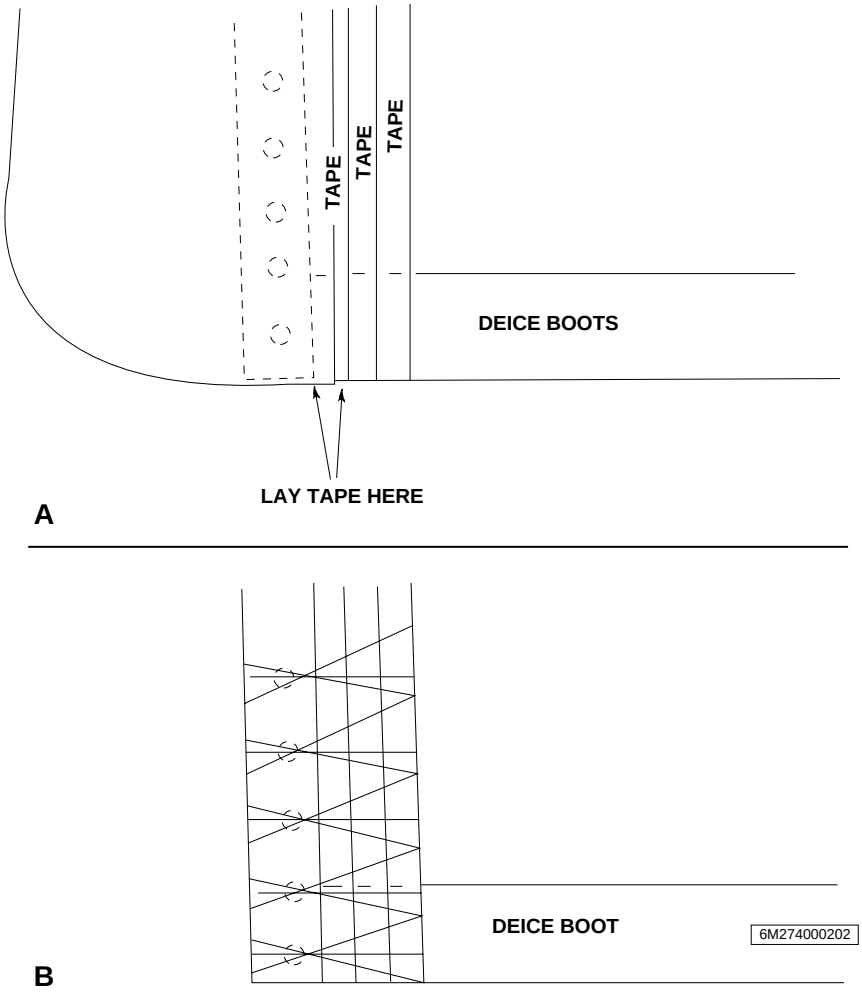
(14) Position tip and secure with tape.

(15) Carefully apply 0.750 inch (19.05 mm) tape over first layer of tape (top and bottom) sandwiching tip between layers. Draw a line along edge of tape onto tip. This line indicates edge of aluminum skin of horizontal stabilizer.

(16) Remove tape and tip. Carefully trim along marked edge.

(17) File tip as necessary for proper fit.

(18) Position tip and secure with tape.



Tip Fairing – Fitting
Figure 202

(19) Mark and trim remainder of tip to fit edge of stabilizer skin.

(20) Using straight edge, draw three lines through center of each screw hole in horizontal stabilizer rib to edge of tape border.

(21) Position tip and secure with tape.

(22) Using straight edge and pencil, continue lines previously drawn from screw holes, from tape back onto tip at all places. Where lines cross on tip indicates where screw holes are located in rib of stabilizer (see Figure 202).

(23) Working from leading edge of tip (at deice boot) predrill top forward hole using size 40 drill bit. Enlarge predrilled hole with size 20 drill bit and secure with screw.

(24) Repeat procedure at bottom foremost mark. Working aft, alternate top and bottom along entire edge of tip.

(25) Remove masking tape.

(26) Support control column and place gust lock lever in unlock position, or remove gust lock belt.

(27) Mark trailing edge of tip allowing 0.125 inch (3.17 mm) clearance when elevator is moved in extreme up or down position.

(28) Remove tip and trim trailing edge.

(29) Remove spacer block.

(30) Install tip.

(31) Position holding clip against stabilizer rib and aft rib tip and draw position mark on stabilizer rib.

(32) Secure clip to stabilizer rib, picking up existing rivet holes in stabilizer rib.

NOTE: If holes in clip do not align with holes in stabilizer, make new clip. Existing holes in stabilizer rib must be used to fasten clip.

(33) Position tip and mark aft rib tip for placement of nutplates and install two nutplates to tip.

(34) Deburr and dimple all screw holes on tip.

(35) Cover holes in ribs with closeout patches.

(36) Secure tip to stabilizer, fourteen (14) places.

(37) Bolt tip to clip, two (2) places.

B. Approved Repairs – Stabilizer Trunnion Bushings

NOTE: This procedure does not require removal of stabilizer to replace bushings.

(1) Tools and Equipment

(a) Brass Punch

(b) 2 ea. 1.250 inch Deep Well Sockets

(c) 2 ea. 0.9375 inch Sockets

(d) 1 ea. 1.125 inch Open End Wrench

(e) 1 ea. 0.750 inch Open End Wrench

(f) 1 ea. 1.00 inch Open End Wrench

(g) 1 ea. 0.875 inch Open End Wrench

(h) Drill Press or Lathe

(i) Small Pneumatic Drill Motor

(j) Pneumatic Rivet Gun

(k) 1 ea. 125 to 150 foot-pounds (169.5 to 203.4 N·m) Torque Wrench

(l) Stabilizer Pivot Bushing Rework Kit:

NOTE: All specialized tools except the reamer can be made locally. Refer to Illustrated Tool and Equipment Manual (ITEM) or 27K43004 for tool specifications.

1 Bullet Tool

2 Reaming Fixture

3 Bushing Puller Tool. Snap-On Tools, Kit CG45.

4 Special Rivet Set

5 Bushing Driver Tool

6 Small Flexible-Shaft Hone

7 1.125 inch Reamer

NOTE: Refer to Figure 203 for item numbers called out in steps **4.**, B., (2) through (47).

(2) Move elevator and pitch trim to neutral position.

(3) Place suitable warnings on cockpit controls, so that pitch trim and elevator control yoke will not be moved:

(a) Tape switches, and pull circuit breakers.

(b) Install warning sign.

(4) Remove access panels on lower side of each stabilizer, to gain access to elevator hinge bolt.

NOTE: Bolt may be installed from either side of elevator.

(5) Disconnect stabilizer deice boot flex lines at one end for better access to bolt and nuts.

(6) Remove first nut (item 1), using 0.937 inch socket wrench.

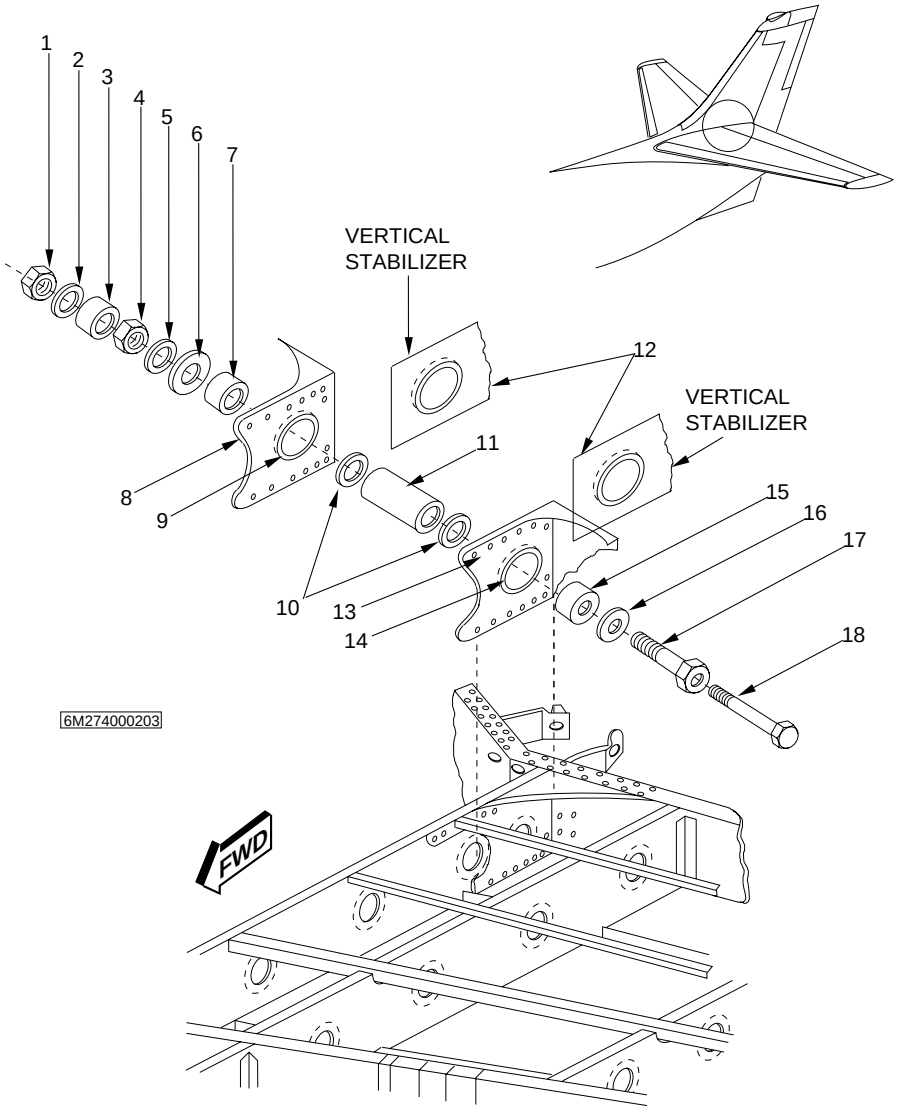
(7) Remove washer (item 2) and chamfered spacer (item 3). Note placement and direction of spacer, with chamfered side towards the bolt head. Save for installation.

(8) Remove second nut (item 4) using 1.250 inch deep well socket. Nut is torqued 108 to 125 foot-pounds (146.45 to 169.50 N·m) and may require an assistant to hold bolt head on other side.

(9) Remove washer (item 5) and washer (item 6). Note order of removal for installation.

(10) Install bolt installer tool onto bolt assembly. This tool is used to protect threads, align spacer and washers and prevent them from falling into vertical stabilizer.

(11) Using bolt installer, drive bolt assembly out just far enough to allow installation of bushing puller assembly. Bolt must retain spacer (item 11), and washers (item 10) inside vertical stabilizer.



Horizontal Stabilizer
Trunnion Bolt Assembly
Figure 203

(12) Remove steel sleeve (item 7). For installation, note that long sleeve is on nut end of bolt, and small sleeve is on head end.

(13) Install bushing puller collet through bushing (item 9), and expand jaws so they catch back or inside edge of bushing. Tighten outer nut against inner nut.

(14) Install steel sleeve, steel plate, and threaded shaft of bushing puller and tighten 1.125 inch nut hand tight until puller is aligned and tight.

(15) Remove old bushing using 0.750 inch wrench and 1.125 inch wrench on puller shaft.

(16) Chuck new bushing into lathe. Adjust lathe speed to 500 RPM.

NOTE: New bushings must be reamed prior to installation.

(17) Ream bushing using piloted 1.125 inch reamer. Make sure reamer is straight and installed in a floating holder if available. Use oil liberally while reaming. Check bushing inside diameter is 1.1259 to 1.1272 inch.

(18) When bushing has been reamed, put bushing and driver in dry ice for at least 30 minutes before installing bushing.

(19) Clean out bushing boss and check for damage to casting (item 8). Apply primer in channel assembly after cleaning and inspection.

(20) Install bushing while primer is wet, and make all preparations so that bushing can be driven immediately, when it is removed from dry ice.

WARNING: DO NOT ATTEMPT TO REMOVE BUSHING FROM DRY ICE WITH BARE HANDS. WEAR INSULATED GLOVES AND GRASP FROZEN BUSHING WITH TONGS OR SIMILAR TOOL.

(21) Line up bushing with boss, and drive bushing as quickly as possible until it seats into casting using rivet set and bushing driver if necessary.

(22) Measure bushing inside diameter after installation and measure outside diameter of new sleeve (item 7). Ensure steel sleeve is 0.0009 to 0.0022 inch (0.0228 to 0.1472 mm) smaller than inside diameter of bushing.

(23) If bushing diameter is too small, hone using small hone with flexible shaft to obtain proper dimension. Use oil liberally during honing.

(24) Install new steel sleeve (item 7).

(25) Install bolt remover tool through steel sleeve and drive out bolt assembly, making sure not to drop spacer (item 11) and washers (item 10) between vertical stabilizer housing halves (item 12). Drive bolt out slowly but steadily, and leave bolt remover tool in place when bolt is removed.

(26) Remove steel sleeve (item 15).

(27) Install bushing puller collet through bushing (item 14), and expand jaws so they catch back or inside edge of bushing (item 14). Tighten outer nut against inner nut until no further tightening is possible.

(28) Install steel sleeve, steel plate and threaded shaft of bushing puller and tighten 1.125 inch nut hand tight, until puller is aligned and tight.

(29) Remove old bushing using 0.750 inch wrench, and 1.125 inch wrench on puller shaft.

(30) Chuck new bushing into lathe. Adjust lathe speed to 500 RPM.

NOTE: New bushings must be reamed prior to installation.

(31) Ream bushing using piloted 1.125 inch reamer. Make sure reamer is straight and installed in a floating holder if available. Use oil liberally while reaming. Check bushing inside diameter is 1.1259 to 1.1272 inch.

(32) When bushing has been reamed, put bushing and driver in dry ice for at least 30 minutes before installing bushing.

(33) Clean out bushing boss and check for damage to casting (item 13). Apply primer in channel assembly after cleaning and inspection.

(34) Install bushing while primer is wet, and make all preparations so that bushing can be driven immediately, when it is removed from dry ice.

WARNING: DO NOT ATTEMPT TO REMOVE BUSHING FROM DRY ICE WITH BARE HANDS. WEAR INSULATED GLOVES AND GRASP FROZEN BUSHING WITH TONGS OR SIMILAR TOOL.

(35) Install rivet set in gun, and bushing driver, and bushing onto set.

(36) Line up bushing with boss, and drive bushing as quickly as possible until it seats into casting using rivet set and bushing driver if necessary.

(37) Measure bushing inside diameter after installation and measure outside diameter of new sleeve (item 15). Ensure steel sleeve is 0.0009 to 0.0022 inch (0.0228 to 0.1472 mm) smaller than inside diameter of bushing.

(38) If bushing diameter is too small, hone using small hone with flexible shaft to obtain proper dimension. Use oil liberally during honing.

(39) Install new shorter sleeve (item 15) on head side of bolt.

(40) Install washer (item 16), and bolts (items 17 and 18) with bolt installer tool still attached and drive bolt through newly installed bushings, driving out the bolt remover tool. Be careful not to drop parts into middle of vertical stabilizer.

(41) Shim as necessary IAW Removal/Installation – Horizontal Stabilizer. Use free-running nut supplied for shimming purposes until dimension required is reached. Torque 1,000 to 1,200 inch-pounds (112.98 to 135.58 N·m) each time shims are removed or replaced.

(42) When proper gap is achieved (0.005 to 0.021 in [0.13 to 5.11 mm], see Figure 201), remove free-running nut, and install a new self-locking nut (item 4). Torque 1,000 to 1,200 inch-pounds (112.98 to 135.58 N·m).

(43) Measure gap. If gap is within tolerance, install chamfered spacer (item 3, chamfer towards bolt head), new washer (item 2) if required, and a new self-locking nut (item 1). Torque 50 to 60 foot-pounds (67.8 to 81.4 N·m).

(44) Lubricate grease fittings.

(45) Connect stabilizer deice boot flex lines.

(46) Install stabilizer access panels.

(47) Remove warning signs, and return cockpit to full operational status.

PITCH TRIM SYSTEM AND CONTROL – DESCRIPTION AND OPERATION

1. General

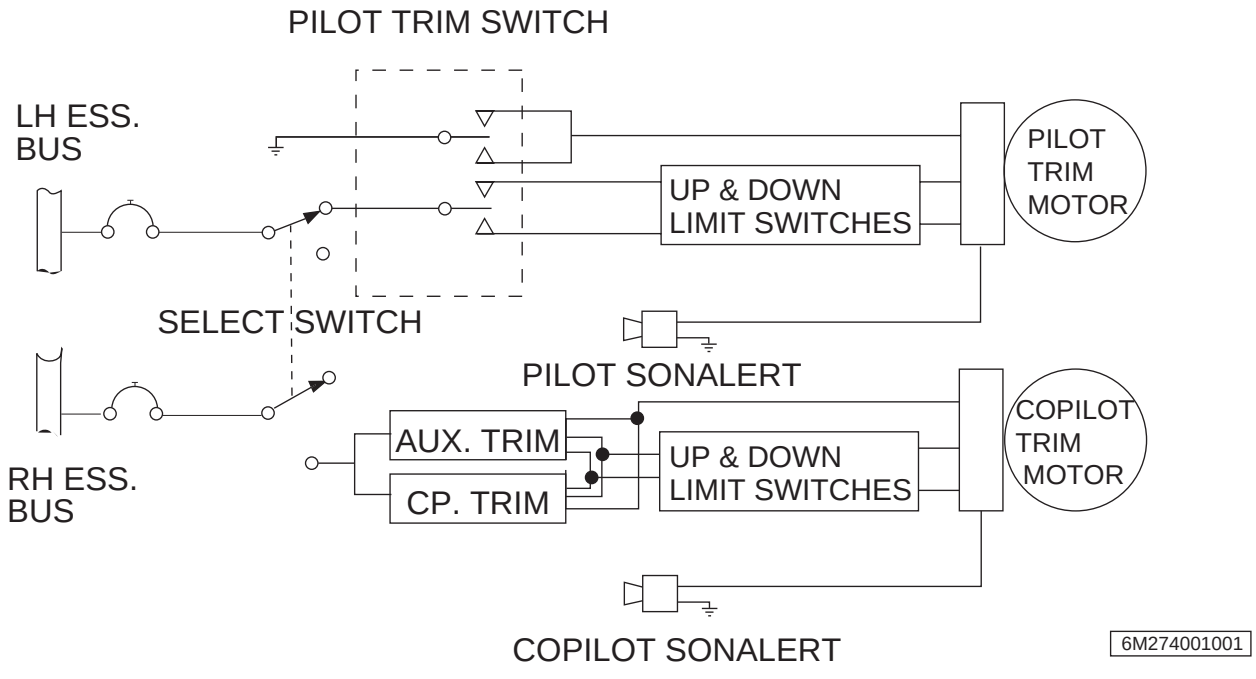
A. The horizontal stabilizer is electrically positioned to provide pitch trim. The stabilizer pivots near its rear spar with the leading edge positioned by a dual jackscrew actuator attached to the stabilizer front spar.

B. The trim system consists of a trim select switch, an auxiliary trim switch, pilot and copilot control wheel trim control switches, pitch trim actuator, up and down travel limit switches, a horizontal position indicating system and an aural trim-in-motion indicator circuit.

C. The pitch trim system is designed to make it impossible for a single failure in the system to cause the trim actuator to operate or become inoperative. Both the power and the ground side of the electrical actuators are controlled so that a failure on one wire due to a short or hot fault, will not hinder system operation. The pilot electrical trim wires are routed separate from the copilot electrical trim wires and also separate from all other control circuits and power circuits in the aircraft.

D. Trim select switch

(1) The master trim switch, located on the pedestal, consists of two separate switches tied together with a gang bar. The gang bar prevents simultaneous pilot and copilot trim operation. The switches have a center OFF position which serves as a master switch. With the trim select switch in the copilot position, the pilot may trim with the auxiliary trim switch.



Pitch Trim System and Control
Figure 1

PITCH TRIM SYSTEM AND CONTROL – DESCRIPTION AND OPERATION

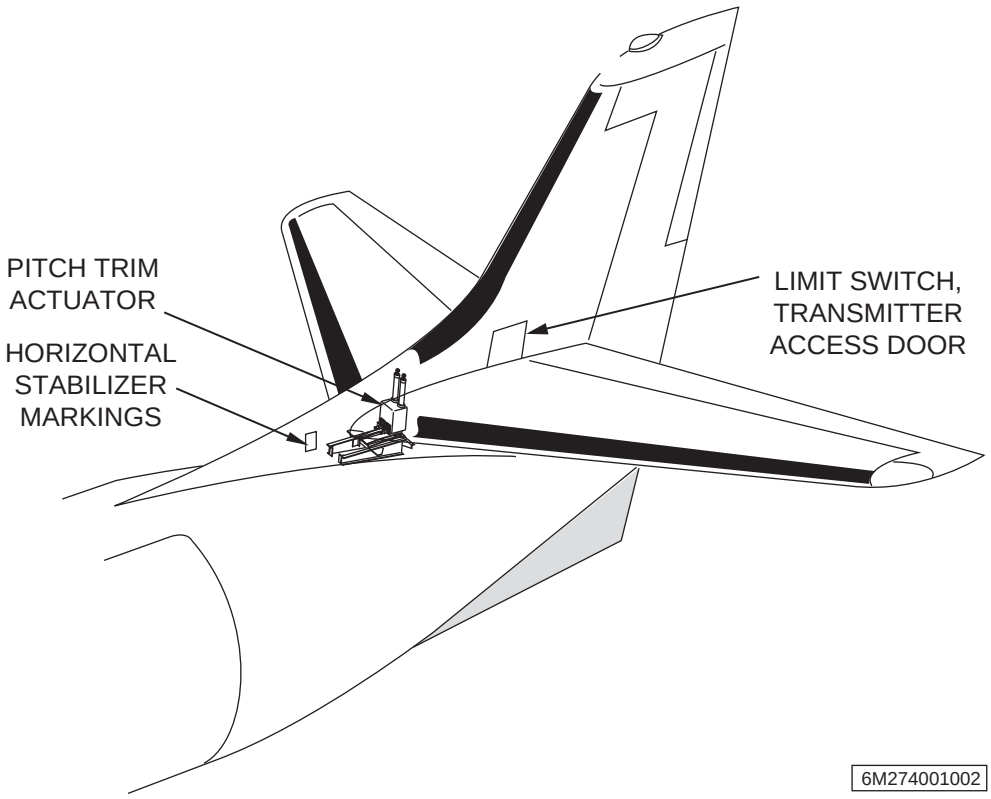
E. Pilot and copilot control wheel trim control switches

(1) The pilot and copilot control wheel trim control switches are special switches which have passed qualification tests for pitch trim control switches. The switches, located on each respective control wheel have double toggle actuators either of which when operated will not cause the trim actuator motor to run. Both halves of the switch must be operated simultaneously in order to provide trim operation.

(2) One half of the switch is the ground side for the trim motor while the other half is the power side for the trim motor. If only one half of the switch is actuated and the trim motor runs then a malfunction exists in the circuitry to that half of the switch which has not been actuated. In this manner, a preflight check of the trim system is performed.

F. Auxiliary trim switch

(1) A pilot auxiliary trim switch, a rocker button switch, located on the lower right section of the pedestal is incorporated to facilitate single-pilot operation should a malfunction occur in the pilot trim control circuitry. The switch is in the copilot trim circuitry and allows the pilot to trim the horizontal stabilizer without having to reach across to the copilot trim switch on the copilot control wheel.



Pitch Trim System - Locator
Figure 2

PITCH TRIM SYSTEM AND CONTROL – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pilot and/or copilot system inoperative.	Circuit breaker (CB61/ CB62) tripped or defective.	Reset or replace circuit breaker.
	Defective wiring between circuit breaker and trim select switch (S57).	Check wiring and terminals and replace wiring and/or switch as necessary.
	Defective wiring between trim select switch and control switches (S58/ S59/S60).	Check wiring and terminals and replace wiring and/or switches as necessary.
	Trim control switches and auxiliary trim not grounded properly.	Check ground wires to pilot and copilot trim switches and auxiliary trim switch.
	Defective ground wire from trim switch (S58/S59/S60) to trim motor (actuator).	Check wire and termination and repair or replace as necessary.
	Sonalert signal wire between actuator (E73) and sonalert (H2/H3) shorted to ground.	Check wire and repair or replace as necessary.
	Actuator (E73) defective.	Replace actuator.
Circuit breaker trips when either control wheel switch is actuated.	Switch (S59/S60) terminals and/or wires shorted to ground in switch housing.	Pull switch and insulate terminals and wires.
Actuator runs when only half of control wheel switch is actuated.	Wire shorted to ground between switch (S59/S60) and actuator (E73).	Remove pilot cockpit bulkhead and check for short.
		Check entire wire along fuselage.
Sonalert inoperative or not functioning properly.	Broken signal wire.	Repair wire.
	Defective sonalert (H2/H3).	Repair or replace sonalert.
	Broken signal wire at actuator (E73).	Repair wire.
Sonalert continues to sound after stabilizer reaches limits.	Mechanical/Electrical Stops not properly adjusted.	See 2. A. Adjustment – Mechanical/Electrical Stops, this section.

PITCH TRIM SYSTEM AND CONTROL – MAINTENANCE PRACTICES

1. Removal/Installation – Pitch Trim System and Control

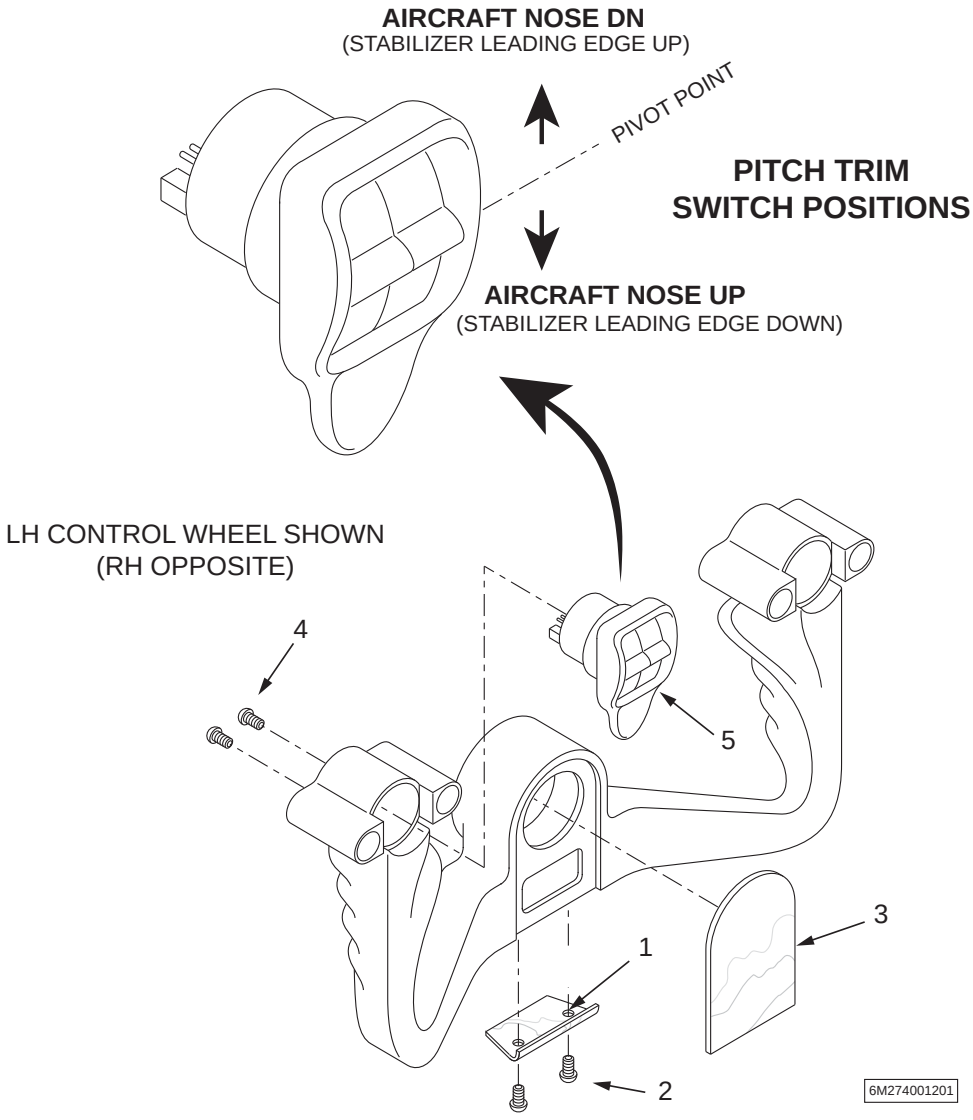
A. Removal – Pilot and Copilot Control Wheel Trim Switches (Figure 201)

(1) Locate retainer (1) at bottom center of control wheel and remove set screws (2).

(2) Remove retainer and escutcheon (3).

(3) Remove screws (4) in rear of switch housing, outboard side of control wheel.

(4) Pull switch (5) out of housing.



Pitch Trim Control Wheel Switches
Figure 201

B. Installation – Pilot And Copilot Control Wheel Trim Switches (Figure 201)

(1) Wire switch IAW SA226 Series Wiring Manual.

CAUTION: ENSURE VINYL SLEEVING IS INSTALLED OVER SWITCH TERMINALS.

(2) Verify proper switch orientation and insert switch (5) into housing while simultaneously pulling wires at center of control wheel.

(3) Secure switch in housing with screws (4).

(4) Position escutcheon (3) in place and secure with retainer (1) and set screws (2).

WARNING: VERIFY STABILIZER LEADING EDGE MOVES IN THE PROPER DIRECTION RELATIVE TO SWITCH POSITION SELECTED AS SHOWN IN FIGURE 201.

(5) Perform Operational Test - Pitch Trim and Control.

C. Removal – Limit Switch Assembly (Figure 202)

(1) Remove access panel located on left side of vertical stabilizer just above horizontal stabilizer (see Figure 2). Tag and identify wires to limit switches.

(2) Remove individual switches by removing screws securing wires and nuts securing switches.

NOTE: Forward limit switches (two) on early assembly are inaccessible unless switch assembly bracket is removed.

(3) Disconnect push-pull rod from switch actuator cam or paddle (depending on model).

(4) Remove position transmitter (see 27-41-03 or 27-42-03 as applicable for airplane S/N).

(5) Remove bolts (four) at base of switch assembly bracket and remove bracket.

D. Installation – Limit Switch Assembly (Figure 202)

(1) Using identification tags, connect wires to limit switches and install nuts to secure switches to bracket assembly.

(2) Secure bracket to vertical stabilizer structure with four bolts.

(3) Install position transmitter (see 27-41-03 or 27-42-03 as applicable for airplane S/N).

(4) Connect push-pull rod to switch actuator (cam or paddle, depending on airplane S/N).

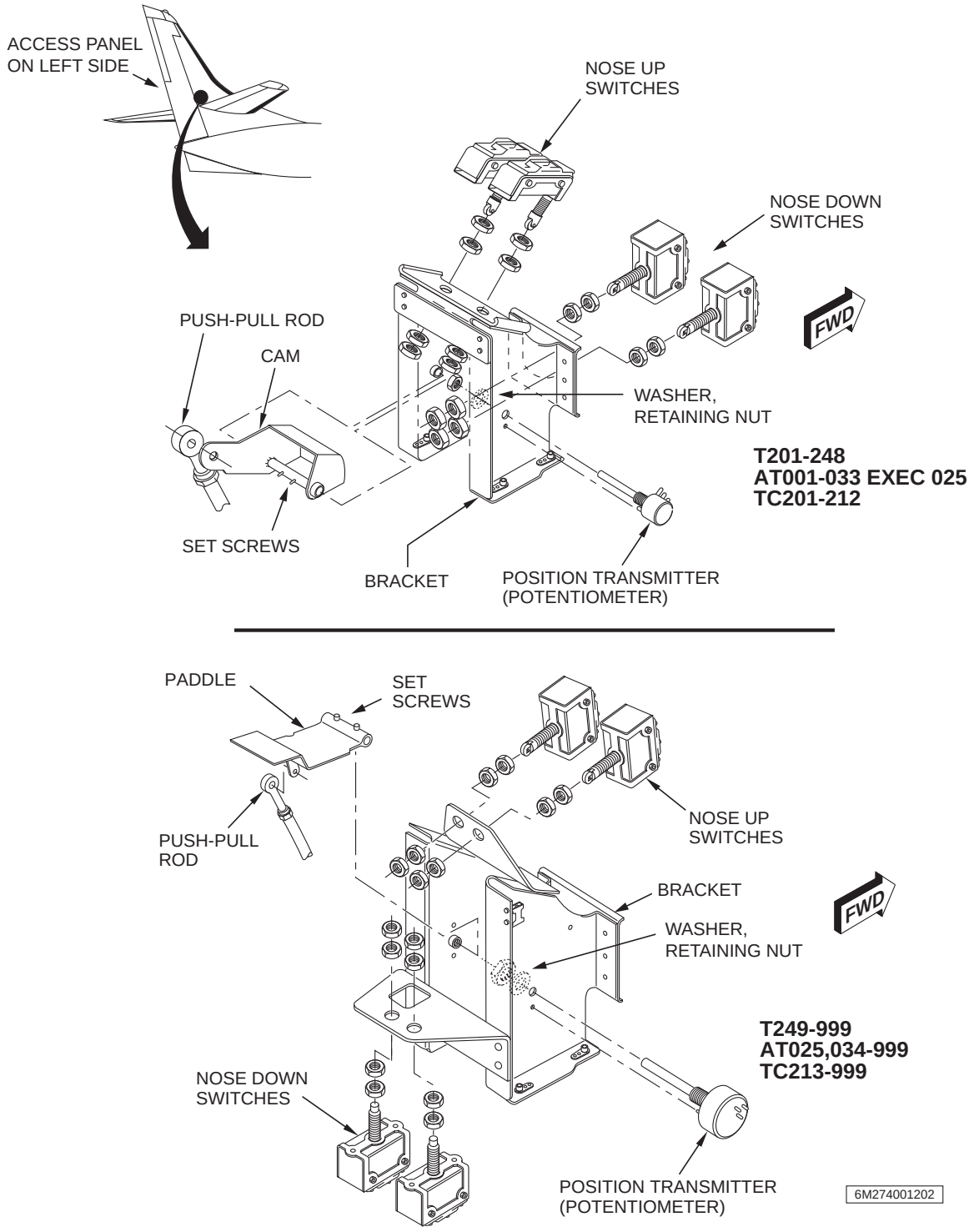
(5) Perform 2. A. Adjustment – Mechanical/Electrical Stops, this section.

(6) Perform 2. A. Adjustment – Pitch Trim Indicator/Warning Circuitry in section 27-41-03 or 27-42-03 as applicable for airplane S/N.

(7) Perform 2. D. Operational Check – Pitch Trim and Control, this section.

(8) Perform 2. C. Operational Test - Indicating and Warning System in section 27-41-03 or 27-42-03 as applicable for airplane S/N.

(9) Install access panel just above horizontal stabilizer (see Figure 2).



Pitch Trim Limit Switch Assembly
Figure 202

2. Adjustment/Test – Pitch Trim System and Control

A. Adjustment – Mechanical/Electrical Stops (Figures 203 and 204)

(1) Gain access to pitch trim limit switch by removing panel located on the left side of vertical stabilizer just above horizontal stabilizer (see Figure 2).

(2) Gain access to trim actuator by removing access panel (saddle). On later series aircraft, remove dorsal fin door.

(3) At limit switch, disconnect the push-pull rod from the switch actuator paddle (or cam). Tie rod loosely enough so that paddle does not activate switches and shut off system while mechanical stops are being adjusted.

(4) Connect ground power unit to aircraft.

(5) Place elevator/horizontal stabilizer gauge on designated station on stabilizer.

(6) Run horizontal stabilizer to zero rivet on dorsal fin. Position so that bottom edge of stabilizer is centered on zero rivet.

(7) On propeller protractor, align zeros on protractor to engage DISC-TO-RING LOCK.

(8) Position protractor at designated place on stabilizer gauge.

(9) With RING-TO-FRAME LOCK on protractor loose, turn RING ADJUSTER until level bubble is centered, then tighten protractor RING-TO-FRAME LOCK.

(10) Release protractor DISC-TO-RING LOCK.

CAUTION: WITH THE ELECTRIC TRIM LIMIT SWITCHES PROPERLY RIGGED PER THIS SECTION, WHEN ATTAINING FULL NOSE UP OR DOWN LIMITS, SOME AUDIBLE RATCHETING SOUNDS MAY BE HEARD FROM THE ACTUATOR. THIS RATCHETING SOUND IS NORMAL AND SHOULD NOT BE REASON TO REJECT THE ACTUATOR AS LONG AS THE ACTUATOR CONTINUES TO DRIVE TOWARD THE COMMANDED POSITION AND STOPS AT AND HOLDS THE COMMANDED POSITION.

(11) Run horizontal stabilizer to full aircraft NOSE DOWN position, stabilizer leading edge full up.

(12) Turn protractor DISC ADJUSTER until level bubble is centered.

(13) Verify protractor reading on horizontal stabilizer is within the following tolerance. Make note of actual reading:

(a) 2.1° ± 0.2° (T Model).

(b) 2.4° ± 0.2° (AT and TC Model).

CAUTION: WHEN BOLTS ARE REMOVED FROM ACTUATOR ROD ENDS, THE HORIZONTAL STABILIZER IS FREE. CARE MUST BE TAKEN NOT TO PLACE ANY WEIGHT ON STABILIZER.

(14) If adjustment is necessary:

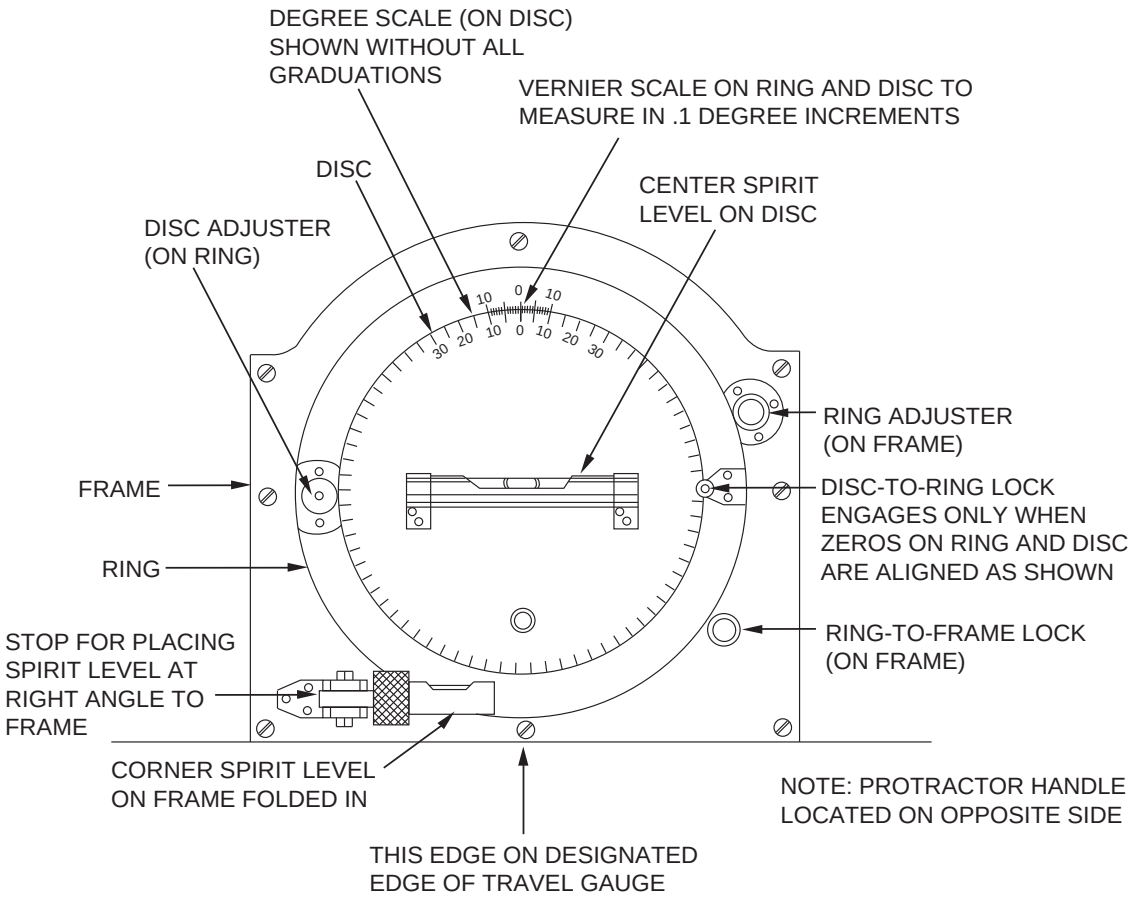
(a) Remove bolts attaching actuator rod ends to stabilizer fittings and remove safety wire from locknuts on rod ends.

(b) Lift safety tabs and turn both rod ends equally in or out as necessary. Make note of the direction and number of turns made. To maintain adequate rod end thread engagement, do not exceed the following rod end adjustment limits for the part number actuator being installed:

1 P/N DL5040M6: Adjust actuator rod ends no more than 5 turns in, and no more than 4 turns out.

2 P/N DL5040M8: Adjust actuator rod ends no more than 3 turns in, and no more than 3 turns out.

3 P/N 27-19008-006 and -007: Adjust actuator rod ends no more than 3 turns in, and no more than 3 turns out.



Propeller Protractor Settings
Figure 203

(15) Run stabilizer to full aircraft NOSE UP position, stabilizer leading edge full down.

(16) Verify protractor reading on horizontal stabilizer is within the following tolerance. Make note of actual reading:

(a) 8.2° ± 0.2° (T Model).

(b) 7.8° ± 0.2° (AT and TC Model).

(17) Verify full nose up and nose down readings fall within tolerance. Repeat adjustments as necessary until both settings are within tolerances.

(a) If one position (nose up or nose down) does not fall within acceptable tolerances, compensation must be made at opposite end of travel.

(b) Example: If NOSE UP is adjusted to 7.9° and NOSE DOWN is adjusted to 2.1° on AT or TC Model airplane. To compensate, the nose up position would have to be adjusted to 7.7° to get a nose down reading of 2.3° and bring both mechanical stops within their tolerance range.

NOTE: Prior to adjusting electrical stops, mechanical stops must be within tolerances.

(18) Attach push-pull rod to paddle (or cam) with bolt, nut, and cotter pin.

NOTE: Push-pull rod length is preset at the factory and is not to be changed to adjust the electrical stop.

NOTE: Prior to adjusting electrical stops, mechanical stops must be within tolerances

(19) Using pilot trim control, run stabilizer to FULL NOSE DOWN position, stabilizer leading edge full up.

NOTE: Electrical stops must prevent contact with mechanical stops.

(20) Using protractor, verify electrical stops (actuator cut-off) are adjusted at 0.2° ± 0.05° before the actuator contacts its mechanical stops.

NOTE: Example: If nose down mechanical stop is set at 2.4°, the nose down limit switch must be adjusted to activate between 2.15° and 2.25°.

(21) Adjust pilot nose down limit switch as necessary.

NOTE: Nose down and nose up limit switches on left side of assembly are for pilot control system and nose down and nose up limit switches on right side of assembly are for copilot control system.

(22) If adjustment of a limit switch is required, ensure that adjacent limit switch is not mechanically interfering with operation of switch to be adjusted. Loosen adjacent switch until first switch is set, then tighten adjacent switch in a position parallel to first switch.

(23) Be sure jam nuts of pitch trim limit switches are secure and torque sealed. Check for structural clearance throughout entire travel of stabilizer.

(24) Using pilot trim control, run stabilizer to full aircraft NOSE UP position, stabilizer leading edge full down.

(25) Using protractor, verify electrical stops (actuator cut-off) are adjusted at 0.2° ± 0.05° before the actuator contacts its mechanical stops.

(26) Adjust pilot nose up limit switch if necessary.

(27) Repeat procedure to adjust copilot limit switches using copilot trim control.

(28) Perform 2. D. Operational Test – Pitch Trim and Control, this section.

(29) Perform 2. A. Adjustment and Test – Pitch Trim Indicator/Warning Circuitry in section 27-41-03 or 27-42-03 as applicable for airplane S/N.

(30) Disconnect external electrical power from aircraft.

B. Adjustment – Locating/Installing Zero Degree Rivet on Dorsal Fin (Figures 203 and 204)

NOTE: The zero degree rivet was located and installed during aircraft manufacture. This procedure should only be necessary if repair or replacement of structure in this area caused the zero rivet location to be lost or become suspect.

NOTE: The airplane should be stabilized with jacks or other means to prevent airplane pitch attitude changes during system adjustments. Clear personnel from aircraft during this procedure except those required for operation of appropriate systems.

(1) Connect external electrical power to aircraft.

(2) Run stabilizer through full up and full down range and check for structural clearance.

(3) Place an 18 inch (45.72 cm) or longer straight-edge on top of fuselage, over wing area, maintaining a flat plane.

(4) Place prop protractor on straight-edge, zero prop protractor and tighten the RING TO FRAME LOCK.

(5) Place elevator/horizontal stabilizer travel gauge (TF27-09) on left horizontal stabilizer. See Illustrated Tool and Equipment Manual (ITEM) for proper gauge.

(6) Place prop protractor, without changing the zero setting, on travel gauge in area marked on gauge.

(7) Using either pitch trim switch in cockpit, run horizontal stabilizer up or down until level bubble in protractor disk is centered.

(8) Mark the dorsal fin along bottom edge of left horizontal stabilizer, from deice boot to approximately two inches (50.8 mm) aft of rub strip.

(9) On zero degree line marked in previous step, at approximately one inch (25.4 mm) aft of rub strip, drill hole and install MS20470A4 rivet.

(10) Disconnect external electrical power from aircraft.

C. Adjustment – Marking Dorsal Fin – Stabilizer Full-Up, Full-Down, Takeoff Bands (Figure 204)

NOTE: The full up, full down and takeoff bands were located and marked with drill dimples on dorsal fin during aircraft manufacture. This procedure should only be necessary if repair or replacement of structure in this area caused the drill dimple marks to be lost or become suspect.

NOTE: These instructions apply to all but early 226 aircraft in which limit marks are punched in the rub strip area. See Figure 204 for limit marks on earlier aircraft.

NOTE: Use contrasting color paint for all pitch trim markings on dorsal fin.

(1) Run horizontal stabilizer to full aircraft NOSE DOWN position, stabilizer leading edge full up.

CAUTION: TAKE CARE NOT TO DRILL MORE THAN 0.120 INCH (3.048 mm) DEEP AND THROUGH FIBERGLASS SKIN. IF DRILL DIMPLES PENETRATE INTO HONEYCOMB, FILL HOLE WITH RESIN AND ALLOW TO SET. REDIMPLE 0.060 TO 0.120 INCH (1.524 TO 3.048 mm) DEPTH.

(2) Start drill a 0.089 to 0.094 inch (2.261 to 2.238 mm) diameter, 0.060 to 0.120 inch (1.524 to 3.048 mm) deep dimple to locate center of nose down stencil marking.

(3) Paint NOSE DOWN limit marking.

(4) If necessary, perform step 2. B. Locating/Installing Zero Degree Rivet on Dorsal Fin, in this section.

(5) Run horizontal stabilizer leading edge down 0.5° (for SA226-T aircraft), or 0.6° (for SA226-AT, and TC aircraft) from zero degree rivet.

(6) Start drill a 0.089 to 0.094 inch (2.261 to 2.238 mm) diameter, 0.060 to 0.120 inch (1.524 to 3.048 mm) deep dimple to mark location of top corner of takeoff band marking.

(7) Run horizontal stabilizer leading edge down 1.5° (for SA226-T aircraft), or 2.7° (for SA226-AT, and TC aircraft) from zero degree rivet.

(8) Start drill a 0.089 to 0.094 inch (2.261 to 2.238 mm) diameter, 0.060 to 0.120 inch (1.524 to 3.048 mm) deep dimple to mark location of bottom corner of takeoff band marking.

(9) Paint TAKEOFF band marking.

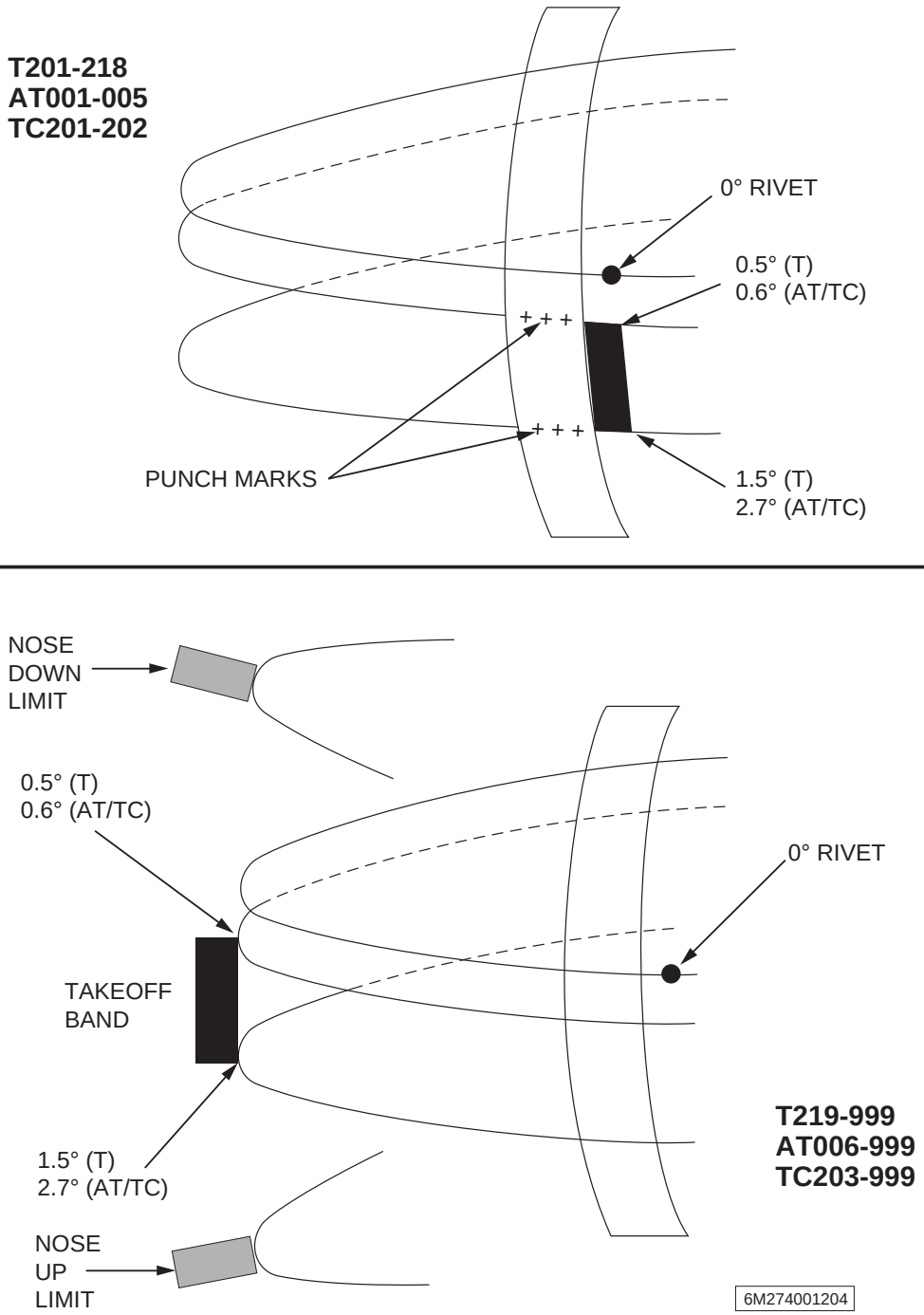
(10) Run horizontal stabilizer to full aircraft NOSE UP position, stabilizer leading edge full down.

(11) Start drill a 0.089 to 0.094 inch (2.261 to 2.238 mm) diameter, 0.060 to 0.120 inch (1.524 to 3.048 mm) deep dimple to locate center of nose up stencil marking.

(12) Paint NOSE UP limit marking.

(13) Perform 2. A. Adjustment and Test – Pitch Trim Indicator/Warning Circuitry in section 27-41-03 or 27-42-03 as applicable for airplane S/N.

(14) Disconnect external electrical power from aircraft.



Horizontal Stabilizer Markings – Dorsal Fin
Figure 204

D. Operational Test – Pitch Trim and Control

(1) Connect GPU to aircraft. Ensure that tail of aircraft is clear of all personnel and equipment.

(2) Position battery switches ON and ensure LH and RH Essential Bus are at 29.0 VDC (+ 0.0 or -0.5).

(3) Place TRIM SELECT switch on PILOT.

(4) Push right toggle of pilot control wheel trim switch in both directions. Verify only pilot's sonalert sounds, with no movement of horizontal stabilizer. Push left toggle of switch. Verify no sonalert sounding and no horizontal stabilizer movement.

(5) Push both toggles of pilot's trim switch to upward position. Verify stabilizer moves toward NOSE DOWN position. Verify only pilot's sonalert sounds as stabilizer is moving and stops sounding when stabilizer reaches FULL NOSE DOWN position.

(6) Push both toggle switches to downward position. Verify stabilizer moves toward NOSE UP position. Verify sonalert sounds as stabilizer is moving and stops sounding when stabilizer reaches FULL NOSE UP position.

(7) Place TRIM SELECT switch on COPILOT.

(8) Repeat steps (4) through (6) using copilot control wheel trim switch.

(9) On pedestal, push auxiliary trim switch to NOSE DOWN until stabilizer reaches TAKEOFF position, then release switch.

(10) Push copilot control wheel trim switch to upward position. While stabilizer is proceeding to NOSE DOWN position, push auxiliary trim switch to NOSE UP position. Verify auxiliary switch overrides control wheel trim switch and horizontal stabilizer proceeds to NOSE UP position.

(11) Push copilot control wheel trim switch to downward position. While stabilizer is proceeding to NOSE UP position, push auxiliary trim switch to nose down position. Verify auxiliary switch overrides control wheel trim switch and horizontal stabilizer proceeds to NOSE DOWN position.

(12) Repeat operational check of auxiliary trim switch several times.

(13) Return horizontal stabilizer to TAKEOFF position.

(14) Select battery switches OFF.

(15) Disconnect GPU.

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PITCH TRIM ACTUATOR – DESCRIPTION AND OPERATION

1. General

A. The horizontal stabilizer is positioned by either of two electric motors within the pitch trim actuator driving a dual jackscrew mechanism.

(1) The motors are individually insulated and contained within one housing.

(2) Both the positive and negative sides of each motor are controlled by their respective trim control switch.

(3) The pilot trim motor is supplied 28 volts DC through a ten amp circuit breaker from the left essential bus. The copilot trim is supplied 28 volts DC through a ten amp circuit breaker from the right essential bus. There is an up limit and a down limit switch associated with each motor. The limit switches are located in the vertical stabilizer.

B. The trim system incorporates an aural trim-in-motion indicator circuit designed to provide aural indication to the pilot and copilot that electrical power is being applied to the trim actuator.

(1) The aural signal is obtained from the positive brush of the actuator, therefore, aural indication may be obtained without stabilizer movement.

(2) The pilot system is completely isolated from the copilot system.

(3) The design furnishes three to five cycles per second (CPS) pulsed 2,800 CPS signal from a sonalert aural transmitter. The frequency is such that trim actuation is easily detectable, yet the indication does not interfere with pilot and copilot aircraft communication and navigation.

(4) The sonalerts are installed adjacent to the pilot and copilot cockpit overhead lights.

C. Some aircraft equipped with autopilot utilize the pitch trim actuator for pitch control and incorporate a sonalert delay system. If so equipped, see avionic electric diagrams for wiring configuration.

PITCH TRIM ACTUATOR – MAINTENANCE PRACTICES

1. Servicing – Pitch Trim Actuator

A. See Airframe Airworthiness Limitations Manual ST-UN-M002 and Chapter 5 in this manual for inspection/checks and overhaul/replacement intervals for each part number pitch trim actuator.

2. Removal / Installation – Pitch Trim Actuator

A. Removal – Pitch Trim Actuator (Figures 201 and 202)

CAUTION: WHEN BOLTS ARE REMOVED FROM ACTUATOR ROD ENDS, THE HORIZONTAL STABILIZER IS FREE. CARE MUST BE TAKEN NOT TO PLACE ANY WEIGHT ON STABILIZER.

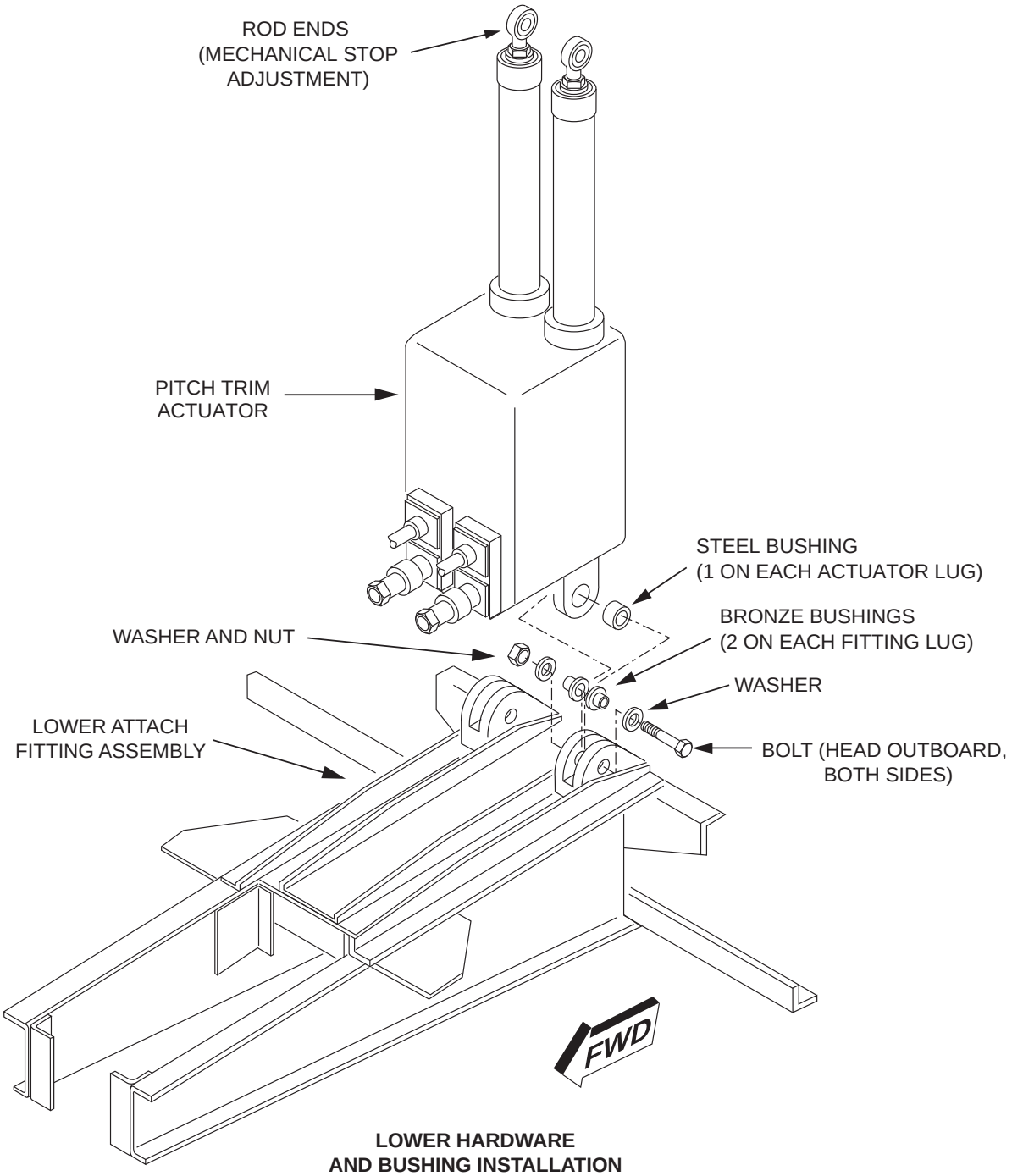
(1) Extend actuator to full aircraft NOSE DOWN position, stabilizer leading edge full up.

(2) Remove screws from bottom portion of access panel (saddle panel on dorsal fin). Later model aircraft, remove screws from left and right access doors on dorsal fin and remove both doors.

INSTALLED ACTUATOR	ACCEPTABLE REPLACEMENT ACTUATORS			
	DL5040M6	DL5040M8***	(W/O Autopilot) 27-19008-006**	(With Autopilot) 27-19008-007
DL5040M5*	X	X	X	X
DL5040M6	X	X	X	X
(W/O Autopilot) 27-19008-001*	X	X	X	X
(With Autopilot) 27-19008-002*	X	X	X	X
(W/O Autopilot) 27-19008-004*	X	X	X	X
(With Autopilot) 27-19008-005*	X	X	X	X
27-19008-006	X	X	X	X
27-19008-007	X	X	X	X
DL5040M8	X	X	X	X

* Do not install as a replacement actuator.
** Airplanes with autopilot will be required to operate with autopilot inoperative.
*** Requires 27K44063-001 Elevator Downspring Pulley Installation Kit prior to installation (AT & TC Models)

Acceptable Replacement Actuators
Table 201



LOWER HARDWARE
AND BUSHING INSTALLATION
(PILOT SIDE SHOWN FOR CLARITY,
COPILOT SIDE IS OPPOSITE)

6M274002201

Pitch Trim Actuator - Lower Hardware/Bushing Installation
Figure 201

(3) Retract to full aircraft NOSE UP position, stabilizer leading edge full down.

(4) Remove remaining screws from saddle panel on dorsal fin and remove saddle.

(5) Inspect upper attach fitting/hardware for correct installation IAW step 4. D., this section.

(6) Remove bolts securing pitch trim actuator rod ends to horizontal stabilizer attach fittings.

(7) Remove bolts securing lower end of actuator. Retain the two steel bushings from actuator lower attach lugs.

(8) Lay actuator back and disconnect electrical connectors.

(9) Remove actuator. See Table 201 for acceptable replacement actuator part numbers.

B. Installation – Pitch Trim Actuator (Figures 201 and 202)

NOTE: Refer to Table 201 for acceptable replacement actuator part numbers.

NOTE: SIMMONDS-PRECISION Actuator P/N DL5040M8 installation on SA226 AT and TC aircraft requires installation of 27K44063-001 Elevator Downspring Pulley Installation Kit.

NOTE: If installing an actuator without autopilot on aircraft with autopilot, cover open end of the aircraft autopilot connector to protect from moisture and tie back to harness to assure no interference with moving components.

(1) Actuator nominal lengths (from center to center of upper and lower attach points) are preset by the actuator manufacturer to the following dimensions:

(a) Fully Retracted - 13.79 inches (350.26 mm).

(b) Fully Extended - 20.25 inches (514.35 mm).

(2) Adjust actuator to same length of removed actuator by adjusting rod ends equally. To maintain adequate rod end thread engagement, do not exceed the following adjustment limits for installed actuator:

(a) P/N DL5040M6: Adjust actuator rod ends from preset nominal length no more than 5 turns in, and no more than 4 turns out.

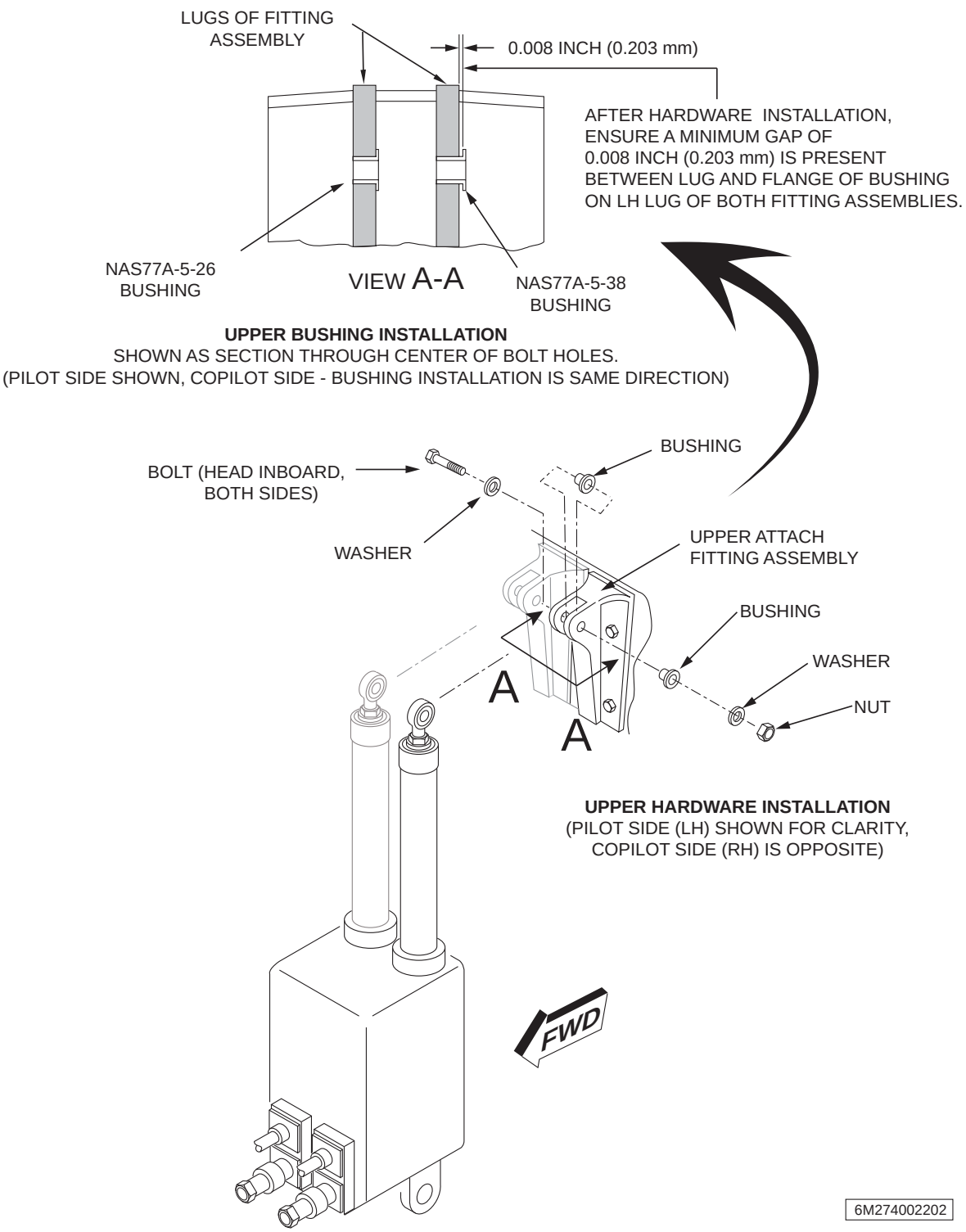
(b) P/N DL5040M8: Adjust actuator rod ends from preset nominal length no more than 3 turns in, and no more than 3 turns out.

(c) P/N 27-19008-006 and -007: Adjust actuator rod ends from preset nominal length no more than 3 turns in, and no more than 3 turns out.

(3) Inspect LH and RH upper and lower actuator attach fittings, bushings and attaching hardware for condition and security. Replace as required.

(4) Verify horizontal stabilizer is not binding or correct as necessary.

(5) Position actuator into lower attach fitting assembly lugs, rotate backwards and connect electrical connections.



Pitch Trim Actuator - Upper Hardware/Bushing Installation
Figure 202

(6) Insert steel bushings into lugs at bottom of actuator and align lugs with fitting lugs.

NOTE: Bolts securing actuator to lower fittings are installed with heads positioned outboard and nuts inboard.

(7) Insert bolts through lower end of actuator with a washer under bolt head and under nut, then install and tighten nuts.

NOTE: Bolts securing rod ends to upper fittings are installed with heads positioned inboard, i.e. head to head with washers and nuts installed outboard. Torque bolts from nut side only while holding bolt head stationary to prevent galling.

(8) Attach actuator rod ends to horizontal stabilizer upper fittings with bolts and nuts, with a washer under bolt head and a washer under nut. On left lug only of both fittings, ensure a minimum gap of 0.008 inch (0.203 mm) is present between fitting lug and flange of bushing as shown at top of Figure 202.

(9) Perform 2. A. Adjustment - Mechanical/Electrical Stops in section 27-40-01.

(10) Perform 3. A. Operational Test - Actuator Time Check in this section.

(11) Perform 2. D. Operational Test - Pitch Trim and Control in section 27-40-01.

(12) Perform 2. C. Operational Test - Indicating and Warning in section 27-41-03 or 27-42-03 as appropriate for airplane S/N.

(13) If test is unsatisfactory, perform step 2. A. Adjustment - Pitch Trim Indication and Warning Control Box in section 27-41-03 or 27-42-03 as appropriate for airplane S/N.

(14) Install saddle or access doors (as appropriate) on dorsal fin.

3. Adjustment/Test – Pitch Trim and Control

A. Operational Test – Actuator Time Check (Figure 203)

NOTE: Make written record of time check every 400 hours. Figure 203 is a sample form for recording actuator travel time.

(1) Clear tail of all personnel and equipment.

(2) Connect external electrical power to aircraft. (Refer to 24-40-00, page 201.)

(3) Ensure LH and RH Essential Bus voltages are 29.0 (+ 0.0 or -0.5).

NOTE: Prior to performing actuator travel time check, verify proper adjustment of horizontal stabilizer electrical stops (limit switches) IAW 2. A. Adjustment - Mechanical/Electrical Stops in section 27-40-01.

(4) Place TRIM SELECT switch on COPILOT.

(5) Using copilot control wheel trim switch, run horizontal stabilizer to full aircraft NOSE UP, leading edge of stabilizer full down.

NOTE: Travel time begins at the first trim in motion beep and ends at the last beep.

(6) Using copilot control wheel trim switch, run stabilizer to extreme aircraft NOSE DOWN (leading edge of stabilizer full up) and record time of travel. Verify travel time is 22 to 27 seconds.

(7) Using copilot control wheel trim switch, run stabilizer to extreme aircraft NOSE UP position (leading edge of stabilizer full down) and record time of travel. Verify travel time is:

(a) 22 to 27 seconds for P/N 27-19008 actuators.

(b) 18 to 27 seconds for P/N DL5040M6 actuators.

(c) 25 to 30 seconds for P/N DL5040M8 actuators.

(8) Place TRIM SELECT switch on PILOT.

(9) Using pilot control wheel trim switch, repeat steps (5) through (7).

PITCH TRIM ACTUATOR TRAVEL TIME CHECK

ACTUATOR SERIAL NUMBER: _____

INSTALLED IN AIRCRAFT S/N: _____

DATE OF TRAVEL TIME CHECK: _____

PILOT PITCH TRIM ACTUATOR TRAVEL TIME

Full NOSE UP to full NOSE DOWN _____ Seconds

Full NOSE DOWN to full NOSE UP _____ Seconds

Difference between down travel _____ Seconds
and up travel

COPILOT PITCH TRIM ACTUATOR TRAVEL TIME

Full NOSE UP to full NOSE DOWN _____ Seconds

Full NOSE DOWN to full NOSE UP _____ Seconds

Difference between down travel _____ Seconds
and up travel

7M274010209

Actuator Travel Time Check Sample Form
Figure 203

(10) Maximum allowable split between travel time from full up to full down and from full down to full up may not exceed four seconds on either side.

(11) If travel time is not within specified limits:

(a) Disconnect actuator rod of faster side (pilot or copilot).

(b) Check stabilizer travel time of the remaining connected actuator rod.

1 If travel time is now within limits, the actuator installation is suspect. Check actuator installation for misalignment and/or check horizontal stabilizer for binding.

(c) Disconnect other actuator rod end and check time required for each actuator rod to extend and retract fully.

1 If actuator rods meet travel time criteria, check actuator installation for misalignment.

2 If either actuator rod does not meet travel time criteria, replace the actuator (see Table 201).

(12) Disconnect external electrical power from aircraft.

4. Inspection/Check – Pitch Trim Actuator

A. Inspection – BARBER COLMAN Pitch Trim Actuators, P/N 27-19008-001, -002, -004 and -005:

(1) Perform step 2. A. Mechanical/Electrical Stops Adjustment, in section 27-40-01 and No-Back Functional Check in accordance with 3. INSTRUCTIONS section in Service Letter 226-SL-014 at the interval specified by Airframe Airworthiness Limitations Manual ST-UN-M002 and Section 05-10-00, Table 201 in this manual.

B. Inspection – SIMMONDS-PRECISION Pitch Trim Actuator, P/N DL5040M5, DL5040M6 and DL5040M8:

(1) Perform Freeplay Measurement Inspection in accordance with 3. INSTRUCTIONS section in Service Letter 226-SL-005 at the interval specified by Airframe Airworthiness Limitations Manual ST-UN-M002 and Section 05-10-00, Table 201 in this manual.

C. Actuator Travel Time Check - All Actuators:

(1) Perform 3. A. Operational Test - Actuator Time Check in this section every 400 hours.

D. Inspection - Actuator Upper Attach Fitting/Hardware

(1) Refer to SA226-SL-031 for correct hardware installation at upper attach fitting.

(2) If upper attach fitting hardware is found to be installed incorrectly, perform procedures in SA226-SL-031, step 4. INSTRUCTION.

E. Inspection - Actuator Upper Attach Fittings

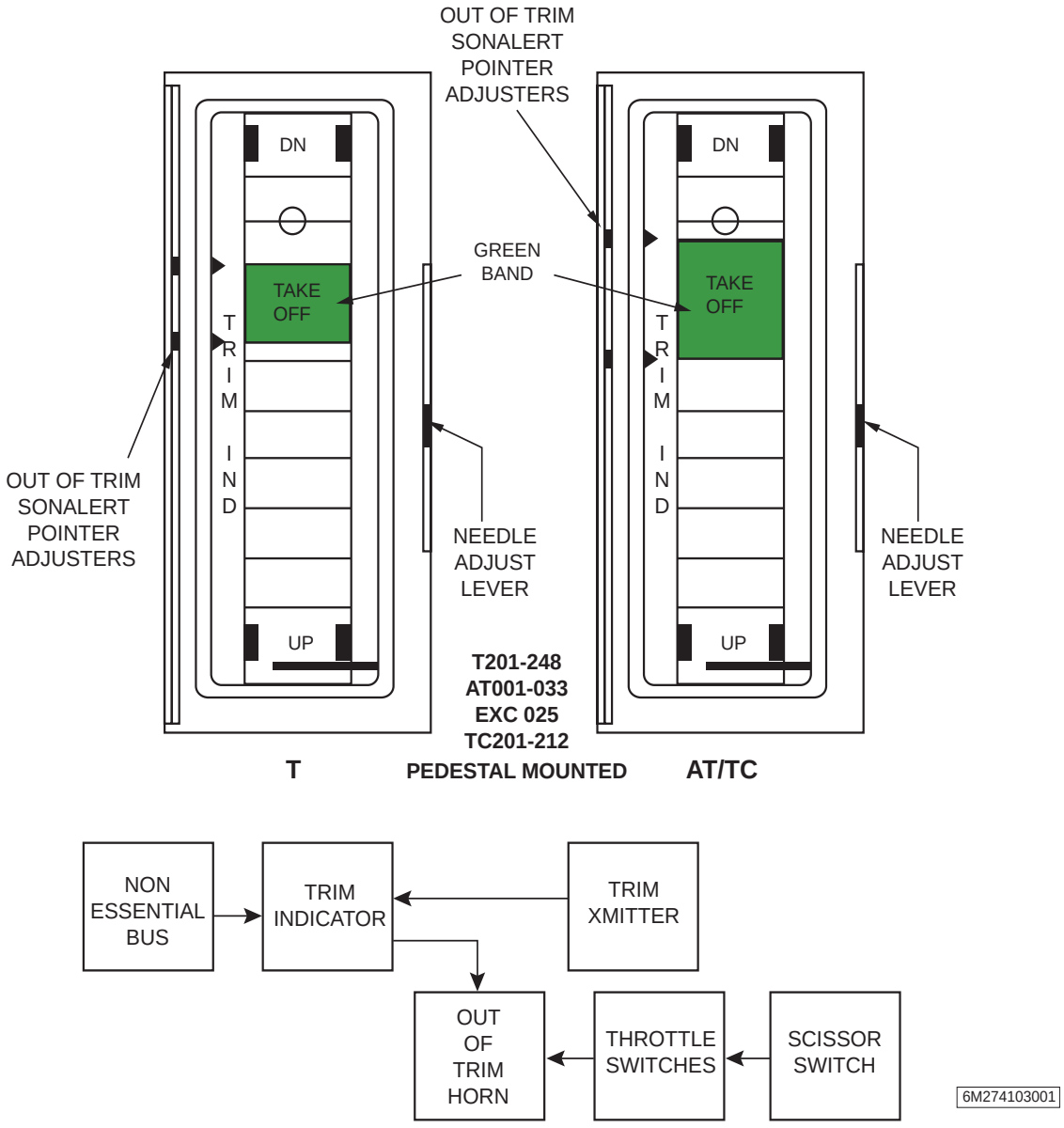
(1) Refer to Service Bulletin 226-27-081 for required repetitive inspections.

PITCH TRIM INDICATING AND WARNING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The pitch trim indicator system includes a vertical scale trim indicator and trim indicating and warning circuit board assembly located in the pedestal, and a position transmitter (potentiometer) located on the limit switch assembly inside the vertical stabilizer.

(1) The trim indicator is mounted on the cockpit pedestal and the circuit board assembly is mounted inside the pedestal on the left side.



Pitch Trim Indicating and Warning System
Figure 1

(2) The limit switch assembly incorporates the position transmitter and is located in the vertical stabilizer. The switches are activated by a push-pull rod attached at the base to the horizontal stabilizer aft spar fitting and at the top to the arm of a cam located within the switch assembly.

(3) The position transmitter is attached to the cam and rotates as the cam rotates. The transmitter signals the position of the horizontal stabilizer to the indicator.

(4) When the power levers are advanced forward for takeoff, if the horizontal stabilizer is not in the takeoff position, a pitch out of trim warning sonalert sounds in the cockpit. The inboard left main landing gear scissor switch disengages the sonalert during flight.

PITCH TRIM INDICATING AND WARNING SYSTEM – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Indicator and horn inoperative.	Circuit breaker tripped or defective.	Reset or replace circuit breaker.
	Defective wiring between circuit breaker and pedestal connector (P2/J2).	Check wiring and plug and repair or replace as necessary.
	Defective wiring between pedestal connector (P2/J2) and turret terminal board (TB1).	Check power on turret board through 510 ohm resistor and zener diode (see Wiring Diagram Manual, 27-40-03).
	Defective wiring between turret terminal board (TB1) and indicator (M22).	Check wiring from turret board indicator.
	Indicator (M22) defective.	Replace indicator.
	Defective wiring at position transmitter (PT5) in limit switch assembly.	Repair wiring or replace potentiometer.
Indicator only inoperative.	Indicator (M22) defective.	Replace indicator.
Horn inoperative or continuously sounding.	Power lever switches (S85/S86) out of adjustment or defective, or broken wiring on switches.	Adjust or replace switches or repair switch wiring.
	Indicator indices out of adjustment, indicator (M22) defective.	Adjust indices at indicator or replace indicator as necessary.
	Scissor switch (S302) out of adjustment or defective, or defective wiring.	Adjust or replace switch and/or repair wiring.
	Defective wiring from indicator (M22) to horn (H4).	Check wiring and repair or replace as necessary.
	Defective horn (H4).	Replace horn.
Indicator fluctuating, unstable, or erratic.	Defective position transmitter (PT5) in limit switch assembly.	Replace transmitter.
	Defective wiring between position transmitter (PT5) and indicator (M22).	Repair or replace wiring as necessary.
	Defective indicator (M22).	Replace indicator.

PITCH TRIM INDICATING AND WARNING SYSTEM – MAINTENANCE PRACTICES

1. Removal / Installation – Pitch Trim Indicating and Warning System

A. Removal – Indicator (Figure 201)

(1) Remove screws securing indicator cover and remove cover.

(2) Remove screws through four shock mounts securing indicator to pedestal.

(3) Remove indicator and disconnect electrical connector.

(4) Remove shock mounts from indicator brackets.

(5) Remove screws attaching brackets to indicator.

(6) Separate brackets and remove indicator. Note position of brackets for reassembly.

B. Installation – Indicator (Figure 201)

(1) Position brackets around indicator and secure with screws.

(2) Insert four shock mounts into brackets.

(3) Connect electrical connector.

(4) Position indicator into pedestal.

(5) Insert screws through shock mounts and secure indicator into place.

(6) Position indicator cover and secure with screws.

(7) Perform 2. A. Adjustment and Test – Pitch Trim Indicator/Warning Circuitry, this section.

(8) Perform 2. C. Operational Test – Indicating and Warning System, this section.

C. Removal – Positioning Transmitter (Potentiometer) (Figure 202)

(1) Remove panel on left of vertical stabilizer just above horizontal stabilizer (see 27-40-01, Figure 2).

(2) Loosen set screws securing cam to potentiometer shaft.

(3) Disconnect push-pull rod from switch cam assembly and slide cam assembly off shaft.

(4) Remove potentiometer retaining nut and washer.

(5) Withdraw potentiometer from mount bracket.

D. Installation – Positioning Transmitter (Potentiometer) (Figure 202)

(1) Position potentiometer into mount bracket.

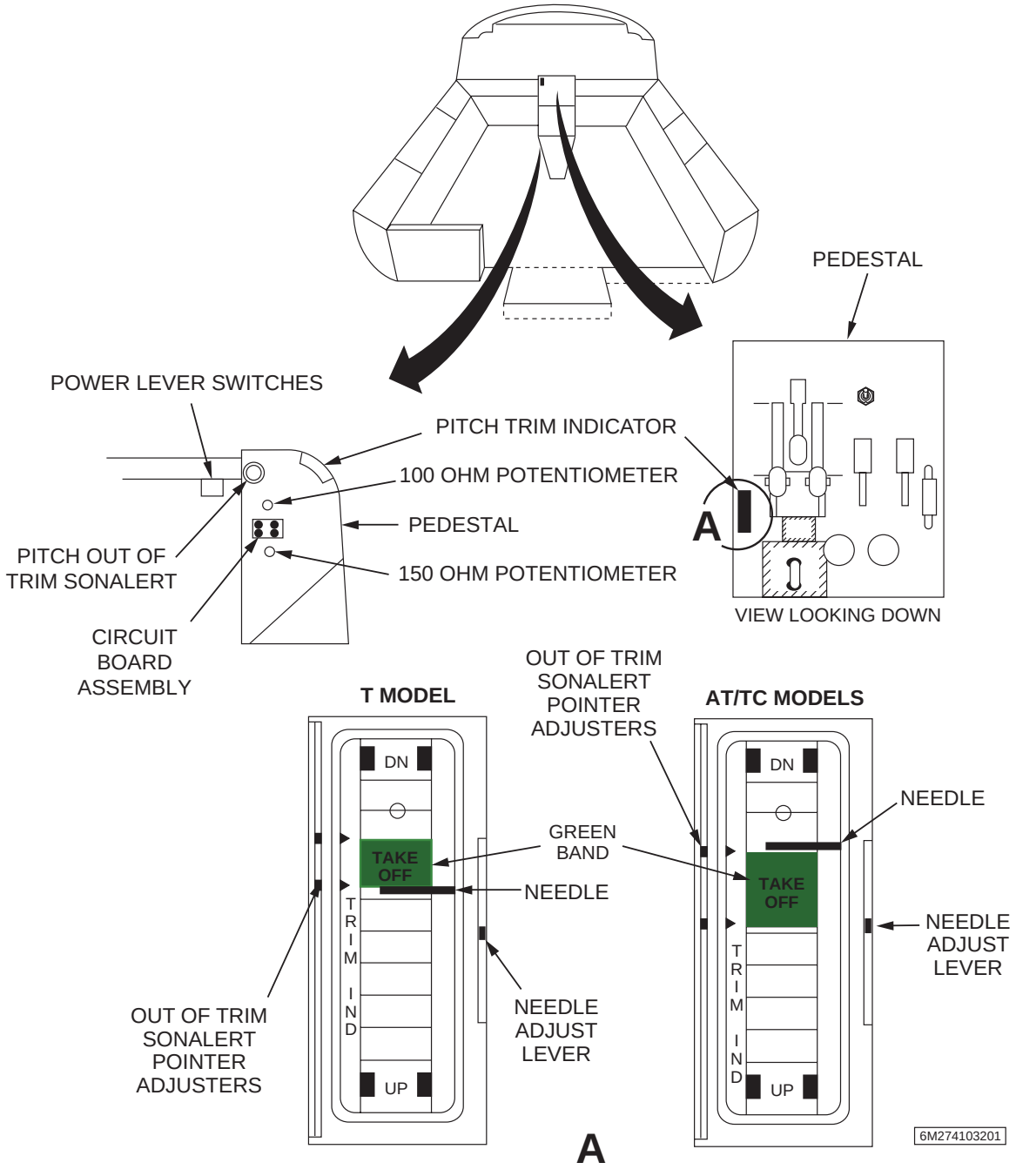
(2) Secure potentiometer to bracket with washer and retaining nut.

(3) Slide cam assembly onto potentiometer shaft and secure to shaft with set screws.

(4) Connect push-pull rod to switch cam assembly.

(5) Perform 2. A. Adjustment and Test – Pitch Trim Indicator/Warning Circuitry, this section.

(6) Perform 2. D. Operational Test - Pitch Trim and Control, in section 27-40-01.



Pitch Trim Indicating And Warning System - Removal/Installation
Figure 201

2. Adjustment/Test – Pitch Trim Indication and Warning System

A. Adjustment and Test – Pitch Trim Indication/Warning Circuitry (Figures 201 and 202)

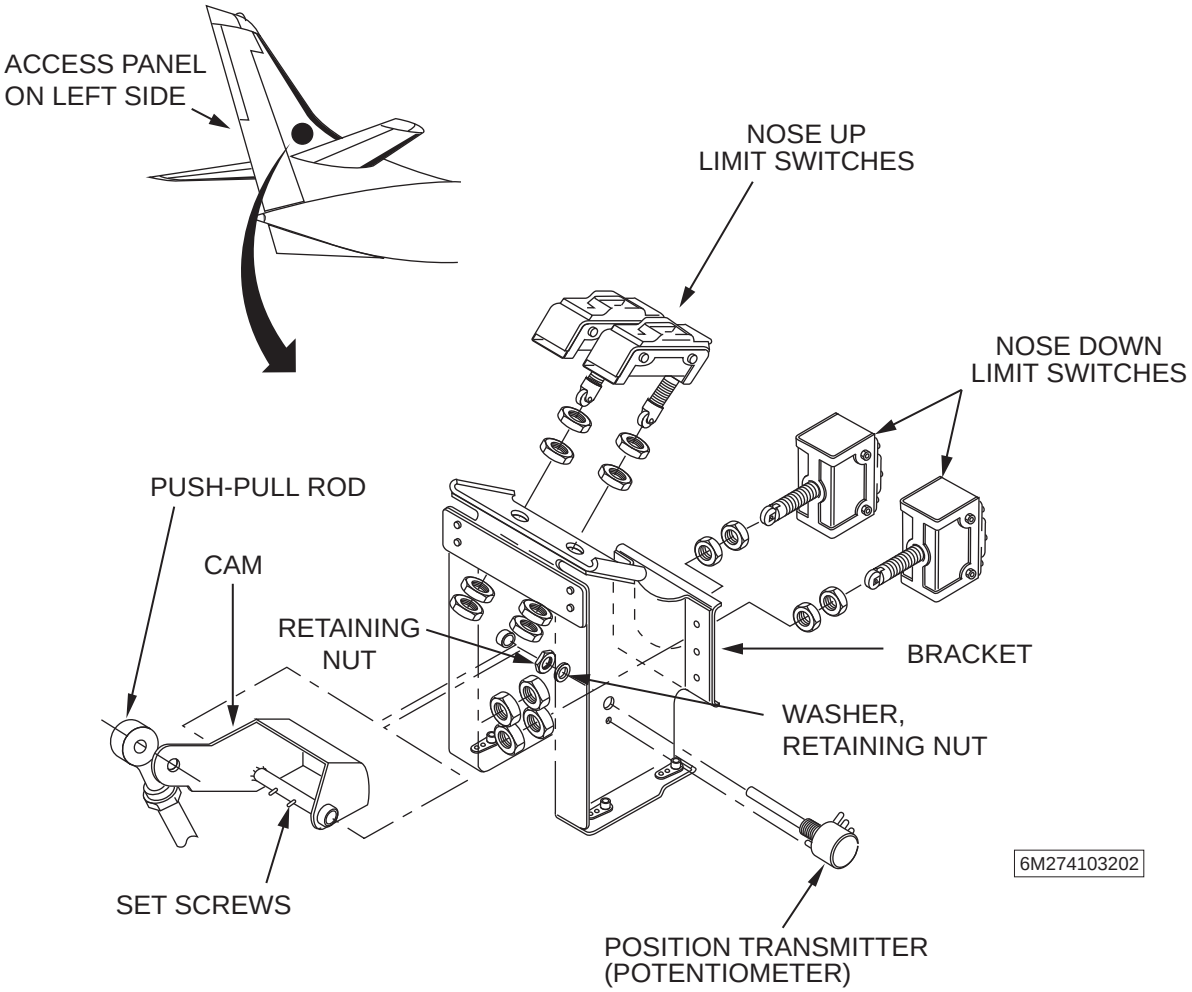
NOTE: Prior to adjusting the pitch trim indication and warning circuitry, adjust the horizontal stabilizer mechanical and electrical stops (see 27-40-01).

(1) Apply external electrical power to aircraft. Ensure tail of aircraft is clear of personnel and equipment.

(2) Run pitch trim to full aircraft NOSE UP position, stabilizer leading edge full down.

(3) Select batteries OFF.

(4) Adjust position transmitter (potentiometer) in vertical stabilizer to obtain a reading of 10 - 15 ohms across pins A and C on right rear pedesta connector P2 (see Wiring Diagram Manual 27-40-03).



Pitch Trim Indicating And Warning System - Adjustment
Figure 202

(5) Tighten position transmitter (potentiometer) retaining nut and connect P2 to J2 at right rear pedestal.

(6) On pedestal mounted pitch trim indicator, move needle adjust lever on right side of indicator until needle is on bottom of UP mark.

(7) Select battery switches ON.

(8) Using pilot trim control switch, run horizontal stabilizer to extreme aircraft NOSE DOWN position, stabilizer leading edge full up.

(9) Below indicator on left side of pedestal, locate two 3/8 inch indication potentiometer adjustment access holes.

(10) Using common screwdriver through lower hole, adjust 150 ohm potentiometer until indicator needle is on nose down mark.

(11) Run horizontal stabilizer to extreme NOSE UP position, stabilizer leading edge full down.

(12) Through upper access hole on pedestal, adjust 100 ohm potentiometer until indicator needle is on nose up mark.

(13) Repeat above procedure until needle is within the up and down marks on indicator when horizontal stabilizer is on extreme up and down limits, and needle is within takeoff band on indicator when horizontal stabilizer is within takeoff band painted on dorsal fin.

NOTE: Because of interaction between potentiometers, it may be necessary to repeat procedure several times.

(14) Push both power levers on pedestal full forward.

(15) On left of indicator locate red and green out-of-trim horn pointer adjustors.

(16) Adjust each pointer individually until horn does not sound when indicator needle is within the green takeoff band on indicator, and does sound anywhere outside the green takeoff band.

B. Adjustment – Power Lever Switches (S85/S86)

NOTE: Prior to adjusting pitch out of trim power lever switches, adjust landing gear warning power lever switches in accordance with Section 32-60-00.

(1) Set power levers at approximately half the travel between flight idle gate and full forward.

(2) Loosen screws in slotted ends of switches and push switches laterally against switch actuator rods until switch actuators are fully depressed.

(3) Move power levers forward and aft. Switches should actuate approximately half the travel between full forward and flight idle gate when the power levers are pulled back.

(4) If switches are installed on vertical bracket, loosen screws attaching bracket to plate. Adjust bracket forward or aft as necessary. If switches are attached directly to plate move switches forward and aft as necessary.

(5) Tighten all switch screws.

(6) Connect external electrical power to aircraft.

(7) With horizontal stabilizer positioned outside of takeoff band, move power levers forward and aft. Adjust switches as necessary until pitch out of trim warning signal sounds at approximately half the travel between full forward and flight idle gate when the power levers are pulled back.

(8) Disconnect external electrical power from aircraft.

C. Operational Test – Indicating and Warning System

(1) Apply external electrical power to aircraft. Ensure tail of aircraft is clear of personnel and equipment.

(2) Select battery switches ON.

(3) Place trim select switch on PILOT.

(4) Using pilot control wheel trim switch run horizontal stabilizer to extreme NOSE UP position. Observe pitch trim indicator needle as it moves toward nose up position. Needle should move smoothly and evenly and stop within nose up mark.

(5) Run pitch trim toward aircraft NOSE DOWN position. Stop indicator needle at zero degrees mark.

(6) At horizontal stabilizer, check to ensure lower edge of stabilizer is near zero degree rivet on vertical stabilizer (Rivet is located immediately aft of rub strip on left side of dorsal fin).

(7) Set power levers at approximately half the travel between flight idle gate and full forward.

(8) Verify pitch out of trim sonalert sounds.

(9) Depress inboard scissor switch on left main landing gear. Verify sonalert goes off.

(10) Release pressure off scissor switch. Verify sonalert sounds.

(11) Pull one power lever back off of 50% position. Verify sonalert goes off.

(12) Push same power lever back up to 50% position. Verify sonalert sounds.

(13) Repeat steps (11) and (12) with opposite power lever.

(14) Run pitch trim toward aircraft NOSE UP position and stop as soon as indicator needle reaches nose down edge of green TAKEOFF band. Verify leading edge of stabilizer is at nose down end of takeoff band marking painted on dorsal fairing.

(15) Run pitch trim toward aircraft NOSE UP position and stop as soon as indicator needle reaches nose up edge of TAKEOFF band. Verify leading edge of stabilizer is at nose up end of takeoff band marking painted on dorsal fairing.

(16) Run pitch trim in and out of both ends of green TAKEOFF band. Verify sonalert remains off while needle is inside green band and sounds when needle exits green band.

(17) Return horizontal stabilizer to TAKEOFF position.

(18) Select battery switches OFF.

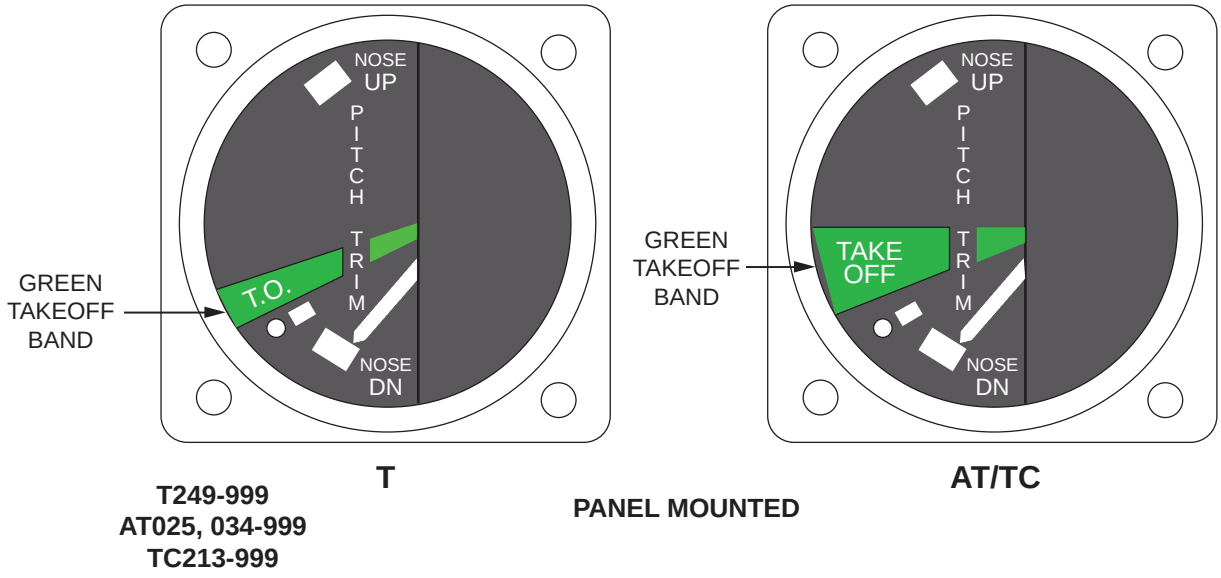
(19) Disconnect external electrical power.

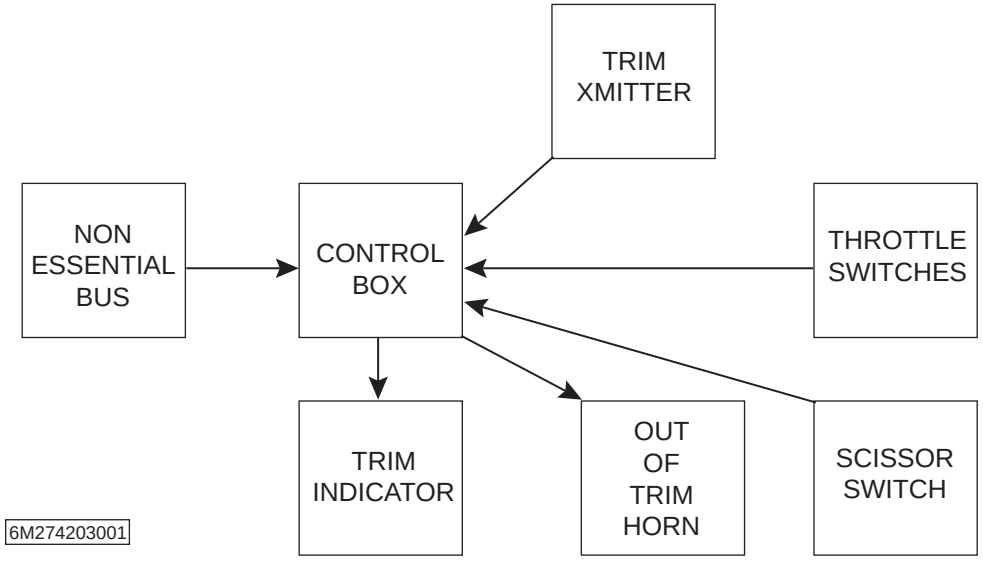
PITCH TRIM INDICATING AND WARNING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The pitch trim indicator system includes an instrument panel mounted indicator, a trim indicating and warning control box located in right console, and a position transmitter located in the limit switch assembly inside vertical stabilizer.

(1) The trim indicator is mounted on the pilot flight instrument panel and the circuit board assembly control box is mounted forward inside the right console.





Pitch Trim Indicating and Warning System Figure 1

PITCH TRIM INDICATING AND WARNING SYSTEM – DESCRIPTION AND OPERATION

(2) The limit switch assembly incorporates the position transmitter and is located in the vertical stabilizer. The switch assembly is activated by a push-pull rod attached at the base to the aft spar fitting and at the top to the switch actuating paddle located within the switch assembly.

(3) The position transmitter is attached to the switch paddle and rotates as the paddle moves. The transmitter signals the position of the horizontal stabilizer through the indicating and warning control box and on to the pitch trim indicator.

(4) When power is applied to the power levers at takeoff and the horizontal stabilizer is not in the takeoff position, a warning horn sounds in the cockpit. The inboard scissor switch on left main landing gear disengages the horn when the aircraft is in flight.

PITCH TRIM INDICATING AND WARNING SYSTEM – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Indicator and horn inoperative.	Circuit breaker tripped or defective.	Reset or replace circuit breaker.
	Defective wiring between circuit breaker and pitch trim indication/warning control box (E75).	Check wiring and plug and repair or replace as necessary.
Indicator only inoperative.	Defective wiring between indicator (M22) and pitch trim indication and warning control box (E75).	Check wiring and repair or replace.
	Loose wire connections at terminals behind indicator (M22).	Tighten terminals at back of indicator.
	Indicator (M22) defective.	Replace indicator.
	Control box (E75) defective.	Replace control box.
Horn inoperative or continuously sounding.	Defective wiring between control box (E75) and horn (H4).	Check wiring and repair or replace as necessary.
	Defective control box (E75).	Replace control box.
	Defective horn (H4).	Replace horn.
	Power lever switches (S85/S86) out of adjustment or defective, or broken wiring on switches.	Adjust or replace switches, or repair switch wiring.
	Scissor switch (S302) out of adjustment or defective, or wiring defective.	Adjust or replace switch and/or repair wiring.
	Defective wiring between scissor switch and small connector (P2/J2) behind pedestal.	Check wiring and plug and repair or replace as necessary.
	Check diode jumper at pedestal connector (J2).	Replace diode or jumper as necessary. (see Wiring Diagram Manual, Section 27-40-03)

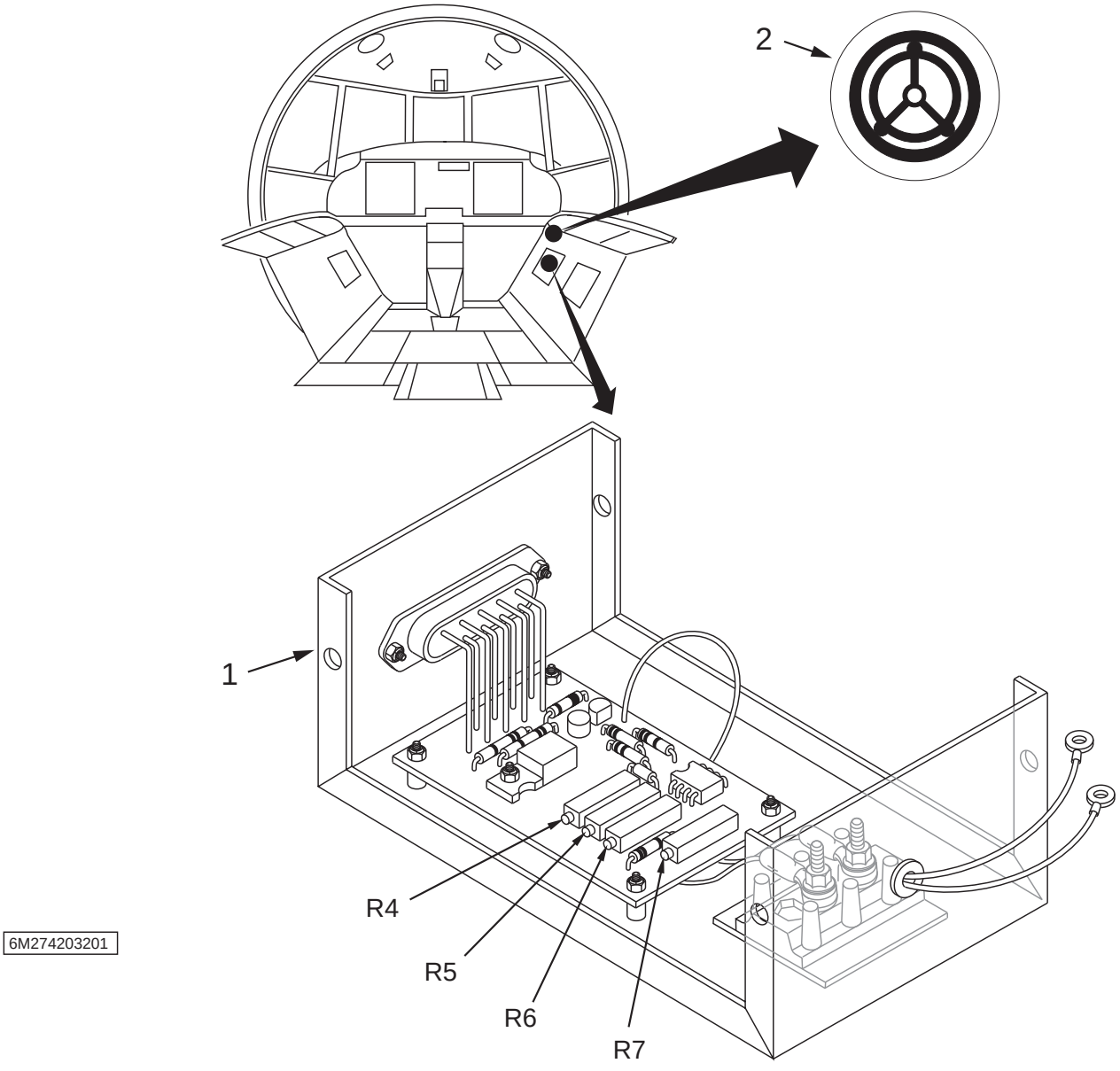
DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Indicator fluctuating, unstable, or erratic.	Indicator (M22) defective.	Replace indicator.
	Loose wire connections at terminals behind indicator (M22).	Tighten terminals at back of indicator.
	Defective wiring at position transmitter (PT5) in limit switch assembly.	Repair or replace wiring as necessary.
	Defective wiring between position transmitter (PT5) and control box.	Repair or replace wiring as necessary.
	Defective position transmitter (PT5).	Replace transmitter.
	Defective wiring between indicator (M22) and control box (E75).	Repair or replace wiring as necessary.

1. Removal/Installation – Pitch Trim Indicating and Warning System

A. Removal – Pitch Trim Indication and Warning Control Box and Horn (Figure 201)

(1) In flight compartment, remove forward right console door.

(2) Remove pitch trim indicating and warning control box by removing two screws attaching cover (1) to base.



Pitch Trim Indication and Warning Control box
Figure 201

(3) Disconnect electrical connector (P24) from cover.

(4) Remove ring-nut on warning horn (2) located at forward right console.

(5) Disconnect wires from horn and remove horn (2).

B. Installation – Pitch Trim Indication and Warning Control Box and Horn (Figure 201)

(1) Connect horn wires to horn (2).

(2) Position horn and install ring-nut finger tight.

(3) Connect electrical connector (P24) to cover.

(4) Perform 2. A. Adjustment and Test – Pitch Trim Indication/Warning Circuitry, this section.

(5) Perform 2. C. Operational Test – Indicating and Warning System, this section.

(6) Install control box cover (1) and console door.

C. Removal – Indicator

(1) Remove post light and screws securing indicator to instrument panel.

NOTE: Depending on aircraft, it may be necessary to remove pilot flight instrument panel or an adjacent instrument prior to pitch trim indicator removal.

(2) Remove indicator from panel.

(3) Disconnect electrical connections.

D. Installation – Indicator

(1) Connect electrical connections.

(2) Position indicator into panel.

(3) Install post light and screws securing indicator to instrument panel.

(4) Perform 2. C. Operational Test – Indicating and Warning System, this section.

E. Removal – Positioning Transmitter (Potentiometer) (Figure 202)

(1) Remove panel on left side of vertical stabilizer just above horizontal stabilizer (see 27-40-01, Figure 2).

(2) Loosen set screws securing switch paddle to potentiometer shaft.

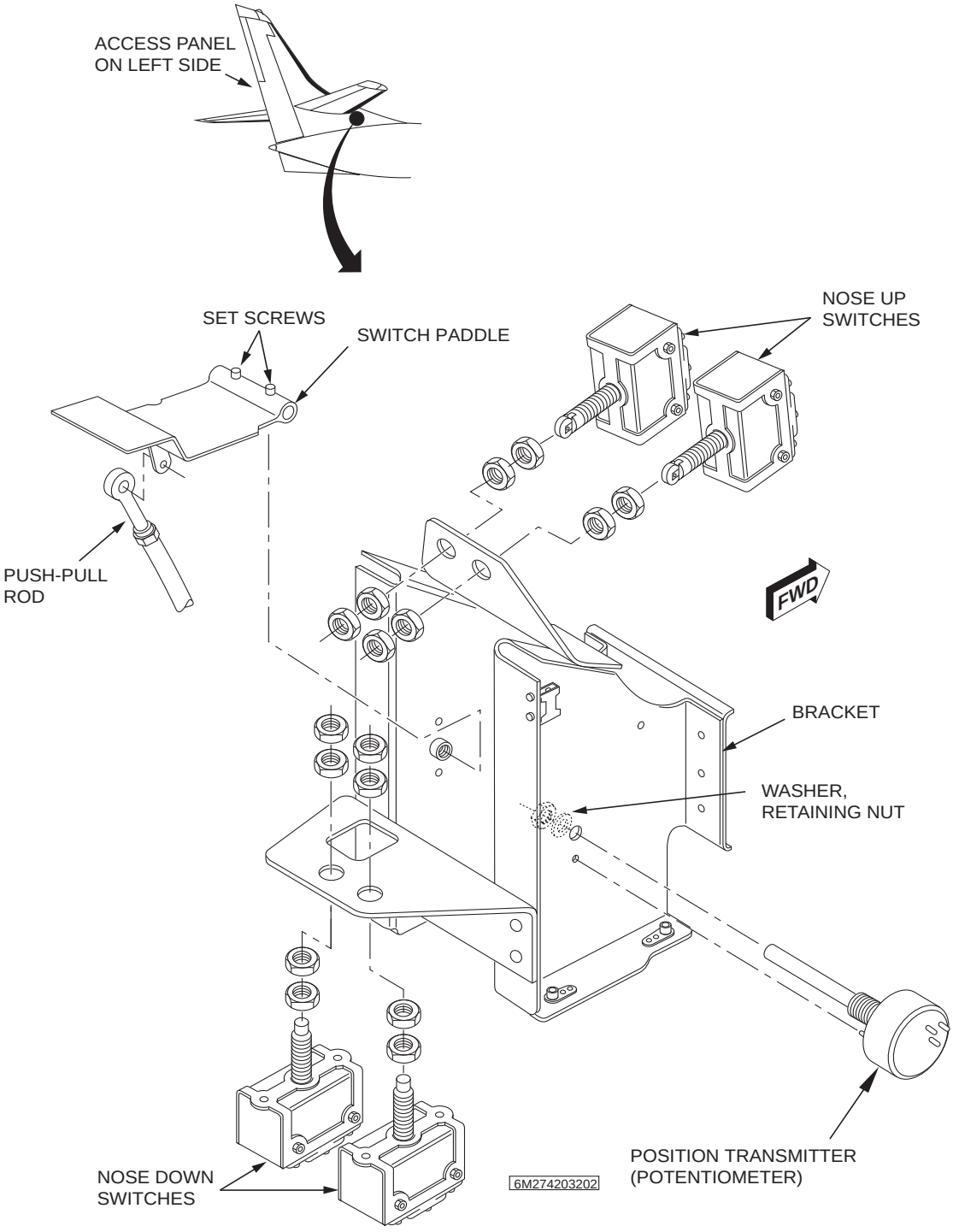
(3) Disconnect push-pull rod from switch paddle assembly and slide switch paddle off shaft.

(4) Remove potentiometer retaining nut and washer.

(5) Withdraw potentiometer from mount bracket.

F. Installation – Positioning Transmitter (Potentiometer) (Figure 202)

(1) Insert potentiometer shaft through switch bracket, washer, retaining nut and switch paddle.



Pitch Trim Indicating System - Position Transmitter
Figure 202

(2) Tighten retaining nut to secure potentiometer to bracket.

(3) Tighten set screws to secure switch paddle to potentiometer shaft.

(4) Connect push-pull rod to switch paddle assembly.

(5) Perform 2. A. Adjustment and Test – Pitch Trim Indicator/Warning Circuitry, this section.

(6) Perform 2. C. Operational Test - Pitch Trim and Control, in section 27-40-01.

2. Adjustment/Test – Pitch Trim Indicator and Warning Circuitry

A. Adjustment and Test – Pitch Trim Indication/Warning Circuitry

NOTE: Prior to adjusting the pitch trim indicator and warning circuitry, adjust the horizontal stabilizer limit switches (see 27-40-01).

Prior to adjusting control box check resistance of position transmitter across pins 1 and 4 of control box connector P24. Measurement should be 100 ohms at full nose down position.

(1) Remove forward right console door.

(2) Connect external electrical power to aircraft. Ensure tail of aircraft is clear of personnel and equipment.

(3) Adjust pitch trim position transmitter:

(a) Select battery switches ON.

(b) Run horizontal stabilizer to full aircraft NOSE DOWN limit, leading edge of stabilizer full up.

(c) Select battery switches OFF.

(d) Disconnect plug P24 from pitch trim indication/warning control box.

(e) Connect multimeter leads to pins 1 and 4 of P24 and set multimeter to OHMS scale.

(f) Loosen set screws securing paddle to position transmitter (potentiometer) shaft (see 27-40-01, Figure 202).

(g) With a small common screwdriver, adjust potentiometer to 100 OHMS.

(h) Tighten paddle set screws and verify 100 OHMS setting. Readjust as necessary to obtain 100 OHMS.

(i) Disconnect multimeter and reconnect P24 to pitch trim indication/warning control box.

(4) Remove pitch trim indicating and warning control by removing two screws attaching cover to base and removing cover (Circuitry is attached to cover).

(5) Place cover in a position for easy access to potentiometer adjustment circuit board.

(6) Select battery switches ON.

(7) Using pilot or copilot trim control switch, run pitch trim until leading edge of stabilizer is at nose down limit of TAKEOFF band painted on dorsal fairing.

(8) In control cover adjust R6 (see Figure 201) until needle of trim indicator is on the nose down limit of green TAKEOFF band.

(9) Run horizontal stabilizer to nose up end of TAKEOFF band on dorsal fairing.

(10) Adjust R7 until needle of trim indicator is on the nose up limit of green TAKEOFF band.

(11) Push both power levers on pedestal full forward and run pitch trim up and down to achieve the following adjustments:

(a) Adjust R4 so that when needle passes out of nose up end of green takeoff band, warning horn sounds.

(b) Adjust R5 so that when needle passes out of nose down end of green takeoff band, warning horn sounds.

NOTE: Horn must sound when stabilizer is out of takeoff band and not sound when stabilizer is within takeoff band.

(12) Perform 2. C. Operational Test - Indicating and Warning System, this section.

(13) Install control box cover and console door.

(14) Disconnect external electrical power.

B. Adjustment – Power Lever Switches (S85/S86)

NOTE: Prior to adjusting pitch out of trim power lever switches, adjust landing gear warning power lever switches in accordance with Section 32-60-00.

(1) Set power levers at approximately half the travel between flight idle gate and full forward.

(2) Loosen screws in slotted ends of switches and push switches laterally against switch actuator rods until switch actuators are fully depressed.

(3) Move power levers forward and aft. Switches should actuate approximately half the travel between full forward and flight idle gate when the power levers are pulled back.

(4) If switches are installed on vertical bracket, loosen screws attaching bracket to plate. Adjust bracket forward or aft as necessary. If switches are attached directly to plate move switches forward and aft as necessary.

(5) Tighten all switch screws.

(6) Connect external electrical power to aircraft.

(7) With horizontal stabilizer positioned outside of TAKEOFF band, move power levers forward and aft. Adjust switches as necessary until pitch out of trim warning signal sounds at approximately half the travel between full forward and flight idle gate when the power levers are pulled back.

(8) Disconnect external electrical power from aircraft.

C. Operational Test – Indicating and Warning System

(1) Apply external electrical power to aircraft. Ensure tail of aircraft is clear of personnel and equipment.

(2) Select battery switches ON.

(3) Position TRIM SELECT switch on pedestal to PILOT.

(4) Using pilot control wheel trim switch, run pitch trim to extreme NOSE UP position. Observe pitch trim indicator needle as it moves toward NOSE UP position. Indicator needle should move smoothly and evenly and stop within NOSE UP mark.

(5) Run pitch trim toward aircraft NOSE DN position. Stop indicator needle at white index mark (zero degrees mark) below green TAKEOFF band.

(6) Verify lower edge of horizontal stabilizer is aligned with zero degree rivet on dorsal fin. Rivet is located immediately aft of rub strip on left side of dorsal fin.

(7) Run pitch trim to align centerline of horizontal stabilizer leading edge with nose down limit of TAKEOFF band marking on dorsal fin. Verify pitch trim indicator needle is on the nose down limit of green TAKEOFF band.

(8) Advance power levers beyond half travel between FLIGHT IDLE and HIGH.

(9) Run horizontal stabilizer towards aircraft NOSE DN and verify out-of-trim sonalert sounds when pitch trim needle passes outside green TAKEOFF band.

(10) Retard power levers to FLIGHT IDLE and verify sonalert stops sounding.

(11) Advance power levers beyond half travel between FLIGHT IDLE and HIGH to get sonalert sounding again.

(12) Run pitch trim towards aircraft NOSE UP and verify out-of-trim sonalert stops sounding as soon as pitch trim needle passes inside green TAKEOFF band.

(13) Run pitch trim to align centerline of horizontal stabilizer leading edge with nose up limit of TAKEOFF band marking on dorsal fin. Verify pitch trim indicator needle is on the nose up limit of green TAKEOFF band.

(14) Run pitch trim towards aircraft NOSE UP and verify out-of-trim sonalert sounds when pitch trim needle passes outside green TAKEOFF band.

(15) Retard power levers to FLIGHT IDLE and verify sonalert stops sounding.

(16) Advance power levers beyond half travel between FLIGHT IDLE and HIGH to get sonalert sounding again.

(17) Depress inboard scissor switch on left main landing gear. Verify sonalert goes off.

(18) Release pressure off scissor switch. Verify sonalert sounds.

(19) Pull one power lever back off of 50% position. Verify sonalert goes off.

(20) Push same power lever back up to 50% position. Verify sonalert sounds.

(21) Repeat steps (19) and (20) with opposite power lever.

(22) Run pitch trim towards aircraft NOSE DN and verify out-of-trim sonalert stops sounding as soon as pitch trim needle passes inside green TAKEOFF band.

(23) Retard power levers to FLIGHT IDLE.

(24) If this operational test reveals any discrepancy with pitch trim indication or out of trim warning, perform 3. F. Adjustment – Pitch Trim Indication and Warning Control Box, in this section.

(25) Disconnect external electrical power from aircraft.

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FLAP SYSTEM – DESCRIPTION AND OPERATION

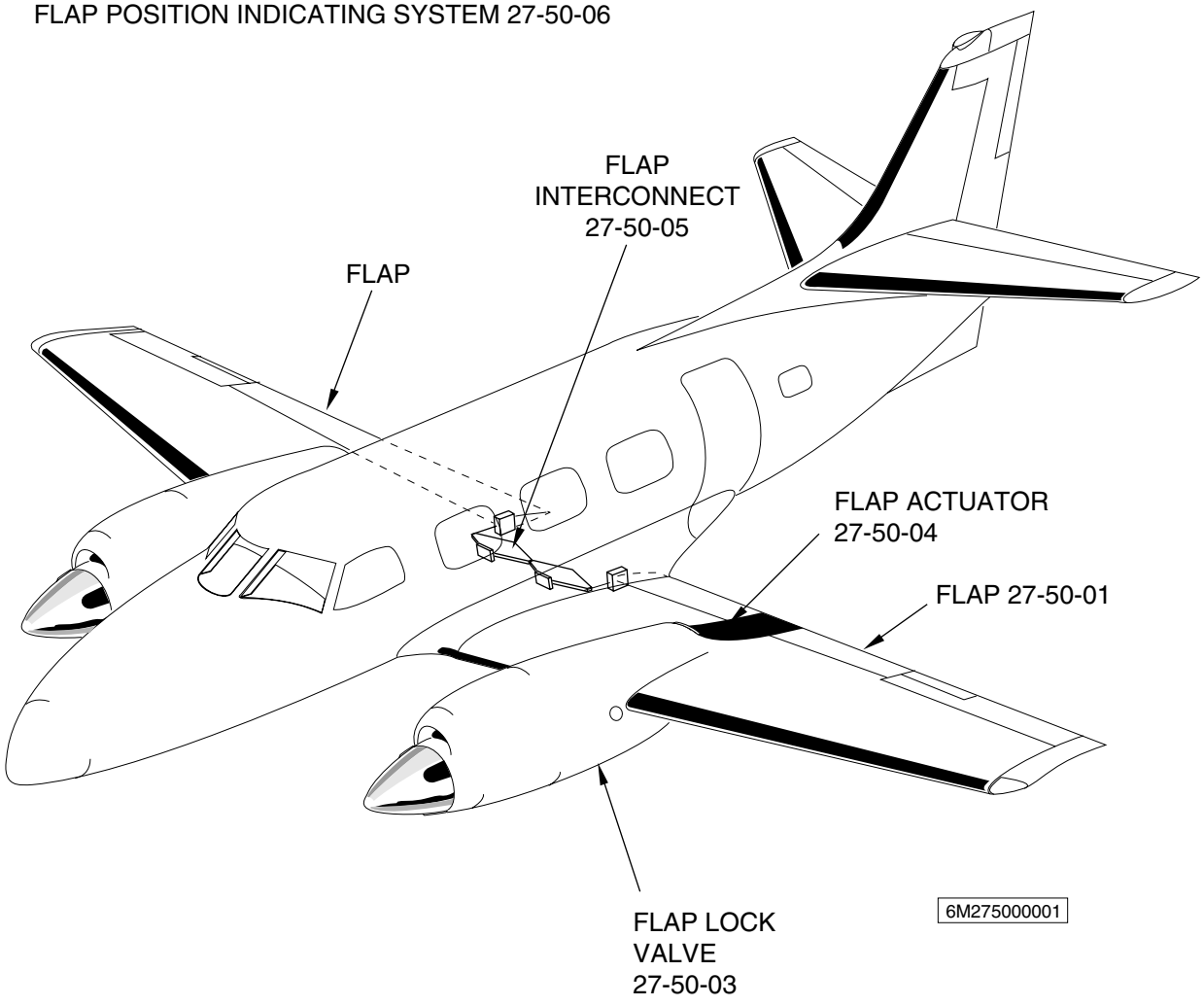
1. General

A. The two wing flaps are electrically controlled and hydraulically actuated. The two flaps are mechanically interconnected to ensure symmetrical flap operation.

B. Flap position is selected by a control lever located on the upper side of the cockpit pedestal.

C. Electrical power from the nonessential bus to a flap selector valve directs hydraulic pressure to the flap actuators. The actuators extend or retract the flaps.

FLAP CONTROL STSTEM 27-50-02
FLAP POSITION INDICATING SYSTEM 27-50-06

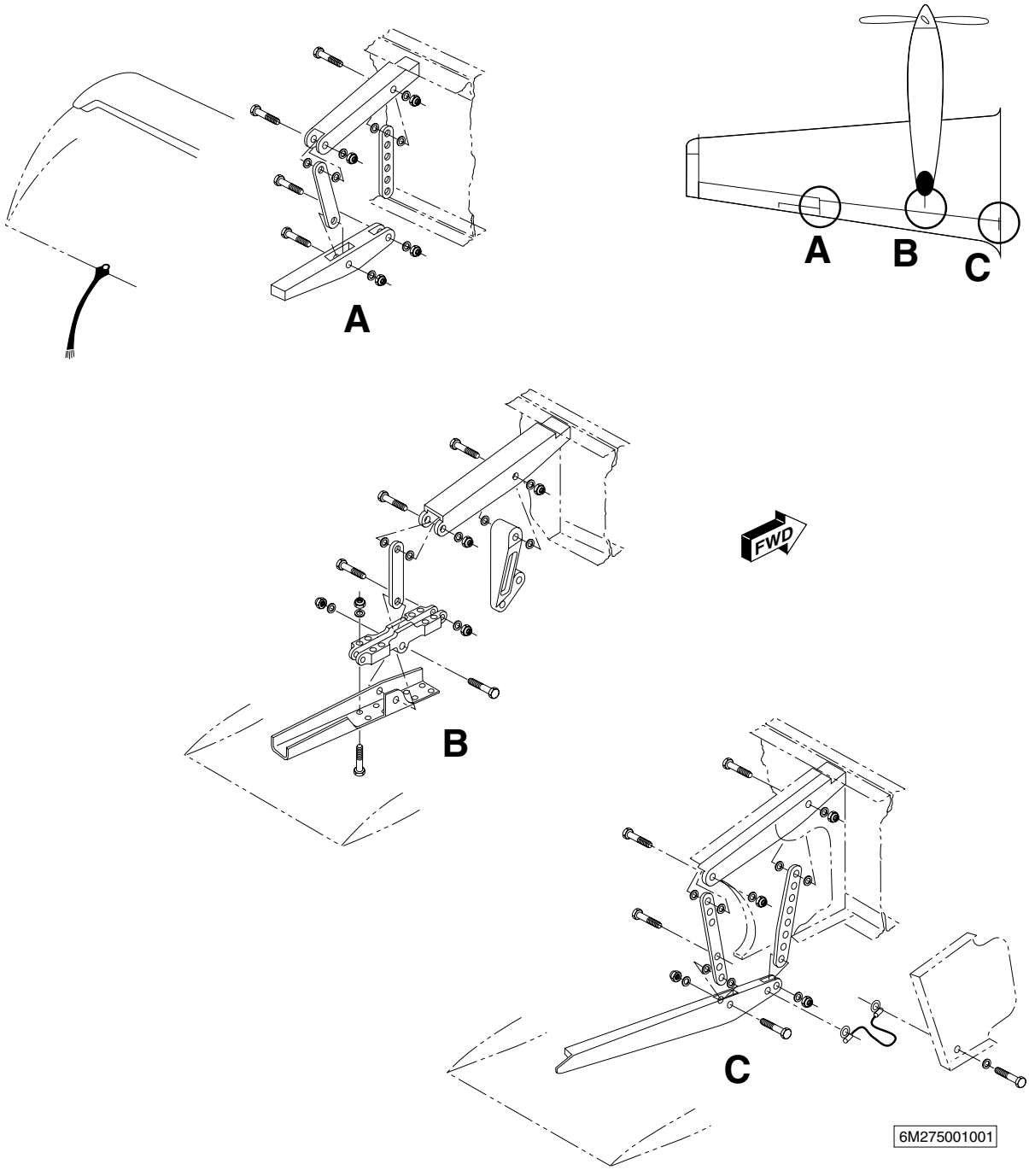


Flap System
Figure 1

FLAPS – DESCRIPTION AND OPERATION

1. General

A. The flaps are manufactured of aluminum skin riveted on a frame of spars and ribs. Each flap is attached to the wing trailing edge at three hinge points. The flap is controlled by a lever on the cockpit pedestal.



Flaps
Figure 1

1. Servicing – Flaps

A. The only maintenance required of the flaps is lubrication of the flap actuator, see Section 27-50-04.

2. Removal/Installation – Flaps

NOTE: When detaching flap channels from flap links, identify locations of washers and spacers to ensure correct washer locations when reinstalling flaps. AN960 series washers go beneath the nut, while 27-32009 series washers fit on either side of the link.

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

CAUTION: WHENEVER ANY HYDRAULIC COMPONENT OF THE FLAP SYSTEM IS REMOVED OR REPLACED, THE FLAP WALKING BEAM CENTERBOLT SHOULD BE REMOVED PRIOR TO PRESSURE BEING APPLIED. AFTER INITIAL OPERATION, REINSTALL THE CENTERBOLT.

A. Removal – Flap

(1) Extend flaps.

(2) Remove access panel located at lower fuselage-to-wing trailing edge.

(3) Support flap.

(4) At center hinge, remove cotter pin, nut, spacers, and bolt connecting flap channel to actuating (forward) link.

(5) At inboard hinge.

(a) Through hole in wing-to-fuselage fairing rib, remove cotter pin, nut, spacers, washer, and bolt connecting flap channel to forward link.

(b) Remove cotter pin, nut, spacers, washers, and bolt connecting flap channel to aft link.

(c) Disconnect bonding strap from flap channel.

(6) At outboard hinge.

(a) Raise aileron for access to bottom of aft link and remove cotter pin, nut, spacers, washer, and bolt connecting flap channel to aft link.

(b) Remove cotter pin, nut, spacers, washer, and bolt connecting flap channel to forward link.

(c) Disconnect bonding strap from flap channel.

(7) At center hinge, where wing channel is attached to top of aft link, remove cotter pin, nut, and slide bolt out far enough to clear channel.

(8) Remove flap.

B. Installation – Flap

(1) Position flap at hinge points and support.

(2) Insert attaching bolts connecting flap channels to links (six places), positioning spacers between channels and links.

(3) Secure all hinge points with washers, nuts, and cotter pins. Be sure to correctly locate washers at flap channel/link junctions. AN960 washers go beneath the nut, and 27-32009 series washers on either side of the link.

(4) At inboard hinge, connect bonding strap.

(5) Replace fuselage access panel.

(6) Check flap through full travel for proper operation.

C. Replacement – Foreflap Brackets

(1) Remove flap assembly from airplane.

(2) Remove foreflap from flap by removing screws and shims from brackets.

NOTE: A new two-piece bracket replaces the older design one-piece bracket. Foreflap half of new two-piece bracket may be made by trimming the existing bracket(s).

(3) With bracket still attached to foreflap, scribe a line on bracket following inner contour of foreflap approximately 0.70 inch from lower foreflap skin.

(4) Cut foreflap bracket along scribe line. Radius corners of cut brackets.

(5) Secure new lower half of foreflap bracket to flap with screws.

(6) Position foreflap templates at W.S. 29.428 and W.S. 97.814 for inner foreflap, or W.S. 101.206 and W.S. 169.025 for outboard foreflap.

(7) Cradle foreflap into templates and secure with "Bungee" cord, masking tape, weighted bags, or suitable equivalent.

CAUTION: ENSURE EDGE OF LOWER BRACKET DOES NOT CONTACT LOWER SKIN OF FOREFLAP OR RADIUS OF UPPER BRACKET.

(8) Trim lower foreflap bracket as necessary to relieve interference with upper bracket radius or lower foreflap skin.

(9) With foreflap cradled on templates, locate four (4) holes through upper and lower brackets and rivet bracket halves together.

(10) Remove "Bungee" cord, tape, weights, and templates.

(11) Install flap assembly to airplane and functional check flap operation, checking for foreflap rub.

3. Inspection/Check – Flaps

A. Flap Freeplay Measurement

(1) Set flaps to about 1/2 travel, hydraulic pressure to zero.

(2) Grasp flap inboard end trailing edge and move flap up and down.

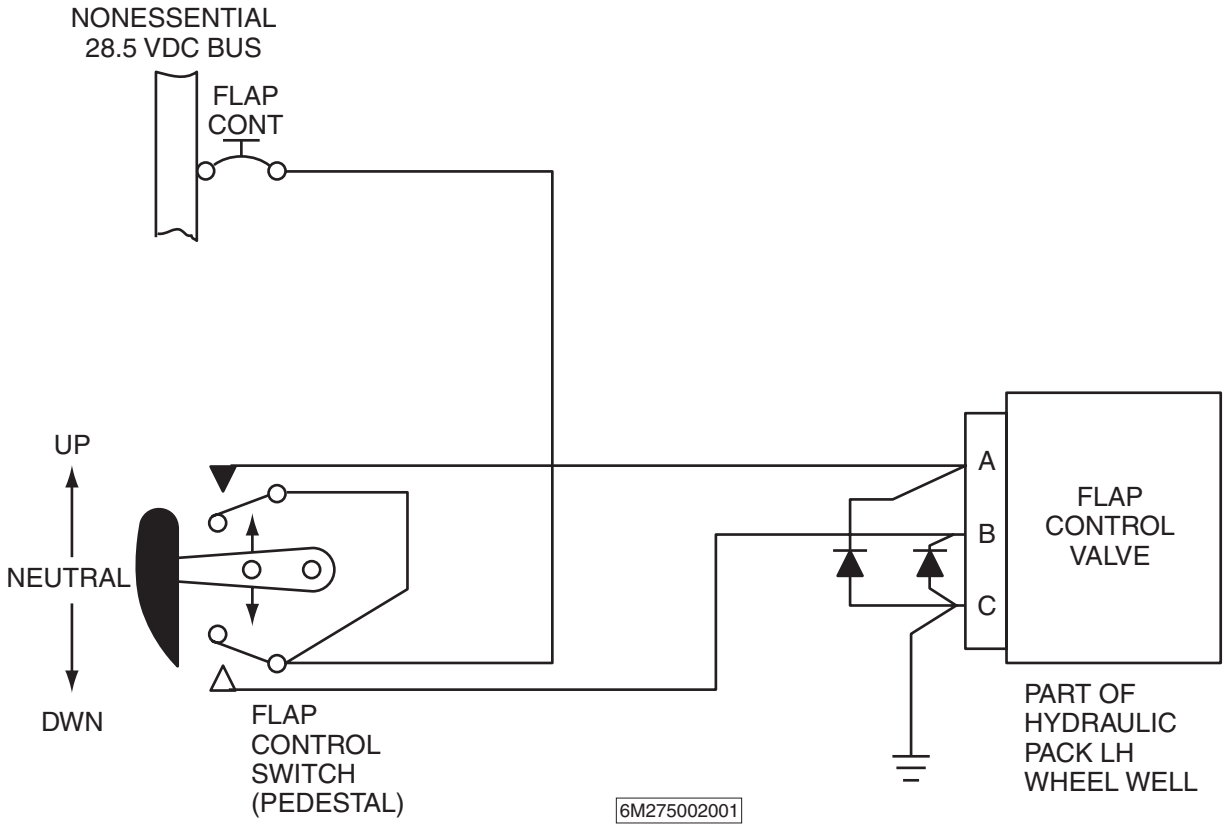
(3) Measure freeplay at outboard end trailing edge.

I (4) If freeplay exceeds 0.25 inch, check flap attach points, hinges, and actuation points for looseness and wear.

FLAP CONTROL – DESCRIPTION AND OPERATION

1. General

A. The flap control system consists of the flap control handle, the flap selector valve, flap lock valve. The flap control handle has three positions; UP. DN. and a center (no power) position. The flap select valve is part of the hydraulic pack. It is located in the left wheel well, attached to the underside of the hydraulic power pack, immediately aft of the gear selector valve. The valve has three positions; extend (down) position directs hydraulic pressure to the extend (down) side of the flap actuators to extend the flaps; retract (up) position which directs hydraulic pressure to the retract (up) side of the flaps; and the closed (off) position creates a hydraulic lock.



Flap Control
Figure 1

FLAP CONTROL – DESCRIPTION AND OPERATION

B. The flap control handle operates in conjunction with two microswitches inside the cockpit pedestal. When the control lever is placed in the UP (retract) position, the up microswitch is closed and electrical power id directed to the up solenoid on the flap selector valve. The selector valve requires approximately 28 volts DC electrical power and approximately 250 psi hydraulic pressure for actuation to either the extend or the retract position. If sufficient hydraulic pressure is available the flap selector valve will then move to the UP position and direct hydraulic pressure to the up side of both flap actuators. Conversely, when the lever is placed in the DN position, the down solenoid microswitch is closed and 28 volts DC is directed to the down solenoid on the flap selector valve. If either electrical power or hydraulic pressure is not available, the selector valve will not move (or remain in) the closed (off) position.

C. When the flap control handle is in the neutral position neither the up nor down microswitch is closed, i.e., no electrical power is directed to the flap selector valve. With the flap selector valve in the closed position, the flaps are hydraulically locked at their existing position.

D. Intermediate flap positions are obtained by moving the flap control lever to the UP or DN position, as appropriate, until the flaps are in the desired position, and then moving the lever to the neutral position.

E. Flap lock valves, one for each flap, perform a two-fold function. First the valves provide an additional hydraulic lock near the actuators to hold the flaps in the selected position in case of hydraulic line or selector valve leakage. Second, they act as a pressure relief valve to relieve excess pressure in the hydraulic lines.

FLAP CONTROL – MAINTENANCE PRACTICES

1. Removal / Installation – Flap Control

NOTE: For Removal/Replacement of Power Pack, Flap, and Gear Valves, refer to Chapter 29, Power Pack – Maintenance Practices.

A. Removal – Flap Handle Assembly

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF.

CAUTION: WHENEVER ANY HYDRAULIC COMPONENT OF THE FLAP SYSTEM IS REMOVED OR REPLACED, THE FLAP WALKING BEAM CENTERBOLT SHOULD BE REMOVED PRIOR TO PRESSURE BEING APPLIED. AFTER INITIAL OPERATION, REINSTALL THE CENTERBOLT.

(1) Turn battery switches OFF.

(2) Remove center overlay/edge-lighted panel on pedestal.

(3) Remove center switch-mounting cover on pedestal.

(4) Remove auxiliary trim switch from cover.

(5) Support cover out of way, taking stress off of wiring.

(6) Remove knob from flap control handle.

(7) Using pointed tool, locate and remove screws (two or four depending on model) upper right side of pedestal.

(8) Maneuver flap handle assembly straight down and out center of pedestal.

(9) Mark and disconnect wires from switch(s).

B. Installation – Flap Handle Assembly

(1) Attach wires to flap handle switch(s).

(2) Insert handle assembly through pedestal opening and position into place.

(3) Secure handle bracket to side of pedestal with screws.

(4) Screw knob to control handle.

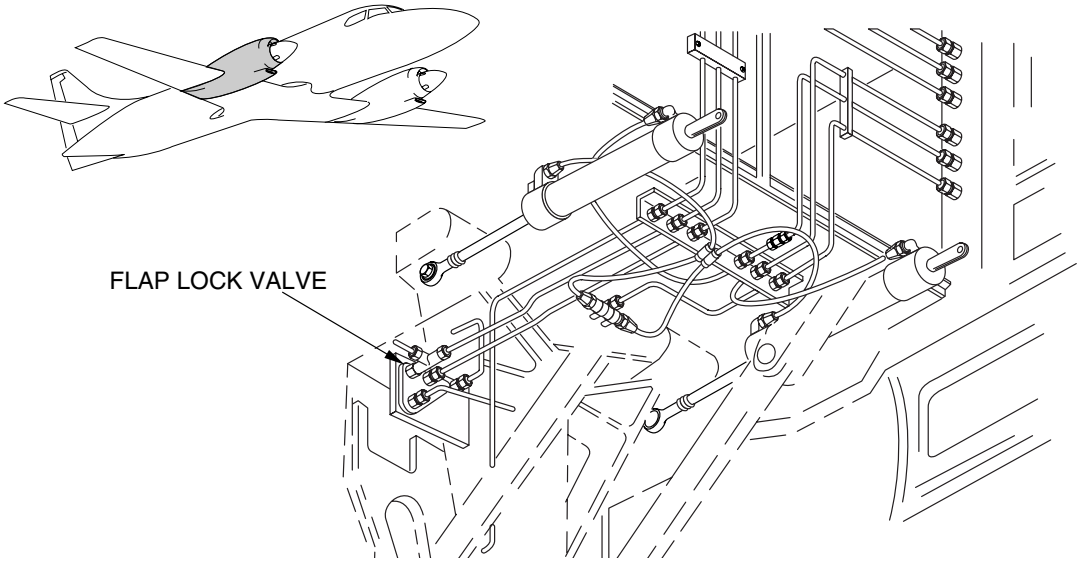
(5) Replace auxiliary switch onto switch cover and replace cover onto pedestal.

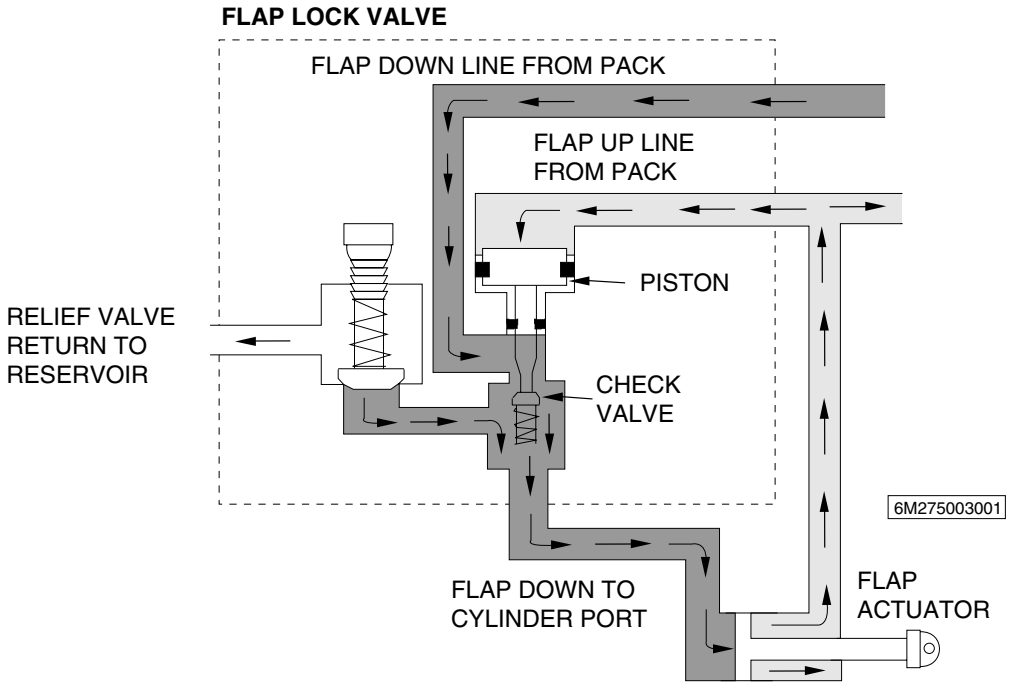
(6) Replace center overlay/edge-lighted panel onto pedestal.

FLAP LOCK VALVE – DESCRIPTION AND OPERATION

1. General

A. There are two flap lock valves, one for each flap. The lock valve for each flap is located in the aft area of the wheel well on the rear spar. The flap lock valve performs a two-fold function. First the valve provides an additional hydraulic lock near the flap actuator to hold the flap in the selected position in case of hydraulic line or selector valve leakage. Second, it acts as a pressure relief valve to relieve excess pressure in the hydraulic line.





Flap Lock Valve
Figure 1

2. Flap Lock Operation during Cool-Down

A. When the flap is retracted, a normally closed limit switch is activated, opening the circuit to the flap control valves causing them to return to a blocked center position. This produces a hydraulic lock for the flap in the UP position by a blocked loop on the rod side of the actuator. The loop is thermally relieved through a relief valve to the reservoir (see Figure 1), and recovers during cool-down through the check valve.

B. The loop to the head side of the actuator is blocked at the same time, and is thermally relieved through the flap lock valve, recovering during cool-down through the same valve arrangement (see Figure 1).

C. During the cool-down contraction, the two blocked loops recover displaced fluid, and the return takes the path of least resistance, either in the loop to the head of the piston or in the loop to the rod side of the actuator. If recovery is in the loop to the head of the piston, the flap will be displaced to an extended position.

D. If the flaps are displaced from the full UP position, the circuit to the control valves closes as soon as the aircraft is powered up, and drives the flaps back to the full UP position.

FLAP LOCK VALVE – MAINTENANCE PRACTICES

1. Removal / Installation – Flap Lock Valve

A. Removal – Flap Lock Valve

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF.

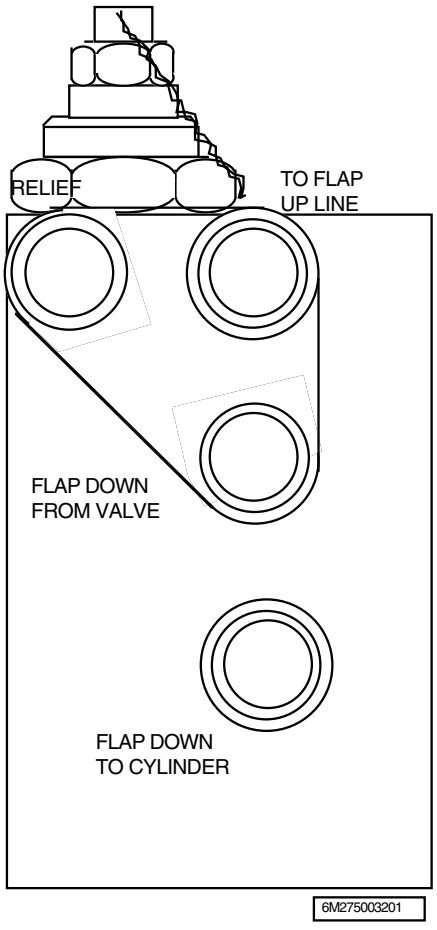
CAUTION: WHENEVER ANY HYDRAULIC COMPONENT OF THE FLAP SYSTEM IS REMOVED OR REPLACED, THE FLAP WALKING BEAM CENTERBOLT SHOULD BE REMOVED PRIOR TO PRESSURE BEING APPLIED. AFTER INITIAL OPERATION, REINSTALL THE CENTERBOLT.

(1) Bleed off any hydraulic pressure by moving flap handle up and down several times with battery switches ON, until hydraulic pressure indicates zero (0).

(2) Bleed off any air pressure on top of hydraulic reservoir by loosening filler cap on outboard side of left nacelle allowing air pressure to escape.

(3) Remove lines and fittings and cap lines. Mark lines for correct reassembly.

(4) Remove bolts securing valve to rear spar and remove valve.



Flap Lock-Out Valve
Figure 201

B. Installation – Flap Lock Valve

(1) Position valve and secure with bolts.

(2) Uncap and attach lines to valve.

(3) Service the hydraulic system with a hydraulic cart. See Hydraulic System Servicing, Section 29-00-00.

2. Adjustment / Test – Flap Lock Valve

A. Test – Flap Lock Valve

NOTE: The following test procedures may be conducted with the valve installed or removed.

(1) Bleed off any pressure by moving flap handle up and down several times with battery switches on, until hydraulic pressure reads zero (0).

(2) Remove plumbing from relief port on lock-out valve.

(3) Using a hydraulic hand pump capable of producing 2,300 psi (161.70 kg/cm^2), slowly apply hydraulic pressure to the flap down cylinder port on the lock-out valve.

(4) Slowly increase pressure until fluid appears at relief port. Relief pressure should be 2,150 ± 50 psi (151.15 ± 3.52 kg/cm^2).

B. Adjustment – Flap Lock Valve

(1) Adjust pressure relief valve as follows:

(a) Cut the safety wire from between the relief valve jam nut and the valve body.

(b) Loosen relief valve jam nut.

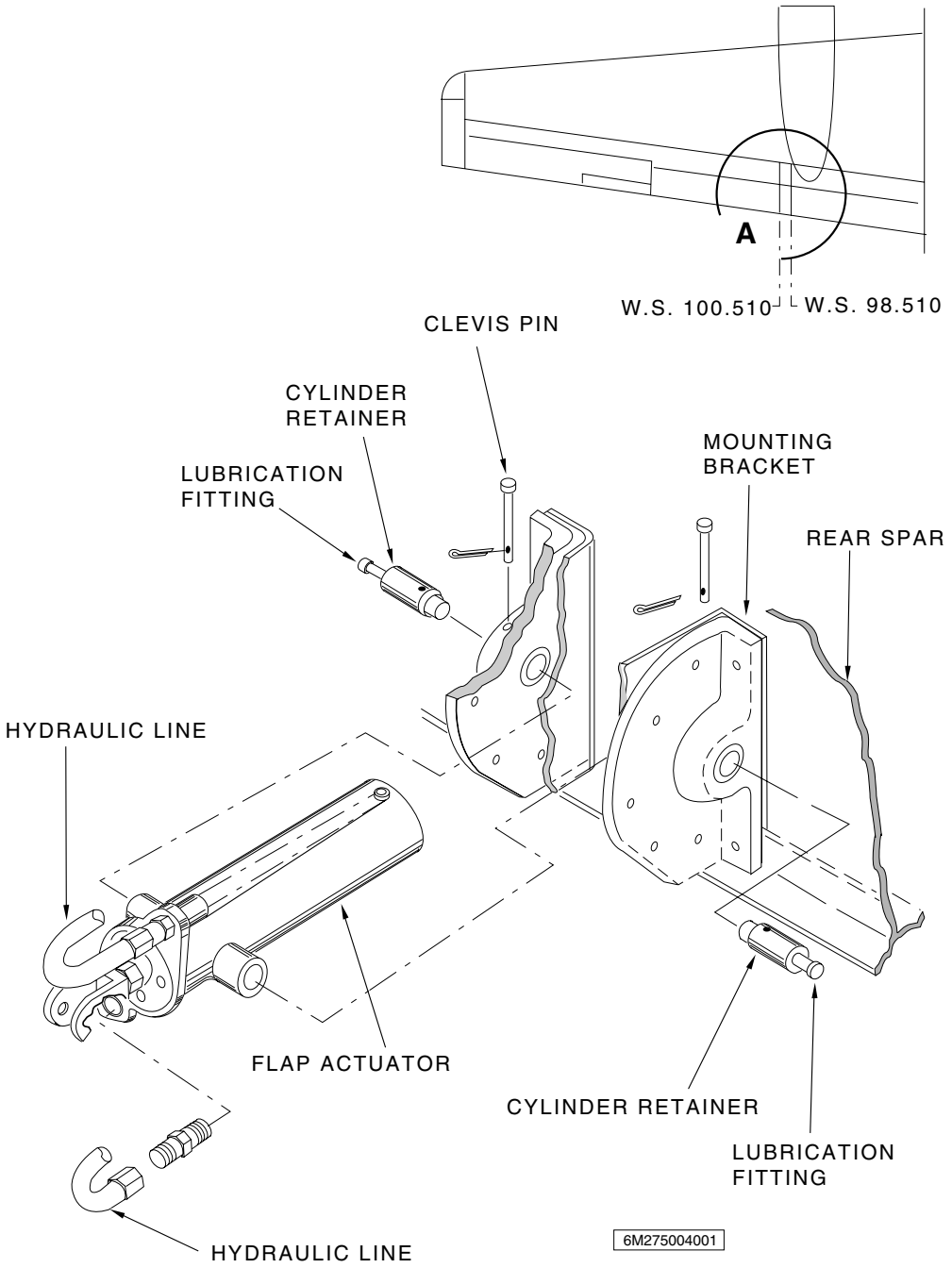
(c) Adjust the relief valve clockwise to increase pressure or counter-clockwise to decrease relief pressure until valve opens at 2,150 plus or minus 50 psi (151.15 ± 3.52 kg/cm^2).

(d) Tighten and safety relief valve jam nut after making adjustments.

FLAP ACTUATOR – DESCRIPTION AND OPERATION

1. General

A. Two dual acting hydraulic actuators are installed, one for each flap at the center hinge point. If either flap actuator fails to operate (but does not freeze in position), the remaining actuator is capable of operating both flaps in conjunction with the mechanical interconnect system. Rigging of flap travel is established by adjusting the flap actuator piston rod ends.



Flap Actuator
Figure 1

FLAP ACTUATOR – MAINTENANCE PRACTICES

1. Servicing – Flap Actuator

A. The only maintenance required of the flap actuator is periodic lubrication. Each actuator has two lubrication fittings (see Figure 1). Lubricate with general type grease (MIL-G-23827 or equivalent).

2. Removal/Installation – Flap Actuator

A. Removal – Flap Actuator

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF.

CAUTION: WHENEVER ANY HYDRAULIC COMPONENT OF THE FLAP SYSTEM IS REMOVED OR REPLACED, THE FLAP WALKING BEAM CENTERBOLT SHOULD BE REMOVED PRIOR TO PRESSURE BEING APPLIED. AFTER INITIAL OPERATION, REINSTALL THE CENTERBOLT.

(1) Extend flaps.

(2) Bleed off any hydraulic pressure by moving flap handle up and down several times with battery switch on, until hydraulic pressure reads zero.

(3) Bleed off any air pressure on top of hydraulic reservoir by loosening filler cap on outboard side of left nacelle allowing air pressure to escape.

(4) Remove access panel located at lower fuselage-to-wing trailing edge.

(5) Remove center bolt and bearing connecting both walking beams (flap interconnect).

(6) Complete steps (3) through (9) – Flap Removal, Section 27-50-01.

(7) Remove cotter pin, nut, spacers, and bolt connecting actuator to actuator link.

(8) Remove cotter pins from clevis pins, left and right of actuator, and remove clevis pins.

(9) Remove cylinder retainers.

(10) Disconnect hydraulic lines from actuator.

WARNING: CAREFUL NOTE MUST BE TAKEN OF POSITION OF ACTUATOR HYDRAULIC LINES. REVERSAL OF LINE WILL CAUSE SEVERE DAMAGE TO FLAPS.

(11) Remove actuator.

B. Installation – Flap Actuator

(1) Attach hydraulic lines to actuator and position actuator into place.

(2) Insert cylinder retainers.

(3) Insert clevis pins and secure with cotter pins.

(4) Approximately adjust actuator clevis to match extension on removed actuator (at least three full threads on shaft must be engaged beyond hole in clevis). Attach actuator to actuating link.

(5) Complete steps (1) through (5), – Flap Installation, Section 27-50-01.

(6) Service the hydraulic system with a hydraulic cart. See Hydraulic System Servicing, Section 29-00-00.

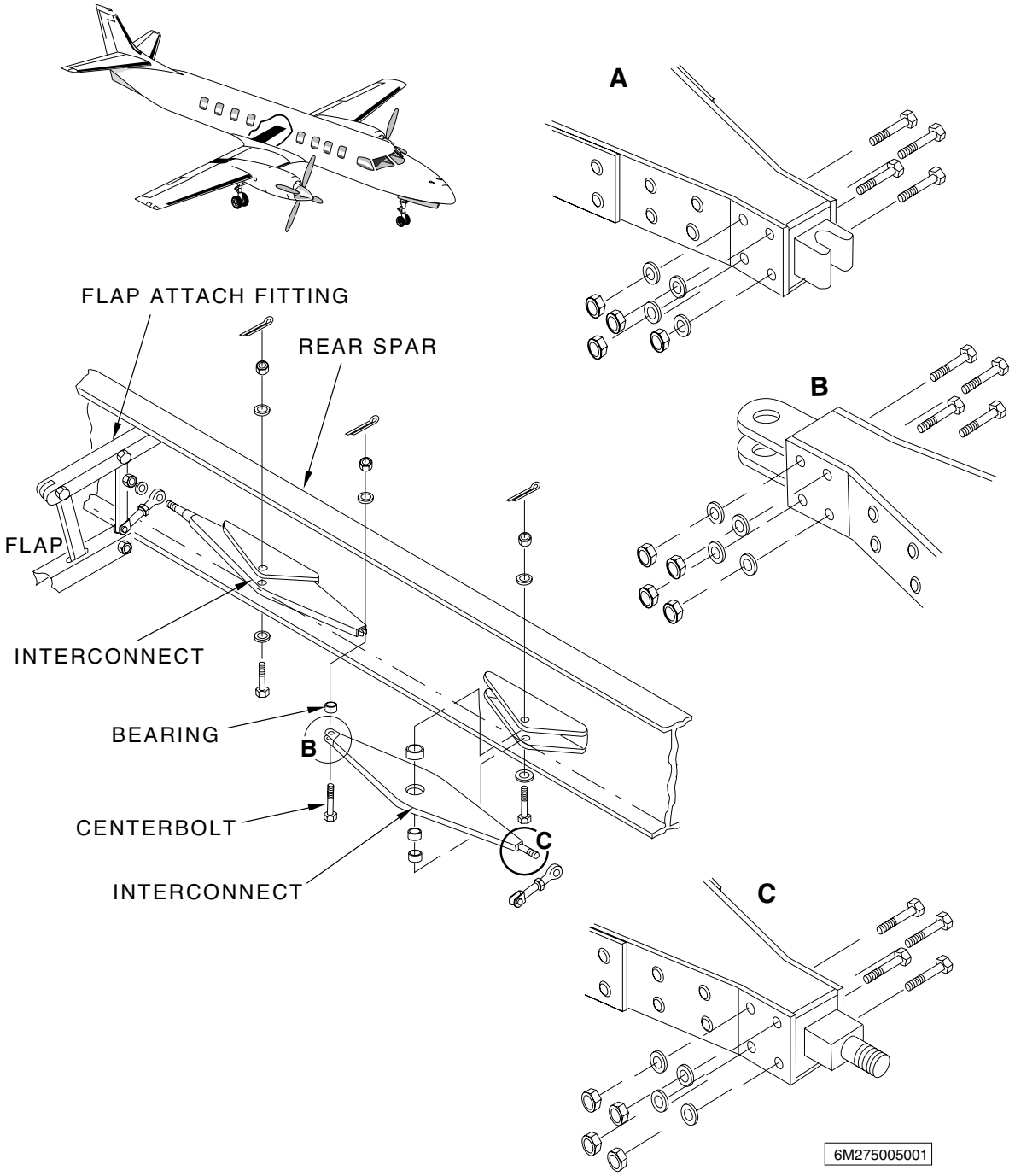
(7) Perform Flap Travel Check in 27-00-00. Follow procedures in this check to connect walking beams and make final actuator clevis adjustments.

(8) Replace all access panels.

FLAP INTERCONNECT (WALKING BEAM ASSEMBLY) – DESCRIPTION AND OPERATION

1. General

A. Two flaps are mechanically interconnected for position symmetrical operation should hydraulic actuation of one flap be lost. The walking beam assembly is attached to the aft side of the wing rear spar.



Flap interconnect
Figure 1

FLAP INTERCONNECT (WALKING BEAM ASSEMBLY) – MAINTENANCE PRACTICES

1. Removal / Installation _ Flap Interconnect

A. Removal – Flap Interconnect

(1) Extend flaps.

(2) Remove access panels located at lower fuselage-to-wing trailing edge.

(3) Remove center bolt and bearing connecting both walking beams.

(4) Between flap and fuselage, remove cotter pin and nut connecting rod end to walking beam.

(5) Separate rod end from walking beam.

(6) Remove bolt and hardware securing walking beam and aileron swing link to walking beam bracket and remove walking beam.

B. Installation – Interconnect

(1) Insert walking beam into rod end and secure with nut and cotter pin.

(2) Position aileron swing link and walking beam at bracket and secure.

(3) Connect both walking beams together with bearing and bolt.

(4) Replace access panels.

(5) Perform operational check.

FLAP POSITION INDICATING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The flap position indicating system is a DC operated system comprised of two main units, the flap position indicator and the flap position transmitter. The flap position indicator depicts flaps at up, 1/4, 1/2, 3/4, and full down positions. The flap position transmitter along with the SAS flap followup transmitter, is mounted on a bracket, which is in turn mounted on the fuselage. The flap position transmitter is located inboard of the SAS transmitter. The flap position transmitter is mechanically linked, by a rod and an arm, to the walking beam. The flap position indicating system contains a flap position warning system. If landing gear is not down and locked and flaps are lowered 20° or more, a landing gear warning horn will sound in the cockpit. Circuit breakers for the landing gear and flap position indicating systems are located on the left or right DC essential buses through a transfer. switch found on the left rear console.

1. Adjustment/Test – Flap Position Indicating System

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF.

A. Operation Check

(1) Connect APU and hydraulic cart to aircraft.

(2) Place flap handle in neutral position.

(3) Ensure flap area is clear of personnel and equipment.

(4) Using jumper wire, jump terminals 6 and 7 on terminal board in left wheel well.

NOTE: If this check is done in conjunction with landing gear check, gear must be in intermediate (unsafe) or UP (retracted) position.

(5) Turn battery switches ON.

(6) Ensure flaps are in UP position. Indicator needle should be on UP mark.

(7) Place power levers forward of flight idle gate.

(8) Run flaps DOWN. Observe indicator. Indicator should move smoothly and evenly. As needle passes 20° mark (slightly past 1/2 position) landing gear warning horn must sound. When flaps reach full down position needle on indicator should be on down mark.

(9) Run flaps UP. Warning horn should stop sounding as needle passes 1/2 mark on indicator. When flaps are UP, needle on indicator should stop on UP mark.

(10) Place flap handle in neutral position.

(11) Turn battery switches OFF.

(12) Remove jumper wire.

(13) Disconnect APU and hydraulic cart.

NOTE: There is nothing wrong if the flaps droop when hydraulic pressure is off. Flaps may extend slowly as the airplane sits and the system cools down. As long as flaps come back up when hydraulic pressure is applied, and they otherwise function correctly there is no need for further maintenance or adjustments.

FLAP POSITION INDICATING – DESCRIPTION AND OPERATION

1. General

A. Flap position indicating for early model aircraft consists of a flap position indicator, mounted on the instrument panel, and a flap position transmitter, located aft of the flap walking beam.

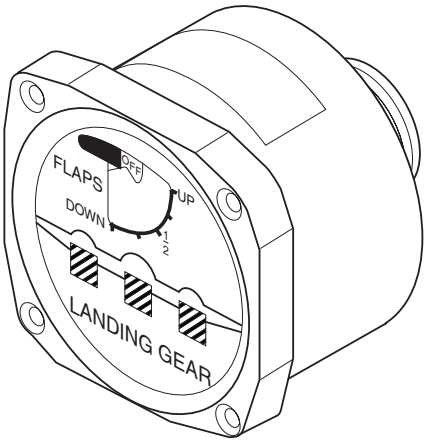
B. The flap position indicator is housed with the landing gear position indicator. The flap position indication pointer is airfoil shaped to resemble a flap. The indicator depicts flaps at up, 1/4, 1/2, 3/4, and full down positions. When power is off, an OFF flag on the pointer appears.

C. The flap position transmitter is a potentiometer whose shaft is connected by a rod and an arm to the walking beam.

D. Movement of the flaps activates the walking beam, which in turn causes the transmitter shaft to rotate. Wires connecting the transmitter and the indicator carry the electrical currents which drive the pointer shaft in the indicator through the same number of degrees that the transmitter shaft rotates. Since both shafts rotate in synchronization, the system is called selsyn (self synchronous).

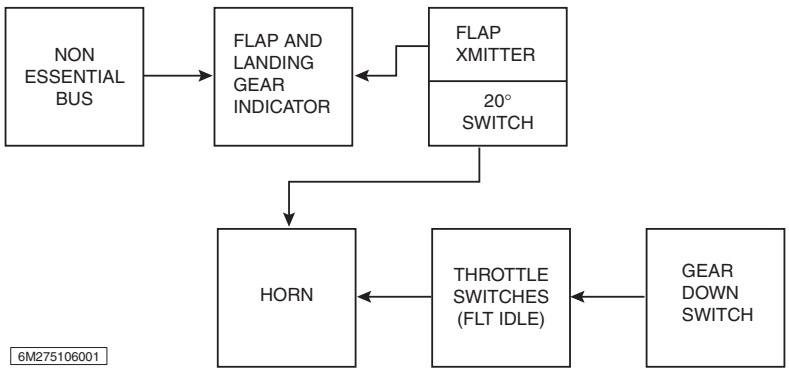
E. When flaps attain 20° down position a cam connected to the drive system of the transmitter and walking beam activates a switch which in turn activates a landing gear warning horn in the cockpit.

NOTE: Aircraft T201-208 and AT001-003 do not incorporate the 20° warning function.



T201-278
AT001-061
TC201-231,233,
234, 236

EARLY FLAP INDICATOR



Flap Position Indicating
Figure 1

FLAP POSITION INDICATING – MAINTENANCE PRACTICES

1. Removal / Installation – Flap position Indicating

A. Removal – Indicator

(1) Remove post lights.

(2) Remove screws holding instrument to panel.

(3) Remove instrument from panel.

NOTE: Some aircraft have rear mounted instruments. If so, work instrument down behind panel to gain access to electrical connector.

(4) Remove safety wire and remove electrical connector.

B. Installation – Indicator

(1) Install electrical connector and safety wire.

(2) Position indicator into panel and secure with holding screws.

(3) Replace post lights.

(4) Perform operational check.

NOTE: Because the flap indicator is within the same case as the landing gear indicator, landing gear must be operationally checked (see Gear Extension and Retraction, Section 32-30-00).

C. Removal – Transmitter

(1) Position flaps in UP position.

(2) Remove access panel located at lower fuselage to wing trailing edge.

(3) Loosen allen-head screw on arm attached to shaft of flap transmitter (inboard transmitter) and remove arm from shaft.

(4) Loosen allen-head screw securing transmitter cam on shaft and remove cam from shaft.

(5) Remove screws holding transmitter to bracket and remove transmitter.

(6) Remove safety wire and remove connector.

NOTE: While transmitter is off, clean electrical connector and any corrosion on transmitter bracket. Also, remove any paint or corrosion from where mounting screws attach transmitter to bracket. Transmitter case must make a good connection to ground.

D. Installation – Transmitter

(1) Position transmitter on bracket and secure with screws.

NOTE: Connector key should face forward.

(2) Connect electrical connector and safety wire.

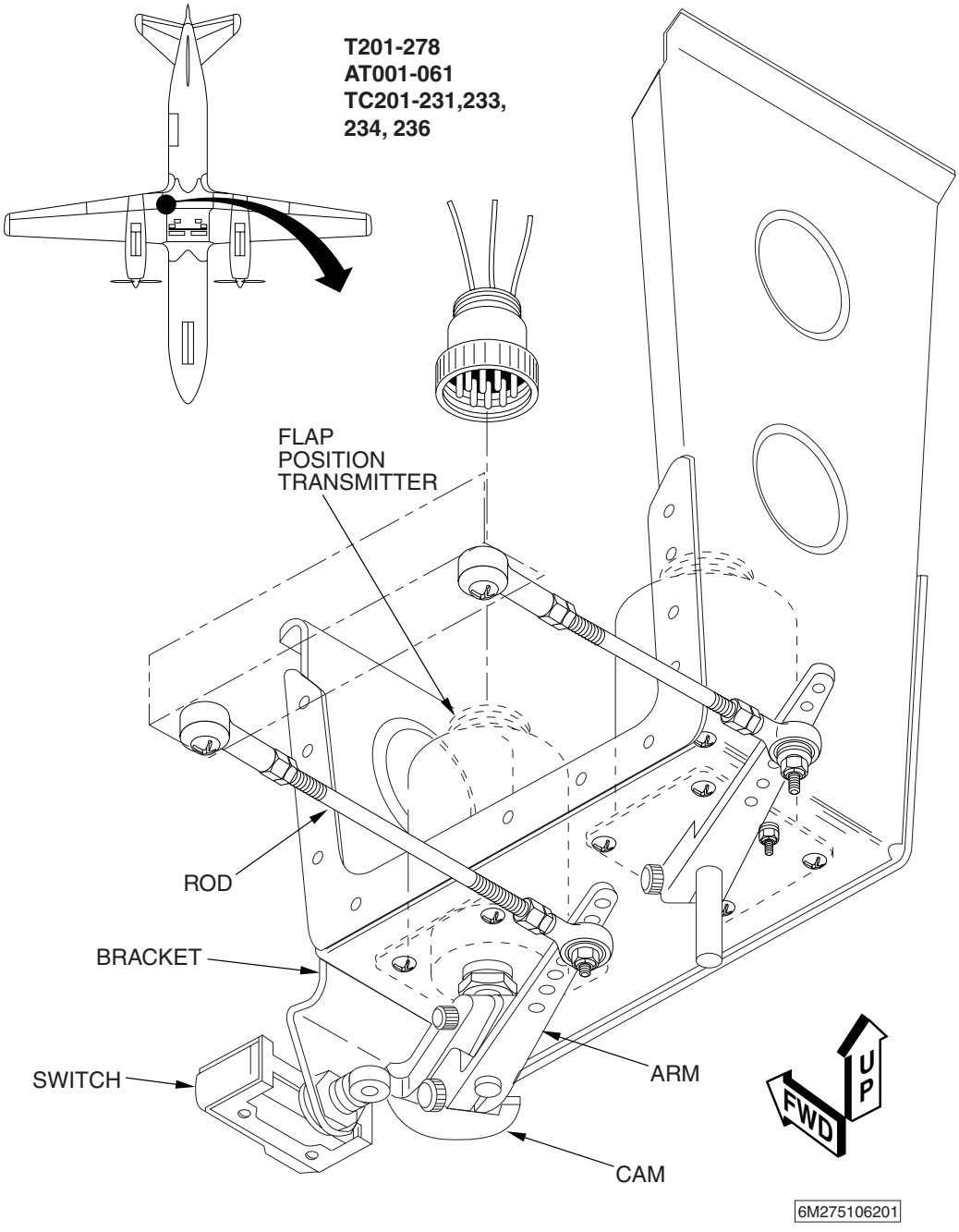
(3) Position and install cam and tighten allen-head screw.

(4) Install arm to shaft and tighten allen screw.

(5) Adjust transmitter (see Adjustment and Test).

(6) Perform operational check.

(7) Install panel.



Flap Position Transmitter
Figure 201

2. Adjustment/Test – Flap Position Indicating

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF.

A. Adjustment/Test – Flap Position Indicating System

(1) Remove access panel located at lower fuselage to wing trailing edge.

(2) Connect APU and hydraulic cart to aircraft.

(3) Ensure flaps are up and flap handle is in neutral position.

(4) Turn battery switches on.

(5) Note reading of indicator. If indicator pointer is not on UP position, loosen allen-head screw on arm. Turn shaft until needle points to UP position on indicator. Tighten all-head screw.

(6) While assistant runs flaps down, observe transmitter arm linkage for proper alignment and movement.

(7) If needle if indicator is not on the down position, move rod end inward or outward on arm of transmitter shaft.

NOTE: Moving arm inward increases the arc of the indicator needle while moving arm outward decreases the arc of the needle.

Some earlier model aircraft (T201-248, AT001-024, TC201-211) have a small potentiometer mounted on the transmitter bracket, inboard of the transmitter. This is for fine adjustment of indicator if necessary.

(8) Run flaps up. Indicator needle should point to UP. If not, split error distance by loosening allen-head screw on arm and adjust.

(9) Repeat procedures until satisfactory adjustments are attained.

(10) Place flaps on 1/2 position.

(11) Loosen allen-head screw and adjust cam so that roller of switch is in depression on cam.

(12) Using jumper wire, jumper terminals 6 and 7 on terminal board on left wheel well.

(13) Place power levers forward of flight idle gate.

(14) Run flaps down. Landing gear warning horn should sound when needle is just past 1/2 on indicator (20°). If not, adjust switch inward until requirement is met.

(15) Remove jumper wire.

(16) Install panel.

FLAP POSITION INDICATING – DESCRIPTION AND OPERATION

1. General

A. The flap position indicating system for later model aircraft consists of a flap position indicator, mounted on the instrument panel, and a flap position transmitter, located aft of the flap walking beam, and a flap position indicating and landing gear warning control box mounted forward of the pilot rudder pedals on the forward pressure bulkhead. On aircraft T292 and up, it is located forward of the copilot rudder pedals.

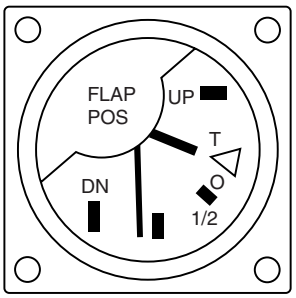
B. The flap position indicator is a separate unit from the landing gear position indicator. The indicator depicts flap position and is marked at UP, 1/4, 1/2, 3/4, and full down positions. When power is off, an OFF, the indicator needle rests on UP scale peg.

C. The flap position transmitter is a potentiometer whose shaft is connected by a rod and an arm to the walking beam.

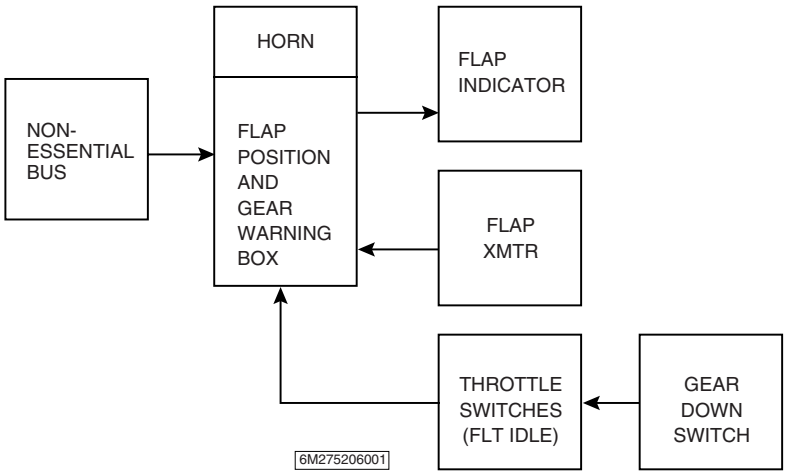
D. The control box, along with the electrical circuitry for flap position indication and landing gear position indication, also contains the landing gear warning horn.

E. Movement of the flaps activates the walking beam, which in turn causes the transmitter shaft to rotate. The transmitter provide flap position information to circuitry in the control box. This in turn provides flap position information to the indicator and also activates the 20° warning horn when landing gear are up and throttle power levers are in flight idle. The up and down positions at the indicator needle as well as the 20° landing gear warning are adjustable by three potentiometers housed within the control box.

T279-999
AT062-999
TC 232, 235,
237-999



LATE FLAP INDICATOR



Flap Positioning Indicating
Figure 1

FLAP POSITION INDICATING – MAINTENANCE PRACTICES

1. Removal / Installation – Flap position Indicating

A. Removal – Indicator

(1) Remove post lights.

(2) Remove screws holding instrument to panel.

(3) Remove indicator from panel.

(4) Disconnect knife disconnects.

NOTE: For installations without disconnects cut wire close to indicator. New indicator has knife disconnects. Cut wire to add knife disconnects to existing wires. Do not remove terminal lugs on indicator being removed. Connect 649-22 to 649A-22 (negative) and 648-22 to 648A-22 (positive). install two inch piece of 1/4 inch vinyl tubing over each disconnect and tie both ends.

(5) Remove spanner nut attaching spacer block to indicator and remove and retain spacer block.

B. Installation – Indicator

(1) Remove spanner nut from new indicator.

(2) Install spacer block and spanner nut onto indicator.

(3) Connect handshake connectors (see note above).

(4) Position indicator into panel and secure with screws.

(5) Replace post lights.

(6) Perform operational check.

C. Removal – Transmitter

(1) Position flap in up position.

(2) Remove access panel located at lower fuselage to wing trailing edge.

(3) Loosen allen-head screw on arm attached to shaft of flap transmitter (inboard transmitter).

(4) Loosen allen-head screw securing transmitter cam on shaft and remove cam.

NOTE: Aircraft with CONRAC SAS installed do not have a cam (T292-999).

(5) Remove nut and washer securing transmitter to bracket.

(6) Remove transmitter from bracket. Identify and cut wires as close to transmitter as possible.

D. Installation – Transmitter

(1) Solder wires to terminals of new transmitter.

(2) Position transmitter and install nut and lockwasher.

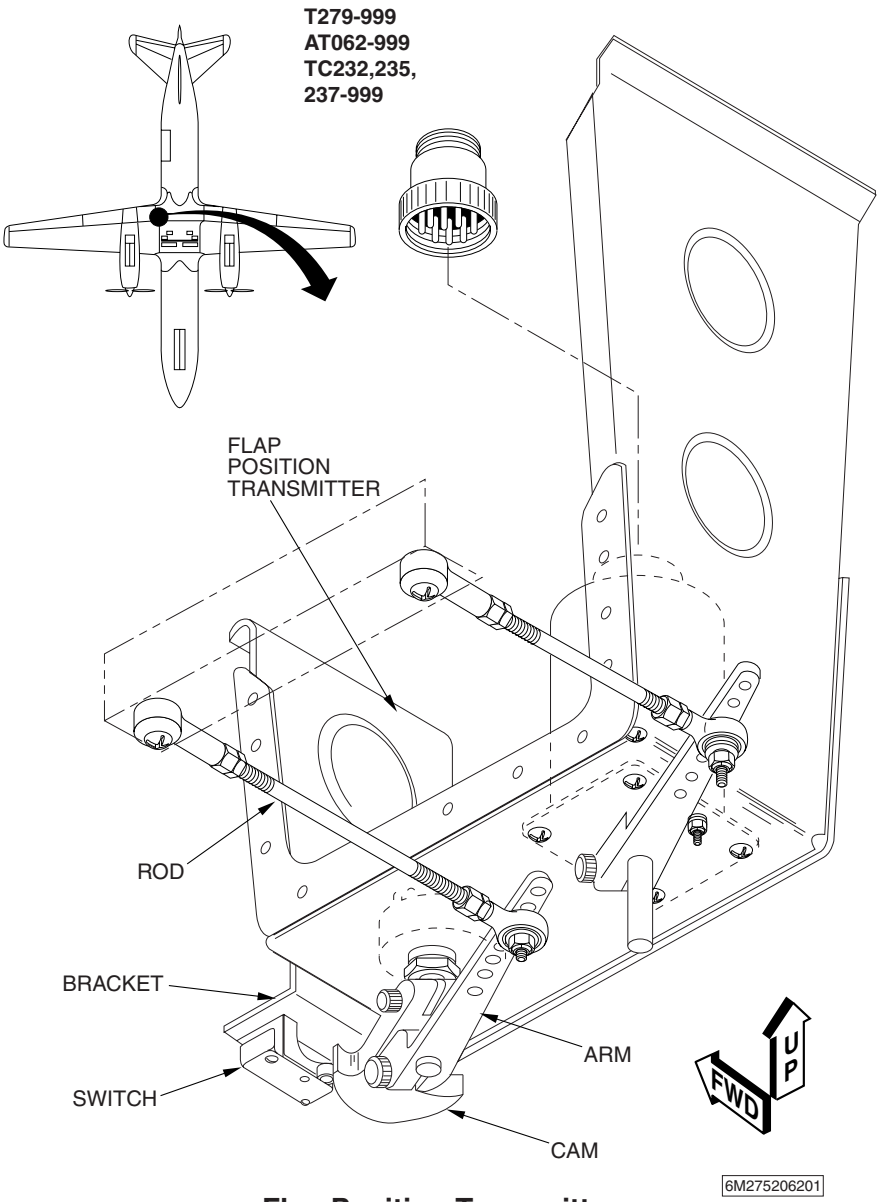
(3) Install cam onto shaft. position cam and tighten allen-head screw only enough to hold cam into place so that when cam meets switch, cam will slip on shaft and not damage switch actuator.

(4) Install arm to shaft and tighten allen-head screw.

(5) Adjust transmitter (see Adjustment and Test procedures).

NOTE: Switch actuates at 20° flap travel. The cam and the switch are SAS components and not for landing gear and flap warning. See Section 27-31-09 for Adjustment of Cam and switch.

(6) Replace walking beam panel.



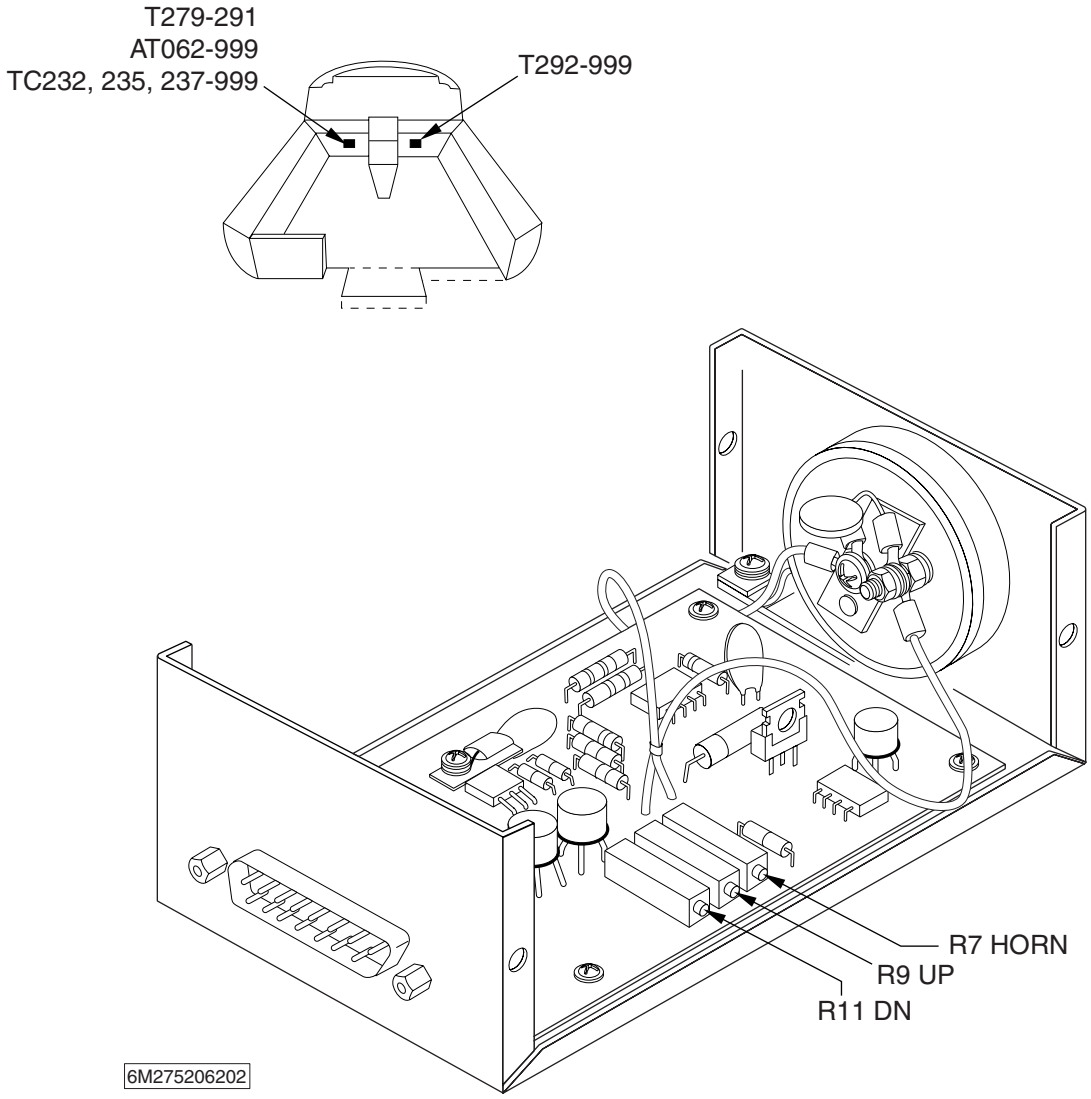
Flap Position Transmitter
Figure 201

E. Removal – Landing Gear and Flap Control Box

NOTE: On aircraft T279-291, AT062-999, TC232. 235, 237-999 landing gear and flap control box is located forward of the pilot rudder pedal on the forward pressure bulkhead. On aircraft T292 and up the control box is located forward of the copilot rudder pedals on the forward bulkhead.

(1) Remove screws securing control box cover containing circuitry.

(2) Disconnect electrical connector.



Landing gear
Warning Control Box
Figure 202

F. Installation – Landing Gear and Flap Control Box

(1) Check and/or adjust transmitter (see Adjustment and Test – Flap Position Indicating System 27-52-06).

(2) Adjust control box (see Adjustment and Test – Flap Position Indicating System (see Adjustment and Test – Flap Position Indicating System, Section 27-52-06)..

(3) Connect electrical connector to new control box.

NOTE: Since control box contains landing gear warning circuitry, an operational check of the landing gear position/warning system must be performed (see Chapter 32).

(4) Secure control box to equipment panel with screws.

2. Adjustment/Test – Flap Position Indicating

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF.

A. Adjustment/Test – Flap Position Indicating System

(1) Remove access panel located at lower fuselage to wing trailing edge.

(2) Locate rod which connects walking beam to flap position transmitter. Adjust rod so that a safety wire (.032) inserted in check hole of rod end will not pass through hole.

(3) Connect APU and hydraulic cart to aircraft.

(4) Ensure flaps are up and flap handle is in neutral position.

(5) Remove electrical connector from control box.

(6) Connect ohmmeter to pins 8 and 15 of electrical connector.

(7) Loosen allen-head screw on arm of transmitter and adjust to read 798 ohms on ohmmeter.

(8) Tighten allen-head screw on arm.

NOTE: If applicable, loosen allen-head screw on cam enough so that cam will slip on shaft and not damage switch actuator.

(9) Turn battery switches on.

(10) While assistant runs flaps down, observe transmitter arm linkage for proper alignment and movement.

(11) With flaps down, ohmmeter should read 580 ohms. If not adjust end in or out on arm of transmitter until flaps up and flaps down readings are attained. Tolerances of these readings are +10 ohms.

NOTE: Switch actuates at 20 degree flap travel. The cam and the switch are SAS components and not for landing gear and flap warning. See Section 27-31-09 for Adjustment of Cam and Switch).

(12) Turn battery switches off.

(13) Remove ohmmeter from control box electrical connector.

(14) Assure all jam nuts and allen-head screws are secure and replace panel.

(15) Remove control box cover exposing circuitry located in cover.

(16) Connect electrical connector to control box.

(17) Ensure flaps are up and flap handle is in neutral position.

(18) Turn battery switches on.

(19) At control box, adjust R9 until needle of indicator is within up mark (see figure 202).

(20) Run flaps down. Adjust R11 until needle of indicator is within down mark.

(21) Run flaps to 1/2 mark on indicator.

(22) Using jumper wire, jump terminals 6 and 7 on terminal board on left wheel well.

(23) Place power levers forward of flight idle gate.

(24) Run flaps down. Landing gear warning horn should sound when needle is just past 1/2 on indicator (20°). If not, adjust R7 until requirement is met.

(25) Remove jumper wire.

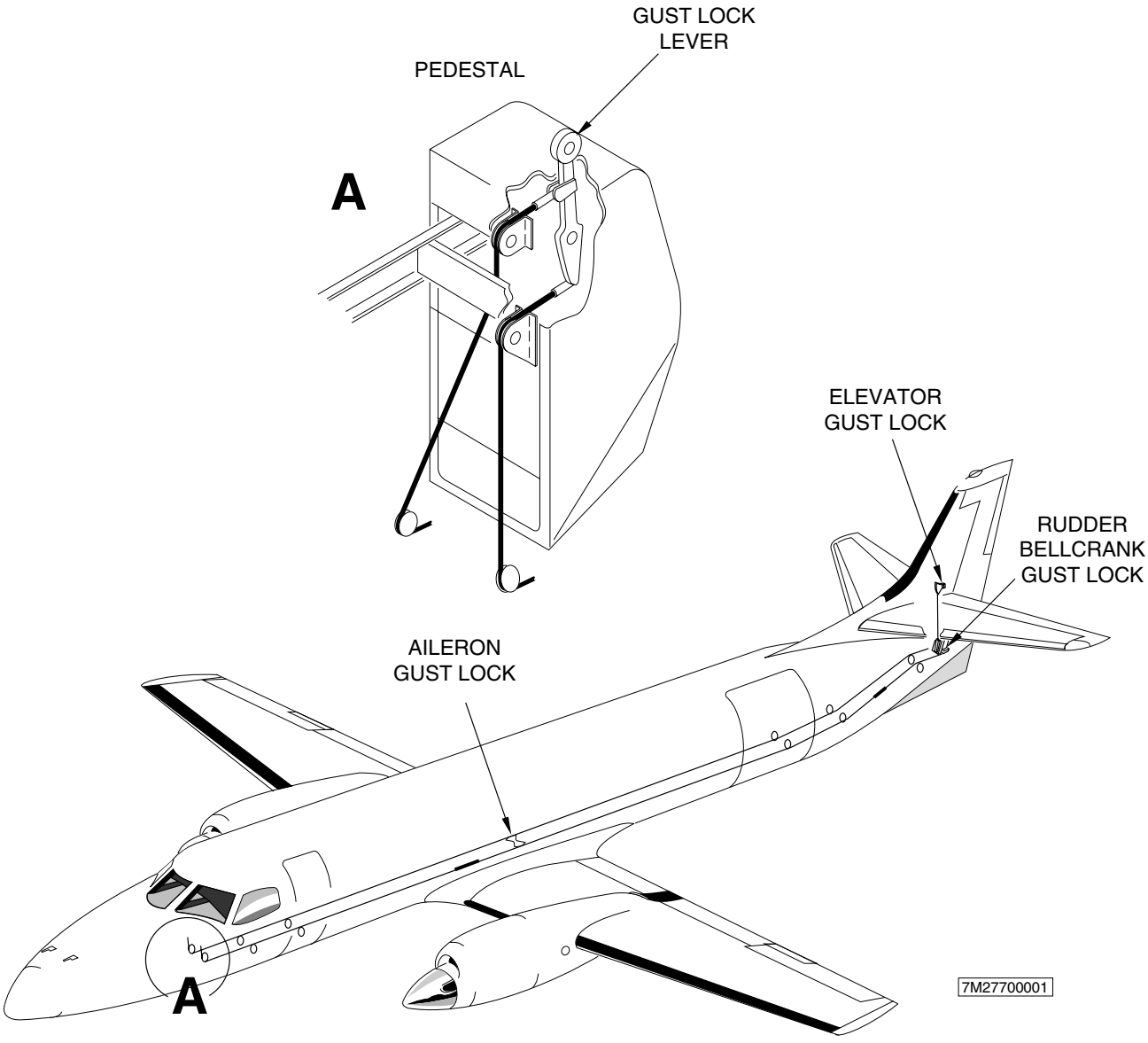
(26) Turn battery switches off.

(27) Install control box.

GUST LOCK SYSTEM – DESCRIPTION AND OPERATION

1. General

A. An internal cable operated gust lock system is provided to lock the flight controls in the neutral (streamlined) position when engaged. The gust lock control lever is located forward of the power levers on the cockpit pedestal. With the gust lock control lever engaged (LOCKED), the power levers are restricted to the retarded (below FLIGHT IDLE) range which prevents application of power for takeoff. When the power levers are in the advanced (FLIGHT IDLE or above) position, the gust lock cannot be moved to the LOCKED position; thus, engagement of the lock lever while in flight is prevented.



Gust Lock System
Figure 1

GUST LOCK SYSTEM – DESCRIPTION AND OPERATION

WARNING: ENSURE GUST LOCK CONTROL LEVER IS ENGAGED BY PUSHING DOWN ON CONTROL LEVER HANDLE WHEN LEVER IS IN LOCKED POSITION. LEVER SHOULD NOT BE ABLE TO BE PUSHED TO CONTROL LOCK OFF POSITION WITHOUT FIRST PULLING UP ON LEVER HANDLE.

B. The gust lock control lever actuates a cable, which is routed to lock pins at the aileron interconnect bellcrank, and the rudder bellcrank, and the elevator quadrant. Cable movement actuates these spring-loaded pins into lock pin holes when the flight controls are in neutral and the gust lock lever is in the LOCKED position. The gust lock pins hold the controls in the neutral or streamlined position until the gust lock lever is moved to CONTROL LOCK OFF position. The lock pins are mounted in spring-loaded housings with the pins spring-loaded toward the CONTROL LOCK OFF position, Hence, each pin is forced toward the UNLOCK position should its gust lock cable break. To engage the gust lock, place the power levers in GROUND IDLE position. Pull up on the gust lock spring-loaded control lever and pull aft to LOCKED position.

WARNING: SUPPORT CONTROL COLUMN WHEN DISENGAGING GUST LOCK THIS WILL PREVENT CONTROL COLUMN FROM THRUSTING FORWARD AND POSSIBLY CAUSING DAMAGE TO THE INSTRUMENT PANEL.

C. When lever is set in LOCKED position, work control wheel left and right until ailerons lock, work rudder pedals up and down until rudder locks, and pull back on control column until elevators lock. All controls can be locked individually. To disengage gust lock, lift up on gust lock control lever handle and push forward to CONTROL LOCK OFF position.

GUST LOCK SYSTEM – MAINTENANCE PRACTICES

1. Adjustment/Test – Gust Lock System

A. Rigging Instructions

(1) Place power levers in GROUND IDLE position.

(2) Place gust lock control lever in UNLOCK position.

(3) Insert 3/32 inch rig pins in aileron bellcrank lock, and rudder bellcrank lock. Refer to figures in Aileron Gust Lock, and Rudder Gust Lock sections of this chapter for rig pin locations.

(4) Adjust turnbuckles to obtain cable tension as shown on cable tension/temperature charts, Section 27-00-00. Check tension with a tensiometer.

(5) Remove rigging pins. Rigging pins should freely insert into the rig pin holes when rigging is complete.

(6) Adjust push-pull cable at the rudder lock arm to allow elevators to unlock before the gust lock handle travels one half the distance toward the unlock position.

(7) Place gust lock levers in lock position. Ensure gust lock control is engaged by pushing down on control lever handle when lever is in locked position. Lever should not push to control lock off position unless lever handle is pulled up first.

(8) Work aileron, rudder, and elevator controls until controls lock.

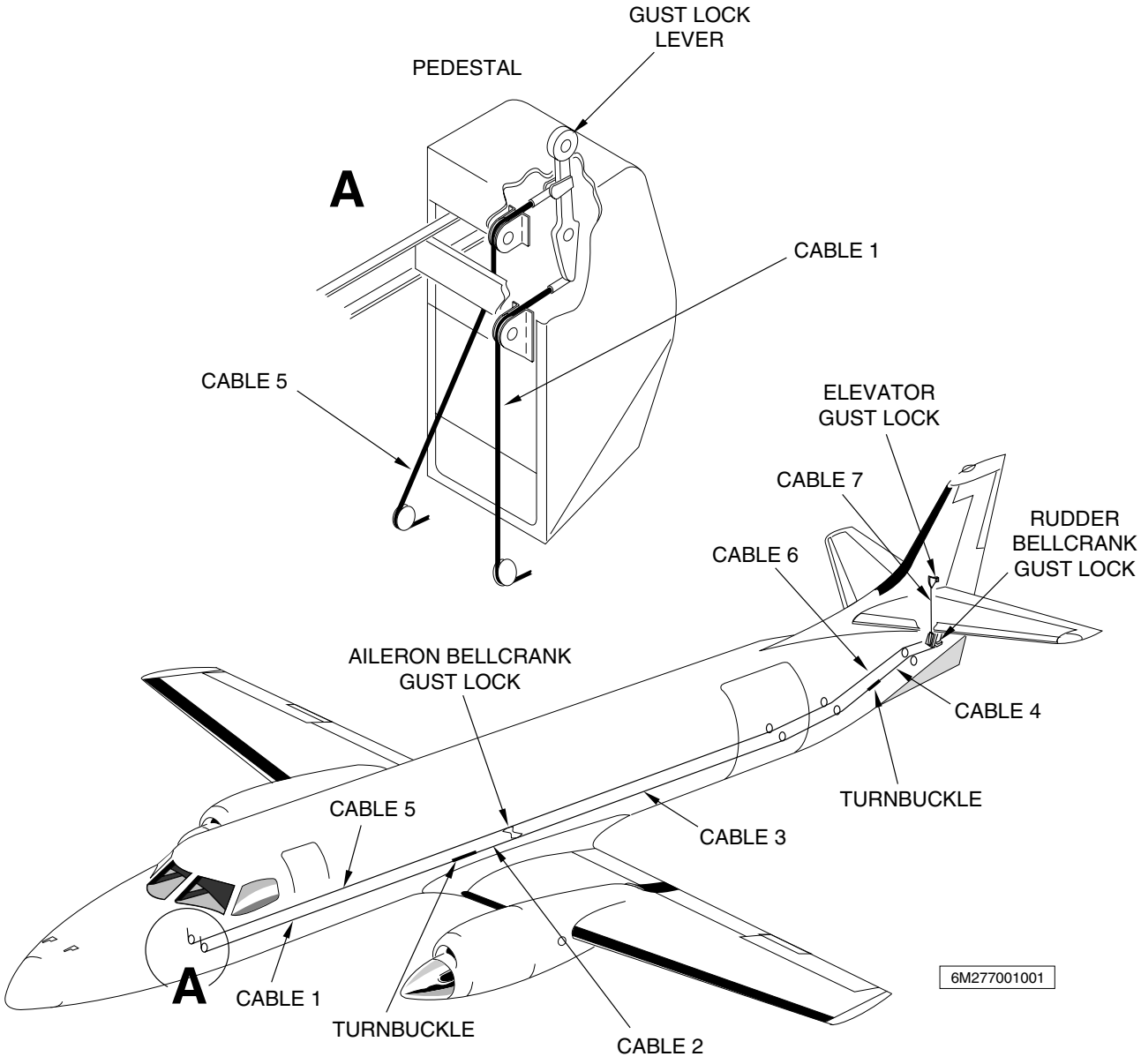
CAUTION: SUPPORT CONTROL COLUMN WHEN DISENGAGING GUST LOCK. THIS WILL PREVENT CONTROL COLUMN FROM THRUSTING FORWARD AND POSSIBLY CAUSING DAMAGE TO THE INSTRUMENT PANEL.

(9) Place gust lock lever in control lock off position. Ailerons, rudder, and elevator must move freely.

GUST LOCK CONTROL CABLE SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The gust lock control cable system is comprised of seven cables (one of which is push-pull type cable) joined together by a series of control arms and turnbuckles.



Gust Lock Control Cable System Figure 1

GUST LOCK CONTROL CABLE SYSTEM – DESCRIPTION AND OPERATION

B. The lock cable system is divided into two sections; the left side and the right side (see Figure 1). The first cable of the left side runs from the pedestal, under the center aisle floorboards to a turnbuckle located forward of the rudder/aileron interconnect at the wing rear spar (see cable 1). The next cable (2) is a short cable connected by a turnbuckle at one end to cable number 1 and bolted at the other end to the aileron bellcrank lock arm. Cable 3 continues from cable 2 (at the aileron bellcrank lock arm) and terminates at turnbuckles aft of the aft interior bulkhead. Cable 4 extends from the turnbuckle aft of the aft interior bulkhead to the bottom of the rudder bellcrank control lock arm.

C. The first cable of the (5) runs from the pedestal, under the center aisle floorboards to a turnbuckle located aft of the aft interior bulkhead. The next cable (6) extends from the turnbuckle aft of the aft interior bulkhead to the top of the rudder bellcrank lock arm. The final section of cable (7) is a push-pull type cable. It connects from the rudder bellcrank lock arm to the elevator lock mechanism.

GUST LOCK CONTROL CABLE SYSTEM – MAINTENANCE PRACTICES

1. Removal/Installation – Gust Lock Control Cable System

A. Removal – Cable 1

(1) Remove top and center panels from pedestal.

(2) Unbolt cable from gust lock control lever.

(3) Remove cable from pulleys at rear of pedestal by removing cotter pins.

(4) Remove cabin center aisle floorboards from pilot and copilot compartment to wing rear spar.

(5) Block cable aft of turnbuckle located forward of rudder/aileron interconnect.

(6) Disconnect cable at turnbuckle.

(7) Remove pulley guards and fairleads as necessary and remove cable.

B. Installation – Cable 1

(1) Position cable and replace fairleads and pulley guards as necessary.

(2) Position cable over pulleys on rear of pedestal and insert cotter pins.

(3) Attach cable end to gust lock lever.

(4) Attach cable to turnbuckle at wing rear spar and take up slack.

(5) Remove cable block.

(6) Replace floorboards.

(7) Replace pedestal panels.

C. Removal – Cable 2

(1) Remove center aisle floorboards at rudder/aileron interconnect and just forward of interconnect.

(2) Block cable forward of turnbuckle and aft of aileron interconnect bellcrank lock arm.

(3) Disconnect cable at turnbuckle.

(4) Unbolt cable from aileron bellcrank lock arm and remove cable.

D. Installation – Cable 2

(1) Attach cable to aileron interconnect bellcrank lock arm.

(2) Attach cable at turnbuckle and take up slack.

(3) Remove cable block.

(4) Replace floorboards.

E. Removal – Cable 3

(1) Remove center aisle floorboards from rudder/aileron interconnect to aft interior bulkhead.

(2) Remove aft interior bulkhead.

(3) Block cable forward of aileron interconnect bellcrank lock arm and aft of turnbuckle located aft of interior bulkhead.

(4) Disconnect cable at aileron interconnect bellcrank lock arm and at turnbuckle aft of interior bulkhead.

(5) Remove pulley guards and fairleads as necessary and remove cable.

F. Installation – Cable 3

(1) Position cable and install fairleads and pulley guards as necessary.

(2) Attach cable to aileron interconnect bellcrank lock arm.

(3) Attach cable to turnbuckle aft of bulkhead and take up slack.

(4) Remove cable block.

(5) Replace floorboards.

G. Removal – Cable 4

(1) Remove aft interior bulkhead.

(2) Block cable forward of turnbuckle located aft of bulkhead.

(3) Disconnect cable at turnbuckle.

(4) Working through aft bulkhead access, unbolt cable from bottom of rudder bellcrank control lock arm and remove cable.

H. Installation – Cable 4

(1) Position cable and attach end to rudder bellcrank lock arm.

(2) Attach other end of cable to turnbuckle and take up slack.

(3) Remove cable block.

(4) Replace aft interior bulkhead.

I. Removal – Cable 5

(1) Remove top and center panels from pedestal.

(2) Unbolt cable from gust lock control lever.

(3) Remove cable from pulleys at rear of pedestal by removing cotter pins.

(4) Remove cabin center aisle floorboards from pilot/copilot compartment to aft interior bulkhead.

(5) Remove aft interior bulkhead.

(6) Block cable aft of turnbuckle located aft of interior bulkhead.

(7) Disconnect cable at turnbuckle.

(8) Remove pulley guards and fairleads as necessary and remove cable.

J. Installation – Cable 5

(1) Position cable and replace fairleads and pulley guards as necessary.

(2) Position cable over pulleys on rear of pedestal and insert cotter pins.

(3) Attach cable end to gust control lock lever.

(4) Attach cable to turnbuckle aft of aft interior bulkhead and take up slack.

(5) Remove cable block.

(6) Replace floorboards.

(7) Replace pedestal panels.

K. Removal – Cable 6

(1) Remove aft interior bulkhead.

(2) Block cable forward of turnbuckle.

(3) Disconnect cable at turnbuckle.

(4) Working through aft bulkhead access, unbolt cable from top of rudder bellcrank lock arm and remove cable.

L. Installation – Cable 6

(1) Position cable and attach end to rudder bellcrank control lock arm.

(2) Attach other end of cable to turnbuckle and take up slack.

(3) Remove cable block.

(4) Replace aft interior bulkhead.

M. Removal – Cable 7

(1) Remove aft interior bulkhead.

(2) Working through aft bulkhead access, locate rudder bellcrank lock arm and attached push-pull cable.

(3) Insert 3/32 inch rig pin into rudder bellcrank lock plunger (see Figure 2, Section 27-00-03).

(4) Straighten end of inner strand of push-pull cable and loosen bolt securing strand end.

(5) Loosen bolts at fitting securing outer housing of push-pull cable.

(6) Remove elevator quadrant access door located on right side of vertical stabilizer.

(7) Locate push-pull cable attached to lock at small end of elevator quadrant.

(8) Straighten end of inner strand and loosen bolt securing strand end.

(9) Loosen bolts at fitting securing outer housing of push-pull cable.

(10) Pull cable out through pressure plate (above rudder bellcrank lock) and remove cable.

NOTE: Normally the outer housing of the cable is not damaged and the inner strand is broken. If the inner strand only is to be replaced, proceed as above omitting step (5) and steps (9) (10) and pull inner strand ends out of outer housing from top and bottom.

N. Installation – Push-pull cable with inner strand

(1) Insert cable up through pressure plate and retrieve end through elevator quadrant access door.

(2) Pass upper end of cable through housing fitting at quadrant lock.

(3) Pass upper end of inner strand through hole in bolt at quadrant lock.

(4) Bend inner strand end around bolt 90° or more and tighten bolt.

(5) Tighten bolts at housing fitting securing cable housing.

(6) Pass lower end of cable through housing fitting at rudder bellcrank lock arm.

(7) Pass lower end of inner strand through hole in bolt at rudder bellcrank lock arm.

(8) Remove rig pin from rudder lock.

(9) Place gust lock lever in LOCK position.

(10) While co-worker pulls back on control column, pull down on cable inner strand until elevator quadrant locks.

(11) Hold fast and bend strand around bolt at rudder lock 90° or more and tighten bolt.

(12) Replace aft interior bulkhead.

(13) Replace quadrant door.

O. Installation – Inner strand only

(1) Measuring from removed strands, cut new strand.

(2) Deburr end of strand to be inserted into housing.

(3) Lubricate strand with silicone spray or light oil.

(4) Working through top at elevator quadrant lock, insert strand through cable housing.

(5) Pass lower end of strand through hole in bolt at rudder bellcrank lock arm.

(6) Pass upper end of strand through hole in bolt at elevator quadrant lock.

(7) Bend strand end around bolt 90° or more and tighten bolt.

(8) Remove rig pin from rudder lock.

(9) Place gust lock lever in LOCK position.

(10) While co-worker pulls back on control column, pull down on cable inner strand until elevator quadrant locks.

(11) Hold fast and bend strand end around bolt at rudder lock 90° or more and tighten bolt.

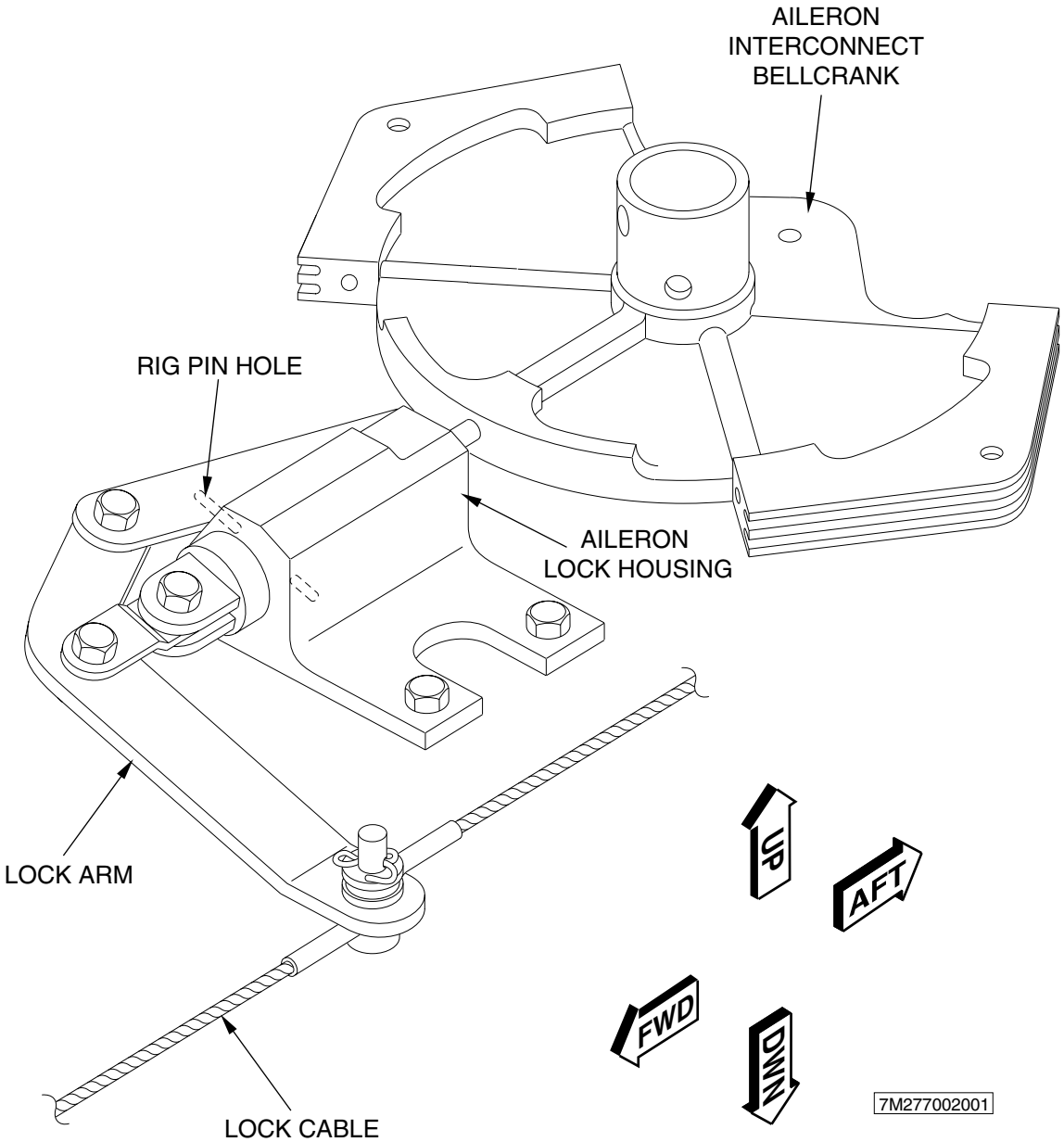
(12) Replace aft interior bulkhead.

(13) Replace quadrant access door.

AILERON GUST LOCK – DESCRIPTION AND OPERATION

1. General

A. The aileron gust lock system is a cable controlled system that activates a spring-loaded plunger mechanism. The plunger mechanism is located forward of the aileron interconnect bellcrank. When the gust lock control lever, located on the cockpit pedestal, is engaged (LOCKED), the lock plunger, or pin, is set into a hole located in the aileron interconnect bellcrank. When the pin is in the bellcrank, both ailerons are locked in the neutral (streamlined) position. When the control lever is set to the CONTROL LOCK OFF position, the pin is disengaged from the bellcrank allowing the bellcrank to move freely for aileron operation.



Aileron Gust Lock
Figure 1

AILERON GUST LOCK – MAINTENANCE PRACTICES

1. Removal / Installation – Aileron Gust Lock

A. Removal – Aileron Gust Lock

(1) Support control column and place gust lock lever in UNLOCKED position.

(2) Remove floorboard at rudder/aileron interconnect (located aft of main spar floorboard).

(3) Separate aileron lock arm from lock cable by removing cotter pin, nut, washer, and bolt.

(4) Remove bolts (four), nuts (four), and washers (four) from lock housing and lift out lock assembly.

(5) Remove lock plunger from lock housing as necessary.

B. Installation – Aileron Gust Lock

(1) Insert lock plunger into lock housing and secure.

(2) Position lock assembly with housing and secure with bolts (four), washers (four), and nuts (four).

(3) Connect cables with bolt and washer, and secure with nut and cotter pin.

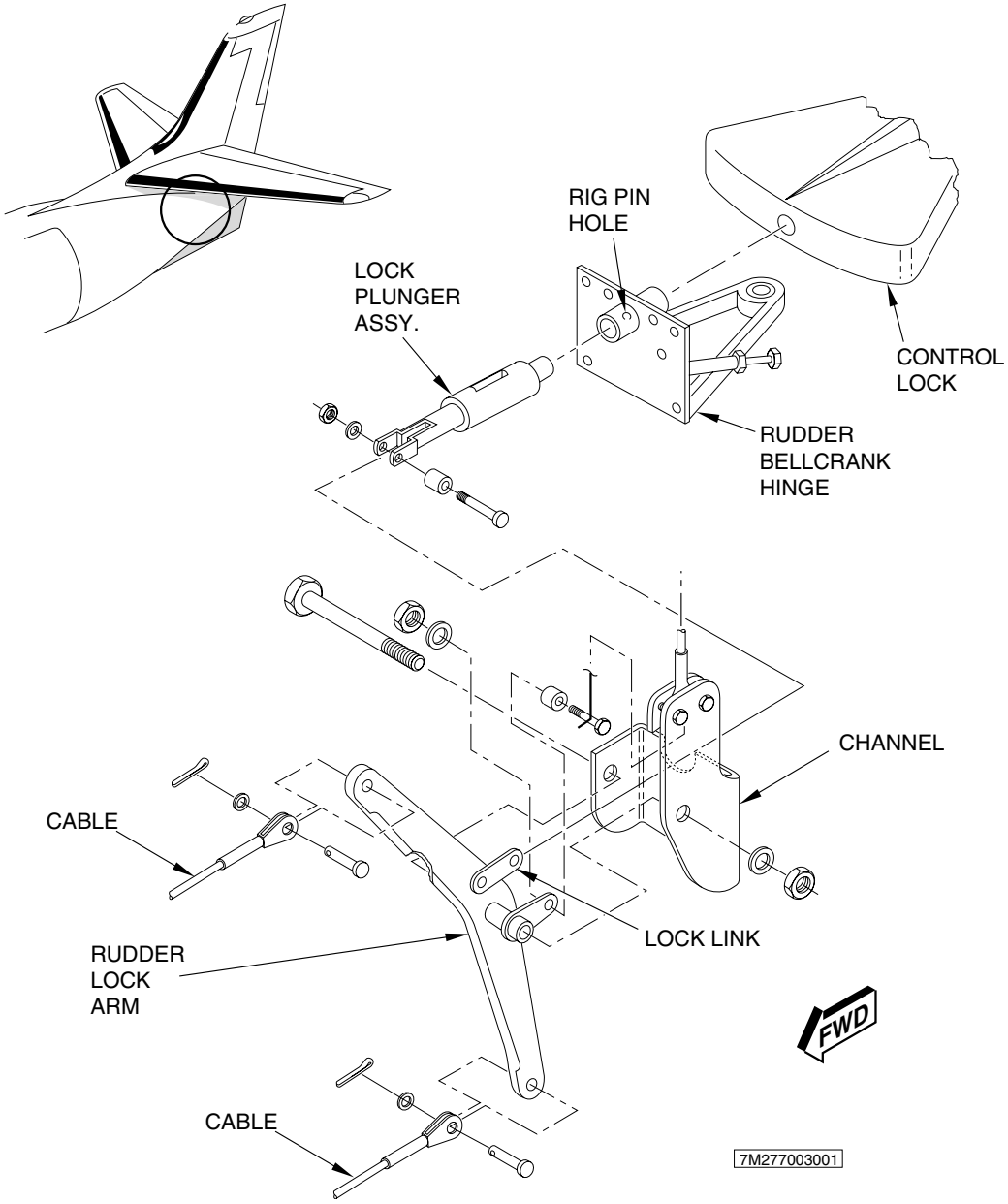
(4) Operational test by engaging gust lock and turning control wheel until ailerons lock.

(5) Replace floorboard.

RUDDER GUST LOCK – DESCRIPTION AND OPERATION

1. General

A. The rudder gust lock system is a cable controlled system that activates a spring-loaded plunger mechanism. The plunger mechanism is housed within the rudder bellcrank hinge located just forward of the rudder control lock, which is connected to the rudder bellcrank. When the gust lock control lever, located on the cockpit pedestal, is engaged (LOCKED), the lock plunger, or pin, is set into a hole located in the rudder control lock. When the pin is in the control lock the rudder is locked in the neutral (streamlined) position. When the control lever is set to the CONTROL LOCK OFF position, the pin is disengaged from the control lock allowing the rudder bellcrank to move freely for rudder operation.



Rudder Gust Lock
Figure 1

RUDDER GUST LOCK – MAINTENANCE PRACTICES

1. Removal / Installation – Rudder Gust Lock

A. Removal – Rudder Gust Lock

(1) Support control column and place gust lock lever in UNLOCKED position.

(2) Gain access to inside aft bulkhead (forward of aft pressure bulkhead) by removing interior aft bulkhead.

(3) Locate rudder lock arm (see Figure 1) by following gust lock cables.

(4) Remove bolt, nut, washer, and spacer connecting lock link to rudder lock arm.

(5) Remove lock plunger assembly and spring from rudder bellcrank hinge by pulling straight out.

(6) Remove lock link from lock plunger assembly if replacing plunger.

B. Installation – Rudder Gust Lock

(1) If removed, attach lock link to lock plunger with spacer, bolt, washer, and nut.

(2) Insert spring into rudder bellcrank hinge.

(3) Insert lock plunger assembly into rudder bellcrank hinge.

(4) Attach lock link to rudder lock arm with spacer, bolt, washer, and nut.

(5) Operational check by engaging gust lock and pushing rudder pedals until rudder locks.

CAUTION: AIRCRAFT EQUIPPED WITH RUDDER GUST LOCK P/N 27-70102-001 (REPLACEMENT FOR P/N 27-70066-001) DO NOT HAVE THE COLLAPSIBLE LOCKING PLUNGER THAT ALLOWS FOR THE GUST LOCK SYSTEM ENGAGEMENT WITH THE RUDDER DISPLACED LEFT OR RIGHT OF LOCKED POSITION. TRYING TO FORCE THE ENGAGEMENT OF THIS NEW RUDDER GUST LOCK WITH THE RUDDER DISPLACED LEFT OR RIGHT OF THE NEUTRAL (FAIRED) POSITION MAY CAUSE DAMAGE (STRETCHING) TO THE GUST LOCK CABLES OR DAMAGE TO RELATED GUST LOCK MECHANISMS. IT IS HIGHLY RECOMMENDED THAT ALL PERSONNEL, BOTH MAINTENANCE AND OPERATIONS, BE AWARE OF THIS CONDITION. MAKE SURE THAT THE RUDDER IS IN THE NEUTRAL (FAIRED) POSITION BEFORE ENGAGING THE GUST LOCK CONTROL HANDLE ON THE PEDESTAL.

(6) Replace bulkhead.

NOTE: Replacing the 27-70066-001 Lock Plunger with the solid 27-70102-001 Lock Plunger removes the 5000 Hour replacement requirement called out in Chapter 05 of this manual.

ELEVATOR GUST LOCK – DESCRIPTION AND OPERATION

1. General

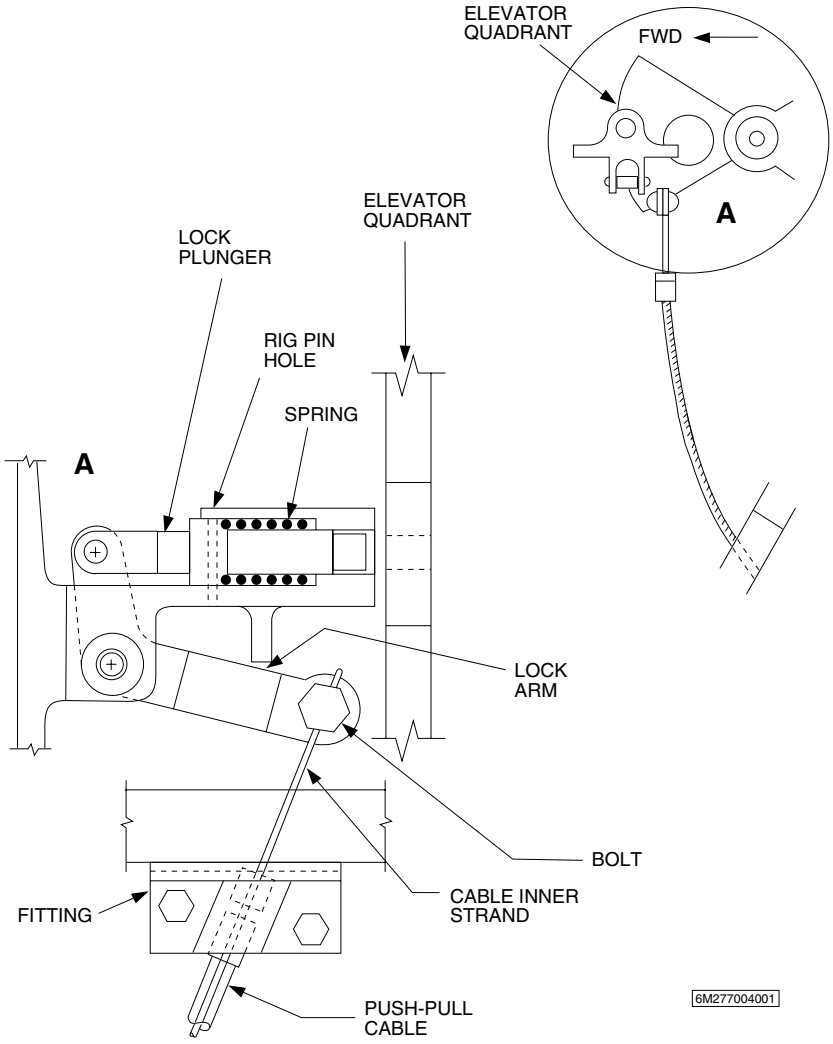
A. SA226 aircraft on which SB 27-041 has not been accomplished, have a cable operated, spring-loaded plunger gust lock system. Aircraft fitted according to SB27-041 are equipped with a gust lock belt.

2. Plunger Gust Lock (Figure 1)

A. A gust lock control lever, located on the pedestal. operates a spring-loaded plunger mechanism by cable. The plunger mechanism is located within the aft spar fitting, just forward of the elevator quadrant.

B When the gust lock control lever is engaged (LOCKED), the lock plunger, or pin is set into a slot located in the narrow end of the elevator quadrant.

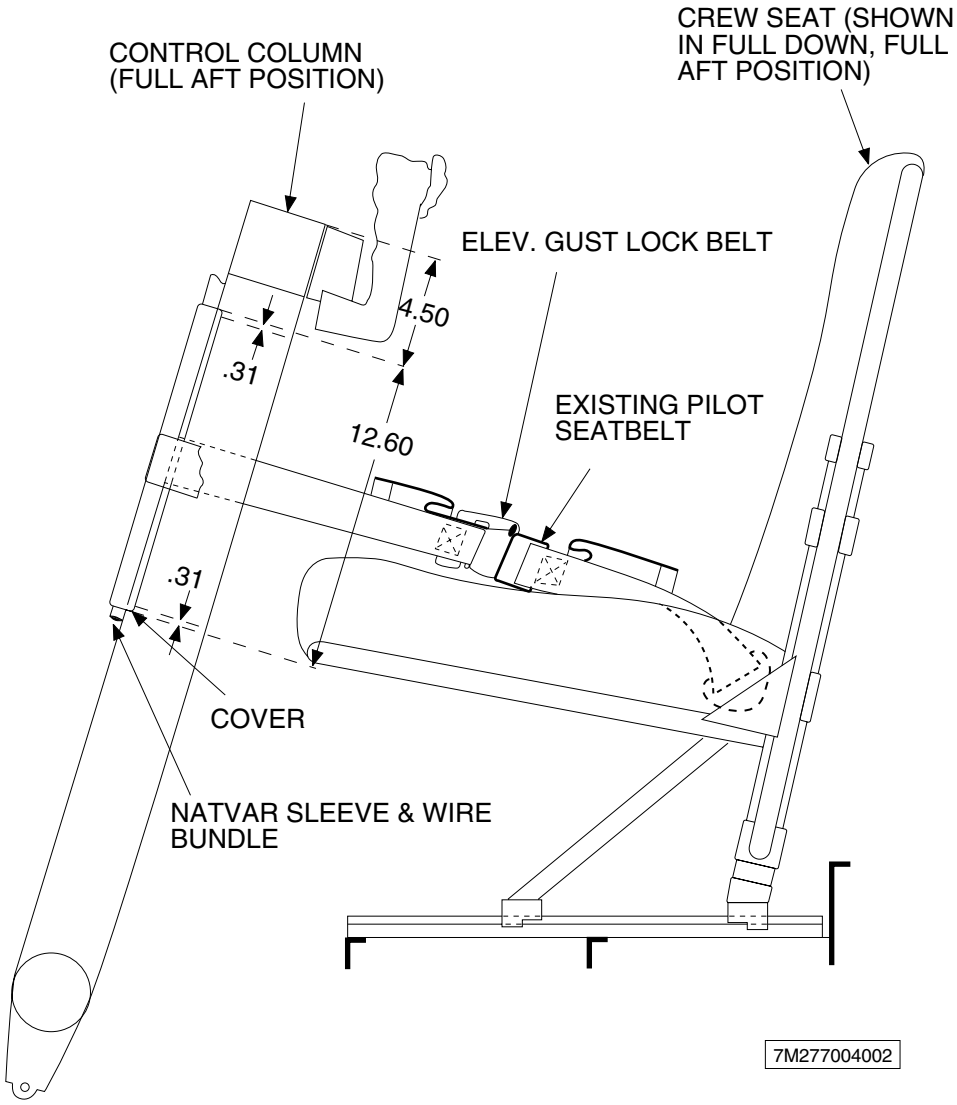
C. When the control is set to the CONTROL LOCK OFF position, the pin is disengaged from the quadrant allowing it to move freely for elevator operation.



Elevator Gust Lock (PlungerType)
Figure 1

3. Elevator Gust Lock Belt (Figure 2)

A. On aircraft conforming to SB 27-041, the spring-loaded plunger mechanism has been removed. In place of the plunger, a gust lock belt secures the LH control column to the LH pilots seat belt (see Figure 2).



Elevator Gust Lock (Alternate)
Gust Lock belt
Figure 2

ELEVATOR GUST LOCK – MAINTENANCE PRACTICES

1. Removal/Installation – Elevator Gust Lock

NOTE: Aircraft on which SB 27-041 has not been accomplished use a cable operated, plunger type gust lock system. Aircraft on which SB 27-041 has been accomplished are equipped with an elevator gust lock belt.

A. Removal – Plunger Gust Lock

(1) Support control column and place gust lock lever in UNLOCKED position.

(2) Place horizontal stabilizer and elevators in streamline (zero) position.

(3) Remove elevator quadrant access panel (large one on right of vertical stabilizer.

(4) Remove right elevator fairing (for easy access to lock mechanism).

(5) Remove bolt, bushing, washer, and nut securing inner cable strand to lock arm.

(6) Remove bolt, bushing, washer, and nut securing lock arm to aft spar fitting (located at center of lock arm).

(7) Remove bolt, bushing, washer, and nut attaching upper end of lock arm to lock plunger.

(8) Remove lock plunger through hole in left of aft spar fitting.

(9) Remove spring inside lock plunger casing of aft spar fitting.

B. Installation – Plunger Gust Lock

(1) Insert spring into casing.

(2) Insert lock plunger into casing.

(3) Position lock arm and attach upper end of lock arm to lock plunger with bolt, bushing, washer, and nut.

(4) Attach lock arm at pivot point to spar fitting with bolt, bushing, washer, and nut.

(5) Attach cable strand to lock arm with bolt, bushing, washer, and nut.

(6) Operational check by engaging gust lock and pulling back on control column until elevators lock.

(7) Install quadrant access panel.

(8) Install elevator fairing.

C. Removal – Gust Lock Belt

CAUTION: BE SURE TO RESTRAIN CONTROL COLUMN WHEN RELEASING ELEVATOR GUST LOCK BELT. FAILURE TO RESTRAIN COLUMN MAY CAUSE FLIGHT INSTRUMENT DAMAGE.

(1) Hold control column firmly.

(2) Release belt and slowly move column to full forward position.

(3) Stow elevator gust lock belt.

D. Installation – Gust Lock Belt

(1) Move pilot's control column full forward. Pilot's seat may be in any position.

(2) Attach elevator gust lock belt into crew seatbelt, and around control column.

(3) Pull control column full aft and tighten belt to hold column aft.

2. Inspection/Check – Elevator Gust Lock

A. Inspection – Plunger Gust Lock System

(1) At regularly scheduled inspections:

(a) Disassemble elevator gust lock assembly. Inspect for corrosion or contamination. Apply a light coat of Dow Corning Molycote 33 to the pins and sockets. Take care not to get lubricant on other moving parts of the gust lock assembly which could increase the operating loads.

(b) Install spring and plunger and operate gust lock several times. Wipe off excess lubricant.

(c) Inspect elevator gust lock push/pull cable for condition of the inner cable strands where cable connects to gust lock arm. Connect push/pull cable to gust lock arm. Check for binding or buckling of cable when gust lock is actuated. Evidence of binding or buckling shall be cause for rejection.

(d) Verify freedom of operation of gust lock controls. Engage gust locks and verify that elevator gust lock unlocks before the gust lock handle travels one half the distance toward the unlock position. If necessary, adjust the push/pull cable at the rudder lock arm.

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MAINTENANCE MANUAL

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FUEL JET PUMP DESCRIPTION AND OPERATION	28-10-30	1	T AT TC	201-999 001-999 201-999
FUEL JET PUMP FAULT ISOLATION	28-10-30	101	T AT TC	201-999 001-999 201-999
FUEL JET PUMP MAINTENANCE PRACTICES	28-10-30	201	T AT TC	201-999 001-999 201-999
FUEL CROSSFLOW DESCRIPTION AND OPERATION	28-10-40	1	T AT TC	201-999 001-999 201-999
FUEL CROSSFLOW FAULT ISOLATION	28-10-40	101	T AT TC	201-999 001-999 201-999

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CHAPTER 28 – FUEL

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FUEL DISTRIBUTION DESCRIPTION AND OPERATION	28-20-00	1	T AT TC	201-999 001-999 201-999
FUEL BOOST PUMP DESCRIPTION AND OPERATION	28-20-10	1	T AT TC	201-999 001-999 201-999
FUEL BOOST PUMP MAINTENANCE PRACTICES	28-20-10	201	T AT TC	201-999 001-999 201-999
FUEL SHUTOFF VALVE DESCRIPTION AND OPERATION	28-20-20	1	T AT TC	201-999 001-999 201-999
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FUEL SERVICE INFORMATION	28-SERVC	1	T AT TC	201-999 001-999 201-999

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FUEL – SERVICE INFORMATION

1. General

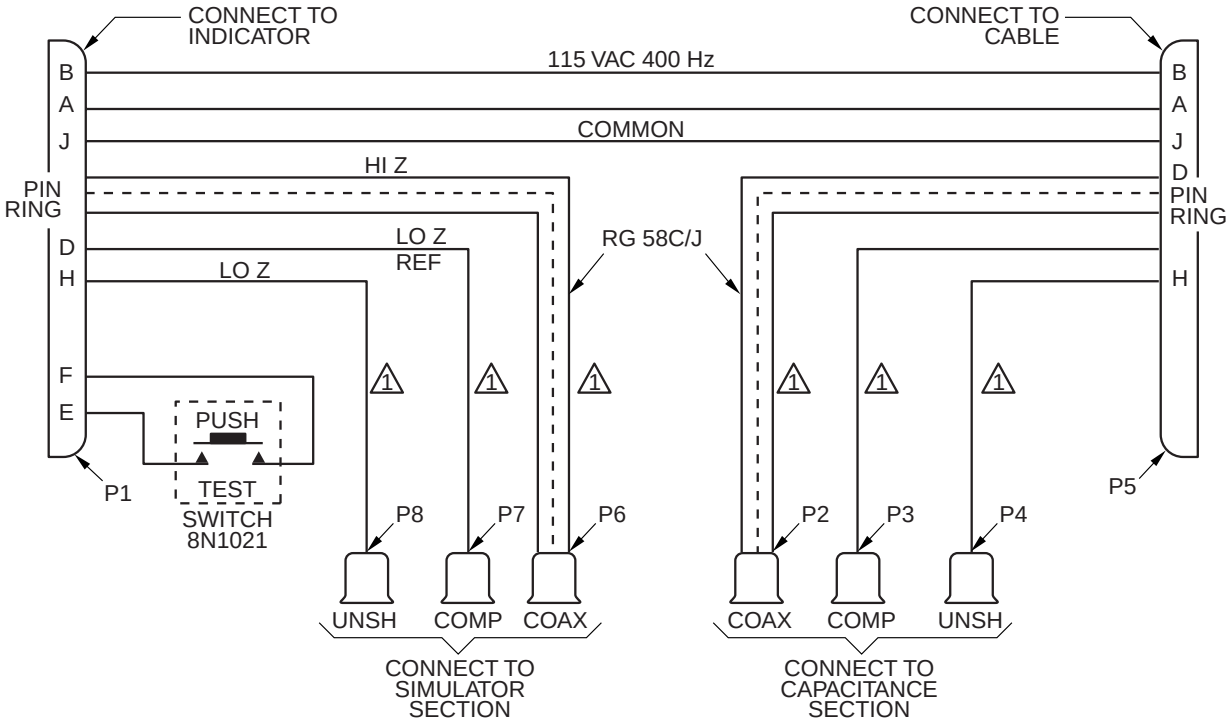
A. The following information is supplied as a supplement to this chapter. Subject matter is included which is not applicable to an ATA 100 format, but is still of interest to Merlin and Metro owner/operators. The information in this section is not part of the basic manual, and may be removed or retained at the discretion of the owner/operator.

B. Revisions and additions to this section will be made as additional information is available. Input from individual owner/operators is encouraged, and may be submitted to:

M7 Aerospace, LP Post Office Box 790490 San Antonio, Texas 78279-0490 Attention: Technical Publications Department

C. 28-01 Adapter A

(1) Adapter A is used to connect through the fuel quantity indicator connector to test the entire fuel quantity system. Individual fuel sensor probes may be isolated only by removing the individual electrical connector to the respective probe. If the fuel quantity tester also has a simulator section, this harness may be used to test the indicator. Wiring details for this harness are shown in Figure 1.



DES	P/N	DES	P/N
P1	165 61 1008	P5	EJC39
P2	EJAO2MO2	P6	EJAO2MO2
P3	EJAO3MO1	P7	EJAO3MO1
P4	EJAO3FO1	P8	EJAO3FO1

Adapter A Schematic
Figure 1

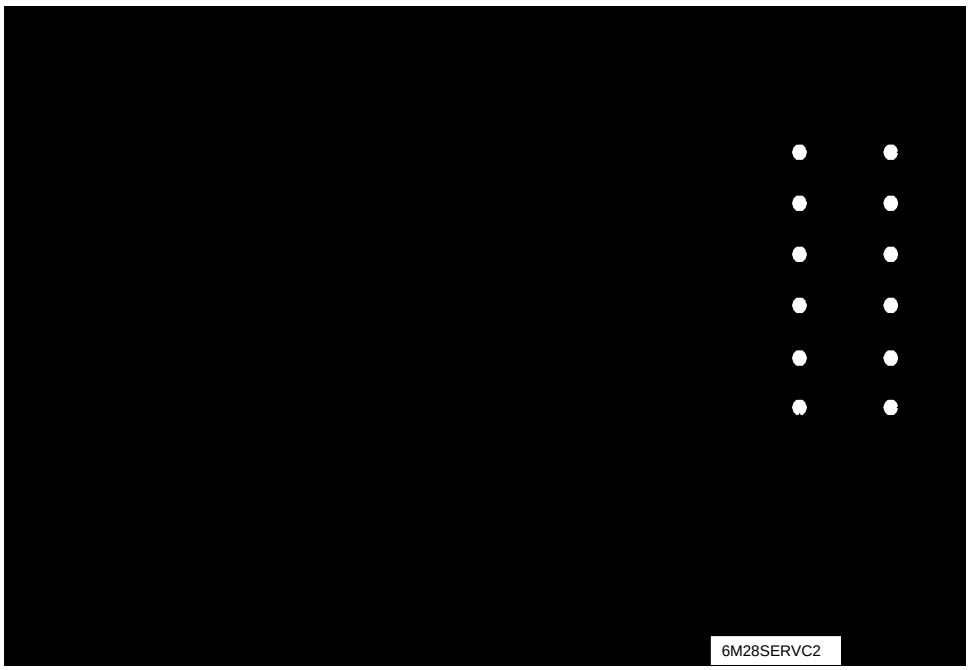
D. 28-02 Adapter B

EFFECTIVITY:
T238-999
AT015-999
TC208-999

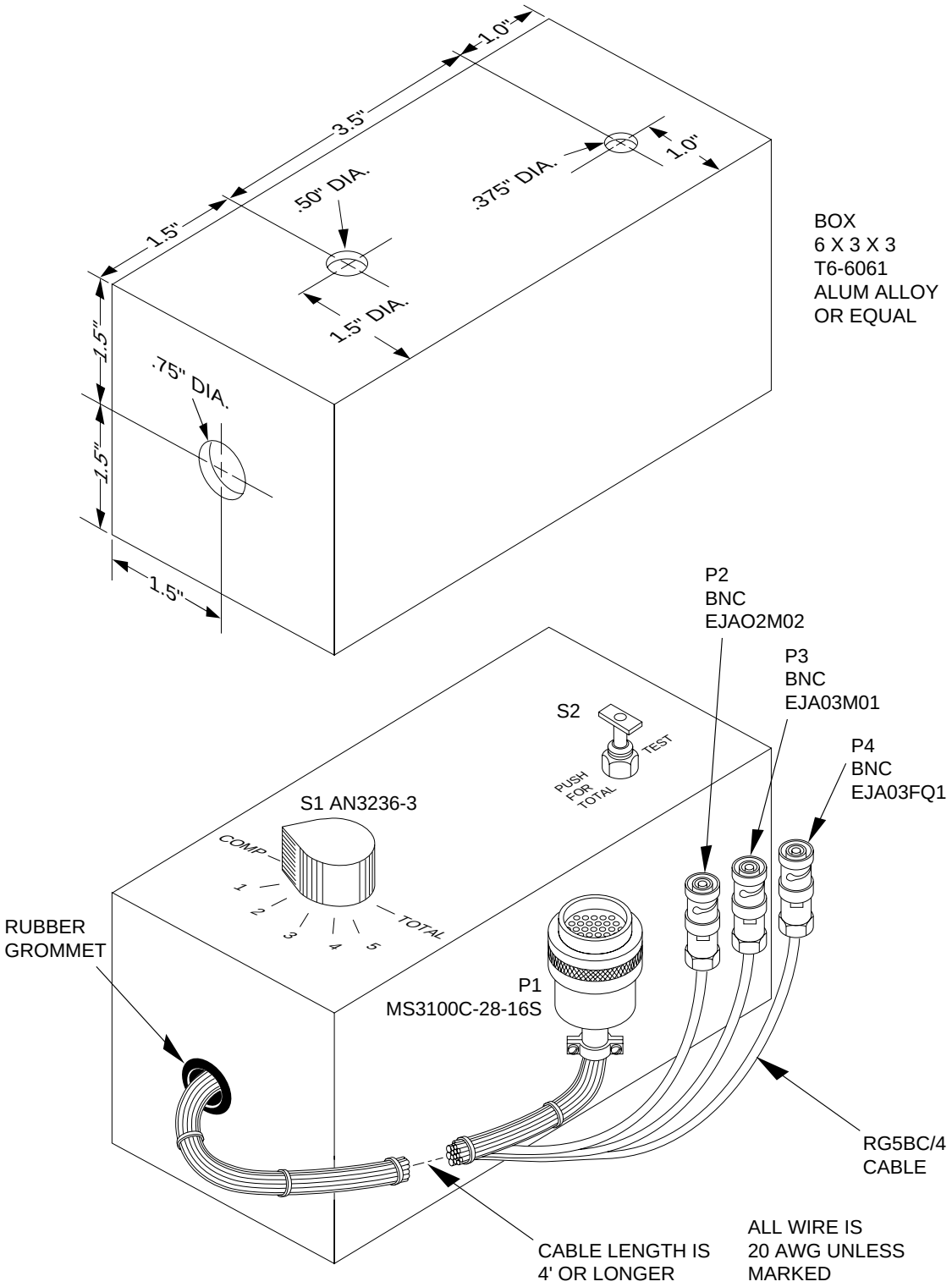
(1) Adapter B allows the entire wing tank fuel sensor system to be checked through one connection. Construction may utilize whatever components are available; however, the jumpers in connector P1 should be kept as short as possible. Figure 2 provides the internal wiring diagram while Figure 3 shows a typical assembly.

E. 28-03 Invermotor Boost Pumps

(1) Installation of the optional invermotor fuel boost pumps (P/N 160-009) has resulted in some undersirable audio characteristics in both the aircraft communications and pitch trim warning circuits. This problem will be apparent by the presence of a hum (approximately 1 KHz) from the cockpit speakers or by continuous operation of the out-of-trim sonalert in flight whenever a boost pump is turned on. To correct this problem, the 28 VDC leads from the boost pump switch to pin A of the respective boost pump should be replaced with 16 gauge shielded wire which bypasses the wing-to-fuselage disconnect plug. The shield should be grounded at the pump connector (reference Wiring Diagram 28-20-01).



Adapter B Schematic
Figure 2



Adapter B Assembly
Figure 3

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FUEL – DESCRIPTION AND OPERATION

1. General

A. The fuel system has a usable capacity of 648 U.S. gallons (2452.68 liters) for all SA226–T and SA226–TC aircraft, and for SA226–AT serial numbers AT001 through AT024. SA226–AT aircraft serial number AT025 and up have a usable capacity of 554 U.S. gallons (2096.89 liters). Fuel is equally contained in integral left and right wing fuel tanks.

B. Each engine is supplied fuel by an independent fuel system. An interconnecting crossflow line is provided for balancing the fuel quantity and for supplying either engine with fuel. This system is electrically controlled by a switch on the instrument panel. Fuel crossflow is controlled by a shut off valve in the crossflow line.

C. Each integral fuel system includes:

(1) A jet transfer pump system for supplying fuel to the boost pumps.

(2) Two boost pumps to supply fuel to the engines.

(3) Three check valves, one from each boost pump to the fuel line and one in the jet transfer pump inlet line.

(4) A fuel shutoff valve in the fuel line.

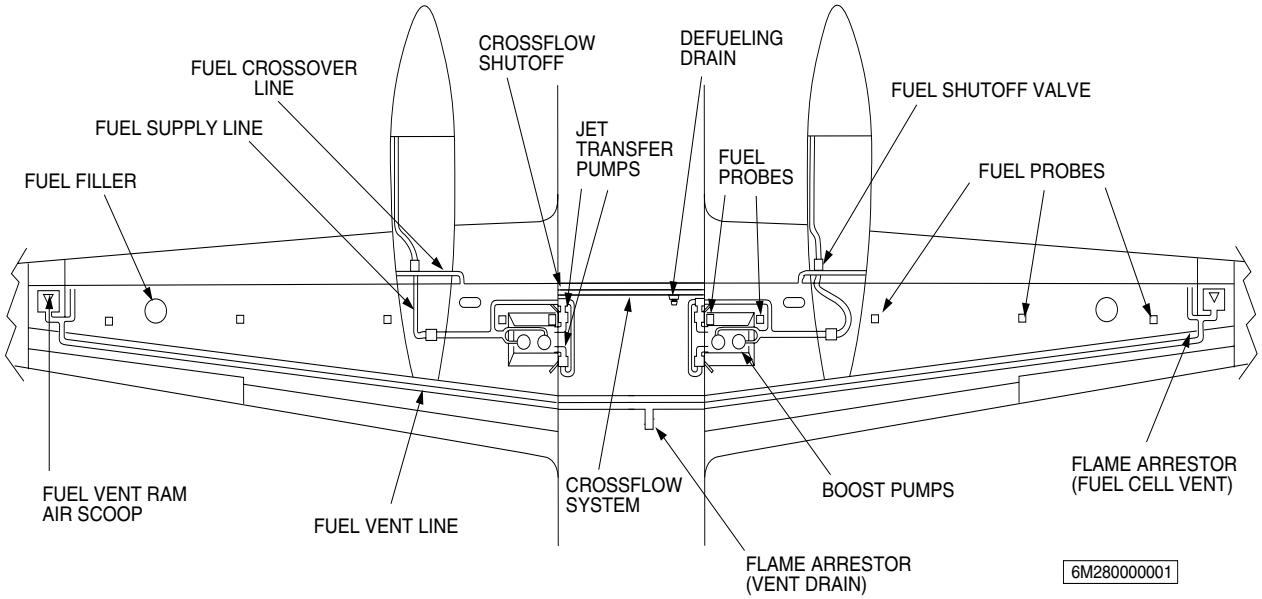
(5) A fuel vent system that allows air to enter fuel tanks to displace used fuel.

(6) A separate capacitance type quantity indicating system.

(7) Two indicators mounted on the instrument panel that display the fuel quantity in pounds.

(8) Magna sticks to provide mechanical indication is a customer option.

D. A transfer pump warning light on the annunciator panel illuminates when the jet pumps fail to maintain the hopper tank at full capacity.



Fuel
Figure 1

1. Servicing – Fuel

A. Defueling Aircraft

NOTE: Aircraft T201-T248, AT001-AT024, and TC201-TC211EEEE may be defueled using the following procedures:

(1) Position aircraft at least 50 feet from any potential source of ignition.

(2) Static ground aircraft. Static ground defueling equipment to ground and to aircraft.

(3) Remove filler cap from tank being defueled.

(4) Remove cowling from left hand side of engine nacelle to gain access to engine fuel supply line connection at the firewall.

(5) Disconnect supply line and connect suction equipment line to firewall fitting.

NOTE: When defueling at engine firewall, disconnect hose at engine side of firewall. When disconnecting hose be sure to secure union which protrudes through the firewall. Should this union turn, fuel lines aft of firewall may loosen and are difficult to retighten.

(6) Turn on aircraft boost pump to prime suction equipment lines.

CAUTION: THE BEARINGS OF THE BOOTS PUMPS ARE LUBRICATED BY THE FUEL, AND DAMAGE TO THE PUMPS WILL RESULT IF PUMPS ARE RUN DRY.

(7) Start suction equipment.

(8) Shut off aircraft boost pump.

(9) Defuel as required.

(10) After defueling operation, install fuel filler cap.

(11) Disconnect suction line and connect fuel supply line at firewall fitting.

(12) Install cowling.

NOTE: Residual fuel may be drained at sump drains located in each wing tank lower skin between nacelle and center section.

B. Defueling Using Crossflow Drain

NOTE: On Aircraft T249 and up, AT025 and up, and TC212 and up, a quick release drain valve is installed in the crossflow line to allow for rapid single point defueling of the aircraft.

Aircraft equipped with crossflow drain fitting may be defueled using the following procedures:

(1) Position aircraft at least 50 feet from any potential source of ignition.

(2) Static ground aircraft. Static ground defueling equipment to ground and to aircraft.

(3) Open crossflow valve (crossflow switch is on pedestal).

(4) At center section, lower wing skin, remove access panel located just forward of right landing light. Locate fuel drain fitting with dust cover (red nipple) attached.

(5) Remove dust cover by depressing end cap and pulling straight out.

(6) Remove fuel filler caps.

(7) Install fuel drain tube assembly (Fairchild part number 27-64043-001) or comparable drain hose assembly.

(8) Defuel as required.

(9) After defueling operation, install fuel filler caps.

(10) Disconnect drain hose assembly and replace dust cover by pressing straight on.

(11) Install access panel.

NOTE: Residual fuel may be drained at sump drains located in each wing tank lower skin between nacelle and center section.

C. Refueling Aircraft

(1) Refuel aircraft IAW Chapter 12.

2. Inspection/Check – Fuel Tank

A. Inspection – Fuel Tank

NOTE: Repetitive inspections are required of all fuel lines in the left and right wheel wells. The following steps shall be taken after every 50 flight hours on aircraft T201-T271, AT001-AT056, and TC201-TC228 (except TC227E) unless Service Bulletin 54-003 has been complied with:

(1) Apply fuel pressure with boost pumps.

(2) Inspect fuel crossover line, main fuel supply line, and fuel shutoff valve for leaks or seepage.

(3) Repair or tighten where necessary.

3. Use of Biobor JF Microbiocide in Fuel Tanks

A. General

(1) Biobor JF is a liquid fuel additive that combats fungus and other microbial life (hydrocarbon utilizing micro-organisms known as HUM Bugs) in hydrocarbon fuels. It eliminates growth of harmful slime-producing fungi that may clog filters, attack rubber fuel system components and whose waste products aid in the corrosion of metal surfaces.

B. Use of Biobor JF

NOTE: Other brands (SOHIO or equivalent) may be used in lieu of Biobor JF at 270 ppm maximum (20 ppm of elemental boron) for fuel pesticide purposes.

(1) Sump fuel tanks to remove as much water as possible. Water at the bottom of the fuel tanks should be kept to a minimum to reduce the possibility of fungi growing in the tank.

It is recommended that the fuel tanks be sumped daily to reduce the possibility of fungus growth in the tank.

(2) Biobor JF may be used at 270 ppm for shock treatment in fuel to effect sterilization and subsequently at 135 ppm as a maintenance treatment. Biobor JF must be added during the addition of fuel to assure adequate mixing. Introduce the Biobor JF while the tank is being filled ***and*** after the tank is approximately 1/2 full. The mixing ratio is as follows:

Fuel (Gallons)	**40**	**50**	**60**	**100**	**300**
Biobor JF (Ounces) for Maintenance Dosage	**1/2**	**3/4**	**7/8**	**1-1/2**	**4**
Biobor JF (Ounces) for Shock Treatment	**1**	**1-1/2**	**1-3/4**	**3**	**8**

CAUTION: **BIOBOR JF IS TOXIC. HANDLE IN ACCORDANCE WITH MANUFACTURER'S INSTRUCTIONS.**

(4) Fuel tanks with heavy fungus contamination could lead to clogged fuel filters following the Biobor treatment. Clean or replace engine fuel filters following Biobor JF shock treatment and after each of three (3) subsequent fuel loads.

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FUEL STORAGE – DESCRIPTION AND OPERATION

1. General

A. Fuel for each engine is stored in two integral fuel tanks, one located in each wing. Each tank serves as an independent fuel system for its respective engine.

2. Interconnect

A. The tanks are interconnected by a crossflow line to balance the fuel quantity and to provide either engine with all available fuel.

3. Collector and Hopper Tanks

A. Two gravity filled collector tanks, one in each wing, coupled with jet transfer pump action, supply a hopper tank with fuel, ensuring boost pumps are submerged in all flight attitudes.

4. Over-wing Fillers

A. The over-wing fuel filler for all SA226–T and SA226–TC aircraft is factory installed at W.S. 196.6. SA226–AT aircraft through AT025 and aircraft that have complied with Service Bulletin 28-004 have the fuel filler installed farther inboard at W.S. 141.5. All SA226–AT aircraft starting with AT066 have the fuel filler installed at W.S. 196.6

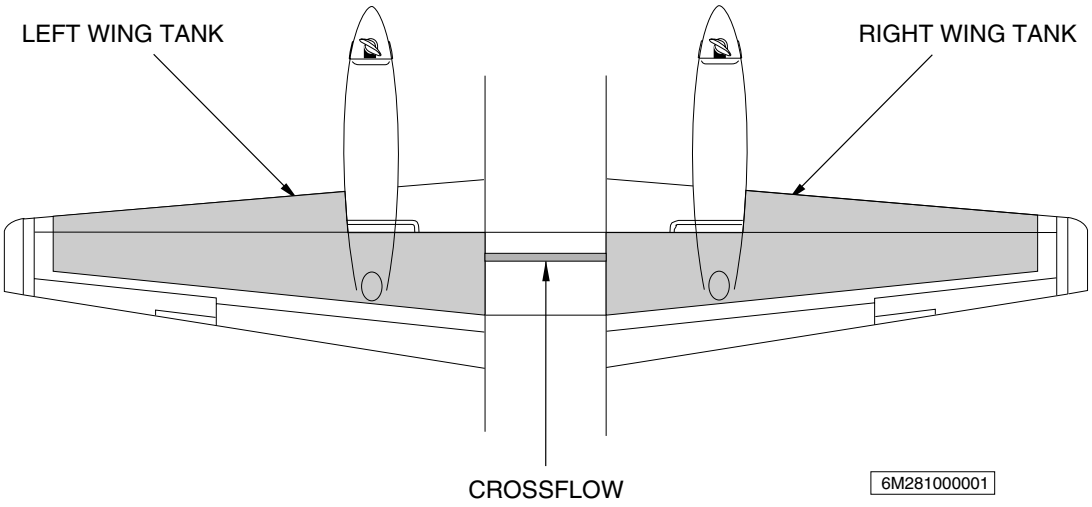
5. Sump Drains and Drain Valves

A. Two sump drain valves are located just outboard of W.S. 27.00, forward of the boost pump access panel. The valves are used to drain accumulated water or to drain residual fuel from the tanks.

6. Fuel Tank Vent

A. A flush-mounted vent is located on the lower surface of the wing tip aft of the main spar. A vent balance line connects with the vent system and is routed along the entire length of the wing behind the rear spar.

B. A flame arrester vent drain is located on the lower aft center wing section aft of rear spar inbd of W.S. 27.00.



Fuel Storage
Figure 1

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FUEL TANK – DESCRIPTION AND OPERATION

1. General

A. The fuel tank in each wing is formed by the upper and lower wing skin, the main and rear spars and the ribs between W.S. 27.00 and 248.00. The intermediate ribs serve as baffles to reduce fuel surges. All surfaces are joined with a sealer having elastic properties.

B. Access panels are provided in the lower wing surfaces. A phenolic insert is installed between each access panel and the tank to serve as a lightening arrestor.

C. An anti-bacteriological coating, compatible with sealants and fuel additives is applied to all inner tank surfaces.

2. Flapper Valves

A. Flapper valves are installed in the ribs at W.S. 132.00, 174.00, and 218.00. These valves permit fuel to flow inboard at all times. During flight operations in a wing low attitude, the weight of the fuel closes the valves and prevents a surge of fuel in the low wing.

3. Wing Tank Divisions

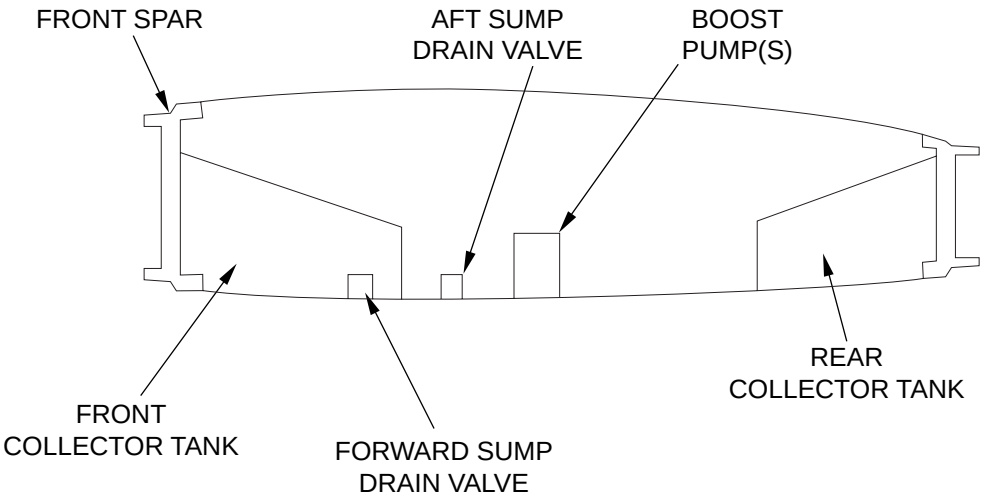
A. The fuel tank area between W.S. 27.00 and 39.00 is divided into three compartments. The fore and aft compartments form collector tanks.

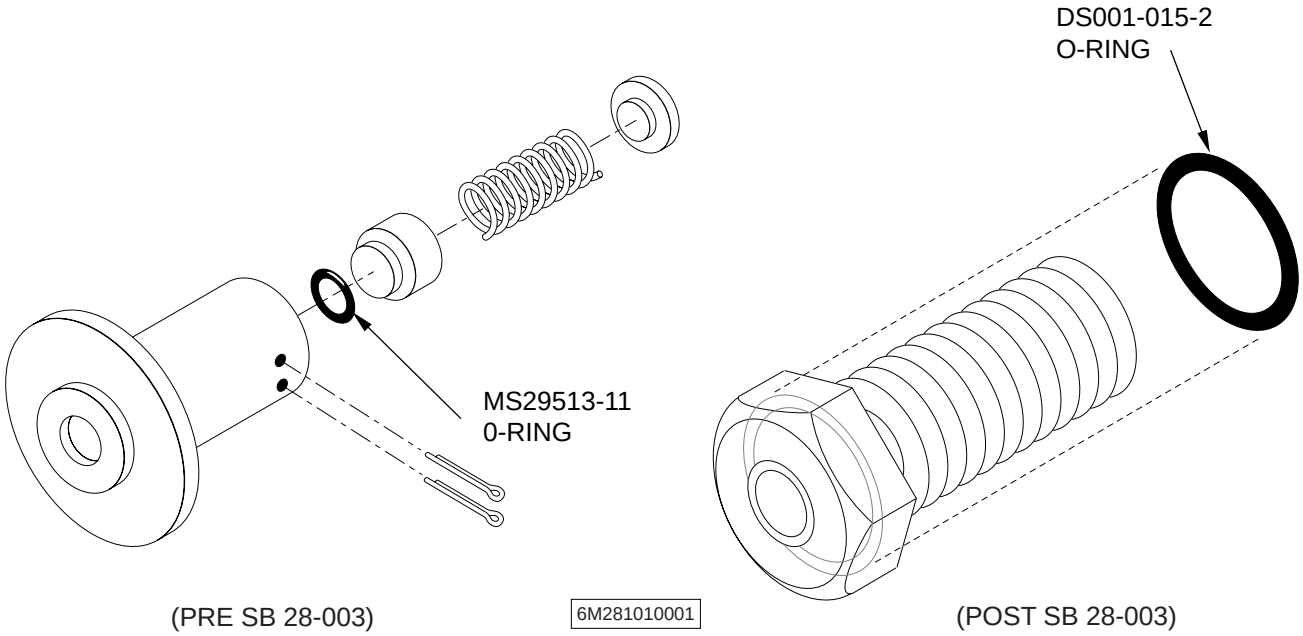
B. The center compartment, called the hopper tank, contains the boost pumps. The hopper tank is sealed from the remainder of the fuel tank by one-way flapper valves. A float switch installed in each hopper tank activates a warning light on the annunciator panel when the fuel level in the hopper tank falls below capacity.

C. The collector tanks are open to the main fuel tank. A jet pump system supplies fuel from the collector tanks to the hopper tank.

D. Each wing tank has two spring-loaded, plunger type sump drains for residual draining. The drain outlets are located on the underside of each wing tank, adjacent to each other in front of the boost pump access panel at W.S.27 and stringer number 12. The aft outlet drains the hopper tank, and the forward outlet drains the forward collector tank.

E. Earlier aircraft were fitted with internal drains riveted to the wing skin. Later aircraft and retrofitted aircraft (post SB 28-003) are fitted with a drain incorporating a removable element. See Figure 1 for illustration of old and new style sump drains.





Sump Drain Valve
Figure 201

FUEL TANK – MAINTENANCE PRACTICES

1. Removal/Installation – Fuel Tank

A. Removal – Sump Drain (Pre SB 28-003)

NOTE: Fitted on aircraft S/N T201-248, AT 001-024, and TC 201-211, on which SB 28-003 has not been accomplished.

(1) Defuel aircraft.

(2) Remove boost pump or collector tank panel to gain access to valve.

(3) The valve may be removed as an assembly by removing rivets and removing valve. The two cotter pins may be removed and the valve disassembled while in the aircraft (see Figure 1).

B. Installation – Sump Drain (Pre SB 28-003)

(1) Install new valve assembly or reassemble old valve as applicable.

(2) Close up access panel or install boost pump.

(3) Service aircraft with fuel IAW Chapter 12.

(4) Check drains for leaks. If drains are seeping, apply shop air pressure through a nozzle tip around perimeter of drain.

C. Removal – Sump Drain (Post SB 28-003)

NOTE: Fitted on aircraft S/N T 249 and up, AT 025 and up, TC 212 and up and aircraft on which SB 28-003 has been accomplished.

It is possible to remove and replace the drain with the aircraft fueled; however, some fuel spillage will result. A new drain and O-ring should be in hand and installed quickly.

(1) Defuel aircraft.

(2) Remove drain valve.

D. Installation – Sump Drain (Post SB 28-003)

(1) Lubricate O-ring with Parker O Lube or similar compound.

(2) Lubricate Sump Drain Valve threads with Anti-seize compound C-601-S, MIL-T-5544.

(3) Install O-ring on new valve (see Figure 1).

(4) Install valve in wing and torque 50 to 70 inch-pounds (5.65 to 7.91 N·m).

(5) Service aircraft with fuel IAW Chapter 12.

(6) Check for leaks.

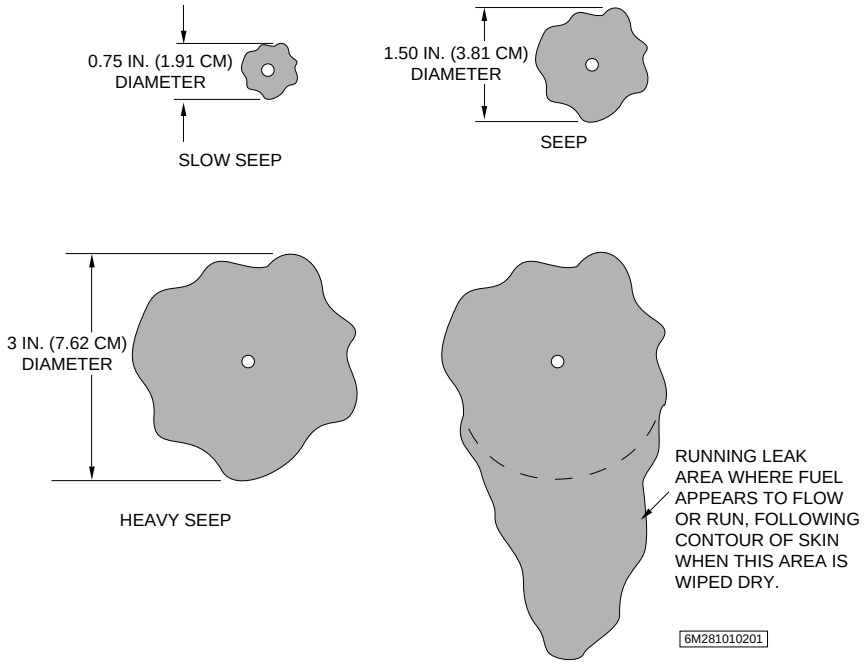
2. Inspection/Check – Fuel Tank

NOTE: Jet fuel is colorless; therefore, leak detection is difficult, especially for minor leaks. Procedures for locating fuel leaks must be followed closely. Always locate the true source of all leaks before beginning repairs.

A. Fuel Leak Evaluation (Figure 201)

(1) To determine leak rate, carefully wipe clean all wetted areas, and inspect leak source for extent of rewetting after thirty (30) minutes. Refer to Figure 201 for definitions of fuel leak classifications.

NOTE: Type I (red) or type II (yellow) fuel dye conforming to Military Specification MIL-D-81298D may be added to the fuel system for the purpose of indicating and readily determining location of fuel leaks, particularly in wet wing area. Add a maximum of 1.6 ounces of concentrated dye per every 100 gallons of fuel. Follow manufacturer's recommended procedures.



NOTE: Surface irregularities can localize fuel and result in dripping even though leak is a seep or heavy seep. When in doubt, classify leak to next higher leak category for repair determination.

SLOW SEEP - Fuel does not wet an area exceeding 3/4 of an inch in diameter.

SEEP - Fuel does not wet an area over 1 1/2 inches in diameter.

HEAVY SEEP - Fuel immediately reappears after being wiped away, spreads on surrounding surface, but does not drip or run from surface.

RUNNING LEAK - Fuel immediately reappears after being wiped away, spreads on surrounding surface, and drips or runs.

Fuel Leak Classification
Figure 201

(2) Determine if leak is in an area that may allow fuel to enter and accumulate in the following critical areas:

(a) Engine Bay.

(b) Tail Pipe Housing.

(c) Wing leading edge between nacelle and fuselage.

(d) Flap well.

(3) Refer to Table 201 to determine inspection and repair status based on leak location and leak rate.

(4) Carefully clean all wet areas after each inspection and repair so that future evaluation of leakage can be accurately determined.

FUEL TANK LEAK LOCATION	SLOW SEEP	SEEP	HEAVY SEEP	RUNNING LEAK
Lower Wing Surface	1	1	2	3
Wheel Well	1	2	3	4
Upper wing surface	1	2	3	4
Wing Center Section (area inboard of LH / RH W.S. 27.00)	1	2	3	4
Critical Areas- Paragraph **2.**, B., (2)	4	4	4	4

1. No repair action required. Frequent inspections must be made to ensure that leak does not increase in intensity.
2. No immediate repair required. Frequent inspections must be made to ensure that leak does not increase in intensity. Repair must be made within next 100 hours. See SA227 Service Note 051.
3. Repair Required. Aircraft may only be flown without passengers to base or to repair facility for repair.
4. Immediate repair required.

Fuel Leak Evaluation and Leak Categories Table 201

C. Locating External Point of Leak

NOTE: Loose rivets, screws, or bolts are the source of fuel leaks. However, internal leak source may be some distance away from the external leak, channelled between attaching members. Locate external leak point first.

(1) Most often a leak is detected by running or dripping fuel at the center section of the wing (dry area), this being the lowest point. From this starting point, determine which wing has the leak.

(2) For complete integrity check top off tanks with recommended fuel, otherwise fill tanks to normal levels used for flight operations.

(3) Dry wet area thoroughly with lint free cloth. Blow out all seams and corners with compressed air. Area must be completely dry.

NOTE: Use compressed air with 30 psi (2.11 kg/cm^2) maximum pressure and keep air hose nozzle at least 0.50 inch (1.27 cm) away from sealant or structure to prevent damage to sealant.

CAUTION: RED TALCUM POWDER IS FOR EXTERNAL USE ONLY AND SHOULD NOT BE MIXED WITH FUEL AS A DYE.

(4) Dust area with talcum powder (Semco P/N 419 or equivalent) immediately after drying. The first trace of fuel will cause the talcum powder to turn blood red.

(5) Continue dusting, working toward external point of fuel leak.

(6) When true external point of exit is determined, remove fuel from all tanks.

D. Locating Internal Point of Leak

NOTE: If leak evaluation indicates a repair is necessary, find the internal leak point. Repair to an external leak point is only temporary at best.

(1) Remove access panel nearest to external leak.

CAUTION: DO NOT USE DROPLIGHT INSIDE FUEL TANK. IF AN APPROVED TANK LIGHT IS NOT AVAILABLE, USE FLASHLIGHT.

(2) Dry area thoroughly with lint free cloth.

(3) Clean tank thoroughly with methyl propyl ketone (MPK) or equivalent to remove fuel.

(4) Inspect for cracks, distortion, corrosion of tank structure, and for loose fasteners.

NOTE: Visible defects in sealant or structure are not necessarily the true source of a leak. Continue visual inspection until entire suspected area has been carefully inspected. By blowing the external point of the leak, you may see trapped fuel blow backwards which will locate the leak.

(5) Prepare Turco soap solution in container. Slightly watered dishwashing liquid is acceptable.

NOTE: Plug fuel passages to prevent soap solution from running down into adjacent cavities.

CAUTION: KEEP SOAP SOLUTION FROM FUEL SENSORS. IF SOAP CONTAMINATES SENSORS, THEY WILL HAVE TO BE FLUSHED.

(6) Apply soap solution with pure bristle brush to suspected leak area inside fuel tank.

WARNING: WEAR PROPER EYE AND EAR PROTECTION WHEN USING HIGH PRESSURE AIR.

(7) Apply air pressure at external leak point (see Figure 203). Air supply should be capable of delivering 90 to 100 psi (6.33 to 7.03 kg/cm^2).

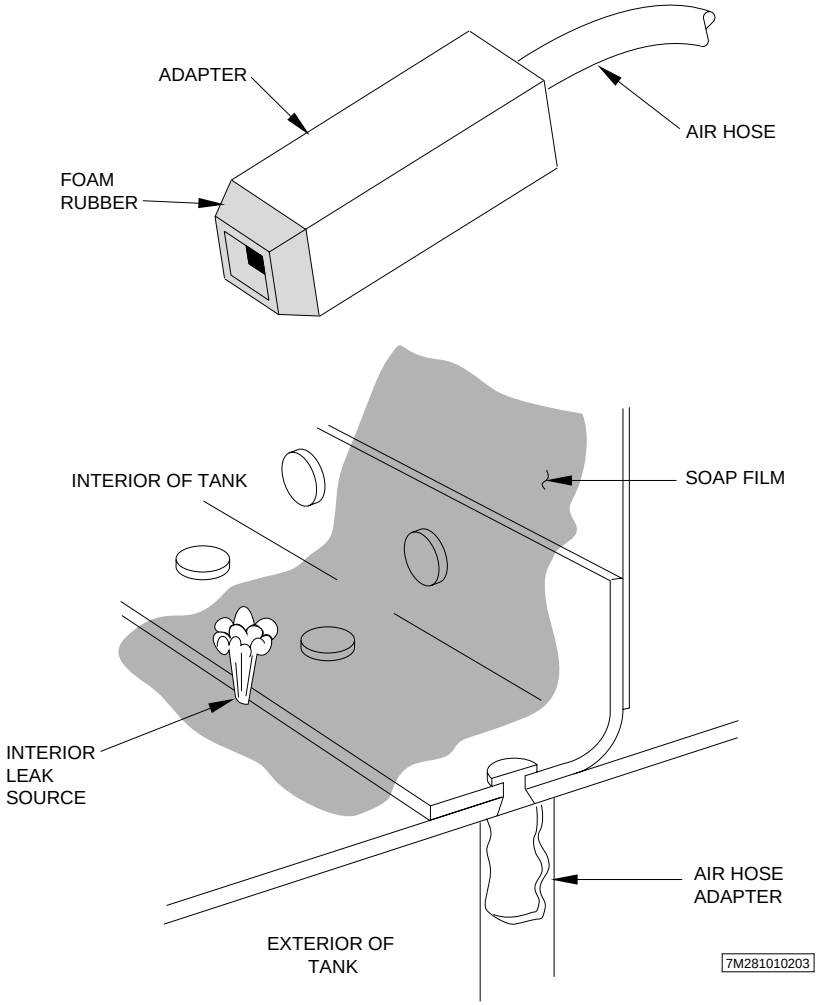
NOTE: Slowly open air valve to regulate air pressure. Too much air pressure may blow soap away not allowing detectable bubbles to form at internal leak area.

(8) When bubbles are observed, mark spot inside of tank.

(9) Check suspected leak spot(s) by applying soap solution on exterior area and applying air pressure from interior.

(10) If no leak is detected in first cavity, proceed to next cavity, either inboard or outboard, and repeat process.

(11) When leak has been identified, mop up soap solution from tank with damp cloth and clean thoroughly with MPK or equivalent solution.



Leak Detection – Fuel Tank
Figure 202

3. Approved Repairs – Fuel Tanks

A. Glossary of Terms

(1) **CLASS A SEALANT** – A pre-coat sealant, suitable for brush application.

(2) **CLASS B SEALANT** – Sealing compound suitable for application by extrusion-gun and spatula.

(3) **CLASS C SEALANT** – A brush or roller applied sealing compound suitable for faying surface sealing.

(4) **FAIRING TOOL** – A tool suitable to smooth and work air bubbles out of sealant.

(5) **FAYING SEAL** – A seal effected by the sandwiching of sealant between mating surfaces. Sealant is applied to one or both surfaces prior to assembly. Surfaces are then placed together and fasteners installed. Sealant should squeeze out from all edges of surfaces being joined.

(6) **SHANK SEAL** – Application of sealant to the hole and/or entire shank and underhead area of a fastener by brush or immersion. Sealant should be applied so entire shank of fastener is completely coated, and sealant will extrude from complete periphery of each end of fastener when fastener is installed.

(7) **INJECTION SEAL** – Application of sealant by the use of pressure (injection-gun) to force sealant into voids or cavities until extruded sealant emerges from opposite side and/or all other openings.

(8) **FILLET SEAL** – Application of sealant after completion of assembly, so that a bead of sealant contacts, covers, and fills all seams or joints.

(9) **BRUSH COAT SEAL** – Brush on application of sealant to provide an overcoat of sealant and to ensure wetting into recesses and/or threaded fasteners.

(10) **POTTING (ENCAPSULATION)** – Sealant applied to electrical wires and connectors for moistureproofing and protection. Normally applied by pressure (injection-gun) to ensure all voids are filled.

(11) **APPLICATION LIFE (WORK-LIFE)** – The period of time that the mixed sealant compound remains at a consistency suitable for application.

(12) **TACK-FREE TIME** – That period of time in the curing of sealant or top coat at which the material does not stick to a dry knuckle pressed lightly against the sealant and withdrawn.

B. Cleaning

NOTE: Sealants will not adhere to a contaminated surface.

(1) Remove old sealant in area of leak source.

(2) Vacuum, if necessary, to remove all dirt, chips, or other contaminants.

CAUTION: USE OF SOLVENT SHOULD BE IN SUCH A MANNER THAT THE SOLVENT SOURCE SHOULD NOT BE CONTAMINATED DURING CLEANING OPERATION. SUCH ITEMS AS PLUNGER-TYPE SAFETY CANS OR POLYETHYLENE SQUEEZE BOTTLES SHOULD BE USED. DO NOT DIP CLOTHS INTO OPEN BUCKETS OR CONTAINERS.

(3) Thoroughly clean the area to be resealed with methyl propyl ketone (MPK) or equivalent.

NOTE: Extreme care is needed to clean inaccessible corners and gaps. A small paint brush may be used for slots and recesses, and for all areas hard to reach. Pipe cleaners or cloth wrapped around wooden or phenolic tools may be used.

(4) Always clean an area wider than the width of the sealant to be applied.

NOTE: Cleaning solvent should never be poured or sprayed on the structure. The solvent may run back between layers of structure, then creep back out again after cleaning operations are completed, bringing some contamination to surfaces where sealant will be applied.

(5) Wipe dry with lint-free cloth prior to solvent evaporation.

(6) Repeat steps **3.**, B., (1) through (5) until there is no discoloration on a clean dry cloth.

(7) After cleaning, any handling should be with white gloves.

C. Preparation of Sealant

(1) Mix sealant thoroughly IAW the vendor's instructions, so the mixture is homogeneous (no streaks).

(2) The suffix of the sealant part number indicates the work life in hours, of the sealant. (Example: in the part number PR-1422B-1/2, the "-1/2" indicates the sealant has a work life of one half (1/2) hour under ideal temperature conditions.

(3) The storage life is printed on each container and should be strictly adhered to. When storage life expires the remaining portion must be discarded.

D. Application of Sealant

CAUTION: BE CAREFUL DURING SEALING OPERATIONS TO KEEP FLAPPER VALVES FREE OF SEALANT. A SMALL QUANTITY OF SEALANT WILL CAUSE IMPROPER SEATING OF THE VALVE.

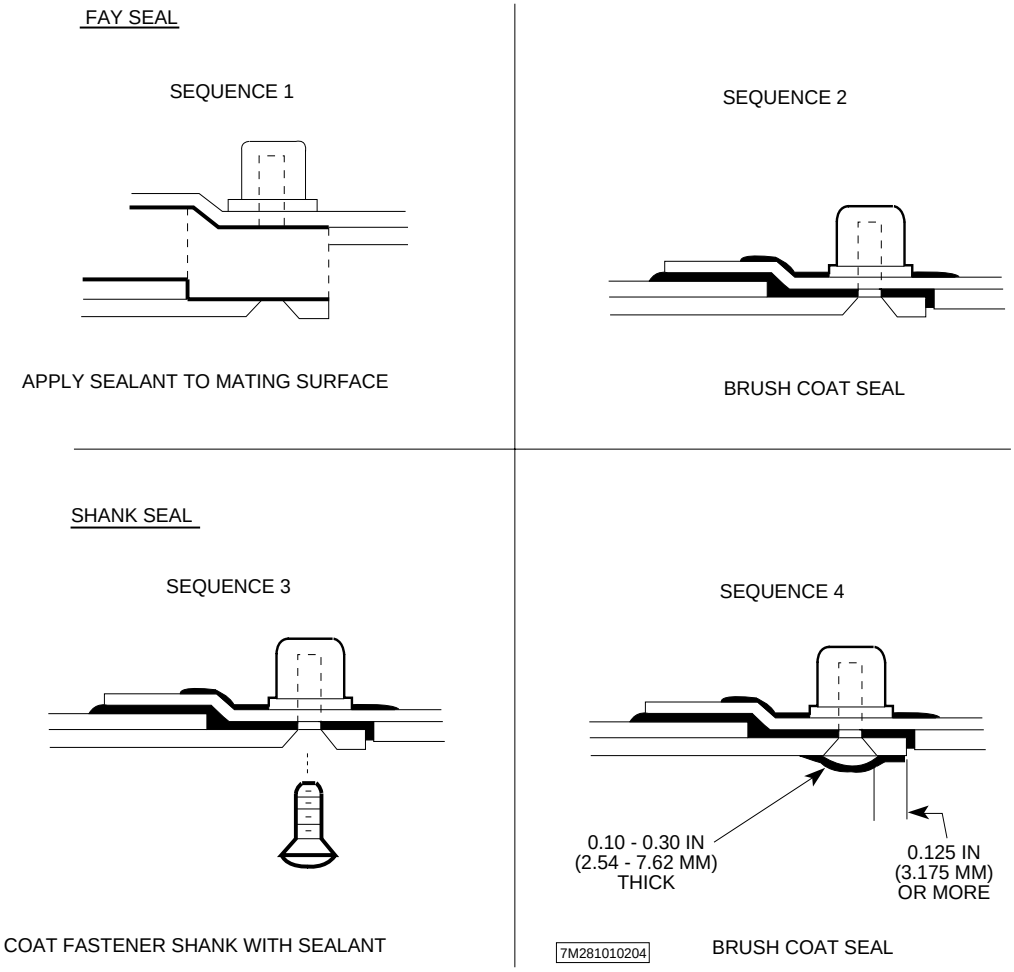
(1) Apply sealant only to areas that have previously been sealed. Repairs should be accomplished using sealant meeting the same specifications (MIL-S-8802) as the original sealant and should be applied by the same method and manner as the original application.

NOTE: PR-1826 may be used as an alternate sealer. PR-1826 must be used IAW manufacturers instructions and with manufacturers primer or adhesion promoter.

E. Faying Surface Seals (see Figure 203)

(1) Clean area to be sealed (see Cleaning, this chapter).

(2) Apply Class B and/or Class C sealant in a uniform bead along mating surface.



Fay Surface Seal and Shank Seal
Figure 203

(3) Spread sealant evenly with spatula until approximately 0.0312 inch thick. Apply enough sealant so that the space between assembled faying surfaces is completely filled with sealant and a small excess is extruded continually along the joint.

NOTE: Faying surface sealant does not have to cure prior to applying fillet seals. It is permissible to apply fillets over the small amount of faying surface sealant extruded from the faying surface.

F. Shank Sealing (see Figure 203)

NOTE: All fasteners must be wet-installed (Shank Seal) and brush coat sealed.

(1) Clean area to be sealed (see Cleaning, this chapter).

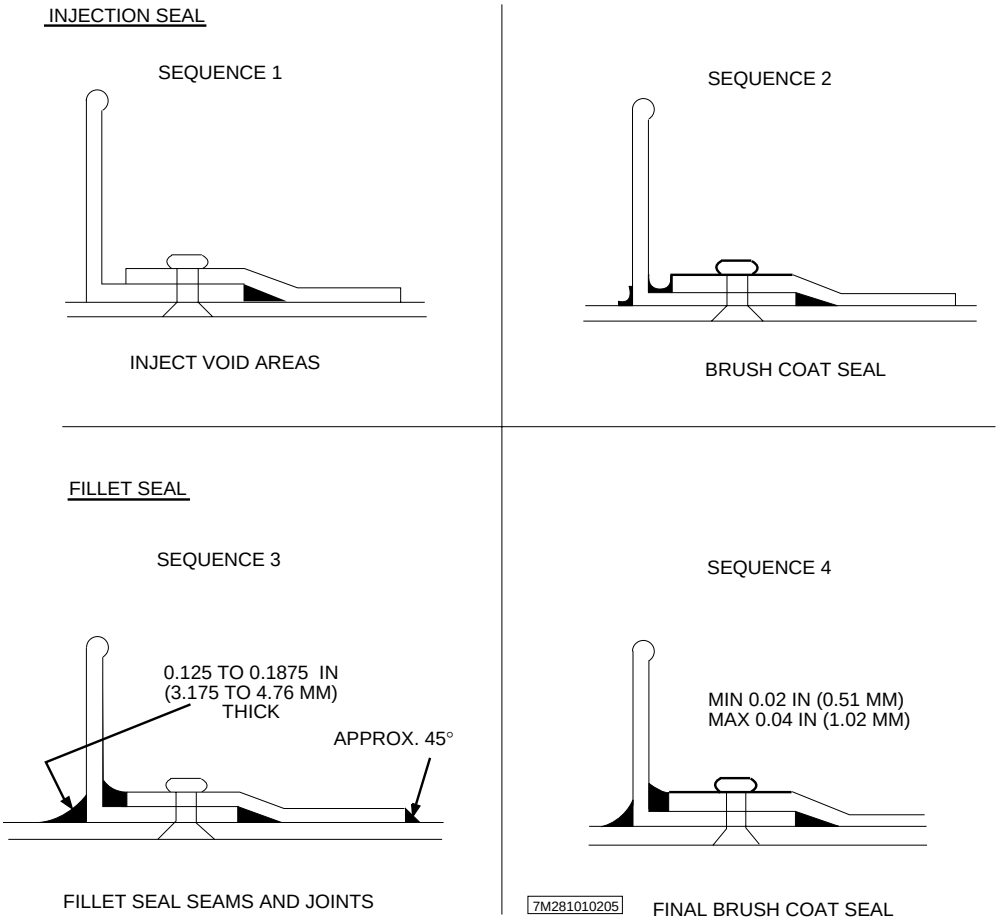
(2) Apply sealant to shank and underhead area of fastener. Sealant may be applied to structure. Apply with a brush, stroking in two (2) directions to apply a coating five (5) to fifteen (15) mils thick.

(3) Eliminate any voids or bubbles.

G. Injection Seals (see Figure 204)

(1) Sealant should be injected through voids and channels until sealant comes out opposite end.

(2) Inject enough sealant to force out all air.



Injection Seal and Fillet Seal
Figure 204

H. Fillet Seal Repairs (see Figure 204)

(1) Repair of damaged or faulty sealant should be accomplished as follows:

(a) Remove all faulty sealant.

(b) Scarf sealant to produce a smooth continuous scarfed face. The sealant may be either completely removed or notched cleanly in affected area. Cutting tools should only be made from non-metallic materials softer than aluminum.

(c) Clean area to be sealed (see Cleaning, this chapter).

(d) Apply new sealant. Slightly overlap the fresh material onto existing fillet.

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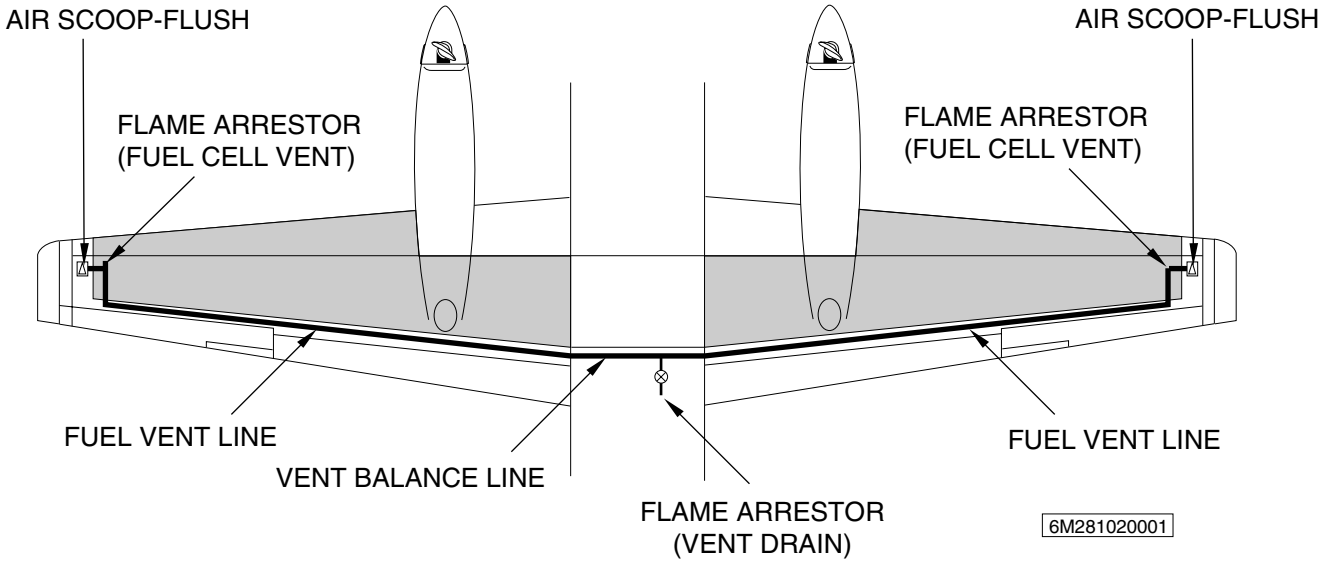
FUEL VENT – DESCRIPTION AND OPERATION

1. General

A. A flush mounted air scoop vent located outboard at the bottom tip of each wing, W.S. 250.00, vents air into the fuel tanks. A vent balance line is routed along the entire length of the trailing edge of the wing outside the fuel tank. The line provides cross venting between fuel tanks.

B. A vent system drain (flame arrester) fitting, located at the center wing section, provides a means of draining condensation (see Figure 1).



Fuel Vent
Figure 1

FUEL JET TRANSFER PUMP – DESCRIPTION AND OPERATION

1. General

A. Fuel jet transfer pumps reduce unusable fuel quantity to approximately two (2) gallons (7.57 liters). The system consists of:

(1) Two collector tanks.

(2) Two jet transfer pumps.

(3) A hopper tank.

B. The two collector tanks and the hopper tank are the most inboard section of the fuel tank.

C. The jet transfer pumps are located in the dry center section immediately adjacent to the fuel tank.

D. A float switch, installed in the hopper tank, is actuated by the level of fuel in the tank.

E. Flapper valves in the hopper tank allow fuel flow into the tank but prevent reverse flow out of the tank.

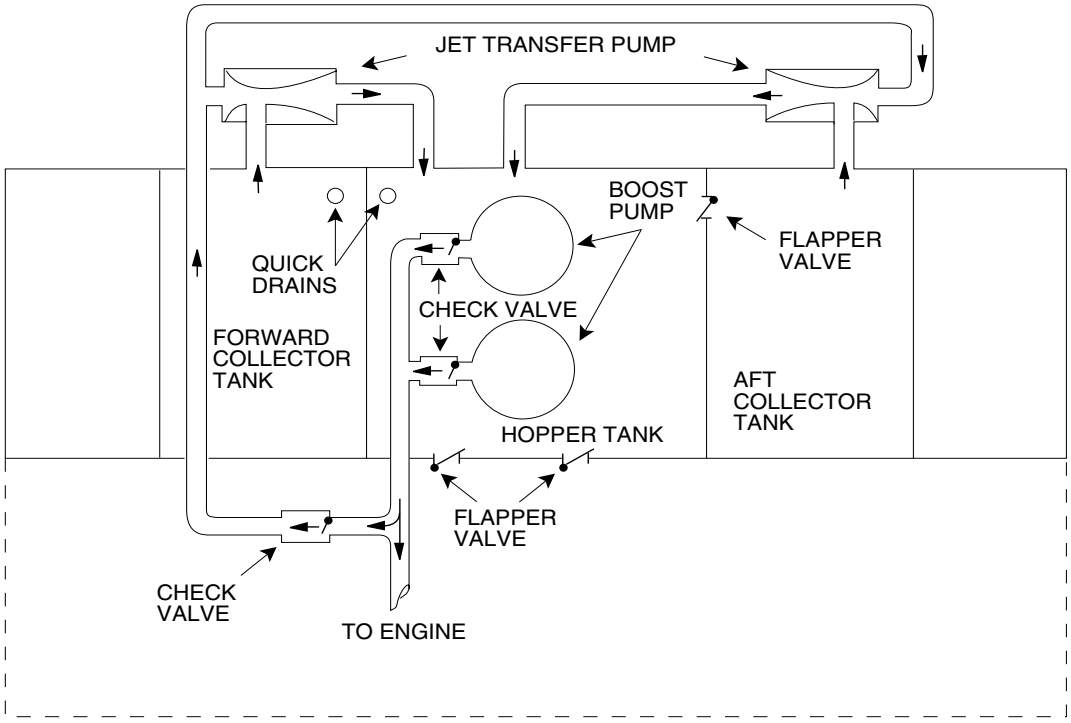
2. Operation

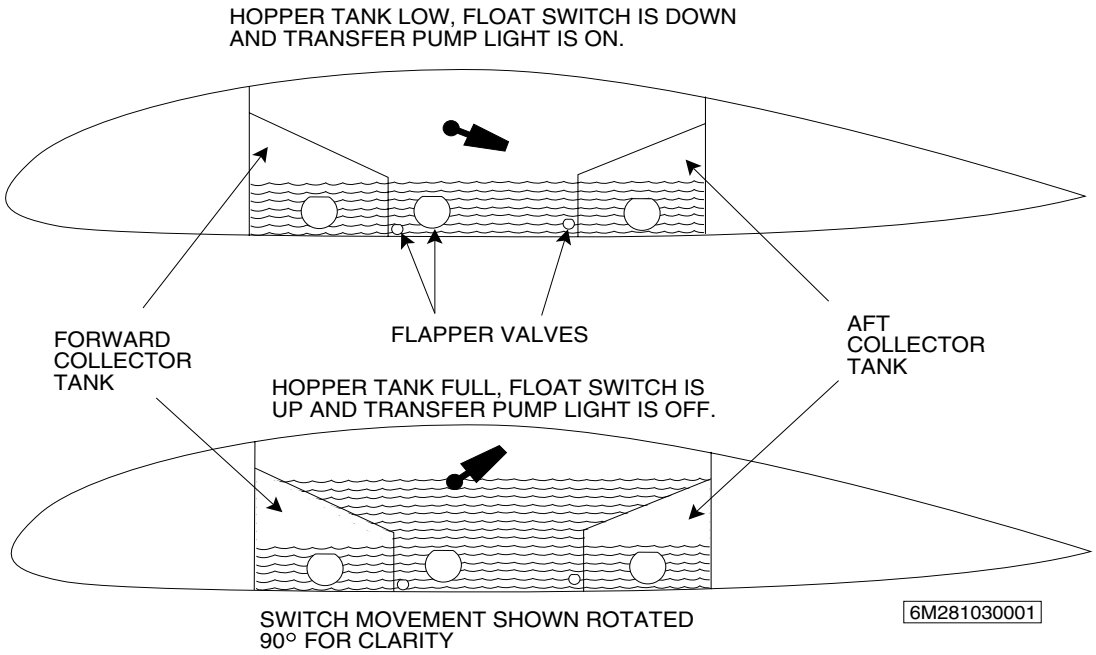
A. Pressurized fuel is supplied by the operating boost pump and delivered to the inlet side of each jet transfer pump (see Figure 1). The suction side of each jet transfer pump is connected to one of the collector tanks. The jet transfer pump outlets empty into the hopper tank.

B. Fuel flow through the jet transfer pump causes fuel to be drawn from the collector tanks and supplied to the hopper tank. The jet pump system has an efficiency factor of 1.5 or better (the jet pumps are capable of filling the hopper tank 1.5 times as fast as the engine driven pumps can drain the tank).

C. A float switch in the hopper tank controls a XFR PUMP light in the annunciator panel. When the tank is full, the switch float opens the switch and the light extinguish.

FUEL JET TRANSFER PUMP – DESCRIPTION AND OPERATION





Fuel Jet Transfer Pump
Figure 1

FUEL JET TRANSFER PUMP – FAULT ISOLATION

1. Troubleshooting – Fuel Jet Transfer Pump

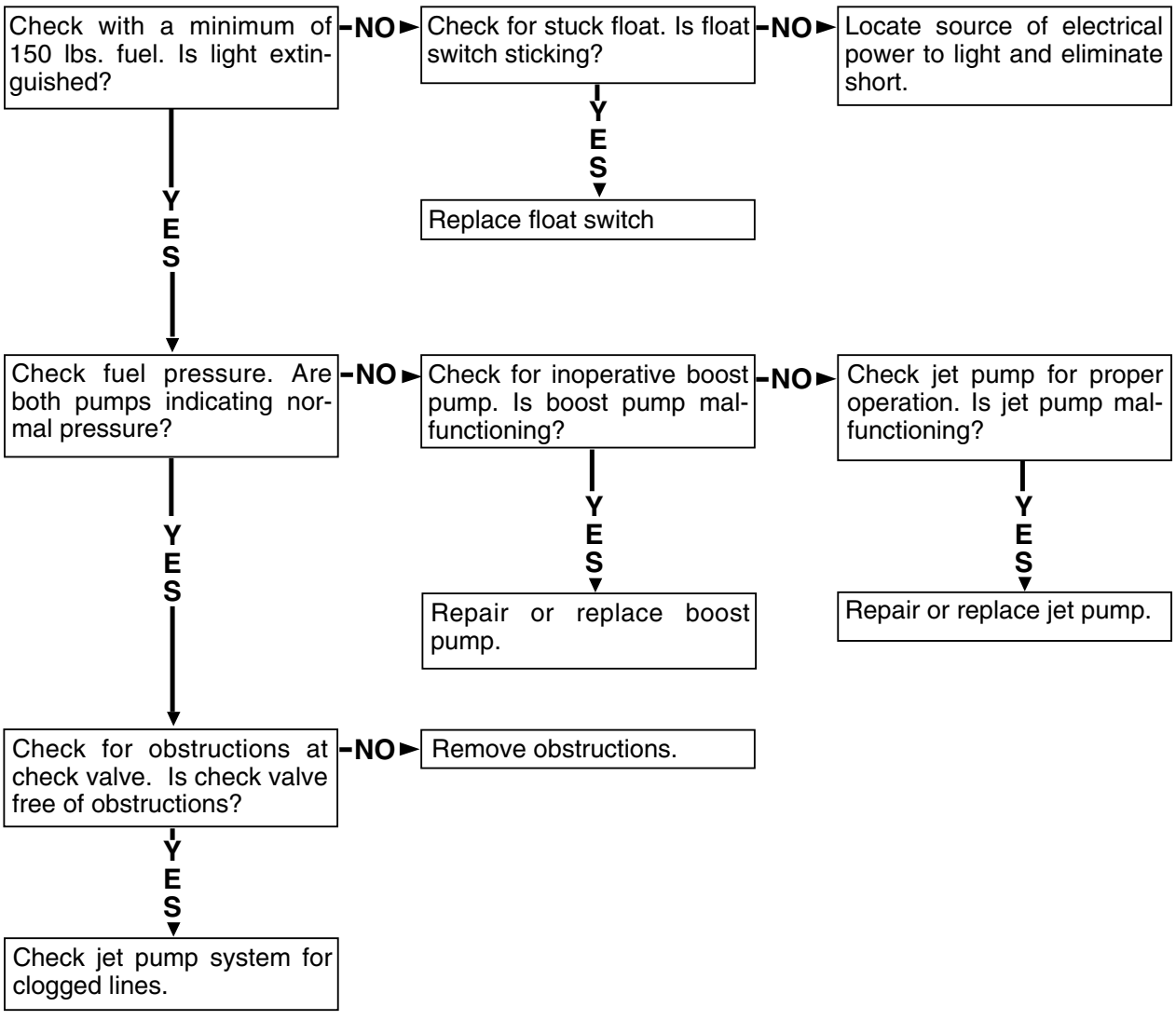
PROBLEM:

TRANSFER PUMP LIGHT ILLUMINATED AT TIMES.

PREREQUISITES:

Ensure there is more than 65 lbs. of fuel in tank. If tank contains less than 65 lbs. the XFR light will remain illuminated acting as a Low Fuel Warning.

START:



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END

FUEL JET TRANSFER PUMP – FAULT ISOLATION

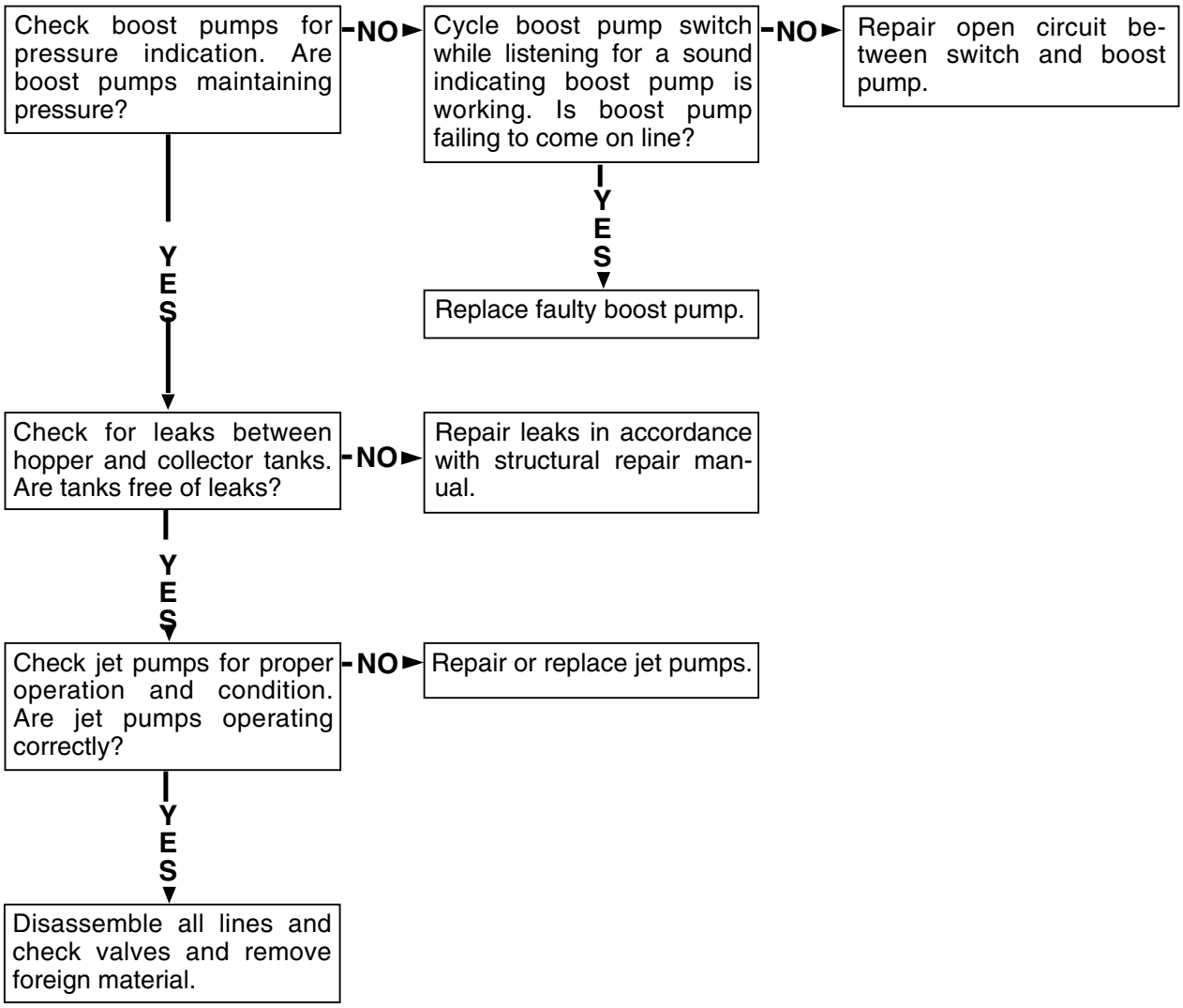
PROBLEM:

JET PUMP SYSTEM WILL NOT PASS FUNCTIONAL TEST.

PREREQUISITES:

With batteries installed ensure battery switches and boost pump switches are ON and circuit breakers in place.

START:



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END

FUEL JET TRANSFER PUMP – MAINTENANCE PRACTICES

1. Adjustment/Test – Fuel Jet Transfer Pump

A. Functional Test – Boost Pump (DC Pump)

(1) Test only the boost pump with lowest indicated fuel pressure. To expedite and improve accuracy when measuring elapsed time, it is recommended two (2) people take the data, one (1) with a stop watch the other operating the boost pump switch and recording data (see Table 201).

(a) Fill each fuel tank with 30, ± 2 gallons (113.55, ± 7.57 liters) of fuel.

(b) Start engine, establish ground idle, allow hopper tank to fill to float level (XFR PUMP light extinguished).

(c) Measure elapsed time between light extinguished and light illuminated by switching boost pump OFF immediately when light is extinguished and then timing until light is again illuminated. This is the time to drain and is referred to as T1.

(d) Measure elapsed time between light illuminated and light extinguished by switching boost pump ON immediately when light illuminates and then timing until light is again extinguished. This is the time to fill and is referred to as T2.

(e) Repeat **1.**, A., (1), (c) and (d) three (3) additional times to obtain a total of four (4) sets of data.

NOTE: The sequence of obtaining data makes no difference. All T1 data may be obtained first and then all T2 data, whichever is most convenient.

(f) Determine each T1 and T2 average values from the data.

(g) Calculate and record the ratio of T1 average to T2 average (T1/T2).

NOTE: If T2/T1 is less than 1.50, system is unacceptable.

B. Functional Test – Boost Pump (AC Pump)

(1) Aircraft equipped with one AC boost pump and one DC boost pump per hopper tank (T201-248 and AT002) will use the same test procedure as given for DC boost pumps with the following exception:

(2) Since the AC boost pump cannot be operated at ground idle, the test will be made using the DC pump. After obtaining the T1/T2 ratio for the test, a comparison of boost pump pressure will be made, and the ratio will be adjusted accordingly (see Table 201).

(3) AC and DC boost pump pressures must be recorded at the same engine RPM and fuel flow for a valid comparison. After recording the pressures on Table 201, use the following formulas for adjusting the ratio:

(a) DC Pump Pressure (P1) - AC Pump Pressure (P2) = P

(b) T1/T2 (obtained from test) - 0.07 (P) = Adjusted T1/T2.

1. RECORD BOOST PUMP PRESSURE AT GROUND IDLE

	LEFT SYSTEM	RIGHT SYSTEM
AUXILARY PUMP PRESSURE		
MAIN PUMP PRESSURE		

NOTE: USE PUMP WITH LOWEST PRESSURE TO RECORD T1 AND T2 VALUES BELOW

2. RECORD T1 AND T2 VALUES

	T1	T2	T1	T2
1				
2				
3				
4				
T0TALS				
AVERAGE (TOTAL/4)				
RATIO (T1/T2)				

NOTE: MINIMUM ACCEPTABLE IS 1.50

3. ADJUSTMENT FOR AC PUMPS (T201-248, AT001-002)

	LEFT SYSTEM	RIGHT SYSTEM
DC PUMP PRESSURE (P1)		
AC PUMP PRESSURE (P2)		
P1 – P2 = P		
ADJUSTED RATIO: T1/T2 – .07 (P)		

NOTE: MINIMUM ACCEPTABLE ADJUSTED RATIO IS 1.50

- AIRCRAFT WITH DC BOOST PUMPS USE STEPS 1 AND 2.
- AIRCRAFT WITH AC AND DC PUMPS USE STEPS 2 AND 3.

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Fuel Jet Transfer Pump
Functional Test
Table 201

2. Inspection/Check – Fuel Jet Transfer Pump

A. Inspection – Fuel Jet Transfer Pump

(1) Gain access to jet transfer pump by removing lower center section access panels in front of landing lights.

(2) Inspect lines for chafing or rubbing.

(3) Inspect lines for kinks, cracks, or other obvious damage.

(4) Inspect all connection points for evidence of fuel leakage.

(5) Inspect threaded parts for damage.

(6) Check clearance between landing light and fuel lines. Note any line that is closer than one-half inch to the light housing, wiring, or any other part of the landing light assembly.

NOTE: The small jumper attached between the two terminals on the landing light may require bending or replacement to achieve the desired clearance.

(7) Install lower center section access panels.

FUEL CROSSFLOW – DESCRIPTION AND OPERATION

1. General

A. Fuel crossflow provides the aircraft with the capability of maintaining a fuel balance between integral wing tanks. The system consists of:

(1) A two inch line that interconnects the wing tanks.

(2) A crossflow valve for regulating fuel flow from each tank.

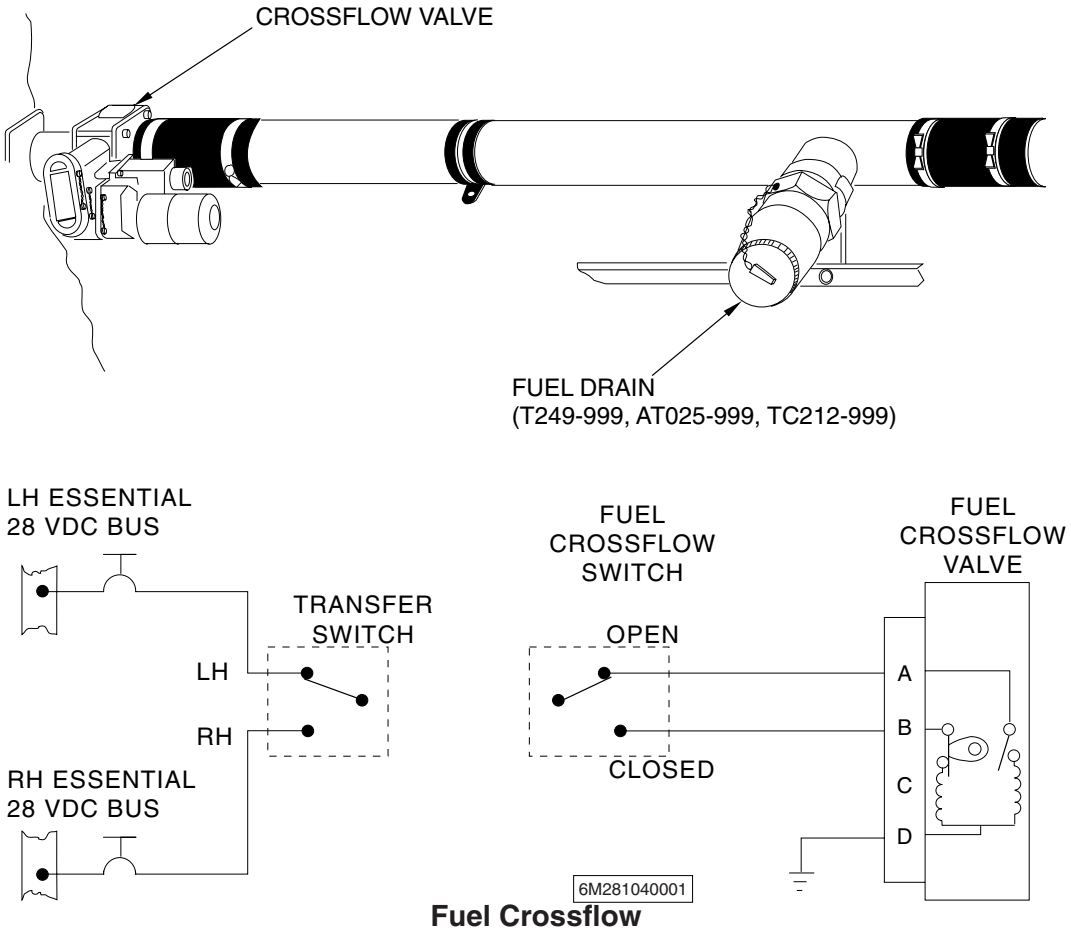
(3) On later series aircraft, fuel drain for single point defueling of the aircraft.

B. The fuel crossflow line is located in the center section aft of the main spar and is accessible through panels in the wing center section.

2. Fuel Crossflow Valve

A. A fuel crossflow valve is installed in the crossflow line connecting the wing tanks. It is located inboard of left W.S. 27.00, aft of the main spar, and accessible through panels in the wing center section.

B. The gate type valve is motor operated and controlled by a switch on the pilots instrument panel. When the valve is open the center light of the three lights on the fluid annunciator panel illuminates. Limit switches are incorporated in the valve to de-energize the motor when the gate reaches full open or full closed position.



Fuel Crossflow
Figure 1

FUEL CROSSFLOW – FAULT ISOLATION

1. Troubleshooting – Fuel Crossflow

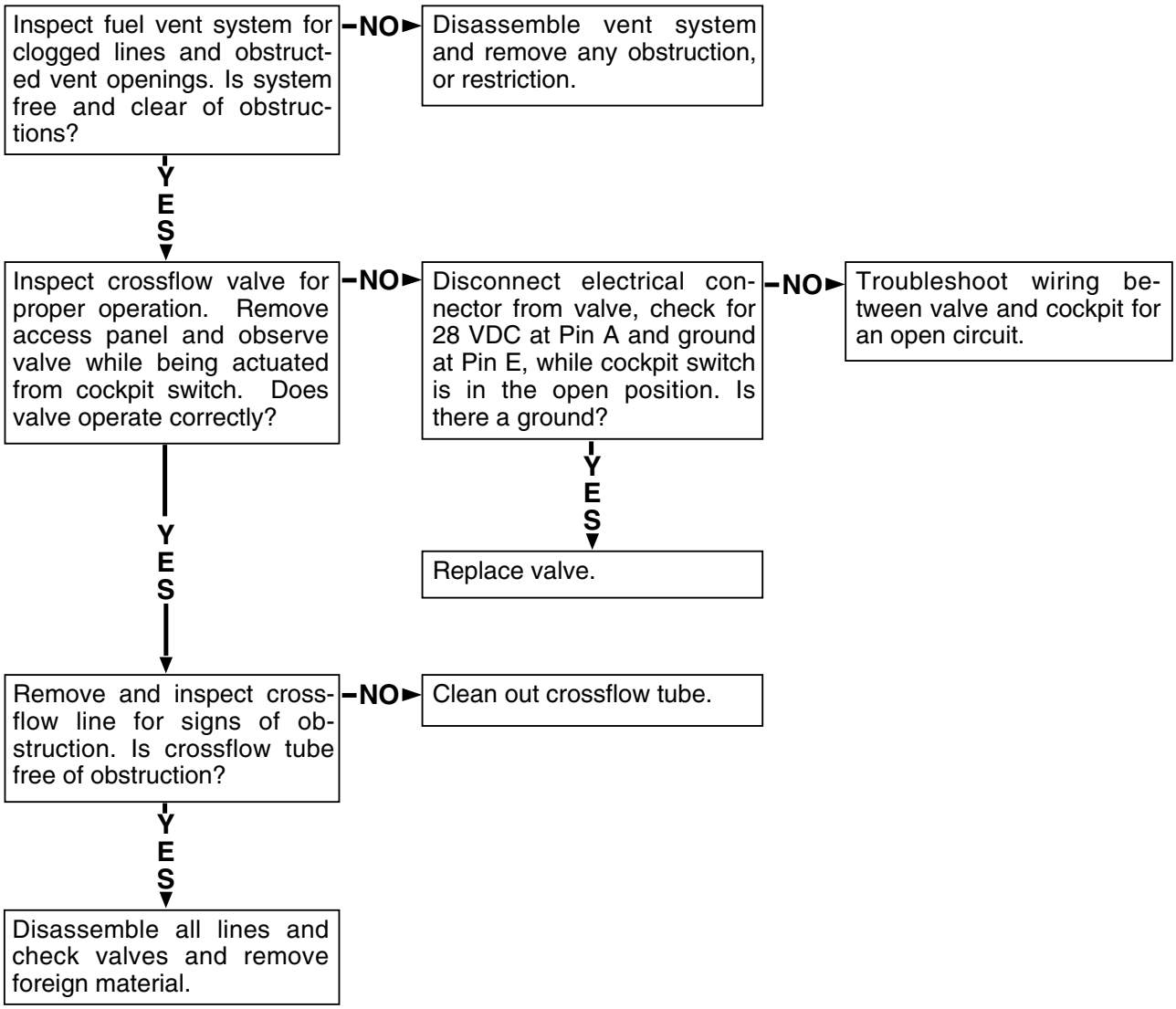
PROBLEM:

FUEL WILL NOT TRANSFER FROM SIDE TO SIDE.

PREREQUISITES:

Ensure batteries are installed and battery switches ON.

START:



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END

FUEL CROSSFLOW – MAINTENANCE PRACTICES

1. Removal/Installation – Fuel Crossflow

A. Removal – Fuel Crossflow Valve

(1) Defuel aircraft.

(2) Remove both access panels forward of the aft, center, wing spar.

(3) Remove safety wire from electrical connector.

(4) Disconnect electrical connector from valve.

(5) Loosen clamps securing rubber hose coupling and slide hose inboard.

(6) Cut safety wire and remove bolts securing valve.

(7) Push end plate inboard and remove valve.

NOTE: O-rings on either side of valve may fall out when removing valve. Ensure new O-rings are properly seated when replacing valve.

B. Installation – Fuel Crossflow Valve

(1) Position valve and end plate in place.

(2) Install and torque bolts 45 to 55 inch-pounds (5.08 to 6.21 N·m). Do not overtorque. Safety wire bolts to each other.

(3) Slide rubber hose outboard and secure with clamps.

(4) Connect and safety electrical connector.

(5) Apply electrical power to aircraft and actuate crossflow switch. Verify valve opens and closes.

(6) Refuel aircraft and check for leaks.

(7) Install access panels.

FUEL DISTRIBUTION – DESCRIPTION AND OPERATION

1. General

A. Fuel is supplied to the engines by an independent fuel system from each wing tank. A crossflow line interconnects the left and right wing tanks providing a means for supplying either engine with fuel from either tank.

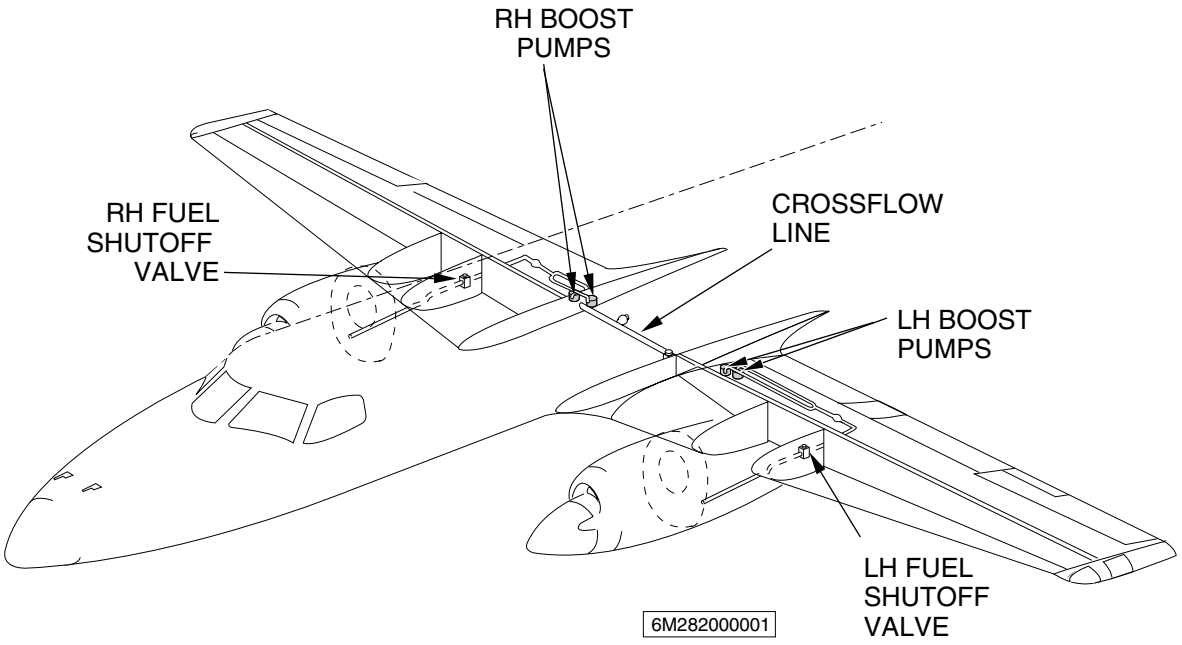
B. Two collector tanks located at the inboard end of each fuel tank remain full at all times by gravity flow inward. A hopper tank between the collector tanks is kept full at all times by a jet pump system.

2. Fuel Boost Pumps

A. Two fuel boost pumps (a main and an auxiliary pump) contained in the hopper tank, supply fuel to the engine. They are connected through check valves to a common supply line.

3. Fuel Shutoff Valve

B. A shutoff valve located in the common fuel supply line between the boost pumps and the engine permits the pilot to terminate fuel supply to the engine.



Fuel Distribution
Figure 1

FUEL BOOST PUMP – DESCRIPTION AND OPERATION

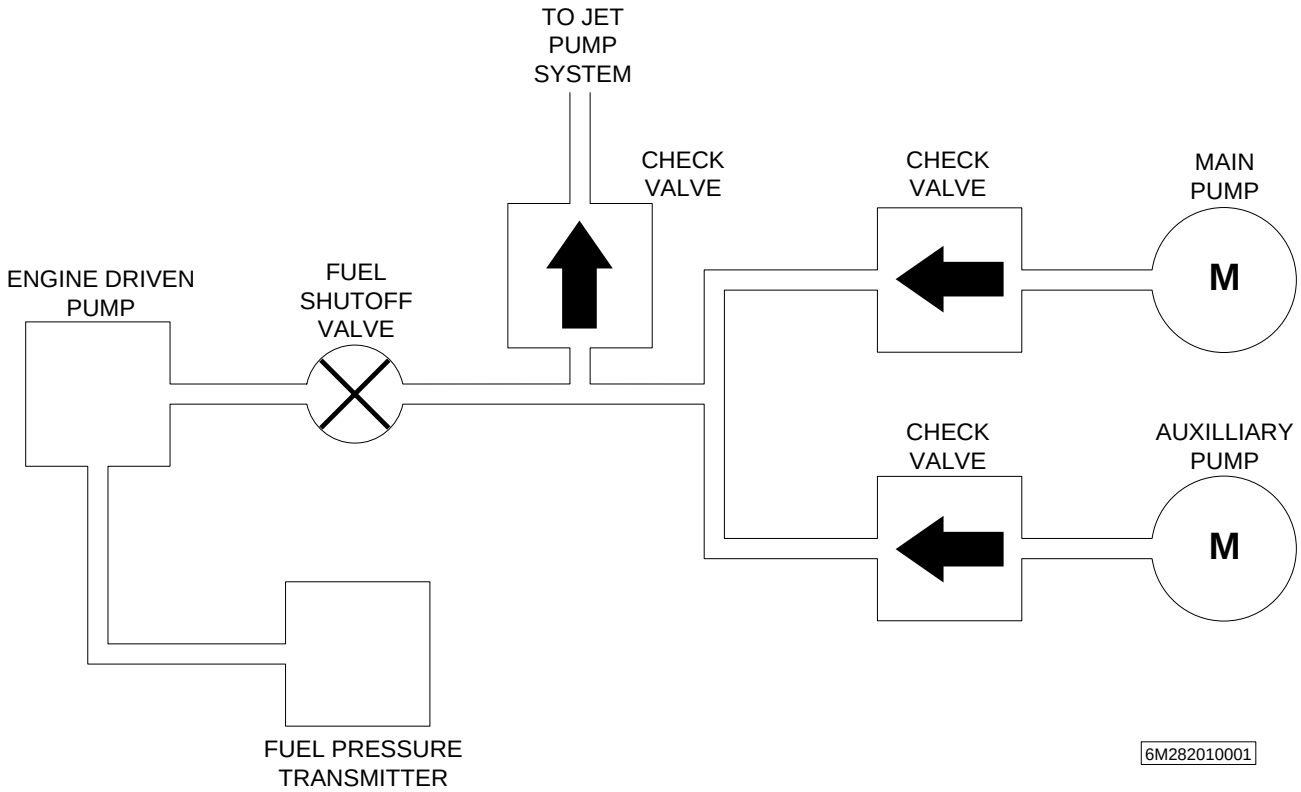
1. General

A. Two submerged boost pumps installed in the hopper tank of each wing are connected through check valves to a common supply line (see Figure 1). Boost pump pressure from the fuel supply line is used in the jet pump system. A fuel shutoff valve installed in each nacelle can be used to stop fuel flow to the engine.

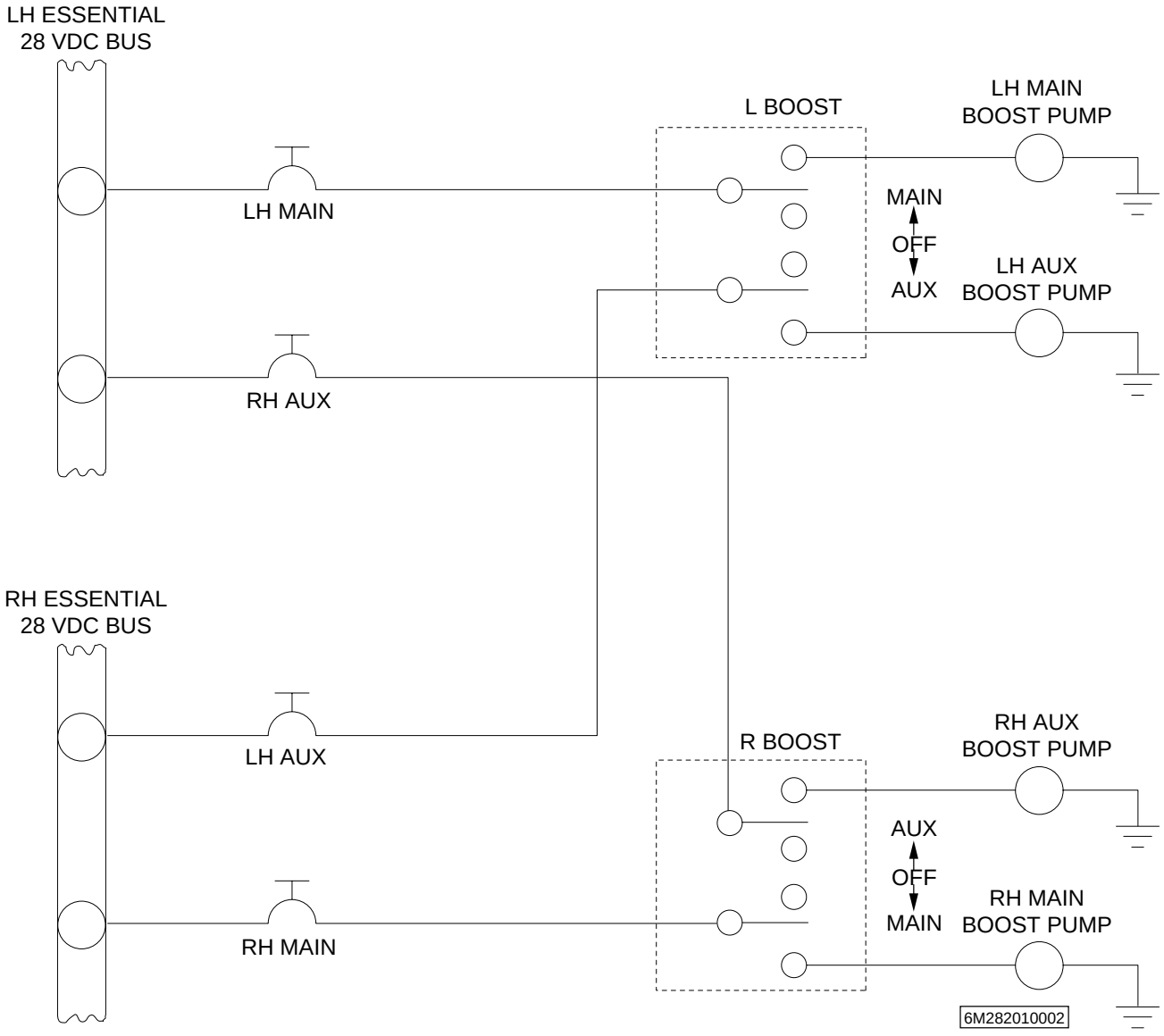
B. A fuel pressure transmitter is connected to the interstage section of the engine driven fuel pump. When the engine is not operating, the fuel pressure gauge indicates boost pump pressure. With the engine operating the fuel pressure displayed is a combination of engine driven pump pressure and boost pump pressure.

C. Early Merlin III and Merlin IV aircraft were equipped with one AC-powered and one DC-powered pump in each tank. Later Merlin III's and Merlin IV's and all Metros are equipped with two DC-powered pumps in each tank

D. Each essential bus provides power for two pumps, one in each wing (see Figure 2). This is a safety feature should either bus be inoperative. The two pumps within each wing are designated main and auxiliary. These designations are for purposes of identification only as the pumps are identical. A three position switch (MAIN-OFF-AUX) is installed in the center pedestal for each pair of pumps.



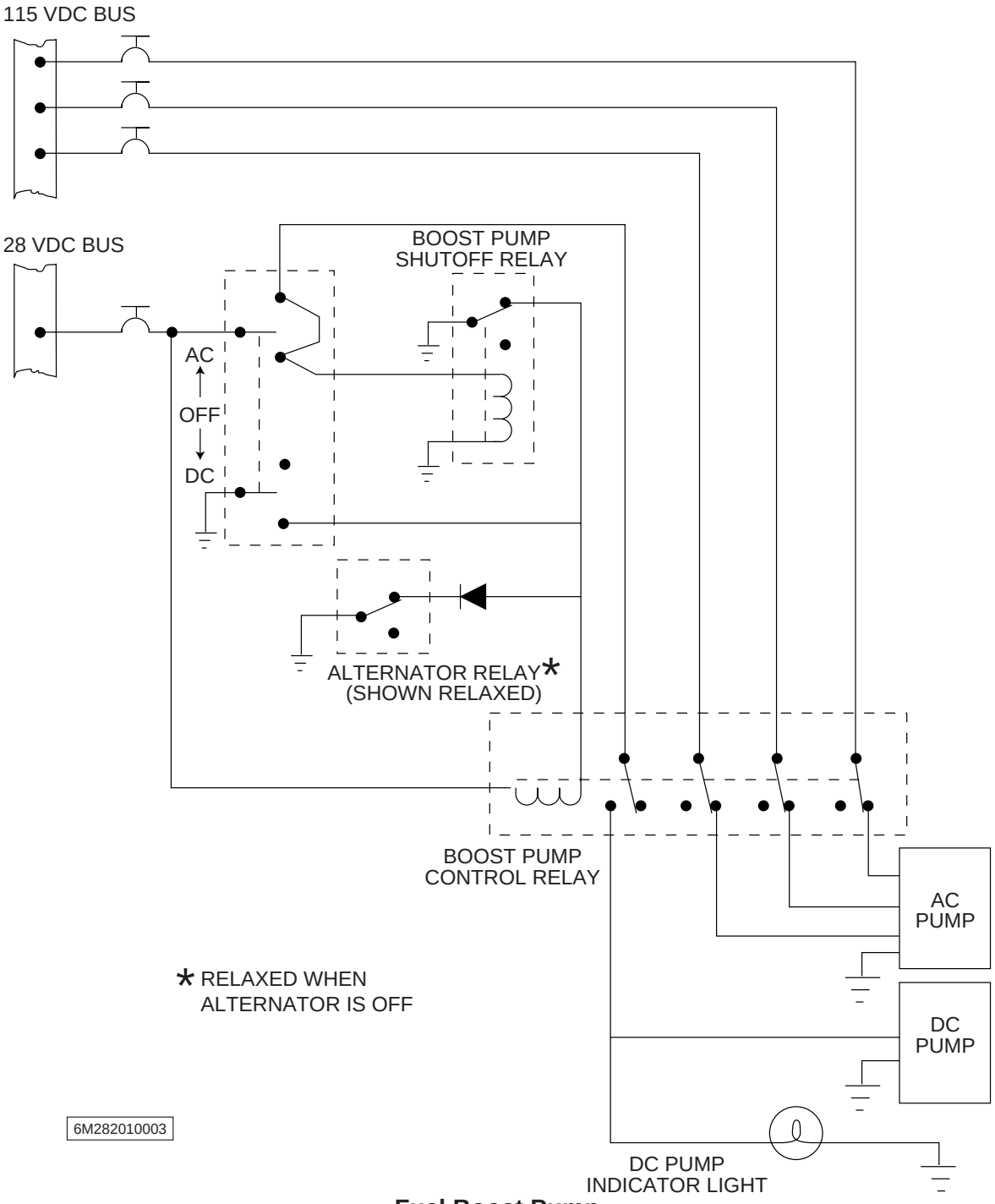
Fuel Boost Pump Schematic
Figure 1



Fuel Boost Pump
Control Circuit
Figure 2

FUEL BOOST PUMP – DESCRIPTION AND OPERATION

E. Each hopper tank contains an AC-powered pump and a DC-powered pump (see Figure 3). Control circuitry consists of a three position switch (AC-OFF-DC) in the center pedestal; a boost pump shutoff relay; a boost pump control relay; and one section of the alternator relay.



Fuel Boost Pump Control Circuit
Figure 3

(1) When the boost pump switch is in the OFF position, the boost pump shutoff relay is relaxed and provides a ground for the boost pump control relay. Actuation of the boost pump control relay breaks the connection between the AC pump to the AC bus. Moving the boost pump switch to the AC position actuates the shutoff relay and removes the ground for the control relay.

(2) An additional ground for the control relay is provided by the alternator relay. If the alternator relay is relaxed (alternator inoperative) the control relay remains actuated. The DC pump receives DC power and operates.

F. Operation of the DC pump is indicated by illumination of a yellow DC pump indicator light located near the boost pump switch.

G. When the alternator control circuit allows the alternator to power the AC bus (above 90% engine speed), the alternator relay closes and removes the ground to the control relay. The control relay relaxes and thereby supplies power to the AC pump while removing power to the DC pump. Should AC power be lost, the system automatically switches back to the DC pump.

H. When the boost pump switch is placed in the DC position, a ground to actuate the boost pump control relay is provided. In the DC position, the boost pump control relay remains actuated regardless of alternator output, and DC power is provided to the DC boost pump.

FUEL BOOST PUMP – MAINTENANCE PRACTICES

1. Removal/Installation – Fuel Boost Pump

A. Removal – Boost Pump

(1) Defuel wing tank containing boost pump to be removed.

NOTE: Defueling the wing tanks should be accomplished with crossflow valve closed to prevent fuel from crossing into opposite tank.

(2) Remove screws securing boost pump access panel.

(3) Remove boost pump access panel.

(4) Place a temporary index mark on boost pump structure.

(5) Cut safety wire and remove boost pump attaching bolts.

NOTE: On aircraft with combination AC and DC boost pumps (T201-248 and AT001-002), AC pump is located outboard and DC pump is located inboard.

(6) Lower boost pump and disconnect electrical connection.

NOTE: Do not allow boost pumps to hang from wiring or flex lines, secure with safety wire or equivalent.

(7) Disconnect fuel line.

(8) Remove pump and gasket.

(9) Remove and retain fittings installed in pump.

B. Installation – Boost Pump

(1) Install previously removed fittings in pump. Coat gasket with sealant (Tite Seal, part number T55 -16, or Hylomar Sealing Compound, part number SQ-32M, or equivalent) and install gasket on pump.

(2) Check boost pump wiring for chafing condition.

(3) Replace any boost pump wiring that is found to be chafing, cracking of insulation material, or any evidence of bare wire.

(4) Wrap wire harness with Heyco-Flex V, Slit Convoluted Tubing, P/N 1634. Secure at both ends with nylon tie strap.

CAUTION: WHEN REPLACING BOOST PUMP, INSTALL PUMP WITH FUEL LINE FACING INBOARD. IF REVERSED, FLAPPER VALVE ON OUTBOARD SIDE OF HOPPER TANK MAY HANG UP ON PUMP FUEL LINE CAUSING VALVE TO REMAIN OPEN AND FUEL TO FLOW OUT OF HOPPER TANK.

(5) Connect fuel line and electrical wiring to pump.

NOTE: Ensure correct electrical connector is connected to corresponding boost pump. Inboard pump is the main pump (electric wire labeled MAIN). Outboard pump is auxiliary pump (electric wire labeled AUX).

(6) Position pump in place.

CAUTION: BOLTS USED TO SECURE THE BOOST PUMP MUST NOT BE LONGER THAN 0.9062 IN. LONGER BOLTS WILL PUNCTURE THE SEALED DOME ON THE BOOST PUMP NUTPLATES WHICH MAY RESULT IN FUEL LEAKS.

(7) Temporarily install one boost pump with minimum of two bolts snugged down.

(8) Looking through opposite pump mounting hole check wiring for chafing.

(9) Reposition pump as required at ensure no chafing condition exists.

(10) Add a permanent index mark on boost pump structure using permanent ink.

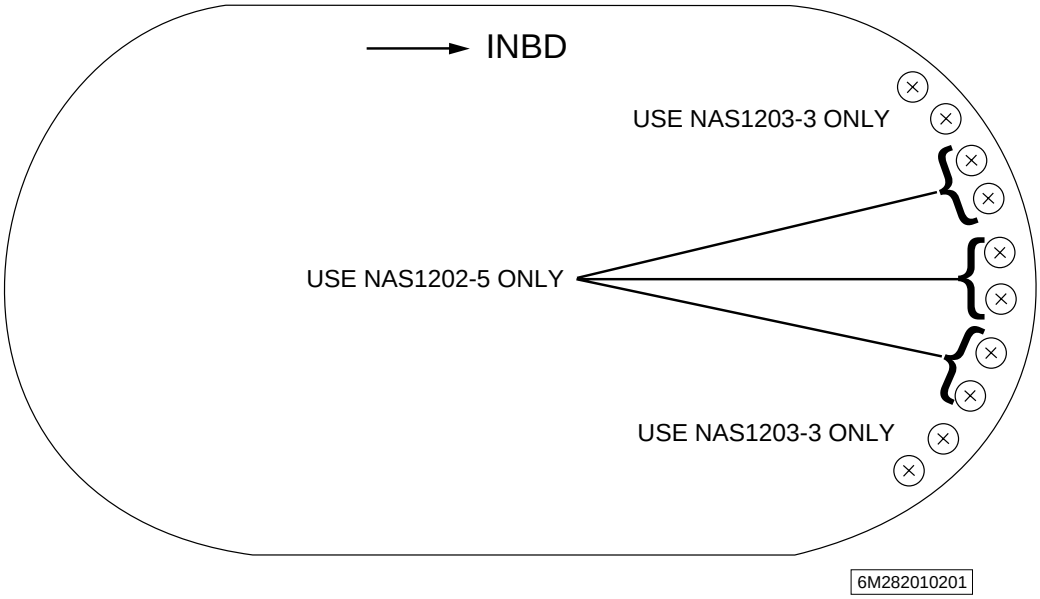
(11) Secure pump with bolts. Torque bolts 50 to 70 inch-pounds (5.65 to 7.91 N·m) and safety.

(12) Fuel aircraft and check boost pumps for leaks and proper operation.

(13) Install boost pump access panel with screws as shown in Figure 201.

CAUTION

FUEL BOOST PUMP ACCESS PANEL, 617 & 618
LOCATE PROPER LENGTH FASTENERS TO AVOID FUEL LEAKS.



Boost Pump Access Panel
Figure 201

C. Removal – Boost Pump Check Valve

(1) Defuel wing tank containing check valve to be removed.

NOTE: Defueling the wing tanks should be accomplished with crossflow valve close to prevent fuel from crossing into opposite tank.

(2) Remove boost pump access panel.

(3) Remove boost pumps.

(4) Remove check valve from affected pump outlet line.

NOTE: Note direction of flow arrow on check valve and install in same direction.

D. Installation – Boost Pump Check Valve.

NOTE: When replacing check valves, ensure correct type valve is used. Note that part numbers of check valves differ. One valve has a hole through the center while the other does not. The hole is to release built-up pressure that could cause vapor lock should aircraft remain idle for an extended period of time.

(1) Position valve in place and connect lines.

(2) Install boost pumps.

(3) Fuel aircraft and perform a functional check of boost pumps.

(4) Install boost pump access panel with screws as shown in Figure 201.

2. Approved Repairs – Fuel Boost Pump

A. Boost Pump Wiring Repair

NOTE: If the boost pump wiring or connectors become damaged, repair IAW the following procedures.

(1) Boost Pump Electrical Connector.

(a) Cut wiring to existing connector (see Figure 202, item A) as close to connector as possible.

(b) Solder wires into new connector.

(c) Clean wires to connector with methyl propyl ketone (MPK) or equivalent.

(d) Prime wire in area to be potted using Product Research Corporation PR 1523 primer or equivalent.

(e) Fill the backshell of the connector with MIL-S-8516 Class 1 potting compound (PR 1201Q, CS 3100, or equivalent).

(f) Cure time may be decreased by allowing connector to air dry for 30 minutes; then cure for three (3) to four (4) hours at not more than 120°F (48.9°C).

B. Wing Connection Damage

(1) Replace entire harness (see Figure 202, item B). New harness may be ordered or locally fabricated as follows:

(a) Using old harness as a guide, cut new wires to length. Label individual wires the same as old wires.

(b) Solder connectors at ends of wire.

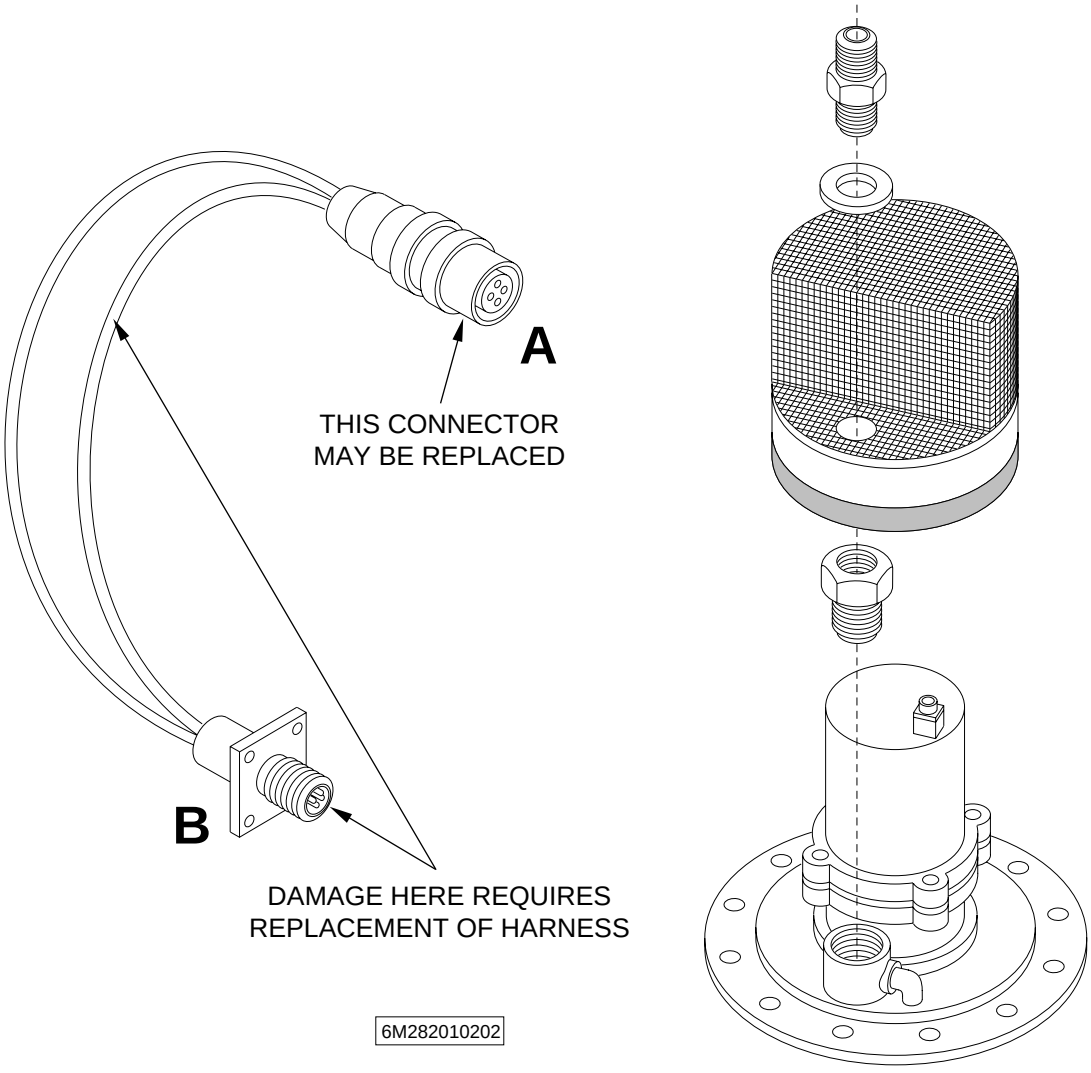
(c) Clean wires using methyl propyl ketone (MPK) or equivalent. Prime wire in area to be potted using PR 1523 primer or equivalent.

(d) Wrap wire harness with Heyco-Flex V, Slit Convoluted Tubing, P/N 1634. Secure at both ends with nylon tie strap.

(e) Fill the backshell of each connector with MIL-S-8516 Class 1 potting compound (PR 1201Q, CS 3100).

(f) Cure time may be decreased by allowing connectors to air dry for 30 minutes; then cure for three (3) to four (4) hours at not more than 120°F (48.9°C).

(g) Coat the receptacle with MIL-S-8802 sealant (PR 1422, P/S 890 or equivalent).



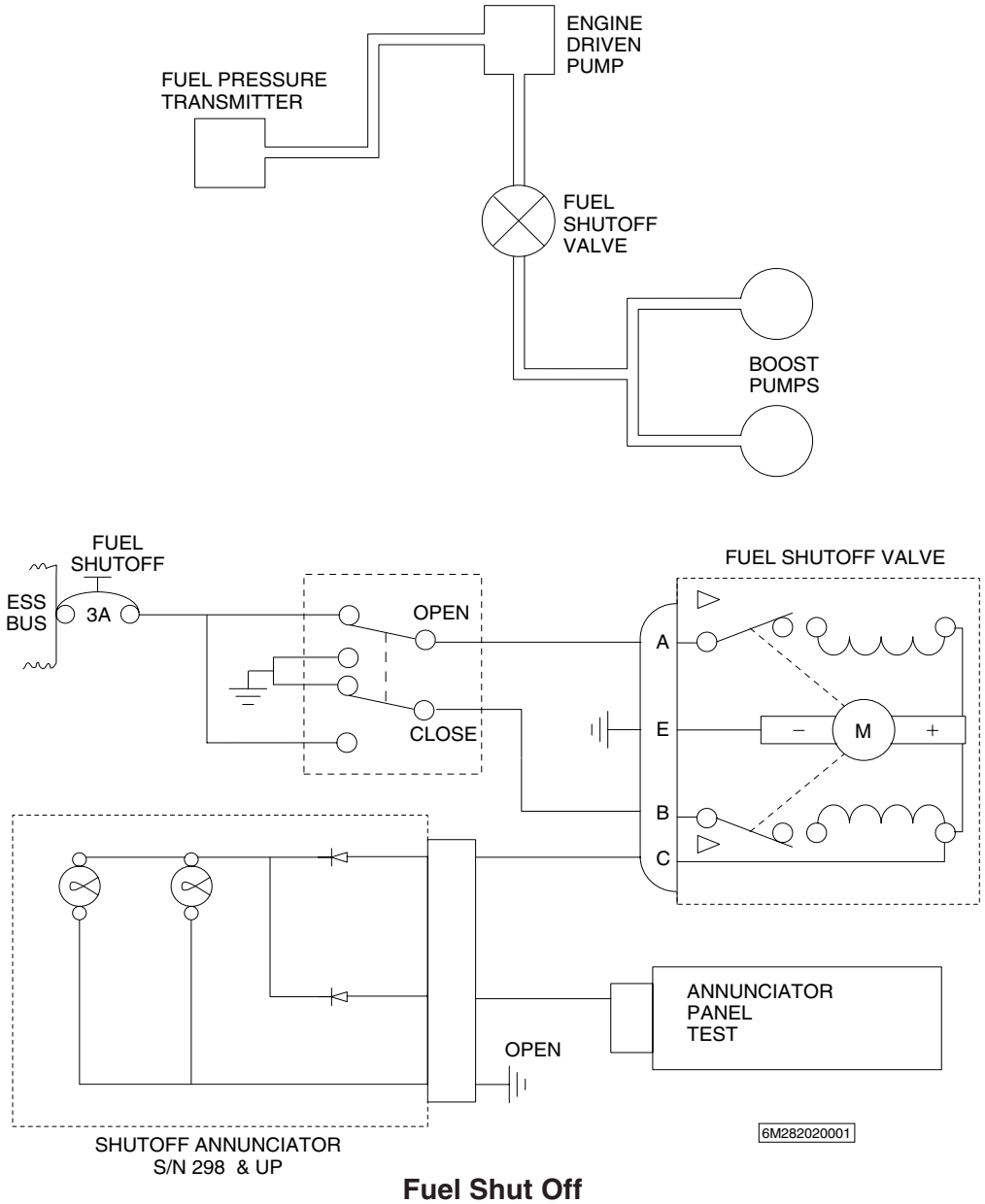
Boost Pump
Figure 202

FUEL SHUTOFF VALVE – DESCRIPTION AND OPERATION

1. General

A. A toggle switch on the center pedestal in the cockpit controls a fuel shutoff valve for each engine. On aircraft prior to T249, AT025, and TC209, the shutoff valve is located inside the left hand keelson web in the nacelle and accessible through the wheel well by removing the large square access panels between nacelle stations 99 and 109. On aircraft T249 and up, AT025 and up, and TC209 and up, the shutoff valve is located in the upper left hand corner of the wheel well along the left hand keelson and the main spar.

B. The shutoff valve is motor operated and controlled by the two position (OPEN-CLOSED) toggle switch. Limit switches are incorporated in the valve to de-energize the motor when the gate reaches the full open or closed position.



Fuel Shut Off
Figure 1

FUEL SHUTOFF VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – Fuel Shutoff

A. Removal – Fuel Shutoff Valve (T201-248, AT001-024, TC201-208)

(1) Defuel tank corresponding to valve to be replaced.

(2) Remove access panel on left hand side of appropriate wheel well.

(3) Disconnect electrical connector.

(4) Disconnect fuel supply lines.

(5) Cap fuel lines and ports.

(6) Remove mounting bolts, washers, and nuts securing valve to mounting bracket.

NOTE: Mounting bolts not only secure the valve assembly to the mounting bracket, but also secure the end plates to the valve body. Take care not to drop the end plates when the final bolt is removed.

(7) Remove valve from wheel well. Retain end plates and attach hardware.

B. Installation – Fuel Shutoff Valve (T201-248, AT001-024, TC201-208)

(1) Replace O-rings in end plates.

(2) Place valve into position against mounting bracket in wheel well. Take care that O-rings do not slip out of position.

(3) Secure valve to mounting bracket with mounting bolts, nuts, and washers. Torque bolts 35 to 40 inch-pounds (3.95 to 4.52 N·m).

CAUTION: DO NOT OVERTORQUE BOLTS AS DAMAGE TO THE END PLATES COULD RESULT.

(4) Remove caps from fuel lines and ports.

(5) Secure fuel lines to valve. Torque fittings 250 inch-pounds (28.25 N·m).

(6) Connect and safety electrical connector.

(7) Fuel aircraft, perform a functional test for leaks and valve operation.

(8) Install clamps and close zipper on shroud if applicable.

C. Removal – Fuel Shutoff Valve (T249-999, AT025-999, TC209-999)

(1) Defuel tank corresponding to valve to be replaced.

(2) Remove tie straps securing shroud to valve.

(3) Remove clamp at each end of shroud; unzip shroud and fold back for access to valve.

NOTE: The shroud is installed on shutoff valves on S/N T272-999, AT057-999, TC229-999, and aircraft retrofitted according to Service Bulletin 540003.

(4) Disconnect electrical connector.

(5) Disconnect fuel supply lines.

(6) Cap fuel lines and ports.

(7) Remove mounting bolts, washers, and nuts securing valve to mounting bracket.

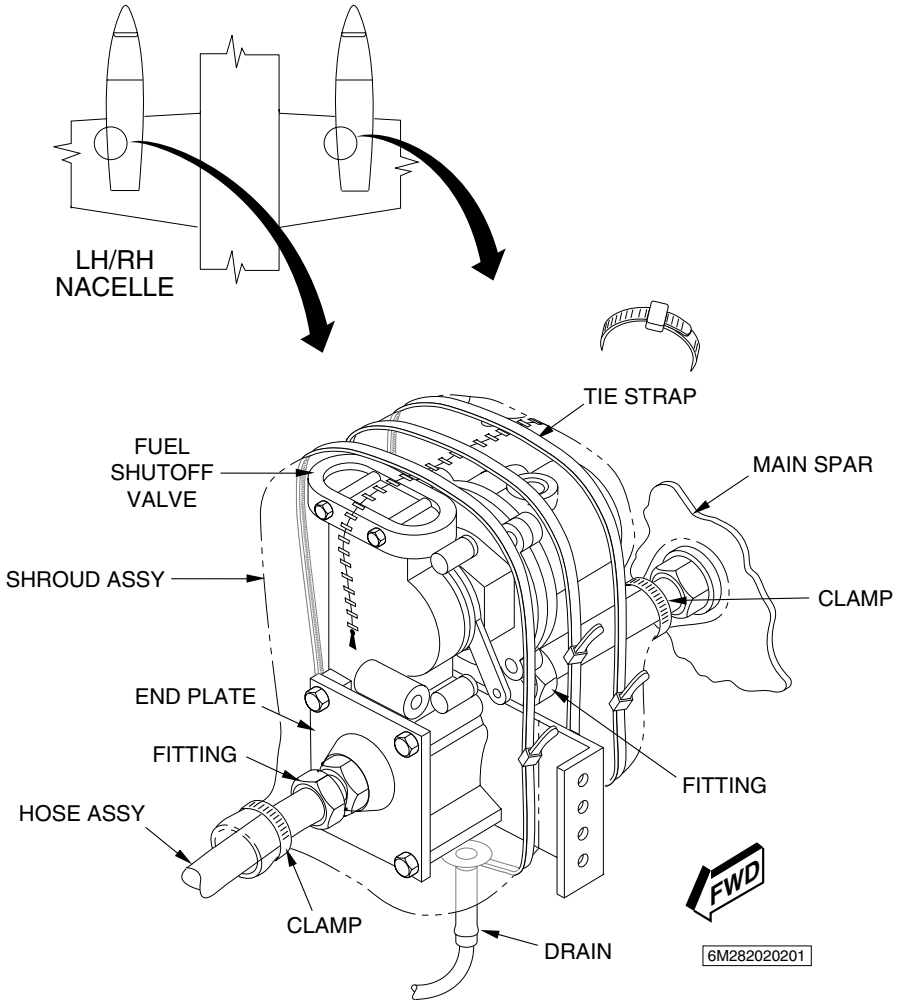
NOTE: Mounting bolts not only secure the valve assembly to the mounting bracket, but also secure the end plates to the valve body. Take care not to drop the end plates when the final bolt is removed.

(8) Remove valve from wheel well. Retain end plates and attach hardware.

D. Installation (T249-999, AT025-999, TC209-999)

(1) Replace O-rings in end plates.

(2) Place valve into position against mounting bracket in wheel well. Take care that O-rings do not slip out of position.



Fuel Shutoff Valve
Figure 201

(3) Secure valve to mounting bracket with mounting bolts, nuts, and washers. Torque bolts 35 to 40 inch-pounds (3.95 to 4.52 N·m).

CAUTION: DO NOT OVERTORQUE BOLTS AS DAMAGE TO THE END PLATES COULD RESULT.

(4) Remove caps from fuel lines and ports.

(5) Secure fuel lines to valve. Torque fittings 250 inch-pounds (28.25 N·m).

(6) Connect and safety electrical connector.

(7) Fuel aircraft and perform a functional test for valve operation; check hoses and valve for leaks.

(8) Install clamps and close zipper on shroud if applicable.

(9) Secure shroud with tie straps.

NOTE: Three (3) ties may be required to clear shroud from main landing gear drag brace.

2. Removal/Installation – Fuel Shutoff Valve Shroud

A. Removal – Fuel Shutoff Valve Shroud (Figure 201)

(1) Remove fuel shutoff valve IAW Step **1.**, C.

(2) Remove clamp from shroud drain line.

(3) Disconnect shroud drain line.

(4) Remove vinyl covered electrical wires from feedthrough.

(5) Remove shroud from fuel lines.

(6) Remove shroud from bracket.

B. Installation – Fuel Shutoff Valve Shroud (Figure 201)

(1) Install shroud on fuel lines. Do not tighten clamps.

(2) Install shroud drain line.

(3) Secure shroud drain line with clamp.

(4) Install vinyl covered electrical wires through feedthrough.

NOTE: Route wires inside shroud behind valve to avoid valve position indicator.

(5) Seal feedthrough with Pro-Seal 890.

(6) Bond shroud flaps to bracket with EC-847.

NOTE: Ensure shroud is positioned correctly for installation of fuel shutoff valve.

(7) Install fuel shutoff valve IAW Step **1.,** D.

3. Inspection/Check – Fuel Shutoff Valve and Hoses

A. Fuel Shutoff Valve and Pressure Check

(1) Set LH fuel shutoff valve to CLOSED.

(2) For aircraft not equipped with indicating lights for the fuel shutoff valves, allow time for motor-operated valves to close. For aircraft equipped with fuel shutoff lights in the valve position indicator assembly (top left), verify that LH fuel shutoff in-transit light illuminates during valve travel, and light extinguishes when valve reaches FULL CLOSED position.

(3) Set RH fuel shutoff valve to CLOSED.

(4) For aircraft not equipped with indicating lights for fuel shutoff valves, allow time for motor-operated valves to close. For aircraft equipped with fuel shutoff valve lights in the valve position indicator assembly (top right), verify that RH fuel shutoff transit light illuminates during valve travel, and that light extinguishes when valve reaches FULL CLOSED position.

(5) Set fuel boost switches to AUX. Verify fuel pressure gauges read zero (0).

(6) Set LH fuel shutoff valve to OPEN

(7) For aircraft without in-transit lights, allow time for motor-operated valves to close. For aircraft equipped with in-transit lights, verify LH light illuminates during valve travel, and extinguishes when valve reaches FULL OPEN position.

(8) Verify and record that LH fuel pressure gauge indicates approximately 20 psi (1.41 kg/cm^2).

(9) Set RH fuel shutoff valve to OPEN

(10) For aircraft without in-transit lights, allow time for motor-operated valves to close. For aircraft equipped with in-transit lights, verify RH light illuminates during valve travel, and extinguishes when valve reaches FULL OPEN position.

(11) Verify and record that RH fuel pressure gauge indicates approximately 20 psi (1.41 kg/cm^2).

(12) Set fuel shutoff switches to CLOSED. Verify valve in-transit lights illuminate.

(13) Set fuel boost pump switches to OFF. Check fuel pressure.

NOTE: A leak in the fuel system or a defective shutoff valve will cause pressure to leak off rapidly (in 2 to 3 seconds). A slow decrease in pressure is acceptable.

(14) Set fuel shutoff switches to OPEN. Verify valve in-transit lights illuminate.

(15) Verify that fuel pressure gauges indicates zero (0).

(16) Set fuel boost switches to MAIN. Verify and record that fuel pressure gauges indicate approximately 20 psi (1.41 kg/cm^2).

(17) Set boost pumps as required.

B. Leak Check of Fuel Shutoff Valve Hoses.

CAUTION: DO NOT RUN BOOST PUMPS WITH WINGS EMPTY OF FUEL. WINGS MUST HAVE FUEL IN HOPPER TANK(S).

(1) LH side leak check (RH side same).

(a) Electrical power ON, engine not running.

(b) Gain access to the LH/RH GCU on the left side of the wheel well (looking forward).

(c) Remove GCU per SA226 Maintenance Manual, chapter 24-35-30.

(d) Ensure that the LH/RH Fuel Shutoff Valve is in the OPEN position.

(e) Place LH/RH Main or LH/RH Auxiliary Boost Pump in the ON position and verify 10-20 psi (0.70 - 1.41 kg/cm^2) registers on LH Engine Fuel Pressure Indicator.

(f) With an explosion proof flashlight, verify that the flexible hose (steel braided) between the fuel shutoff valve and the firewall is not leaking or seeping fuel. Any evidence of leakage/seep is cause to replace line.

(g) Place LH/RH Main or LH/RH Auxiliary Boost Pump in the OFF position.

(h) Reinstall GCU per SA226 Maintenance Manual, chapter 24-35-30.

(i) Close access opened in step (b).

4. Approved Repair – Fuel Shutoff Valve Containment Shroud

A. Temporary Repair Fuel Shutoff Valve Containment Shroud

(1) The flap on the containment shroud may be glued down to help keep the zipper closed as a temporary measure.

(2) Glue flap with 3M 847 or equivalent pliable glue, or PR-1422.

(3) Temporary repair is for no larger than one quarter (1/4) inch hole.

(4) If hole is larger than one quarter (1/4) inch, shroud must be replaced.

FUEL INDICATING – DESCRIPTION AND OPERATION

1. General

A. The fuel indicating system consists of; sensors, indicators, float valve switches, a transfer light system and associated electrical wiring and connectors.

(1) Both integral wing tanks contain identical fuel indicating systems.

(2) An optional mechanical fuel level indicator (Magna-Stick) is also located in each fuel tank.

B. The low level warning (XFR PUMP) light is activated by a float valve switch located in each wing hopper tank. When tank is full, the float will rise, the transfer pump electrical power is interrupted and XFR PUMP light extinguishes.

C. With more than 700 lbs of fuel, the hopper tank is filled by gravity. With less than 65 lbs of fuel in the tank, XFR PUMP light is illuminated and serves as a low fuel warning.

D. Each wing tank contains five fuel quantity probes. As the tank is filled, the fuel probes are immersed in fuel.

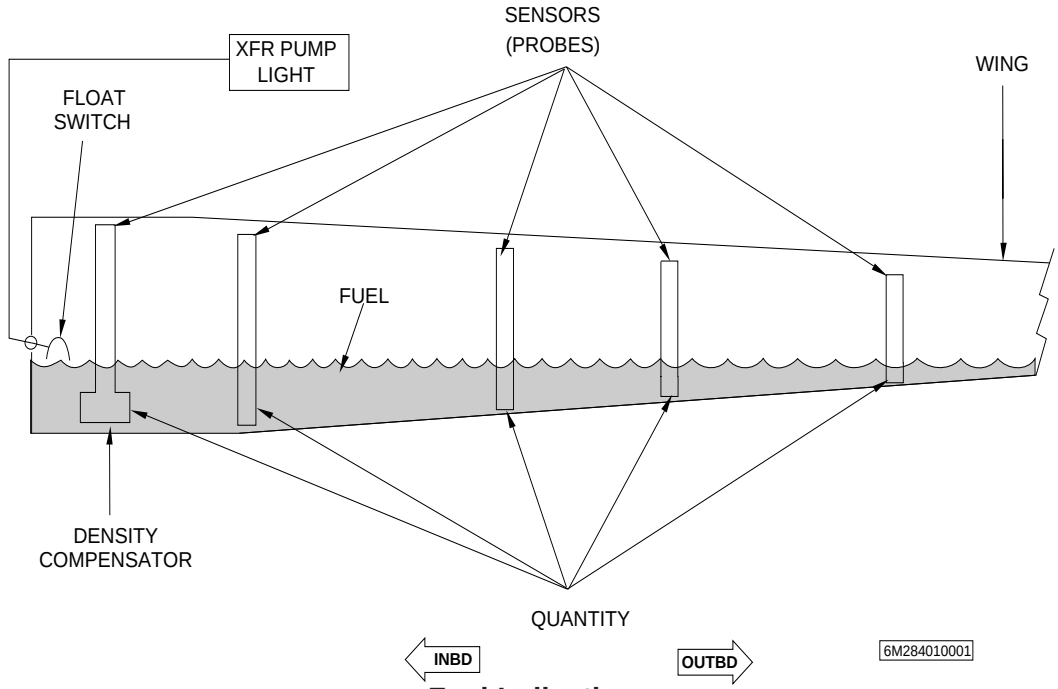
(1) Capacitance of each sensor varies directly to the amount the probe is immersed in fuel.

(2) The system is calibrated to measure fuel quantity of each tank and display this information on an indicator as pounds of fuel.

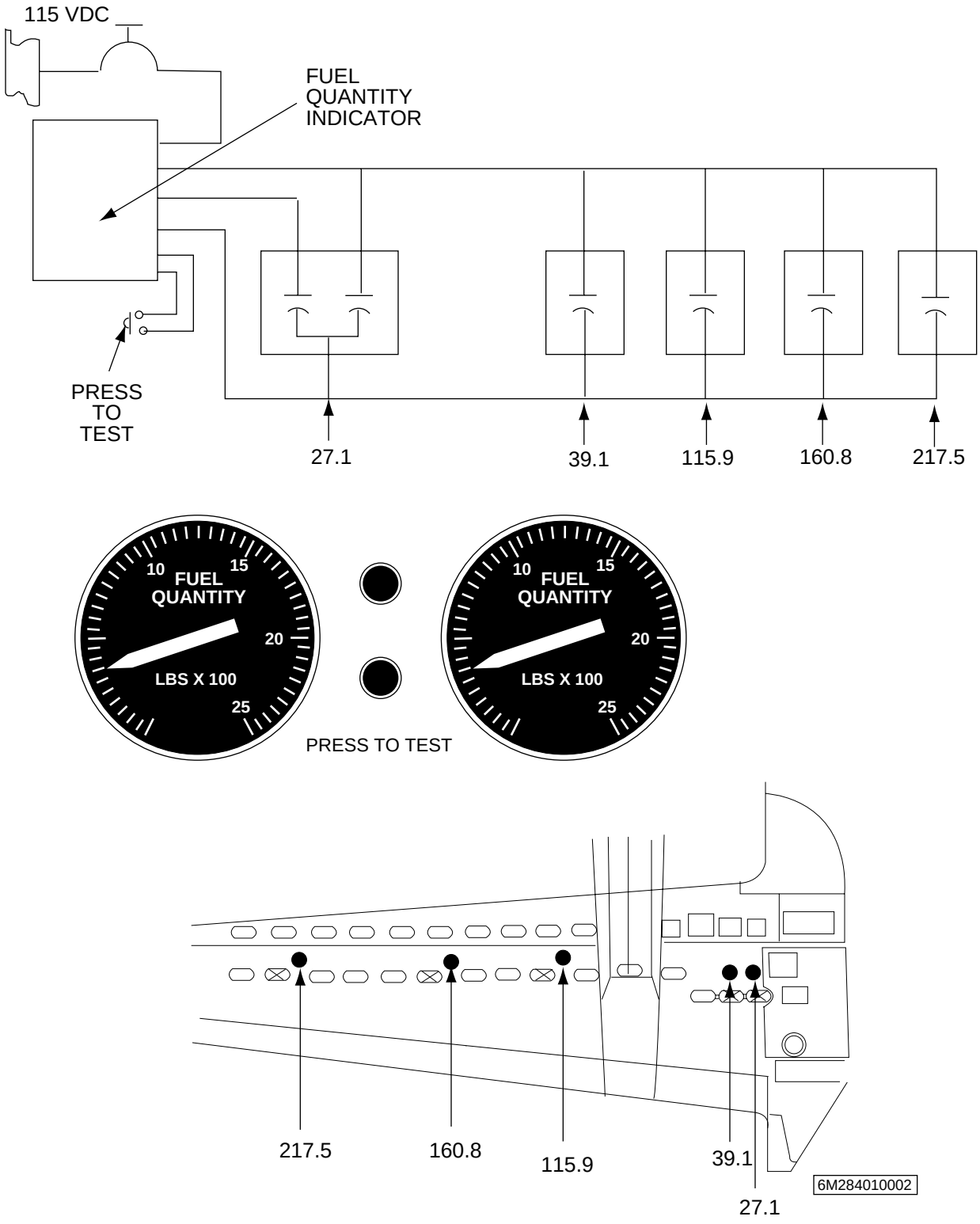
(3) Fuel quantity in each wing is displayed on fuel quantity indicators mounted in the instrument panel.

(4) The inboard probe contains a temperature compensator section. As fuel density varies with temperature, the compensator capacitance varies.

(5) Output of the compensator is used to bias capacitance of tank probes, thereby ensuring weight shown on indicator will be accurate under all temperature conditions.



Fuel Indicating
Figure 1



Fuel indicating
Figure 2

FUEL INDICATING – DESCRIPTION AND OPERATION

2. Fuel Tank Probes

A. Fuel tank probes are made of two concentric tubes. Fuel is introduced into the space between the tubes, forming the variable nonconductor of direct electric current.

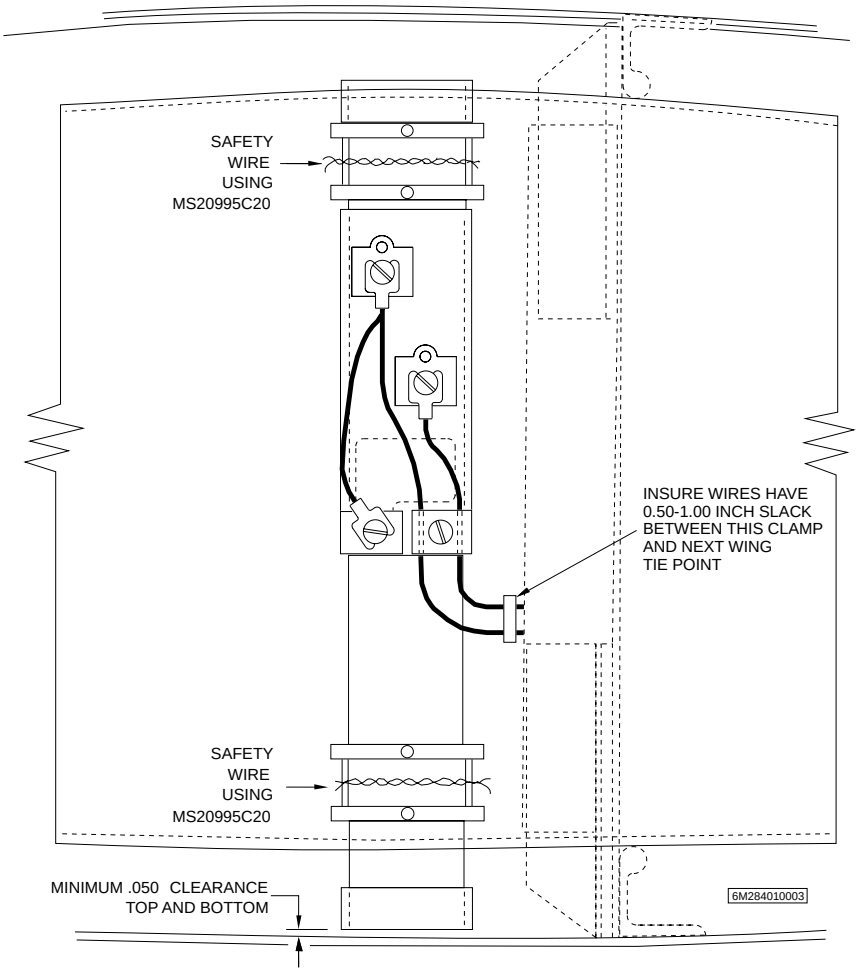
(1) Five probes are located in each wing tank, attached to the structure by clips. Access for maintenance purposes is gained through lower wing access panels (see Figure 3).

3. Fuel Quantity Indicator

A. Two fuel quantity indicators located on the instrument panel display fuel quantity for each integral fuel tank.

B. The indicators display, in pounds, quantity of fuel in the tanks.

C. A test switch mounted near each indicator, will cause indicator needle to move to the 5 o'clock position when pressed. Releasing the switch causes the needle to return to quantity of fuel in tanks.



Typical Fuel Sensor Probe Installation
Figure 3

FUEL QUANTITY SYSTEM – FAULT ISOLATION

1. Troubleshooting – Fuel Indicating

A. If the resistance to ground is lower than specified, the indicator response will be sluggish and a nonlinear error may result.

B. If the capacitance of the system falls out of tolerance, the EMPTY adjustment on the indicator may not have enough range to bring the indicator pointer to zero.

C. All fuel quantity testing should begin with switching the indicator or installing a new one for testing purposes. Always replace the indicator in its original position so that EMPTY and FULL will not have to be calibrated for the operative system.

D. No change in fuel indicator when tanks are filled could be an indication of no power or disconnected indicator.

E. An oscillating indicator may be caused by a loose fuel probe connection or defective indicator.

F. An indicator not adjusting to FULL within the allowed five turns may be a result of the indicator out of calibration or systems capacitance being out of range.

FUEL INDICATING – MAINTENANCE PRACTICES

1. Removal/Installation – Fuel Indicating

A. Removal – Fuel Tank Probe

(1) Defuel tank containing probe being removed.

(2) Remove access panel.

(3) Cut safety wires and remove probe from clips.

(4) Tag and remove wires to probe.

B. Installation – Fuel Tank Probe

(1) Connect electrical wiring to probe. Place a small amount of MIL-S-8802 sealer (Product Research Corporation PR-1422 or equivalent) on terminal screws after installation of wires to ensure that they are locked.

(2) Carefully position probe in place and safety wire clamps using 0.020 inch diameter wire. Ensure there is at least 0.050 inch clearance between the top and bottom of the fuel probe and the wing skin.

NOTE: If the probe was replaced because of a fuel quantity problem and a fuel quantity tester is available, check the system before closing the wing access panel.

(3) Install wing access panel.

(4) Fuel aircraft, observe fuel quantity system for proper operation.

NOTE: Replacement of a fuel probe requires a Capacitance Measurement, Cockpit and Indicator Calibration performance.

C. Removal – Fuel Quantity Indicator

(1) Remove two post lights.

(2) Loosen indicator retaining screw.

(3) Pull indicator straight out.

(4) Cut safety wire and disconnect electrical connector.

D. Installation – Fuel Quantity Indicator

(1) Connect electrical connector and safety wire.

(2) Position indicator in place in panel.

(3) Secure indicator with retaining screw.

(4) Install post lights.

(5) Tighten retaining screw and post lights.

NOTE: Replacement of indicator requires Indicator Calibration Performance.

2. Adjustment/Test – Fuel Indicating

A. Adjustment Test – Indicator

(1) Turn battery switch(es) ON.

(2) After a minimum of ten seconds, press and hold the test switch.

(3) Observe that the indicator needle drives counterclockwise toward zero.

(4) When the needle has passed zero and is at approximately the five o'clock position (moving counterclockwise), release the test switch.

(5) Observe that the needle continues moving counterclockwise until stopping at the actual fuel quantity.

(6) If the test switch is released before the needle reaches the five o'clock position, observe the needle stops, reverses direction, and moves clockwise to the actual fuel quantity.

B. Calibration – Indicator

NOTE: If the indicator or any other parts of the fuel quantity gauging system are repaired or replaced, calibrate the indicator.

(1) Fuel tank with five (5) gallons of fuel.

(2) Remove indicator from panel leaving electrical connector intact.

(3) Turn power ON.

(4) With insulated screwdriver, adjust EMPTY screw setting on back of indicator until indicator needle bisects the zero (0) position.

(5) Turn power OFF.

(6) Fuel tank to full capacity.

(7) With insulated screwdriver, adjust FULL screw setting on back of indicator until indicator needle is set to full capacity in pounds.

NOTE: Use the following capacities for indicator calibration:

SERIES	EFFECTIVITY	CAPACITY
SA226-T	ALL	2170
SA226-TC	ALL	2170
SA226-AT	001-024* 025-999	2170 1850
*If SB28-004 is complied with: 1850		

Aircraft Fuel Capacity
Table 201

C. Bench Check – Indicator

(1) Various brands and models of fuel quantity testers may be used to check the fuel quantity indicator. The following models have been evaluated by Fairchild Aircraft and found satisfactory:

Model TF-889
Consolidated Airborne
Systems, Inc.
115 Old Country Rd.
Carle Place, LI, NY 11514

Model 2548-G
Barfield Instrument Corp.
4101 NW 29th Street
Miami, FL 33142

(2) In general, any tester capable of the following measurements is acceptable:

(a) Leakage resistance 0-500 megohms plus or minus 10%.

(b) Capacitance -500 pf plus or minus 0.2%.

(c) The ability to simulate capacitances in this range is also helpful.

(3) Section SERV-28 provides instructions for assembling two different adapters for connecting the tester to the indicator. For bench check of the indicator, use adapter A.

(a) Remove the indicator to be tested and disconnect the electrical connector.

(b) Connect adapter A as follows: P1 to indicator; P2, P3, and P4 to tester; P5 to indicator cable.

(c) Set tester probe output at 84 pf plus or minus one and compensator output at 57.75 pf. Check indicator for zero reading. If zero is obtained, proceed to **2.**, C., (3), step (e); if zero is not obtained, perform **2.**, C., (3), step (d).

(d) With insulated screwdriver, adjust EMPTY screw on back of indicator until zero is obtained. If zero cannot be obtained, replace indicator; if zero is obtained, proceed to **2.**, C., (3), step (e).

(e) Set tester probe output at 183 pf plus or minus one and compensator output at 57.75 pf. Check indicator for reading of 2500. If desired reading is obtained, proceed to **2.**, C., (3), step (g); if desired reading is not obtained, perform **2.**, C., (3), step (f).

(f) With insulated screwdriver, adjust FULL screw on back of indicator until a reading of 2500 is obtained. If desired reading cannot be obtained, replace indicator; if desired reading is obtained, proceed to **2.**, C., (3), step (g).

(g) Push and hold test switch; indicator should run counterclockwise. If indicator does not run counterclockwise when test switch is depressed, replace indicator; if indicator reacts properly, perform **2.**, C., (3), step (h).

NOTE: When replacing indicator, perform **2.**, C., (3), steps (c) and (e) before installing new indicator.

(h) Disconnect test set from indicator. Connect aircraft electrical connector to indicator. Install indicator. Test other indicator if necessary.

D. Fuel Quantity Test

(1) Various brands and models of fuel quantity testers may be used to check the fuel quantity system. For satisfactory testers see **2.**, C.

(2) Section SERV-28 provides instructions for assembling two different adapters for connecting the tester to the indicator.

E. Fuel Quantity Test: T201-237, AT001-014, TC201-207

(1) Defuel tank corresponding to system to be tested.

(2) Remove the indicator for the system being tested. Using adapter A, connect P1 to indicator; P5 to fuel quantity indicator cable; P2, P3, P4 to tester. If necessary, connect tester to source of power.

(3) Read total system capacitance which must be 80 to 89 pf. The capacitance of the compensator should be 26.20 plus or minus five pf. Total system resistance to ground should be 500 megohms or greater. If these readings are not obtained, proceed with **2.**, E., step (4).

NOTE: In each of the following steps, one probe at a time will be disconnected. At each step the system should be checked for the listed capacitance and resistance to ground. If disconnecting a probe causes the system to read good, the probe or the associated wiring is defective.

(4) Disconnect electrical connector at WS 27.1. If total capacitance reading is 80.98 plus or minus five pf, replace probe. If system resistance is now correct (500 megohms or greater), replace probe. If readings are not correct, install connector and proceed to **2.**, E., step (5).

(5) Disconnect electrical connector at WS 39.1. If total capacitance reading is 65.99 plus or minus five pf, replace probe. If system resistance is now correct (500 megohms or greater), replace probe. If readings are not correct, install connector and proceed to **2.**, E., step (6).

(6) Disconnect electrical connector at WS 115.9. If total capacitance reading is 58.89 plus or minus five pf, replace probe. If system resistance is now correct (500 megohms or greater), replace probe. If readings are not correct, install connector and proceed to **2.**, E., step (7).

(7) Disconnect electrical connector at WS 160.8. If total capacitance reading is 62.21 plus or minus 5 pf, replace probe. If system resistance is now correct (500 megohms or greater), replace probe. If readings are not correct, install connector and proceed to **2.**, E., step (8).

(8) Disconnect the plug at WS 217.5. If total capacitance reading is 71.78 pf, replace probe. If system resistance is now correct (500 megohms or greater), replace probe. (9) Install electrical connector. Disconnect adapter from indicator and aircraft connector. Attach aircraft connector to indicator. Install indicator. Test other system if necessary.

F. Fuel Quantity Test: T238-999, AT015-999, TC208-999

(1) Defuel tank corresponding to system to be tested.

(2) Remove the landing light on the side to be tested.

FUEL PROBE CAPACITANCE RECORD

FUEL CAPACITANCE TANK PROBE & SYSTEM CAPACITANCE CHART TO BECOME PART OF AIRCRAFT LOG BOOK.

AIRCRAFT SN: _____ MECHANIC SIGN OFF: _____

DATE OF TEST: _____ Q.A. ACCEPTANCE: _____

PROBE CIRCUIT UNDER TEST	PROBE PART NO.	LH & RH PROBE LOCATION W.S.	ACCEPTABLE CAPACITANCE	RECORD CAPACITANCE READINGS	
				LH	RH
1	PAB575-1	27.103	4.02 ± 0.5 pf		
2	PAA574-4	39.134	19.01 ± 0.5 pf		
3	PAA574-1	115.994	25.11 ± 0.5 pf		
4	PAA574-2	160.889	21.79 ± 0.5 pf		
5	PAA574-3	217.500	12.22 ± 0.5 pf		
CAL-IND	ALL ABOVE	ALL ABOVE	82.15 ± 3.0 pf		
COMP	PAB575-1	27.103	26.20 ± 1.0 pf		

6M284010201

Fuel Probe Capacitance Record
Figure 201

(3) Disconnect electrical connector located on the aft collector tank. Check connector for looseness, wear, dirt, broken or frayed wiring. Ensure that connector is clean and dry. Using adapter B, connect P1 to the portion of the connector in the wing rib. Connect P2, P3, and P3 to the tester.

(4) Starting with position 1 on the adapter selector switch, select and record the readings of each position 1 through 5 and the COMP position. Set the selector switch to TOTAL and push total button; record the reading. Compare the readings obtained with those in Figure 201. If any reading is not within the tolerance, proceed to **2.**, F., step 6. If all readings are within tolerance, perform **2.**, F., step 5.

(5) Read system total resistance to ground. Total resistance must be 500 megohms or higher. If desired reading is not obtained, read resistance through each position of the selector switch until defective probe is isolated.

(6) Remove the access panel to expose the desired probe. Check probe and wiring for dirt, broken or frayed wires, or foreign matter. Probe may be washed with alcohol if contaminated with water or leak detecting soap. If no obvious defect in wiring or installation is discovered, replace probe, install access panel, and proceed to **2.**, F., step 7.

(7) Disconnect adapter and tester. Install aircraft electrical connector at wing rib and install landing light. Remove indicator for system being tested. Disconnect the electrical connector on back of indicator. Using adapter A, connect P1 to indicator; P5 to fuel quantity indicator cable; P2, P3, P4 to tester. If necessary, connect tester to source of power.

(8) Read total system capacitance which must be 80 to 89 pf. The capacitance of the compensator should be 26.20 plus or minus 5 pf. Total system resistance to ground should be 500 megohms or greater. Compare these readings with those obtained in **2.**, F., step 4. If a significant difference exists between the results obtained in the two tests, the wiring and connectors between the cockpit and wing may be defective. The connectors at either end of the cable are the most suspect and should be inspected first.

(9) After all defects have been corrected, disconnect adapter from indicator and aircraft cable. Connect aircraft electrical connector to indicator and install indicator.

(10) Fuel aircraft.

G. Wiring Repair – Fuel Indicating

(1) Repairs to the fuel indicating system wiring must be made very carefully to avoid compromising system accuracy. The following procedures must be followed:

H. Single Shielded Wire Terminations

(1) Remove 0.40 to 0.50 inch outer insulation (see Figure 202, item A). Be careful not to cut shield.

(2) Fold shield back over outer insulation to expose inner conductor (see Figure 202, item A).

(3) Make jumper from inner conductor of shielded wire. Solder jumper to shield (see Figure 202, item A). Place solder sleeve over wire so that soldered joint is covered. Use a heat gun to shrink solder sleeve.

(4) Prepare inner conductor as necessary.

I. Jumpers at Electrical Connectors

(1) Pin-to-jumpers at electrical connectors must be 20 AWG tinned solid copper wire. Length of jumpers will be determined as shown in Figure 202, item B.

J. Potting Electrical Connectors

(1) The terminal portion of electrical connectors outside of the fuel tank shall be potted as follows:

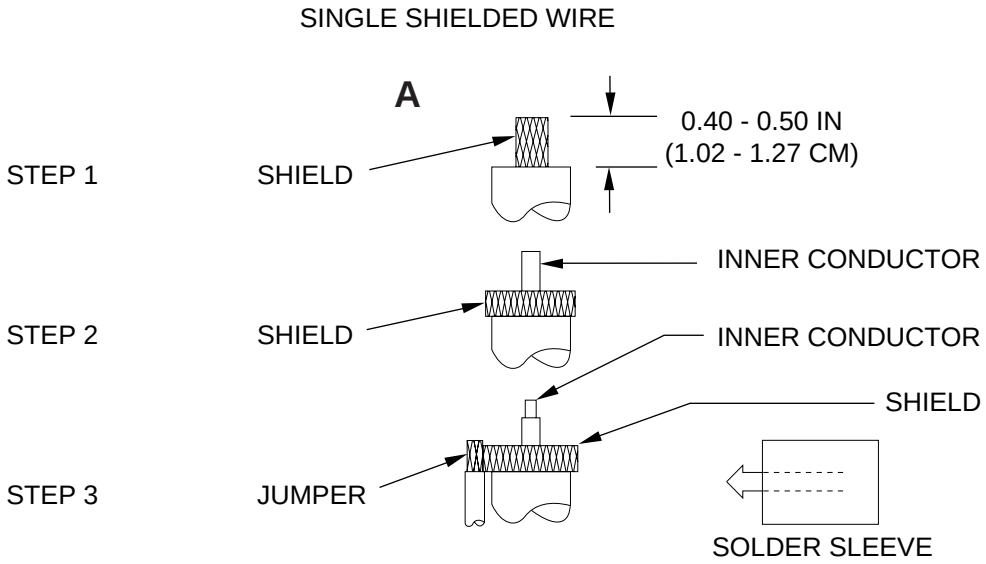
(2) Clean wires using methyl propyl ketone (MPK), or equivalent, or trichloroethane.

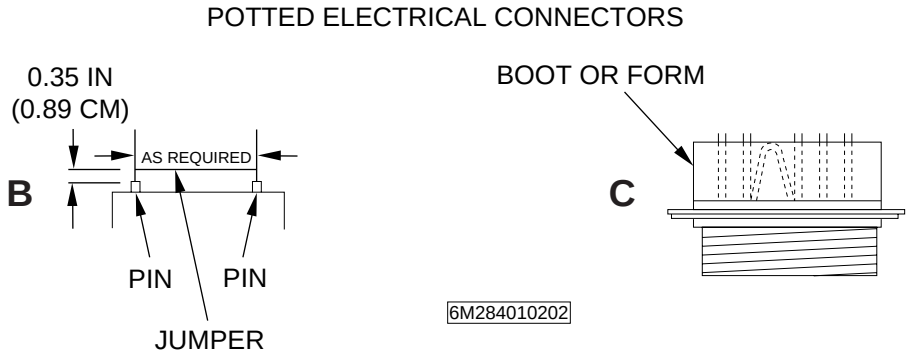
(3) Prime wires using PR1523 primer (Product Research Corporation) or equivalent.

(4) Using a boot or form, apply potting compound meeting MIL-S-8516 (Coast Pro-Seal 727 or equivalent). Potting must be deep enough to completely cover any jumpers in connector (see Figure 202, C).

(5) Cure potting. Cure time may be shortened by allowing the connector to air dry 30 minutes; then cure three to four hours at not more than 120°F (48.89°C).

NOTE: Aircraft with an electrical connector for each probe may have potting on the portion of the electrical connector inside the fuel tank. When replacing these connectors, do not pot the portion inside the fuel tank.





Electrical Connections/Terminations
Figure 202

FUEL INDICATING – MECHANICAL – DESCRIPTION AND OPERATION

1. General

A. The magna-stick fuel level indicator is a customer option for all SA226 aircraft, serving as a backup fuel quantity indicator system. It provides a mechanical measure of fuel quantity in gallons, for each wing tank.

2. Magna-Stick Description

A. Refer to Figure 201, this section for location and structure of the magna-stick.

B. The magna-stick indicator consists of a sealed aluminum tube, base mounted on a fuel tank access panel. The aluminum tube projects vertically up into the fuel tank. A doughnut shaped plastic float containing a magnet, is free to slide up and down on the outside of the tube as fuel level changes.

C. The tube contains a calibrated stick equipped with a permanent magnet. When the stick is released by rotating the lock tab, it slides down the tube and latches magnetically to the float, so that its scale moves with the float to indicate fuel level.

3. Magna-Stick Readings

A. The magna-stick is located on the underside of each wing tank between wing stations 66.584 and 81.714 (see Figure 1). To take a reading;

(1) Push up lock tab in center of access panel with a screwdriver.

(2) Turn 90 degrees in either direction.

(3) Let indicator scale drop down.

(4) Read fuel level flush with the base of the instrument.

B. The scale moves into position without assistance, and registers fuel quantity from 20 to 160 gallons.

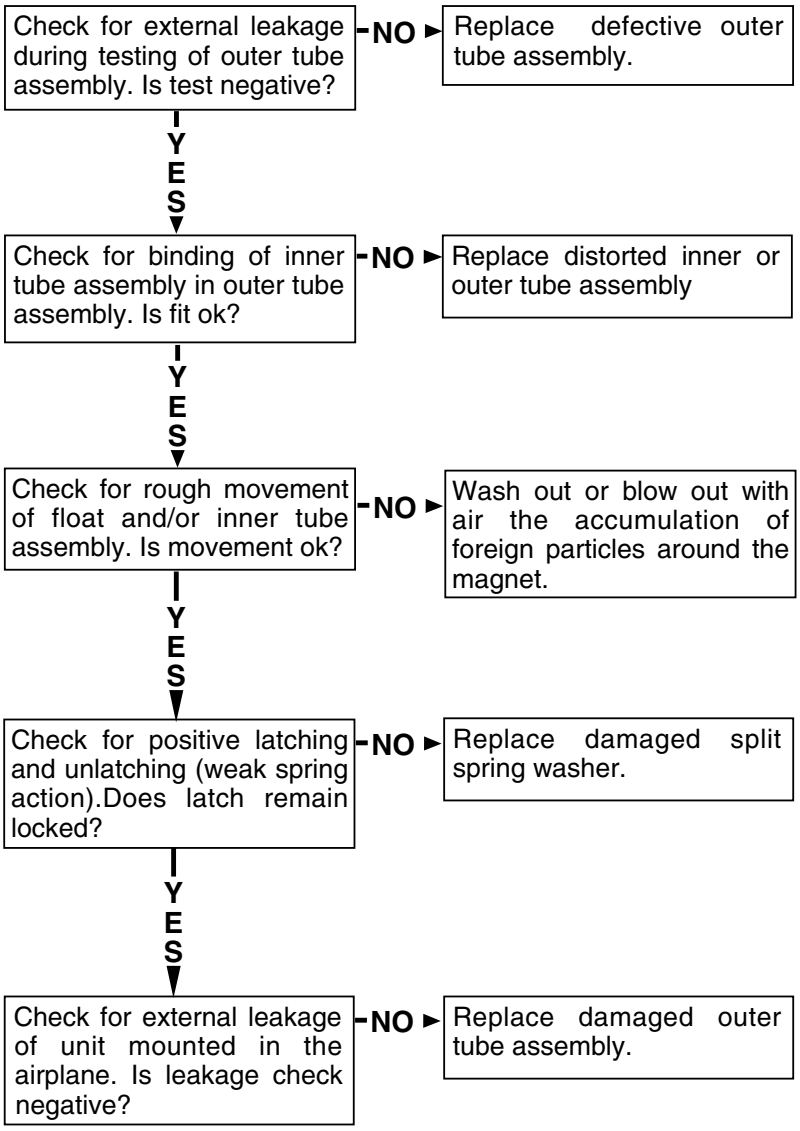
B. Magna-stick readings do not require temperature adjustment.

C. It is possible to have differences between left and right indicators of as much as 200 pounds with equalized fuel levels. It is possible to have differences between indicators and Magna-Sticks of as much as 120 pounds.

1. Troubleshooting

A. Troubleshoot the magna sticks as follows:

START:



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END

1. Removal/Installation – Fuel Indicating – Mechanical

A. Removal – Magna–Sticks

(1) Defuel the tank containing the indicator to be removed.

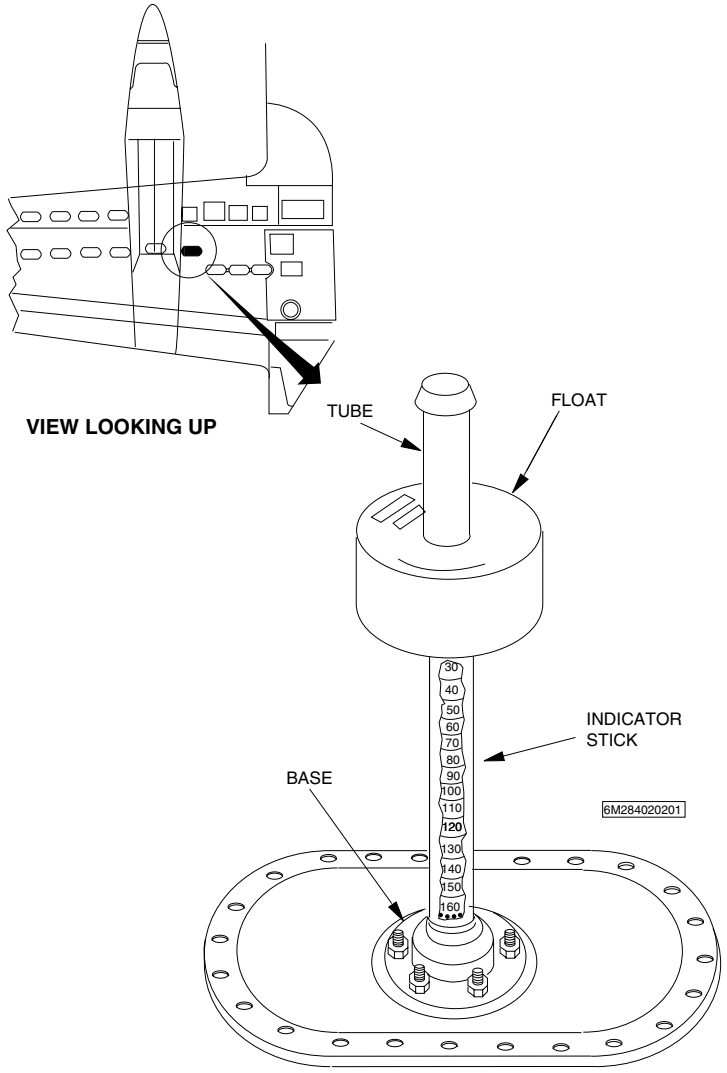
(2) Remove the access panel housing the indicator.

(3) Remove the indicator from the panel by removing screws, nuts, and washers.

B. Installation – Magna–Sticks

(1) Coat the indicator base with MIL-S-8802 sealant (PR-1422, P/S 890 or equivalent) and install using original screws, nuts, and washers. Torque attaching hardware to 35 inch-pounds (3.95 N·m).

(2) Seal and install the access panel.



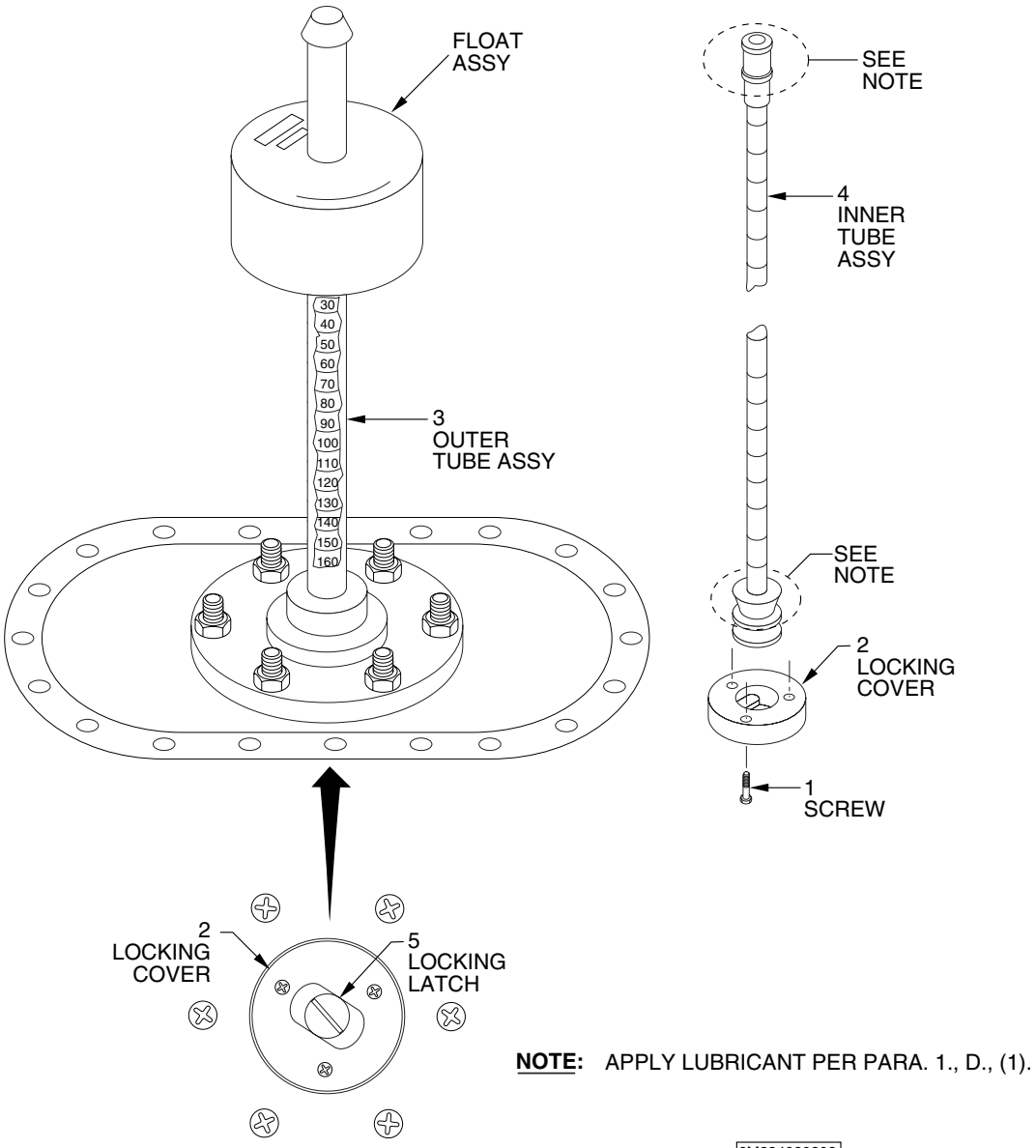
Magna-Stick Installation
Figure 201

C. Removal – Magna–Stick Inner Tube Assembly

(1) Remove the screws (Figure 202, item 1) from the locking cover (Figure 202, item 2) at the bottom of the outer tube assembly (Figure 202, item 3).

(2) Slide the inner tube assembly (Figure 202, item 4) from the outer tube assembly.

NOTE: Do not attempt to disassemble the float assembly and outer tube assembly. Detail parts are non-procurable separately. For repair of inner tube assembly refer to para. **4.** Repairs.



Magna-Stick Inner Tube Assembly
Figure 202

D. Installation – Magna–Stick Inner Tube Assembly

CAUTION: BEFORE INSTALLING THE INNER TUBE ASSEMBLY (FIGURE 202, ITEM 4) ENSURE THAT NO FERROUS PARTICLES HAVE BEEN ATTRACTED TO THE MAGNETS.

(1) Apply a light lubricating grease (DOW CORNING #4 compound or equivalent (MIL-S-8660B) to the surfaces as shown in Figure 202.

(2) Install the inner tube assembly (Figure 202, item 4) into the outer tube assembly (Figure 202, item 3).

(3) Install the locking cover (Figure 202, item 2) with the screws (Figure 202, item 1).

NOTE: Make sure that during the installation of the inner tube assembly (Figure 202, item 4) into the outer tube assembly (Figure 202, item 3), the spring washer (Figures 203 and 204 item 1) is centered to avoid deformation when tightening the cover screws (Figure 202, item 1).

(4) Lock the locking latch (Figure 202, item 5) at the lower end of the inner tube assembly (Figure 202, item 4) to the locking cover (Figure 202, item 2).

NOTE: Check to make sure that there is no (free) axial movement of the inner tube assembly (Figure 202, item 4) within the outer tube assembly (Figure 202, item 3 after installation. If there is (free) axial movement replace the spring washer (Figure 204, item 1).

To lock or release the inner tube assembly requires an axial push of approximately 4.0 lbs (1.814 kg) and a torque of 3.0 inch-pounds (0.34 N·m).

2. Inspection/Check – Fuel Indicating – Mechanical

A. Inspection – Magna Stick Inner tube Assembly

NOTE: Conduct the following inspections under a strong light.

(1) Inspect the inner tube assembly (Figure 202, item 4) for obvious wear, damage, legibility of scale, security of attaching parts, and deformed spring washer.

B. Check – Magna Stick Inner tube Assembly

(1) Check the fit of the inner tube assembly within the outer tube assembly. Make sure that the fit is free sliding.

(2) Check to insure that the magnet in the inner tube assembly locks onto the magnet in the float assembly.

(3) Check to insure that the inner tube assembly is snugly locked within the outer tube assembly as indicated in para. **1.**, D., (4).

3. Cleaning – Fuel Indicating – Mechanical

A. Cleaning – Magna Stick Inner tube Assembly

WARNING: USE THE CLEANING SOLVENT IN A WELL-VENTILATED AREA. AVOID BREATHING THE FUMES AND AVOID EXCESSIVE SOLVENT CONTACT WITH SKIN. KEEP THE SOLVENT AWAY FROM OPEN FLAMES.

(1) Wipe all parts with a clean cloth moistened with a cleaning solvent, Federal Specification P-D-680, or equivalent.

(2) Dry all parts thoroughly with a clean, lint-free cloth or with compressed air at a maximum of 20 psig (1.41 kg/cm^2).

4. Repairs – Fuel Indicating – Mechanical

A. Repairs – Magna Stick Inner tube Assembly

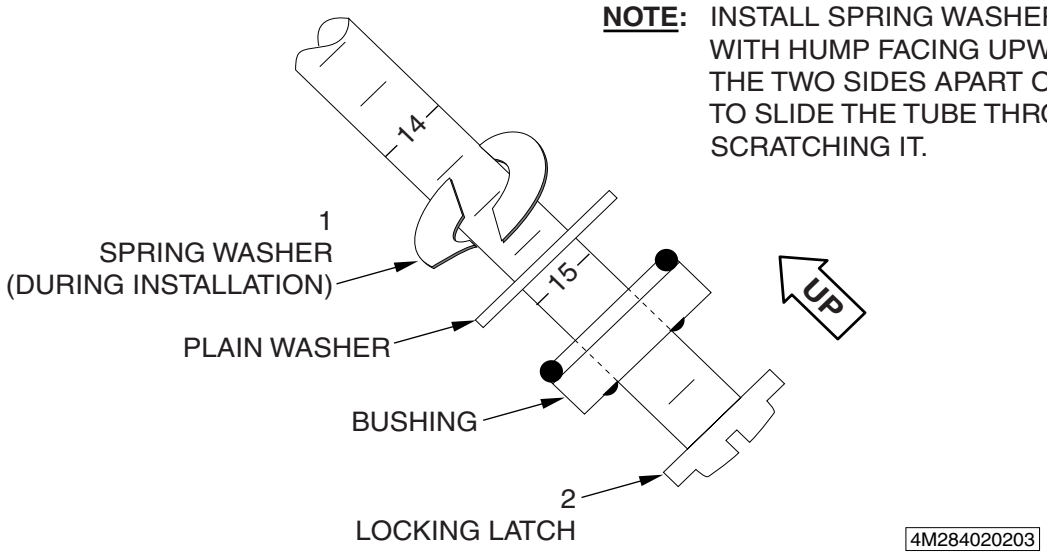
(1) Polish out minor nicks and scratches on aluminum alloy parts with number 600 to 800 grade aluminum oxide abrasive cloth, Federal Specification P-C-451, or equivalent, provided that critical mating surfaces are not damaged.

NOTE: After polishing is complete, clean the parts in accordance with para. **3.** Cleaning.

(2) Repair damage to anodized finish on aluminum parts by applying a protective chemical film per Military Specification MIL-C-5541, or equivalent.

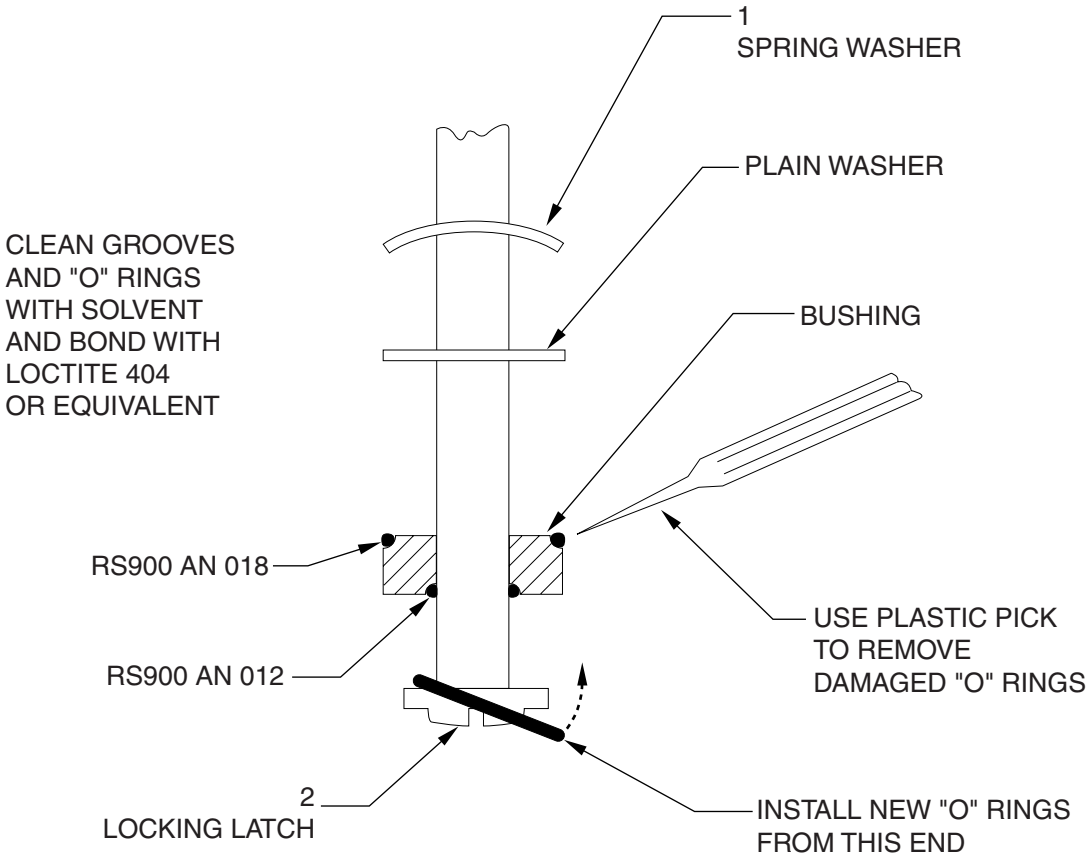
(3) Replace the inner tube assembly if calibrations are worn beyond the state of legibility, or if other damaged components are not available for replacement.

(a) Damaged spring washers of the inner tube assembly may be replaced as shown in Figure 203.



Magna Stick Inner Tube Assembly
Spring Washer
Figure 203

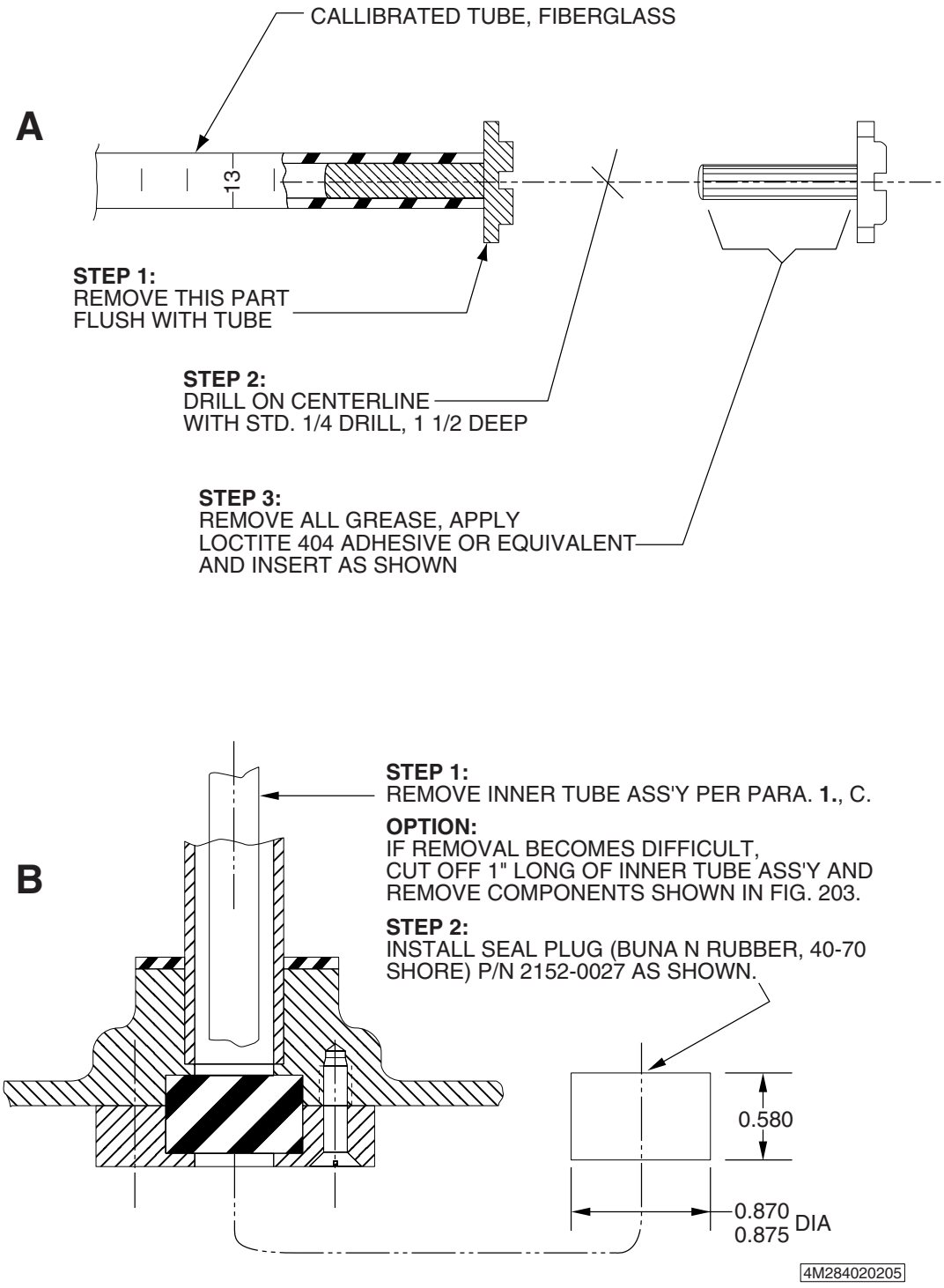
(b) On the inner tube assembly containing redundant "O" ring seals, replace the damaged seals as shown in Figure 204. (Cure date is not required for these "O" ring seals).



NOTE: AFTER PLACING "O" RINGS APPLY A THIN COAT OF DOW CORNING NO.4 GREASE OR EQUIVALENT TO THEIR SURFACES.

Magna Stick Inner Tube Assembly
"O" Ring Seals
Figure 204

(c) A broken latch on the inner tube assembly may be replaced as shown in figure 205.



Magna Stick Inner Tube Assembly and Locking Latch
Figure 205

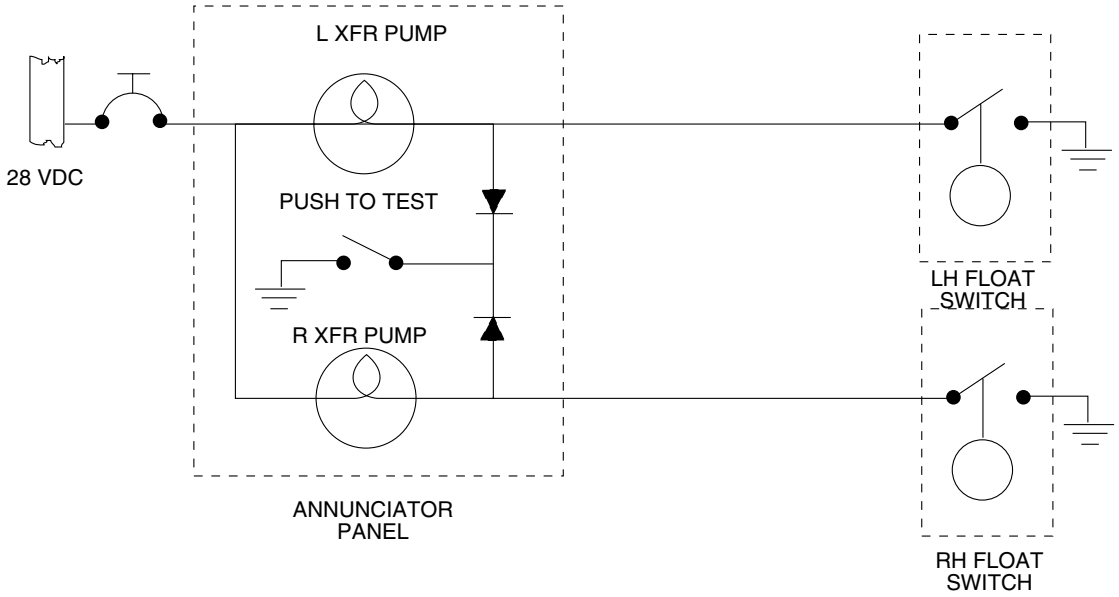
TRANSFER LIGHT – DESCRIPTION AND OPERATION

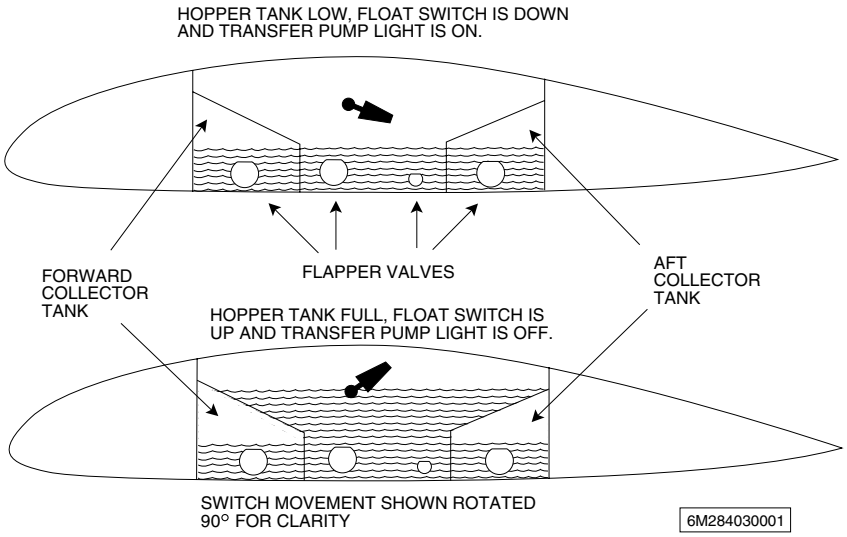
1. General

A. The transfer pump warning light for each integral fuel tank is located on the cockpit annunciator panel (LH XFR PUMP, RH XFR PUMP).

(1) Each XFR PUMP warning light is activated by a float switch located in the hopper tank (see Figure 1). During normal operations, illumination of a XFR PUMP warning light indicates jet transfer pumps are failing to maintain the hopper tank to full fuel capacity.

B. A float switch, which indicates full capacity in each hopper tank, is attached to the inboard wall of the hopper tank. The float is on the inside of the hopper tank rib. The holding nut and electrical connection is on the outside of the rib.



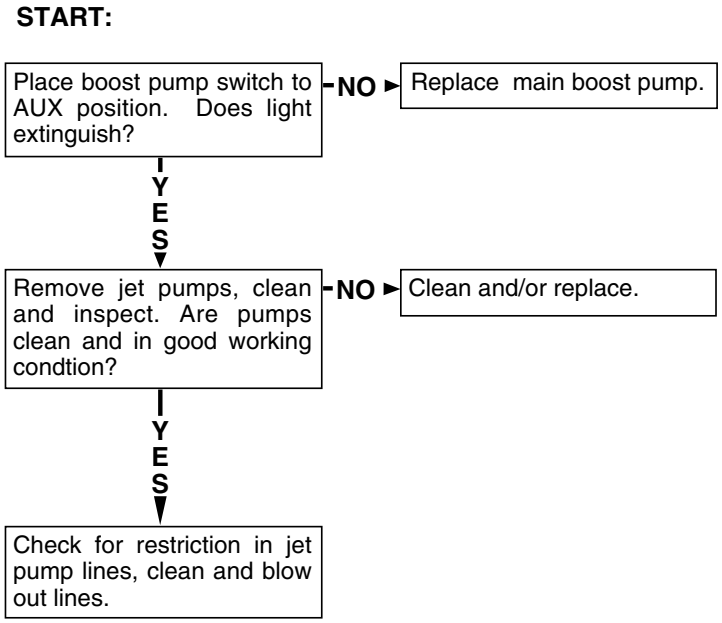


Float Switch
Figure 1

TRANSFER LIGHT – FAULT ISOLATION

1. Troubleshooting – Transfer Light

A. The XFR PUMP warning light will always be illuminated during normal operation with less than 600-700 pounds of fuel remaining in each tank and boost pumps OFF, and with less than 65-70 pounds of fuel remaining in each tank and boost pumps ON. If the XFR PUMP light illuminates during normal operation under circumstances other than this, troubleshoot as follows:



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END

TRANSFER LIGHT SYSTEM – MAINTENANCE PRACTICES

1. Removal/Installation – Transfer Light

A. Removal – Float Switch

(1) Defuel tank containing switch being removed.

(2) Remove center wing section access panel forward of landing light.

(3) Remove inboard boost pump IAW Section 28-20-10 this chapter.

(4) Disconnect electrical wiring from float switch.

(5) Remove holding nut.

(6) Remove float switch from hopper tank.

B. Installation – Float Switch

(1) Position float switch and secure holding nut.

NOTE: Install float switch with side marked TOP up.

(2) Splice electrical wiring to new float switch.

(3) Install boost pump IAW Section 28-20-10 this chapter.

(4) Fuel tank and inspect for leaks.

(5) Perform boost pump operational test.

(6) Install access panels.

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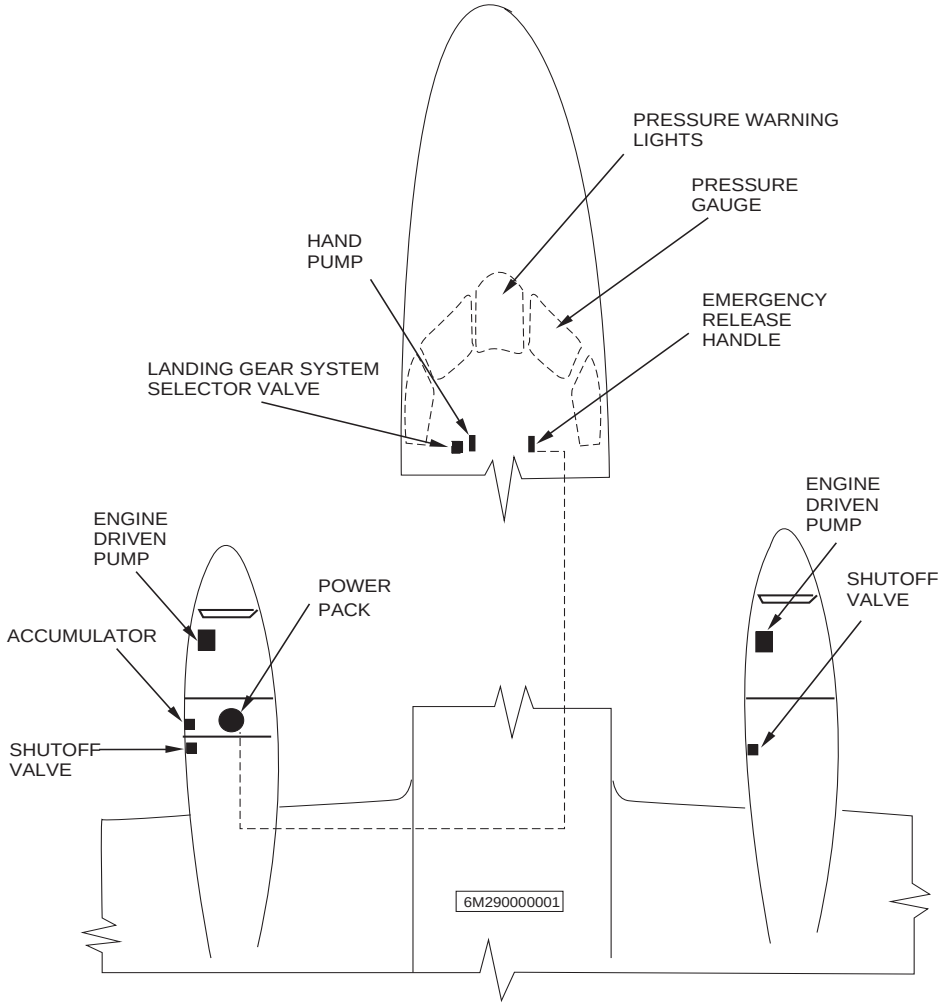


Hydraulics
Figure 1

1. General

A. The aircraft hydraulic system provides pressure to raise and lower the wing flaps and landing gear during normal operation, and to lower the landing gear by emergency means.

(1) Two variable displacement pumps, one driven by each engine, provide pressure for use in the main hydraulic system.

(2) A hand pump in the cockpit provides pressure for the auxiliary hydraulic system.

(3) A direct reading pressure gauge will show the pressure in either the main or auxiliary system.

(4) Pressure distribution and control is accomplished by a hydraulic power pack.

B. The hydraulic power pack, located in the left hand nacelle, stores fluid for use in the system and controls the pressure provided by the engine driven pumps.

(1) Two selector valves in the bottom of the pack control operation of the gear and flaps.

(2) The upper cylindrical portion of the pack is used to store fluid for use in the system.

(3) The hand pump is supplied from a sump in the fluid storage area; the engine driven pumps are supplied through stand pipes.

(4) A shutoff valve is installed in the supply line to each engine driven pump.

C. An accumulator is installed near the left side of the hydraulic pack. The accumulator helps smooth out pressure fluctuations within the system.

D. The auxiliary system consists of the hand pump, a system selector valve, and a manual hydraulic pressure dump system.

(1) The system selector valve, located adjacent to the hand pump, is a two position manually controlled valve. With the selector valve in the NORM GEAR position, the hand pump handle is blocked and the pump outlet is connected to the inlet. With the selector valve in the EMER GEAR position, the hand pump supplied pressure to lower the gear.

(2) The manual hydraulic pressure dump system is controlled by a handle located adjacent to the copilot's inboard seat track.

E. Two warning lights in the annunciator panel and an electric pressure gauge are used to monitor the hydraulic system.

(1) Each warning light is controlled by a pressure switch on the output side of each engine driven pump.

(2) The pressure gauge is connected to a shuttle valve. The shuttle valve allows the gauge to display main system pressure or auxiliary system pressure, whichever is greater.

F. On aircraft equipped with hydraulic nose wheel steering, main system hydraulic pressure is tapped off of the nose landing gear down line to power the nose steering actuator.

G. On aircraft equipped with power brakes, main system hydraulic pressure is tapped off of the nose landing gear down line to provide power to the brake system.

H. The engine driven hydraulic pumps deliver 2,000 psi (140.6 kg/cm^2) pressure to the hydraulic system.

(1) When gear or flap selection is made with the cockpit switches, a solenoid actuated pilot valve on the power pack directs pump pressure to the gear or flap actuators. Fluid from the return side of the actuators is routed through the opposite side of the selector valve and through a filter in the reservoir.

(2) Limit switches actuated by the landing gear retraction mechanism open the circuit to the selector valve, permitting the spring loaded valve to return to the neutral position and trap the fluid. The overcenter uplock mechanism coupled with the trapped fluid mechanically and hydraulically lock the landing gear in the up position.

(3) With the landing gear extended, the selector valve stays in the gear down position and pressure is continuously exerted on the actuators until the engines are shut down and the battery switch turned off.

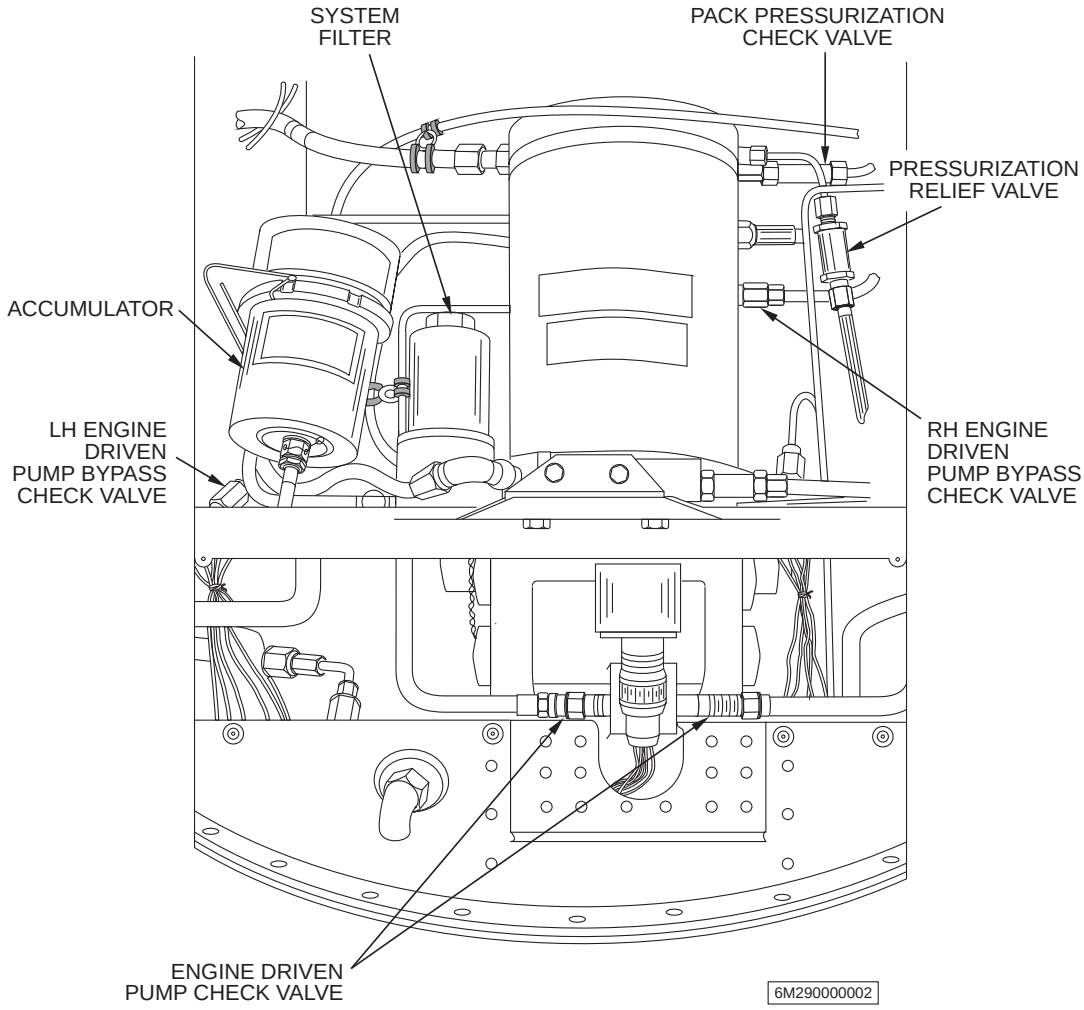
(4) Normal gear or flap extension or retraction time is four to eight seconds. With one pump inoperative, the time increases by two to three seconds.

I. In the event of failure of the main hydraulic system, fluid is drawn from the suction sump in the reservoir by actuating the hand pump in the cockpit.

(1) The landing gear system selector valve is moved forward to the EMER GEAR position and diverts fluid pressure to the emergency gear down cylinders. A shuttle valve permits the reading of auxiliary hydraulic pressure on the cockpit pressure indicator. A hand pump relief valve prevents excessive pressure.

(2) Approximately 800 psi of pressure is required to lower and lock the gear. After lowering the gear, the system pressure should not bleed below 500 psi; however, should the pressure bleed to zero, the overcenter down locks will hold the gear in the down and locked position.

(3) A minimum of one quart of hydraulic fluid is available in the reservoir for use by the auxiliary system.



Hydraulic Components
Figure 2

J. With the engine driven pumps disabled, the shuttle valve located under the pilot's floorboards opens on the hand pump pressure side to display that pressure.

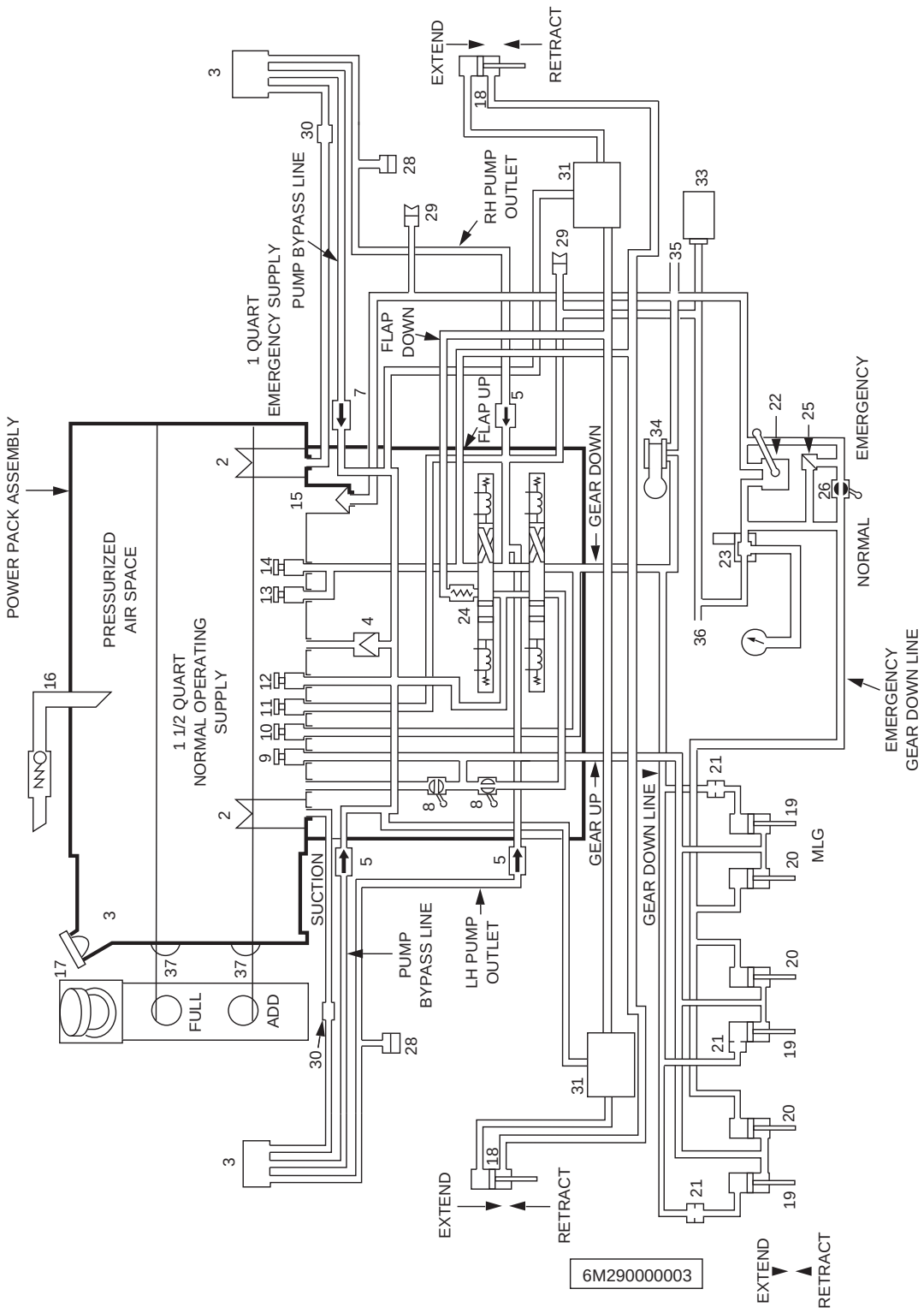
(1) The gauge pressure relief valve located under the pilot's floorboards serve to relieve any trapped pressure above normal system pressure.

(2) The low pressure switches located in the left wheel well behind the firewall close at approximately 1,600 to 1,700 psi (112.48 to 119.51 kg/cm^2) to illuminate low hydraulic pressure warning lights on the annunciator panel.

FIGURE 3 LEGEND – HYDRAULIC SCHEMATIC

1 HYDRAULIC FLUID RESERVOIR
2 PUMP SUCTION STAND PIPE
3 ENGINE DRIVEN HYDRAULIC PUMP
4 SYSTEM RETURN FILTER
5 ENGINE DRIVEN PUMP CHECK VALVE
6 FLAP POSITION SELECTOR VALVE
7 GEAR POSITION SELECTOR VALVE
8 GEAR BYPASS VALVE
9 GEAR UP THERMAL RELIEF VALVE
10 GEAR DOWN THERMAL RELIEF VALVE
11 SYSTEM RELIEF VALVE
12 FILTER RELIEF VALVE
13 FLAP UP THERMAL RELIEF VALVE
14 FLAP UP SUCTION CHECK VALVE
15 HAND PUMP SUCTION SUMP
16 PRESSURIZATION RELIEF VALVE
17 FLUID FILTER & SCREEN
18 FLAP ACTUATING CYLINDER
19 GEAR UP & NORMAL GEAR DOWN ACTUATING CYLINDER
20 GEAR UP & EMERGENCY GEAR DOWN ACTUATING CYLINDER
21 GEAR DOWN RESTRICTOR VALVE
22 EMERGENCY HAND PUMP (A)
23 SHUTTLE VALVE (1)
24 FLAP DOWN RESTRICTOR VALVE
25 HAND PUMP RELIEF VALVE (A)
26 LANDING GEAR SYSTEM SELECTOR VALVE (A)
27 PRESSURE GAUGE (1)
28 PRESSURE WARNING SWITCH (1)
29 EXTERNAL CONNECTION ASSY
30 SHUTOFF VALVE
31 FLAP CONTROL VALVE
32 PRESSURE RELIEF VALVE (1)
33 ACCUMULATOR
34 NOSE GEAR STEERING ACTUATOR
35 POWER BRAKE RETURN
36 POWER BRAKE PRESSURE SUPPLY
37 SIGHT GUAGE (1)

NOTE: Components designated (a) comprise the auxiliary system. Components designated (1) comprise the indicating system. All other components comprise the normal system.



Hydraulic Schematic
Figure 3

1. Connect/Disconnect - External Hydraulic Power

A. Connect External Hydraulic Power

(1) Remove hydraulic access panel 961 from left nacelle (see Figure 201).

NOTE: External hydraulic power cart requirements: 2-1/2 gpm capacity at 2,000 psi, variable below 2,000 psi, a manual on/off control valve and a 10 micron filter.

(2) Remove protective caps on aircraft quick disconnect fittings and hydraulic power cart quick disconnect fittings.

(3) Clean quick disconnect fittings on aircraft and external hydraulic power cart with clean cloth.

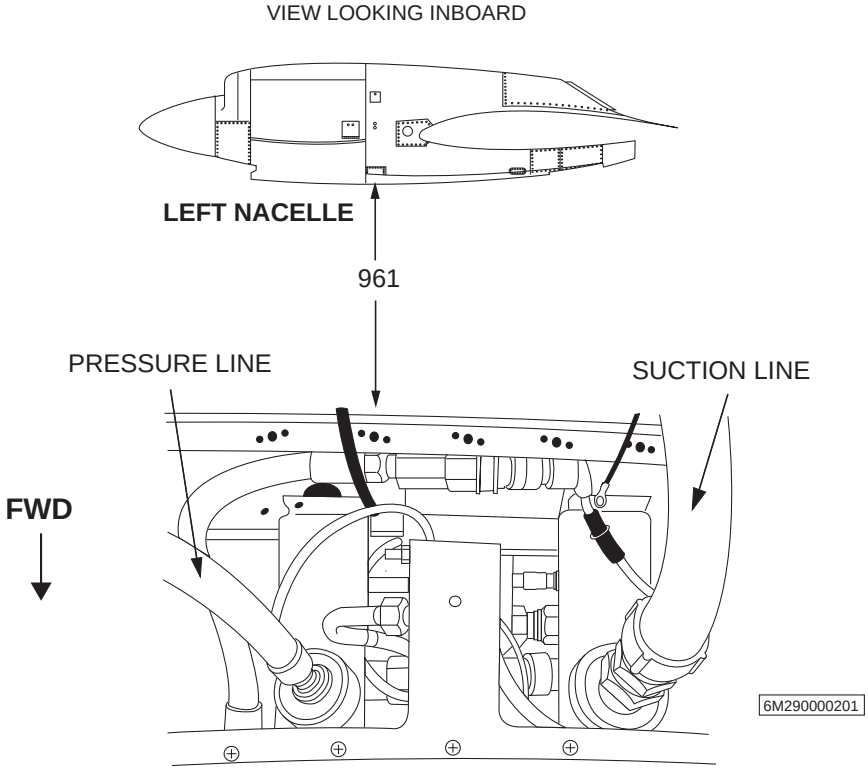
(4) Connect external hydraulic power cart quick disconnect fittings to aircraft quick disconnect fittings.

(5) Verify landing gear handle is in DOWN position and turn battery switches ON.

(6) Verify landing gear indicator shows all three gear down and locked.

(7) Following external hydraulic power cart manufacturer's instructions, turn on external hydraulic power cart and fill hydraulic reservoir to full mark with flaps up. (Refer to 12-10-20, page 201)

(8) Following external hydraulic power cart manufacturer's instructions, apply hydraulic pressure to aircraft system as required for the maintenance procedure being performed.



Quick Disconnect Fittings For External Hydraulic Power Cart
Figure 201

B. Disconnect External Hydraulic Power

(1) Following external hydraulic power cart manufacturer's instructions, fill hydraulic reservoir to full mark with flaps up. (Refer to 12-10-20, page 201)

(2) Reduce external hydraulic power cart pressure to zero.

(3) Disconnect external hydraulic power cart quick disconnect fittings from aircraft quick disconnect fittings.

(4) Verify hydraulic reservoir fluid level is still at full mark. Fill if necessary.

(5) Install protective caps on aircraft quick disconnect fittings and hydraulic power cart quick disconnect fittings.

(6) Install hydraulic access panel 961 on left nacelle (see Figure 201).

2. Removal/Installation – Hydraulics

A. Removal/Installation – Hard Tubing and Lines

MATERIAL	OUTSIDE DIAMETER	WALL THICKNESS	MATERIAL	SPECIFICATION
ALUMINUM	0.250	0.035	5052-0 [1]	WW-T-700/4
	0.250	0.035	6061-T6	MIL-T-7081
	0.375	0.035	6061-T6	MIL-T-7081
	0.500	0.035	5052-0 [1]	WW-T-700/4
STAINLESS STEEL	0.187	0.028	304 CRES	MIL-T-8504
	0.250	0.035	304 CRES	MIL-T-8504
	0.375	0.035	304 CRES	MIL-T-8504
	0.500	0.035	304 CRES	MIL-T-8504

[1] Tubing of this material is only permitted on lines with 1500 psi pressure or less.

Tubing Material
Table 201

TUBE O.D.	RECOMMENDED BEND RADII		OPTIONAL BEND RADII
	3D	4D	6D
0.125	0.380	0.500	0.750
0.188	0.560	0.750	1.130
0.250	0.750	1.000	1.500
0.313	0.940	1.250	1.880
0.375	1.130	1.500	2.250
0.438	1.310	1.750	2.630
0.500	1.500	2.000	3.000
0.625	1.880	2.500	3.750
0.750	2.250	3.000	4.500
0.875	2.630	3.500	5.250
1.000	3.000	4.000	6.000
1.125	3.380	4.500	6.750
1.250	3.750	5.000	7.500
1.375	4.130	5.500	8.250
1.500	4.500	6.000	9.000
1.625	4.880	6.500	9.750
1.750	5.250	7.000	10.50
1.875	5.630	7.500	11.25
2.000	6.000	8.000	12.00

Tube Bend Radii Data
Table 202

(1) Replace hard line tubing with parts manufactured of like material described in Table 201 using bend data in Table 202.

NOTE: It is acceptable to use 304 MIL-T-8504 stainless steel tubes with aluminum end fittings to replace existing aluminum tubes.

When using stainless steel tubes with aluminum end fittings, use the aluminum torque values for installation.

Galvanic corrosion may result due to the use of dissimilar metals.

B. Removal/Installation – Flexible Hose Assemblies

CAUTION: DO NOT TWIST FLEXIBLE HOSES DURING INSTALLATION.

(1) Replacement hose assemblies may be ordered by part number or locally manufactured by the operator. Flexible hose assemblies procured/manufactured by the operator, must be certified IAW Chapter 20, this manual, and must be installed and supported IAW FAA Publication AC43.13-1B (Aircraft Inspection and Repair.)

CAUTION: FLEXIBLE HOSE ASSEMBLIES MAY NOT BE USED TO REPLACE HARD LINE TUBING BETWEEN BRAKE MASTER CYLINDERS AND BRAKES.

(2) Replacement hoses used inside the engine compartment are seamless extruded Teflon inner tube, reinforced with one stainless steel wire braid and protected by one layer of asbestos braid and conform to TSO-C53a or TSO-C75. Replace hoses when they become brittle or collapse.

CAUTION: TEFLON TUBING USED IN THE ENGINE COMPARTMENT TENDS TO ASSUME A PERMANENT SET WHEN EXPOSED TO HIGH TEMPERATURES. DO NOT ATTEMPT TO STRAIGHTEN A HOSE THAT HAS BEEN IN SERVICE, AS ANY BENDING OR TWISTING MAY CAUSE WEAKENING OF THE TUBING WALL.

(3) Replacement hoses used outside the engine compartment are seamless synthetic rubber inner tubing, reinforced with one (1) fiber braid, one (1) braid of high tensile steel wire and covered with an oil resistant rubber impregnated fiber braid.

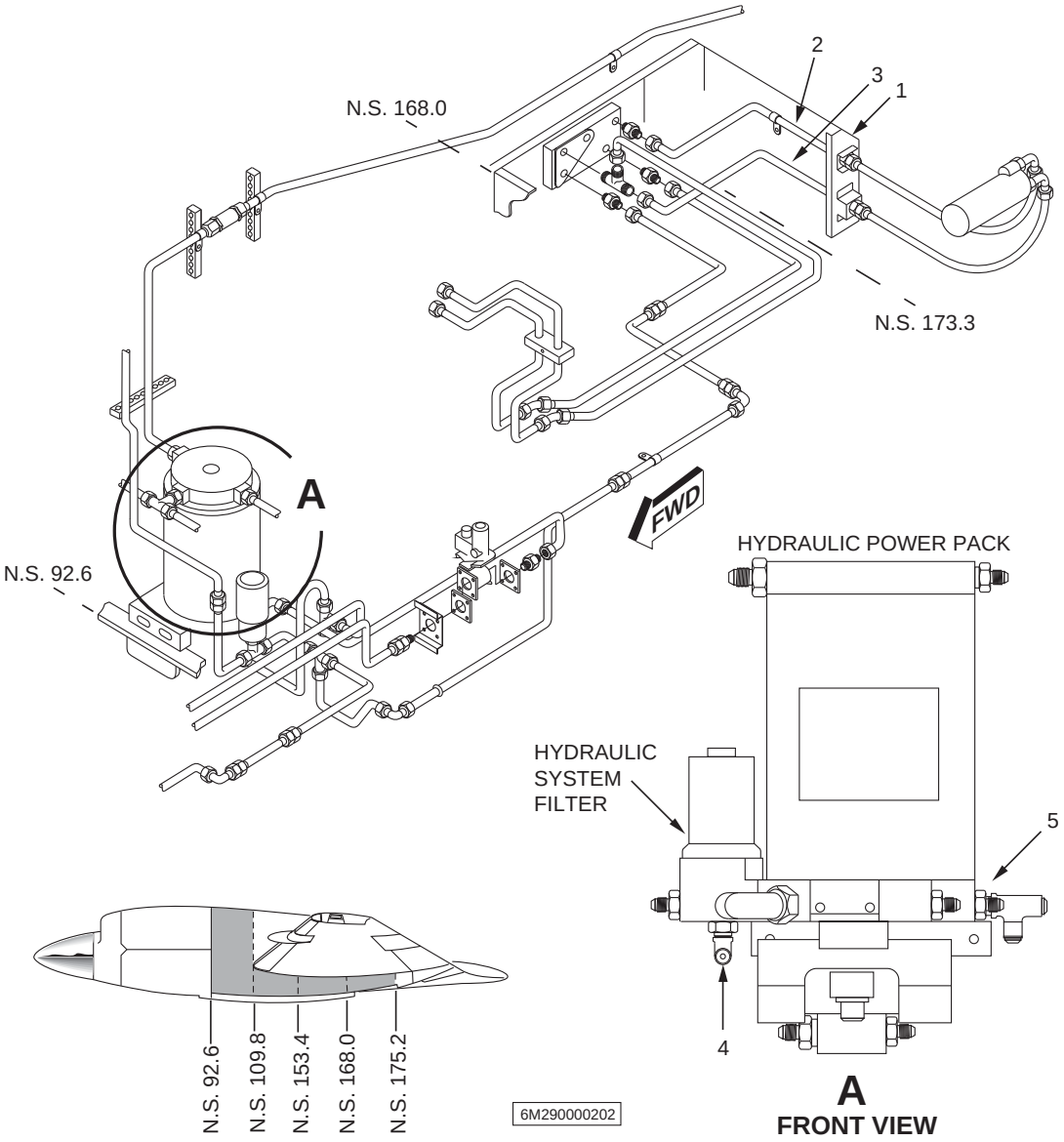
C. Removal/Installation – Clear Flexible Tubing

(1) Where clear flexible tubing is utilized (fluid drainage, etc.), replace with TYGON, Specification B-44-3 or equivalent. Diameter and wall thickness shall match original part.

D. Torque – Hard lines, Tubing, and Hoses

(1) Refer to 20-10-00 Table 202 for all tube coupling nut torque values.

(2) All hydraulic flexible hoses and/or hard lines must be supported using proper size and type clamps.



Purging Hydraulic Disconnects
Figure 202

3. Adjustment/Test – Hydraulics (Figure 202)

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF.

CAUTION: USE ONLY A HYDRAULIC CART SERVICED WITH MIL-PRF-83282 HYDRAULIC FLUID. DO NOT SERVICE AIRCRAFT WITH CART THAT HAS PREVIOUSLY CONTAINED MIL-PRF-5606 HYDRAULIC FLUID.

A. Purging Hydraulics

NOTE: If hydraulic fluid contamination is suspected, perform this purge procedure.

NOTE: Refer to 29-00-00, Description and Operation Figure 2 for hydraulic system schematic diagram and to Figure 202, this section for disconnects.

NOTE: Refer to 20-10-00 Table 202 for all tube coupling nut torque values.

(1) Jack aircraft. (Refer to 07-10-00)

(2) Disconnect linkage at landing gear doors and secure doors open.

(3) Apply external electrical power to aircraft. (Refer to 24-40-00)

(4) Apply external hydraulic power to aircraft. (Refer to 12-10-20)

(5) Move flaps to an intermediate position.

(6) Retract landing gear sufficiently to unlock all landing gear and place a support under each landing gear to maintain gear unlocked position.

(7) Turn off external hydraulic power cart and reduce hydraulic pressure to zero.

(8) Select battery switches OFF.

(9) Drain hydraulic reservoir as follows:

(a) Remove hydraulic power pack access covers on left nacelle.

(b) Turn battery switches to ON and move flap handle up and down several times until hydraulic pressure reads zero.

(c) Bleed off air pressure on top of hydraulic reservoir by loosening filler cap on outboard side of left nacelle.

(d) Gain access to hand pump suction and reservoir drain line (5) at base of power pack on RH side between gear down and flap up lines. Disconnect reservoir drain line at base of power pack and drain reservoir.

(e) Reconnect reservoir drain line (5) at base of power pack and torque coupling nut.

(f) Fill hydraulic reservoir according to procedure in Servicing, this section. Do not disconnect GPU.

(10) Flush out LH landing gear actuators as follows:

(a) Emergency Gear Selector Valve to NORM GEAR.

(b) Disconnect LH landing gear hydraulic lines at actuators. Direct lines into a container to catch expelled hydraulic fluid.

(c) Connect an overboard drain to all LH gear actuator fittings.

(d) Lift each gear to near full up and near full down to drain actuators.

(11) Flush out LH pump pressure line as follows:

(a) LH hydraulic shutoff valve to OPEN.

(b) Landing gear control lever to DOWN.

(c) Hand motor LH propeller. Pump at least one pint of fluid through LH pump.

(12) Flush out landing gear down lines and up lines as follows:

(a) Select LH hydraulic shutoff valve to CLOSED.

(b) Position landing gear control lever to DOWN, apply external electrical and hydraulic power to aircraft, and flush down lines to outboard actuator.

(c) Move landing gear control lever to UP, and flush UP lines.

(13) Flush out LH emergency gear down lines as follows:

(a) Move Emergency Gear Selector Valve to EMER GEAR, and flush emergency down line to inboard actuator.

(b) Using emergency hand pump, assist flushing of emergency system if necessary.

(c) Reconnect all LH main gear hydraulic lines to actuators and torque coupling nuts.

(14) Flush out LH flap actuator lines as follows:

(a) Remove walking beam access panel located at lower fuselage-to-wing trailing edge and remove center bolt and bearing connecting both walking beams. (See Section 27-50-05 in this manual for removal procedure).

(b) Gain access to vertical mounting bracket (1) supporting flaps up and flaps down line fittings on LH wheel well wall, just aft of NS 173.3.

(c) Disconnect flaps down line (2) and flaps up line (3) where each exits bracket at wheel well wall. Attach tubing to disconnected lines and direct them into a container to catch expelled hydraulic fluid.

(d) Cycle flaps by hand until no more fluid comes out of disconnected lines.

(e) Attach drain lines to flaps up and flaps down line fittings at mounting bracket.

(f) Select flaps UP, and flush out flaps up line.

(g) Select flaps DOWN, and flush out flaps down line.

(h) Reconnect all disconnected lines and torque coupling nuts.

(15) Flush out nose landing gear hydraulic lines as follows:

(a) Emergency Gear Selector Valve to NORM GEAR.

(b) Disconnect nose landing gear hydraulic lines at actuators. Direct lines into a container to catch expelled hydraulic fluid.

(c) Connect an overboard drain to all nose gear actuator fittings.

(d) Move gear to near full-up and near full-down to drain nose gear actuators.

(e) Move landing gear control lever to DOWN, and flush down line to LH actuator.

(f) Move landing gear control lever to UP, and flush UP lines.

(16) Flush out emergency nose gear down lines as follows:

(a) Move Emergency Gear Selector Valve to EMER GEAR, and flush emergency down line to RH actuator.

(b) Using emergency hand pump, assist flushing of emergency system if necessary.

(c) Reconnect all nose gear hydraulic lines to actuators and torque coupling nuts.

(17) Flush nose wheel steering pressure and return lines as follows:

(a) Emergency Gear Selector Valve to NORM GEAR.

(b) Disconnect nose steering return line at nose steering actuator Tee fitting on return side of actuator.

(c) Attach a drain to disconnect at Tee fitting and direct nose steering line into a catch tank.

(d) Apply hydraulic pressure to flush out nose steering pressure and return lines.

(e) Reconnect nose steering return line at nose steering actuator Tee fitting on return side of actuator, and torque coupling nut.

(18) Flush out RH landing gear actuators as follows:

(a) Emergency Gear Selector Valve to NORM GEAR.

(b) Disconnect RH landing gear hydraulic lines at actuators. Direct lines into a container to catch expelled hydraulic fluid.

(c) Connect an overboard drain to all RH gear actuator fittings.

(d) Lift each gear to near full-up and near full-down to drain actuators.

(19) Flush out RH pump pressure line as follows:

(a) RH hydraulic shutoff valve to OPEN.

(b) Landing gear control lever to DOWN.

(c) Hand motor RH propeller. Pump at least one pint of fluid through landing gear down line.

(20) Flush out RH landing gear down-line and up-lines as follows:

(a) RH hydraulic shutoff valve to CLOSED.

(b) Landing gear control lever to DOWN, and flush down line to outboard actuator.

(c) Move landing gear control lever to UP, and flush UP lines.

(21) Flush out RH emergency gear down lines as follows:

(a) Move Emergency Gear Selector Valve to EMER GEAR, and flush emergency down line to inboard actuator.

(b) Using emergency hand pump, assist flushing of emergency system if necessary.

(c) Reconnect all RH main gear hydraulic lines to actuators and torque coupling nuts.

(22) Flush out RH flap actuator lines as follows:

(a) Gain access to flaps-up and flaps-down lines on RH wheel well wall, just aft of NS 173.3.

(b) Remove flaps-up line and flaps-down line where each exits wheel well wall. Attach tubing to disconnected lines, and direct them into a catch tank.

(c) Cycle flaps by hand until no more fluid comes out of removed lines.

(d) Attach drain lines to flaps-up and flaps-down disconnects at mounting bracket.

(e) Select flaps UP, and flush out flaps UP line.

(f) Select flaps DOWN, and flush out flaps DOWN line.

(g) Connect both walking beams together with bearing and bolt. (see Section 27-50-05 in this manual for removal procedure).

(h) Replace walking beam access panel.

(i) Reconnect all disconnected lines and torque coupling nuts.

(23) Flush out pump bypass lines as follows:

(a) Disconnect Tee fitting (4) at base of filter and attach drain line to a catch tank.

(b) LH and RH hydraulic shutoff valves to OPEN.

(c) Hand motor each propeller and catch fluid from return lines at filter. Pump at least one pint of fluid through each pump. DO not reconnect return lines at filter.

(24) Flush out flap return lines as follows:

(a) LH and RH hydraulic shutoff valves to CLOSED.

(b) Disconnect flap return line (5) at LH flap lock valve. Attach hydraulic pressure source at flap lock valve, and flush out LH return line.

(c) Reconnect LH flap return line at flap lock valve and torque coupling nut.

(d) Disconnect flap return line at RH flap lock valve. Attach hydraulic pressure source at flap lock valve, and flush out RH return line.

(e) Reconnect RH flap return line at flap lock valve and torque coupling nut.

(f) Reconnect return line at filter and torque coupling nut.

(25) Change hydraulic system filter. For procedures, refer to maintenance practices in this section for procedures.

(26) Clean nose steering actuator filters and replace nose steering system filter (if installed). Follow procedures in Section 32-52-35 in this manual.

(27) Connect external hydraulic unit. Service hydraulic reservoir with MIL-PRF-83282 fluid.

CAUTION: DRAIN HYDRAULIC FLUID INTO A CATCH TANK. DO NOT RETURN IT TO HYDRAULIC CART.

(28) Actuate all portions of system and bleed as required. Operate flaps and gear through several complete cycles. Check operation of landing gear free-fall system.

(29) Open hydraulic shutoff valves. Motor engines. Check for pressure indication.

(30) When both pumps are primed and gear and flap retraction and extension are satisfactory, wash down reservoir area to remove fluid residue, using varsol or equivalent. Blow reservoir area dry using compressed air.

(31) Disconnect external hydraulic power cart. (Refer to 12-10-20)

(32) Disconnect external electrical power from aircraft. (Refer to 24-40-00)

(33) Down jack aircraft. (Refer to 07-10-00

(34) Connect linkage at landing gear doors.

(35) Operate engines and check system for leaks.

(36) Install all removed panels.

4. Approved Repairs – Hydraulics

A. Temporary Repair of Hydraulic Lines

NOTE: When a leak is discovered during out-station operation and where maintenance is unavailable, temporary repairs may be accomplished. Comply with the following standards when making temporary repairs.

M7 Aerospace approves the use of flexible hose (not to exceed 150 hours), in place of metal tubing for temporary repairs.

(1) Flexible hose assemblies may be manufactured by the operator or purchased from an approved flexible hose supplier, however they must be certified IAW Chapter 20, this manual.

(2) Install and support IAW FAA Publication AC43.13-1B (Aircraft Inspection and Repair).

(3) Support all flexible hoses using proper size and type clamps.

(4) To ensure adequate fatigue life of temporary hoses, maintain the “as delivered” tube support as closely as possible. Add an additional clamp at each end of tubes to which hose is to be attached.

(5) Temporary hoses must be replaced within 150 hours operating time.

(6) Upon installation of flexible hoses, perform functional test and proof pressure test of system.

(7) If two (2) or more landing gear hydraulic lines are affected, perform landing gear extension and retraction check IAW Section 32-00-00. Cycle landing gear five (5) times minimum. Perform one (1) emergency landing gear extension.

(8) If only one (1) landing gear hydraulic line is affected (repaired or replaced), perform taxi check, making at least two (2) 90° turns left and two (2) 90° turns right

(9) Verify fluid level and service if required.

B. Permanent Repair of Hydraulic Lines

(1) In event of hydraulic tube failure, an airworthy repair may be accomplished by using one of the following procedures.

(a) Replace damaged tube assembly with identical part number assembly. Locally fabricated assembly may be used, if fabricated from proper type tubing material and end fittings are used.

(b) Remove damaged section from the tube assembly and replace with a splice section or a union type fitting.

1 It may be possible to remove the damage in a section short enough that the ends of the assembly may be rejoined by the insertion of a union fitting.

2 Fabricate a splice section of tubing to fit the removal section of damage, using steel tubing for the splice section, if a union fitting cannot be used to joint the tube ends in the airframe.

C. Hydraulic Tubing Allowable Damage Limitations

(1) Systems with working pressure above 500 psi (35.15 kg/cm^2);

(a) There shall be no wrinkle or kinks deeper than 1% of tube outside diameter.

(b) There shall be no scratch, chafe, or gouge deeper than 5% of nominal tube wall thickness.

(2) Systems with working pressure below 500 psi (35.15 kg/cm^2);

(a) There shall be no wrinkle or kinks deeper than 2% of tube outside diameter.

(b) There shall be no scratch, chafe, or gouge deeper than 10% of nominal tube wall thickness.

(3) Tube Flatness;

NOTE: Tubing shall meet the flatness requirements specified in Table 201 formula.

Flatness = Max. O.D. Minus Min. O.D. Divided By Nominal O.D. X 100%

Flatness Formula
Table 204

(a) Systems with working pressure above 1,000 psi (70.30 kg/cm^2);

1 Flatness shall not exceed 5%.

(b) Systems with working pressure below 1,000 psi (70.30 kg/cm^2);

1 Flatness shall not exceed 10%.

D. Repair of Hydraulic Containment Shrouds

NOTE: Moisture may be present in containment shrouds from mechanical failure. Any condition allowing leakage must be corrected as soon as possible by repair or replacement of mechanical parts.

Damage to shrouds in most conditions is repairable as long as there is sufficient material outside of damaged area for specified overlap.

(1) Cut away damaged area of shroud. Leave sufficient material outside damaged area for overlap.

(2) Clean surface of damaged area to be cemented by washing thoroughly with Brulin Super 816 detergent or equivalent IAW AMS 1526, 1527, or 1550 mixed with hot water, using a nylon scrub brush.

(3) Use clear water and shop towels to dry surface area thoroughly. Remove all detergent residue and loose particles.

E. Repair of Hydraulic Containment Shrouds

NOTE: Moisture may be present in containment shrouds from mechanical failure. Any condition allowing leakage must be corrected as soon as possible by repair or replacement of mechanical parts.

Damage to shrouds in most conditions is repairable as long as there is sufficient material outside of damaged area for specified overlap.

(1) Cut away damaged area of shroud. Leave sufficient material outside damaged area for overlap.

(2) Clean surface of damaged area to be cemented by washing thoroughly with Brulin Super 816 detergent or equivalent IAW AMS 1526, 1527, or 1550 mixed with hot water, using a nylon scrub brush.

(3) Use clear water and shop towels to dry surface area thoroughly. Remove all detergent residue and loose particles.

(4) Lightly abrade or roughen around damaged surface area to be cemented.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(5) Wipe surface thoroughly with lint free cloth and Methyl Propyl Ketone (MPK) solvent, Fed Spec. TT-M-261B. Use clean cloths and rotate cloth often. Repeat at least once, or until cloth remains clean after wiping.

(6) Cut a repair patch of nylon reinforced Buna-n BN5049, (B.F. Goodrich), to approximately the same shape as the repair area. The patch should overlap the damaged area by 2.0 inches (5.08 cm) (see Figure 203).

(7) Apply a thin even layer of Scotchgrip No. 847 or 826 (3M Co.) adhesive to both surfaces to be bonded, and join the adhesive coated surfaces together.

(8) Apply light pressure to surfaces and allow cure time of twenty four (24) hours to obtain good bond strength.

NOTE: For maximum repair strength, stitch around edges of repair area using nylon thread Style 86055, (Conso). Sew approximately 0.125 inch (3.18 mm) per stitch and edge distance of approximately half of the overlap.

(9) Apply an extra layer of adhesive over repair area seams to ensure proper sealing.

NOTE: 3M 847 glue or PR-1422 sealant may be used to hold zipper flap down and keep zipper closed.

F. Replacement of Hydraulic Containment Zipper.

NOTE: No repairs can be made to the containment shroud zipper. If the zipper is defective it must be replaced.

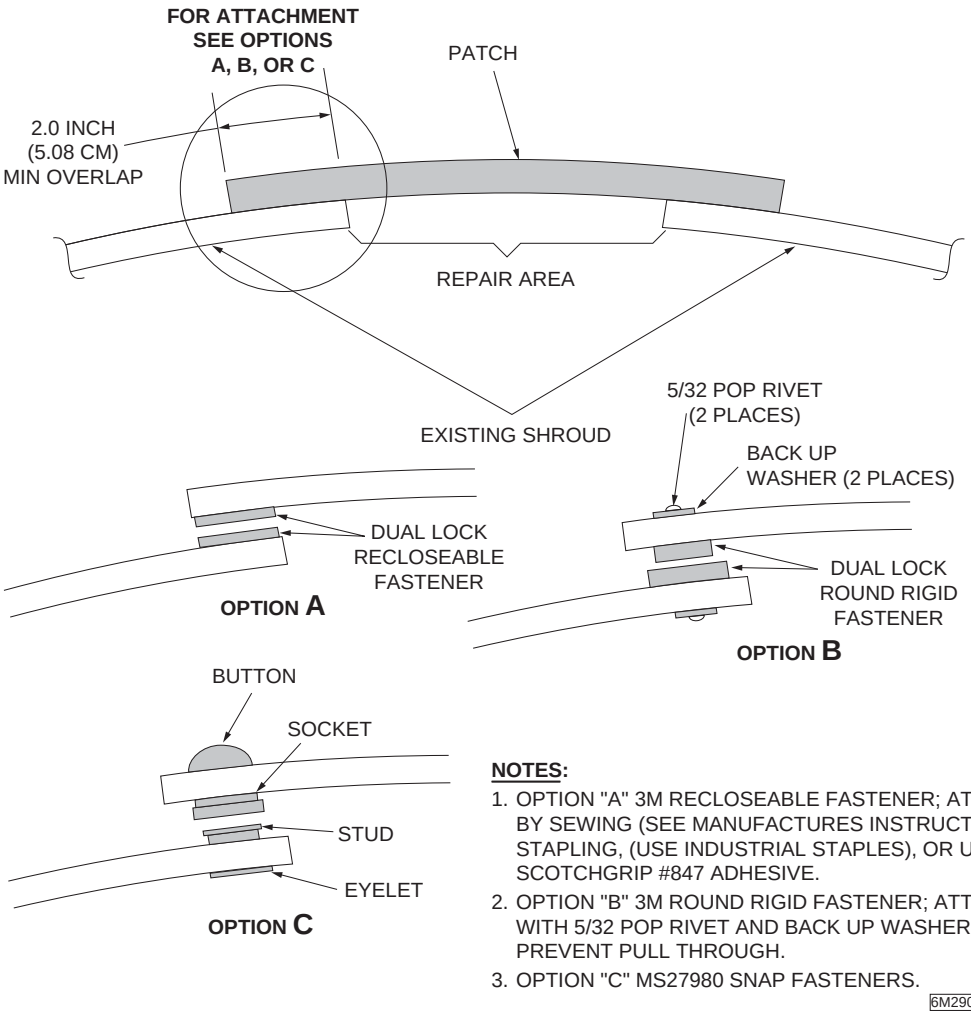
(1) Remove old zipper and follow steps **4.**, E., (1) through (9), Repair of Hydraulic Containment Shrouds, to replace zipper.

NOTE: 3M 847 glue or PR-1422 sealant may be used to hold zipper flap down and keep zipper closed.

G. Alternate Repair In Place of Hydraulic Containment Zipper Replacement.

(1) To prevent replacing defective zippers fasten patch to original shroud using one or more of the methods as shown in Figure 203.

NOTE: For adhesive method comply with cleaning procedures in **4.**, E.



NOTES:

1. OPTION "A" 3M RECLOSEABLE FASTENER; ATTACH BY SEWING (SEE MANUFACTURES INSTRUCTIONS), STAPLING, (USE INDUSTRIAL STAPLES), OR USING SCOTCHGRIP #847 ADHESIVE.
2. OPTION "B" 3M ROUND RIGID FASTENER; ATTACH WITH 5/32 POP RIVET AND BACK UP WASHER TO PREVENT PULL THROUGH.
3. OPTION "C" MS27980 SNAP FASTENERS.

6M290000203

Alternate Repair to Prevent Hydraulic Containment Zipper Replacement
Figure 203

MAINTENANCE MANUAL

MAIN HYDRAULICS – DESCRIPTION AND OPERATION

1. General

A. The main hydraulic system consists of two engine driven pumps, a shutoff valve for each pump, and associated lines and check valves. Later aircraft and those retrofitted by Service Bulletin 29-002 are equipped with a hydraulic accumulator.

B. The hydraulic power pack reservoir supplies fluid for use by engine driven pumps.

(1) Two standpipes within the power pack ensure that a minimum of one U. S. quart (0.9463 liter) fluid will be left in the power pack for hand pump operation.

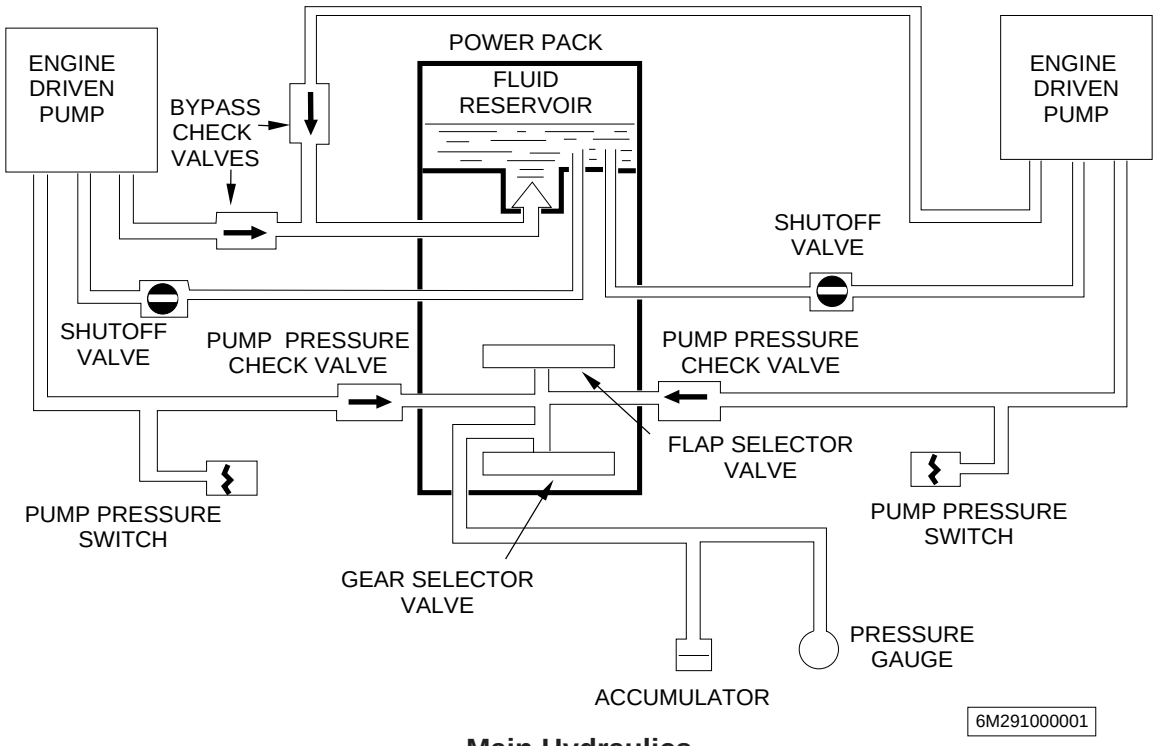
(2) An electrically powered shutoff valve is located just aft of the firewall in each engine driven pump supply line. The valve may be used to stop the flow of fluid to the respective engine driven pump.

C. The engine driven pump is installed on each engine between the starter-generator and fuel control.

(1) The pump output pressure is routed into the power pack where two check valves control the fluid pressure. A pressure switch, installed in the output of each pump, controls the low pressure warning light in the annunciator panel.

(2) The third line exiting each pump is a bypass line. A small amount of fluid flows through the pump, mainly for cooling purposes, when the pump is not supplying fluid for either landing gear or wing flap operation. The bypass fluid is returned to the reservoir and passes through the system filter.

D. Fluid pressure is contained in a manifold at the bottom of the power pack. Each pump output is connected through a check valve to this manifold. A separate tap provides pressure to the hydraulic gauge. On aircraft so equipped, the hydraulic accumulator is teed into the gauge line.



Main Hydraulics
Figure 1

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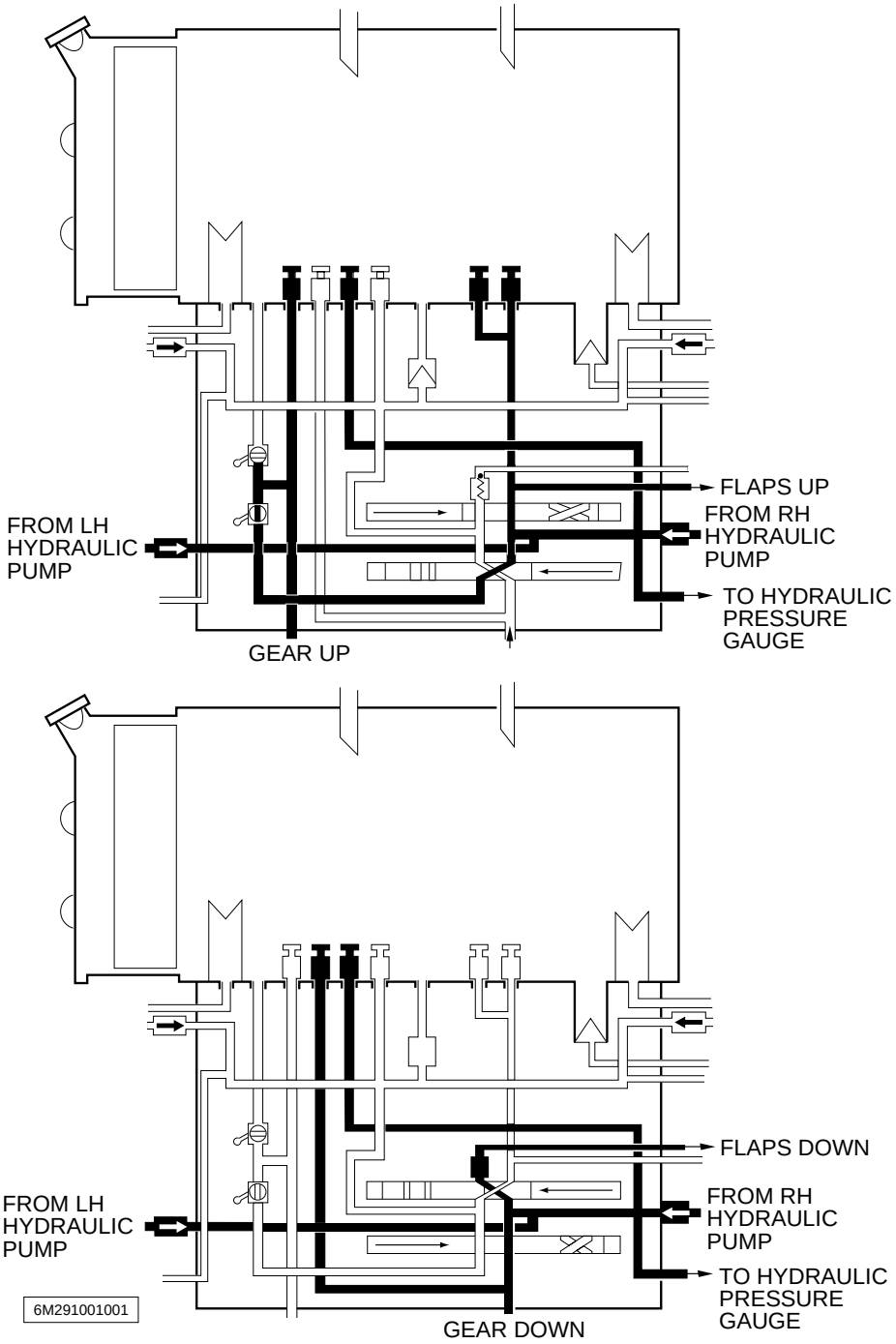
HYDRAULIC POWER PACK – DESCRIPTION AND OPERATION

1. General

A. The hydraulic power pack, located in the left nacelle, is used to store and distribute hydraulic fluid for use in the system. The power pack is installed just aft of the left engine firewall, in forward end of left wheel well.



Hydraulic Power Pack
Figure 1

(1) Connections to the hydraulic power pack include engine driven pump pressure; landing gear and flaps lines; a manual release cable controlled from the cockpit; electrical connections to actuate the landing gear and flap solenoids and an air pressure connection to pressurize the reservoir.

(2) The hydraulic power pack is supported on two beams located on FWD and aft sides.

B. The hydraulic power pack has 2.5 U.S. quarts, (2.3658 liters) hydraulic fluid stored in the upper cylindrical reservoir. Air pressure, supplied by a regulator in the right wing root, is supplied to the reservoir to ensure an adequate supply to the pumps in all flight attitudes.

(1) A pressurization relief valve, set at 43 psi (3.02 kgm/cm^2), protects the reservoir should the air regulator malfunction.

(2) Standpipes in the reservoir restrict engine driven pumps to 1.5 U.S. quarts (1.4195 liters) operating fluid. Should a leak in the system occur, the remaining one U.S. quart (0.9463 liter) of the fluid is reserved for hand pump operation.

C. The lower portion of the hydraulic power pack contains landing gear and flap selector valves, manual hydraulic pressure dump valve, relief and check valves for each system and system filter.

(1) Landing gear and flap selector valves are electrically controlled from the cockpit.

(2) The landing gear bypass valve is actuated by a release handle in the cockpit that also releases the three landing gear uplocks.

HYDRAULIC POWER PACK – FAULT ISOLATION

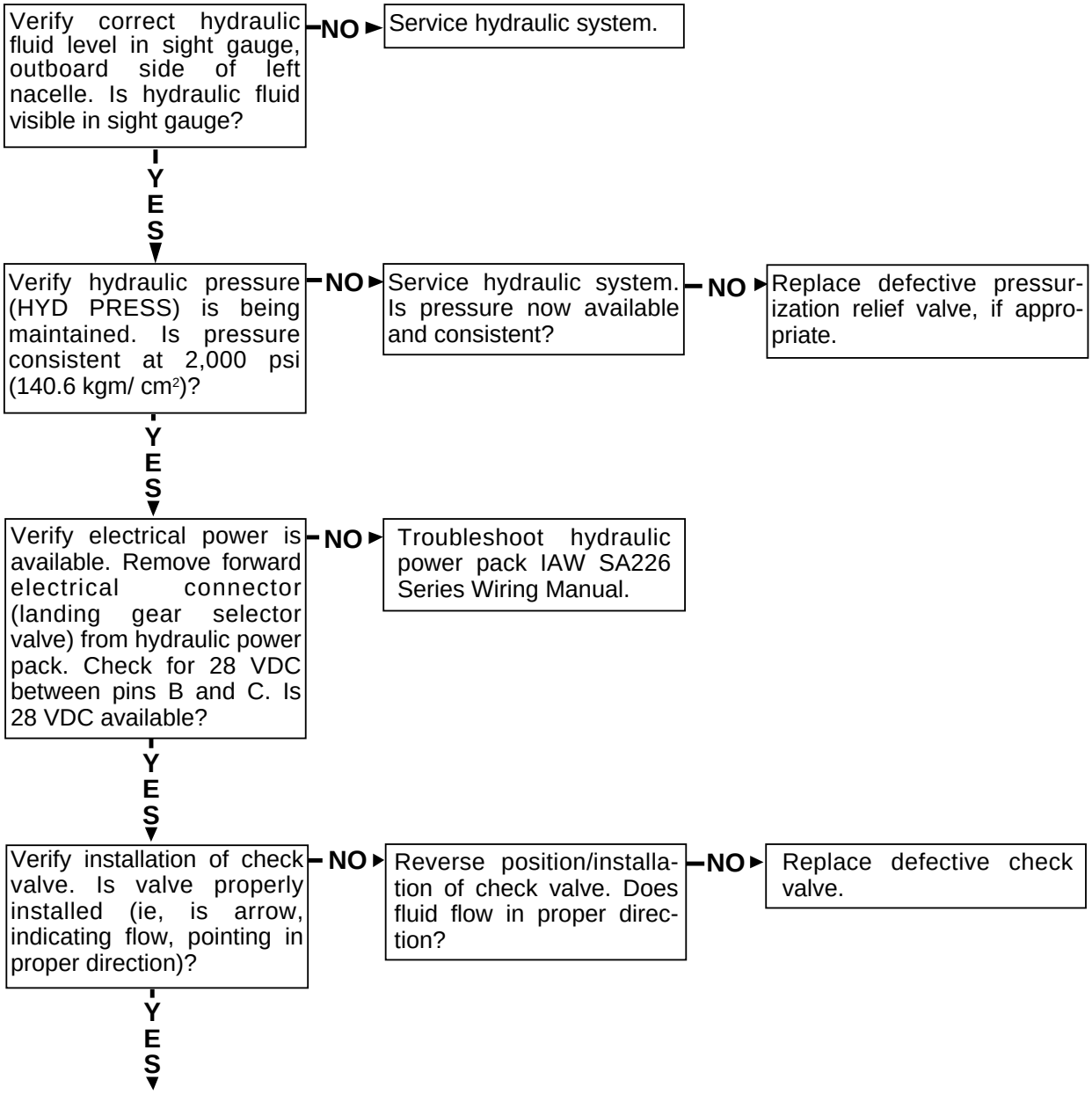
PROBLEM:

LANDING GEAR WILL NOT EXTEND DURING NORMAL OPERATION.

PREREQUISITES:

Emergency gear selector NORM, landing gear control handle DOWN, electrical power applied, hydraulic power pack head pressure at 31 + 1 psi (2.18 + 0.0703 kg/cm^2) , and aircraft on jacks.

START:



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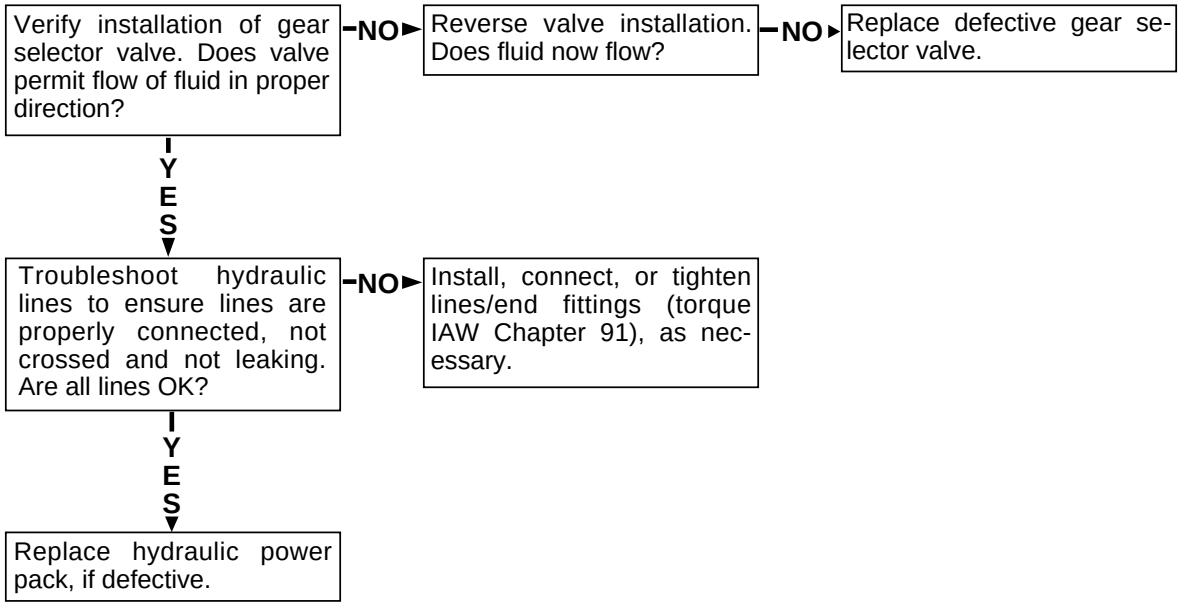

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END

HYDRAULIC POWER PACK – FAULT ISOLATION

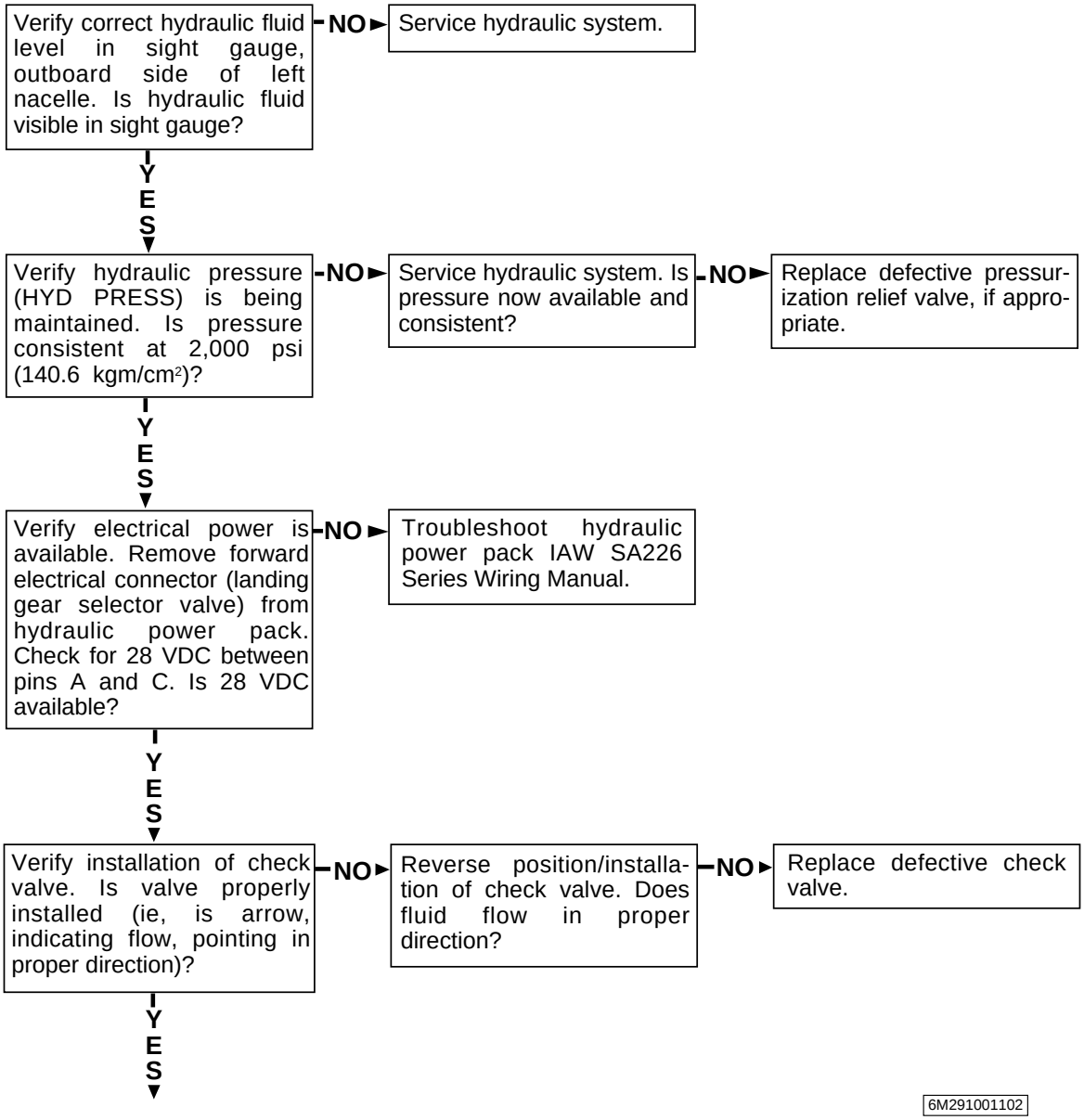
PROBLEM:

LANDING GEAR WILL NOT RETRACT DURING NORMAL OPERATION.

START:

PREREQUISITES:

Emergency gear selector NORM, landing gear control handle UP, electrical power applied, hydraulic power pack head pressure at 31 + 1 psi (2.18 + 0.0703 kg/cm^2), and aircraft on jacks.



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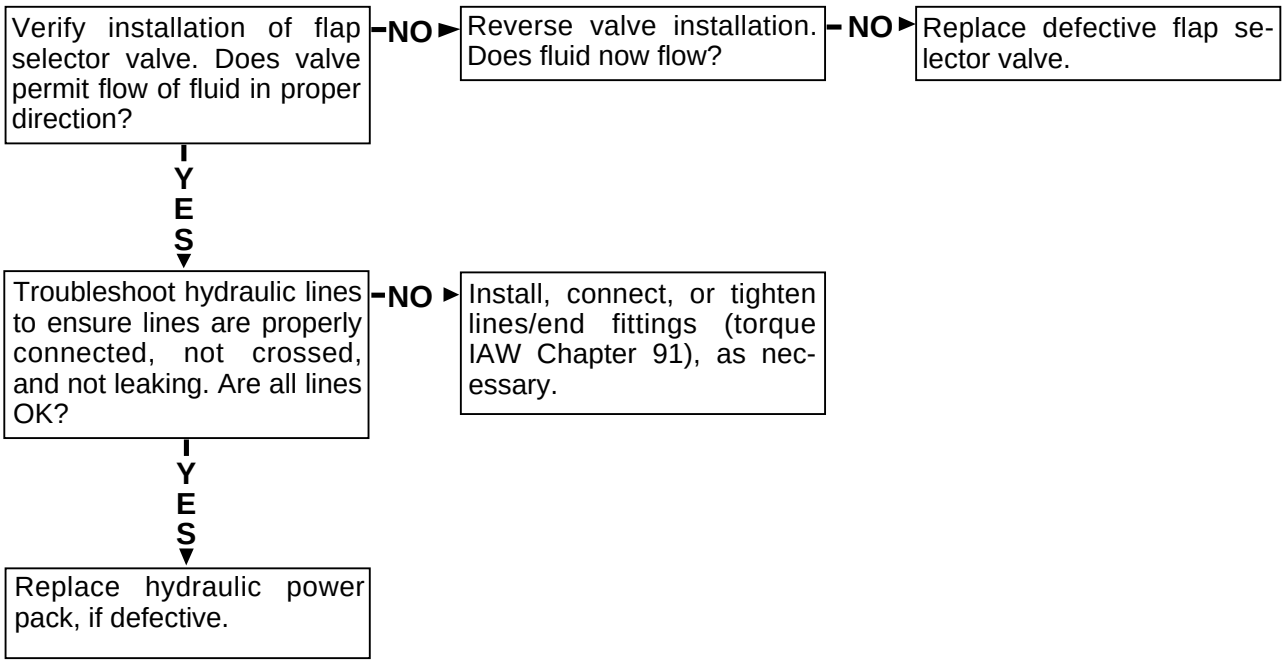

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HYDRAULIC POWER PACK – FAULT ISOLATION

PROBLEM:

LANDING GEAR WILL NOT EXTEND DURING NORMAL OPERATION.

PREREQUISITES:

Flap switch in UP position, electrical power ON, and hydraulic power pack head pressure at 31 + 1 psi (2.18 + 0.0703 kg/cm^2)

START:

Verify correct hydraulic fluid level in sight gauge, outboard side of left nacelle. Is hydraulic fluid visible in sight gauge?

NO ► Service hydraulic system.

YES ▼

Verify hydraulic pressure (HYD PRESS) is being maintained. Is pressure consistent at 2,000 psi (140.6 kgm/ cm^2)?

NO ► Service hydraulic system. Is pressure now available and consistent?

NO ► Replace defective pressurization relief valve, if appropriate.

YES ▼

Verify electrical power is available. Remove forward electrical connector (landing gear selector valve) from hydraulic power pack. Check for 28 VDC at pin A. Is 28 VDC available?

NO ► Troubleshoot hydraulic power pack IAW SA226 Series Wiring Manual.

YES ▼

Verify installation of check valve. Is valve properly installed (ie, is arrow, indicating flow, pointing in proper direction)?

NO ► Reverse position/installation of check valve. Does fluid flow in proper direction?

NO ► Replace defective check valve.

YES ▼

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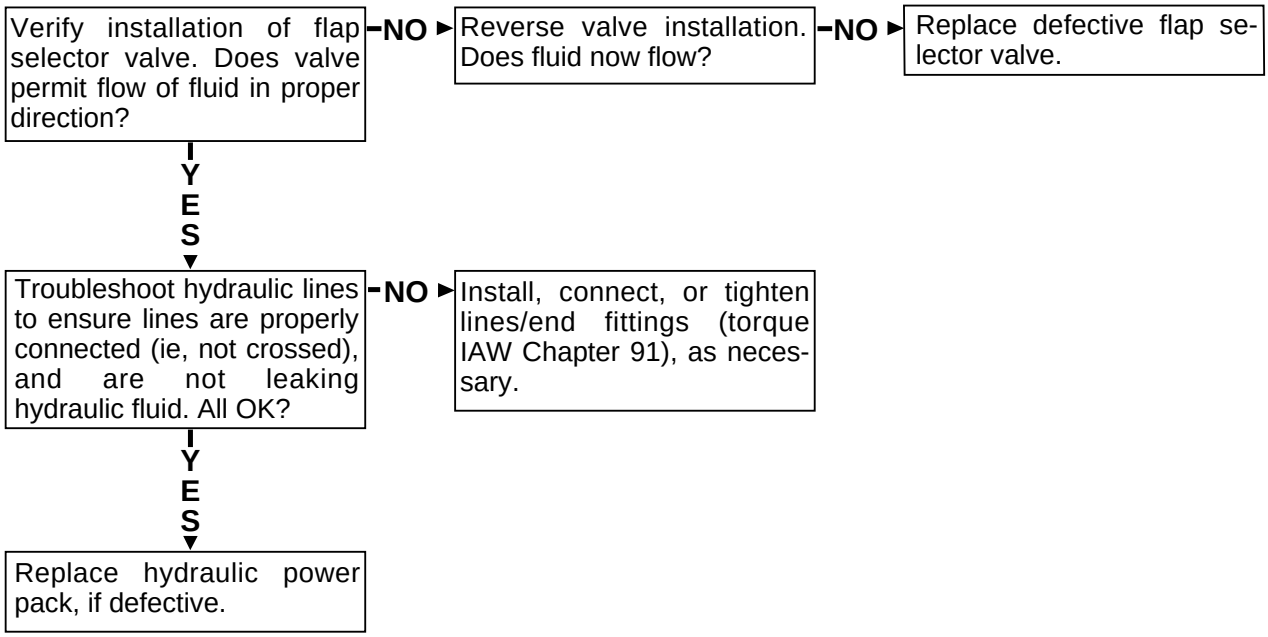

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END

HYDRAULIC POWER PACK – FAULT ISOLATION

PROBLEM:

FLAPS WILL NOT RETRACT DURING NORMAL OPERATION.

PREREQUISITES:

Flap switch in UP position, electrical power applied, and hydraulic power pack head pressure at 31 + 1 psi (2.18 + 0.0703 kg/cm^2).

START:

Verify correct hydraulic fluid level in sight gauge, outboard side of left nacelle. Is hydraulic fluid visible in sight gauge?

- NO ► Service hydraulic system.
- YES ▼

Verify hydraulic pressure (HYD PRESS) is being main-tained. Is pressure con-sistent at 2,000 psi (140.6 kgm/cm^2)?

- NO ► Service hydraulic system. Is pressure available and consistent?
  - NO ► Replace defective pressurization relief valve.
- YES ▼

Verify electrical power is available. Remove aft electrical connector (flap selector valve) from hydraulic power pack. Check for 28 VDC at pin B. Is 28 VDC available?

- NO ► Troubleshoot hydraulic power pack IAW SA226 Series Wiring Manual.
- YES ▼

Verify installation of check valve. Is valve properly installed (ie, is arrow, indicating flow, pointing in proper direction)?

- NO ► Reverse position/installation of check valve. Does fluid flow in proper direction?
  - NO ► Replace defective check valve.
- YES ▼

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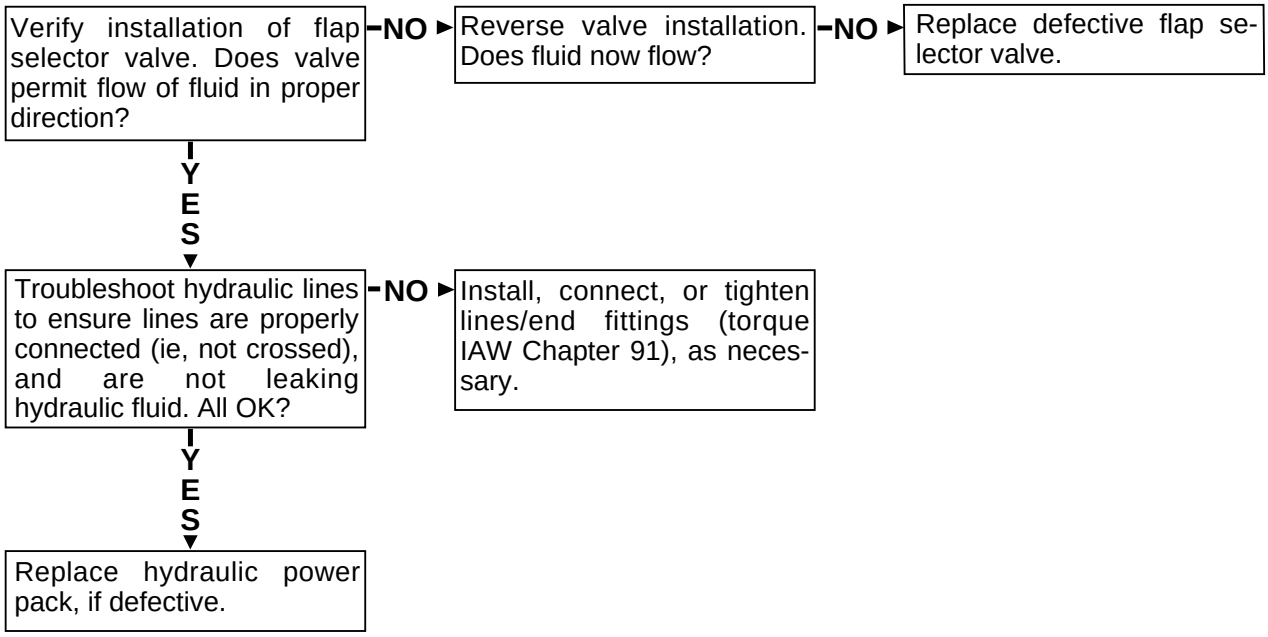

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END

HYDRAULIC POWER PACK – MAINTENANCE PRACTICES

1. Removal/Installation – Hydraulic Power Pack

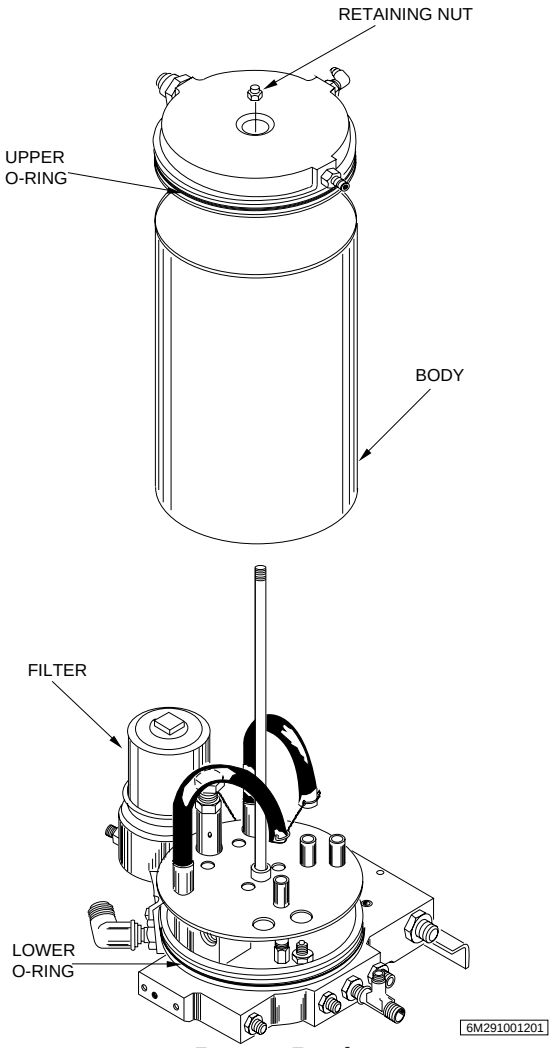
WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

NOTE: The following maintenance practices may be performed with the power pack installed in the aircraft.

A. Removal – Reservoir Body O-Ring

WARNING: HYDRAULIC SYSTEM IS UNDER PRESSURE. USE CAUTION WHEN LOOSENING OR REMOVING FILLER CAP.

(1) Relieve air pressure in hydraulic power pack by loosening filler cap on outboard side of left nacelle.



Power Pack
Figure 201

(2) Drain fluid from hydraulic power pack.

(3) Remove lines connected to top of reservoir.

(4) Remove reservoir retaining nut.

(5) If only upper O-ring is being replaced, support reservoir body and remove top.

(6) Remove upper O-ring.

(7) If lower O-ring requires replacement, pull reservoir body up to expose O-ring.

(8) Remove lower O-ring.

B. Installation – Reservoir Body O-Rings (Figure 201)

(1) Coat new O-ring with hydraulic fluid. Drop O-ring through body.

NOTE: Never reuse old O-rings as they may be swollen from exposure to fluids and become set from being under pressure, or have minor cuts or abrasions not easily seen by visual inspection.

(2) Support body up and away from base. Carefully install O-ring around base, seating O-ring into lower groove.

(3) Carefully slide body down over base and O-ring.

(4) Coat upper O-ring with hydraulic fluid and install reservoir top.

(5) Tighten reservoir retaining nut.

(6) Service hydraulics IAW Chapter 12.

C. Test O-Ring After Replacement (Figure 201) as follows:

NOTE: Failure of either upper or lower body O-ring is rare and may be indicative of problems with air pressure source to hydraulic power pack or hydraulic power pack pressurization relief valve.

(1) Remove hydraulic power pack pressurization relief valve.

(2) Using an air pressure regulator and gauge, slowly apply air pressure to valve in direction indicated by arrow on valve. Valve should open at 43 + 1 psi (3.02 + 0.0703 kg/cm^2.) Replace any valve found defective.

(3) Connect pressure gauge in place of relief valve. The gauge should be in the line coming from the hydraulic power pack. Operate right engine at 96% and observe that gauge reads 31 + 1 psi (2.18 + 0.0703 kg/cm^2.)

(4) Install relief valve and operate both engines.

(5) Check hydraulic power pack area for leaks.

(6) Install all access panels and covers.

D. Removal – Filter (Figure 202)

WARNING: HYDRAULIC SYSTEM IS UNDER PRESSURE. USE CAUTION WHEN LOOSENING OR REMOVING FILLER CAP.

(1) Relieve hydraulic pressure by loosening filler cap on outboard side of left nacelle.

(2) Remove safety wire attaching filter bowl to filter base.

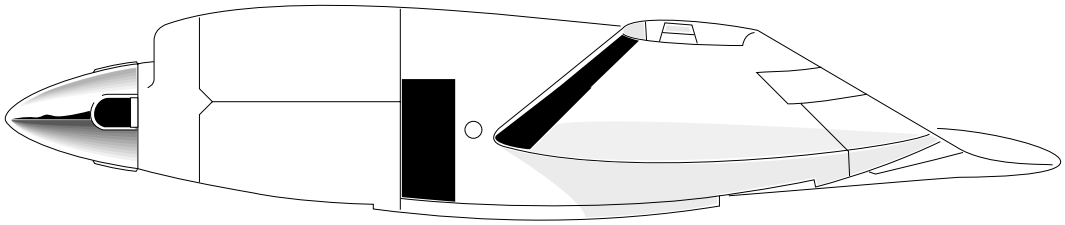
E. Installation – Filter

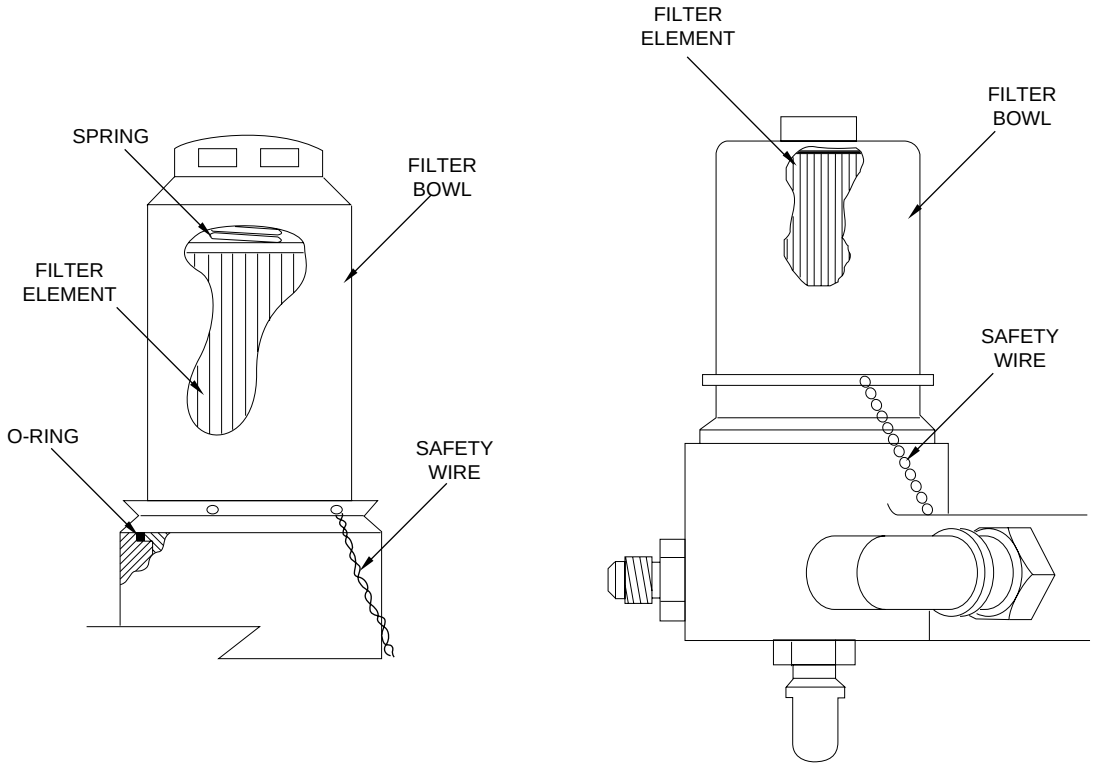
CAUTION: DO NOT INSTALL AN 188303-1 SPRING ABOVE THE FILTER ELEMENT IN THE FLAT-TOPPED AS18828-2 FILTER BOWL ASSEMBLY. IT WILL CRUSH THE FILTER ELEMENT. USE THE SPRING ONLY WITH THE AS18828-1 DOME FILTER BOWL ASSEMBLY.

(1) Reinstall filter and housing, torque housing finger tight plus 1/4 turn.

(2) Safety housing assembly.

(3) Apply hydraulic pressure to the system and check for leaks.





Hydraulic Fluid Filter Assembly
Figure 202

E. Removal – Pressure Manifold O-Ring

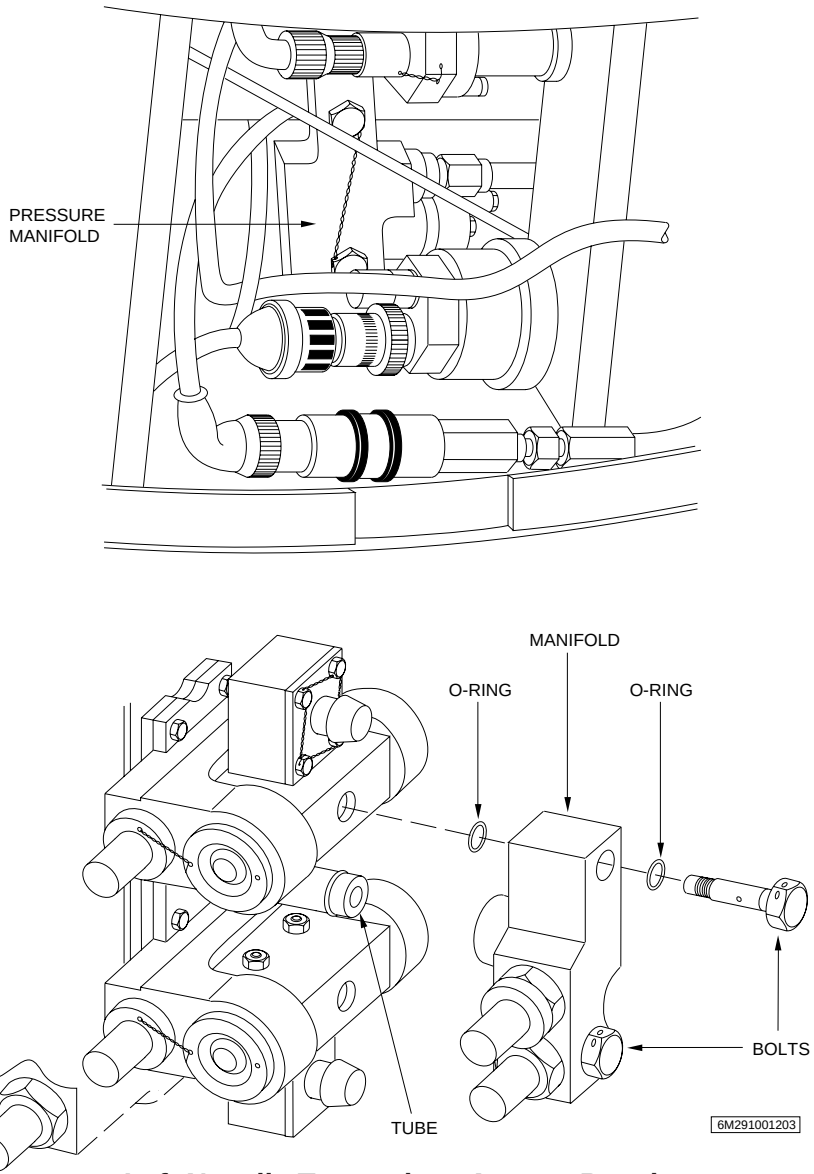
NOTE: External leakage around the bottom of the pack may be caused by leakage past the pressure manifold O-rings. These O-rings may be replaced in the field.

(1) Remove transmitter access panel on left hand nacelle.

(2) Remove pump pressure and gauge lines from bottom of pack.

(3) Locate the two large bolt heads on lower portion of pack (see Figure 204). Cut safety wire and unscrew bolts.

(4) Pull manifold down and remove.



Left Nacelle Transmitter Access Panel and Pressure Manifold
Figure 203

F. Installation – Pressure Manifold O-Ring

(1) Replace O-rings on the tube. Replace O-rings on two bolt fittings. MS28775-111 is installed under the head, and MS28778-4 around the threads.

(2) Reinstall manifold and torque bolts 70 inch-pounds (7.91 N·m). Reconnect pump pressure and gauge lines.

(3) Move aircraft to a run up area and operate both engines. Check power pack for leaks.

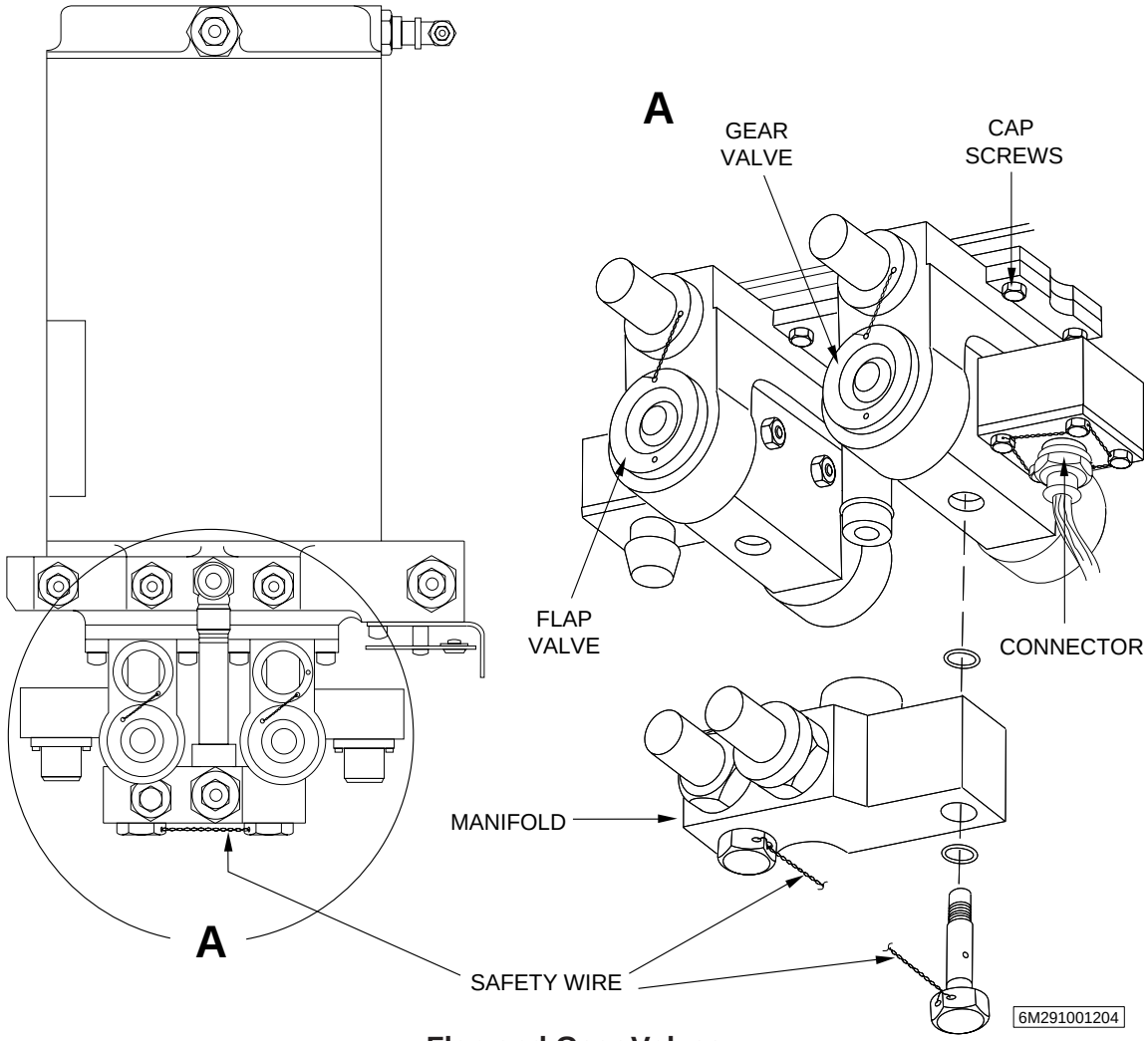
(4) Reinstall transmitter access panel and power pack cover.

G. Removal – Flap or Gear Valve

(1) Remove transmitter access panel from bottom of left hand nacelle.

(2) Remove pump pressure and gauge lines from bottom of power pack.

(3) Remove pressure manifold.



Flap and Gear Valves
Figure 204

(4) Remove fluid transfer tube. Cap open ports to prevent contamination.

(5) Disconnect flap or gear valve electrical connector.

(6) Cut safety wire and remove four (4) valve retaining capscrews and washers.

(7) Lower flap or gear valve and remove six (6) O-rings. Remove valve.

H. Installation – Flap or Gear Valve

(1) Position replacement flap or gear valve onto bottom of hydraulic power pack. Install O-rings and valve with solenoid connector forward (gear valve) or aft (flap valve).

(2) Secure valve to hydraulic power pack housing with capscrews and washers. Torque capscrews 15 to 20 inch-pounds (0.56 to 2.26 N·m) and safety.

(3) Connect flap or landing gear valve electrical connector.

(4) Remove dust covers from ports; position fluid transfer tube and install pressure manifold.

(5) install pump pressure line. Torque end fitting 75 to 125 inch-pounds (8.47 to 14.12 N·m).

(6) Install pressure gauge line. Torque end fitting 40 to 65 inch-pounds (4.52 to 7.34 N·m).

(7) Install transmitter access panel.

(8) Operate landing gear and/or flaps a minimum of six (6) times.

(9) Perform hydraulic power pack integrity check as follows:

(a) With landing gear down (gear selector handle down) and locked and flaps up, normal hydraulic pressure applied, select electrical power off. There should be no movement in either landing gears or flaps.

NOTE: This check must be performed following any maintenance or replacement of landing gear valve or hydraulic power pack.

I. Removal – Power Pack

(1) Jack aircraft.

(2) Remove power pack access covers. Close both hydraulic shutoff valves.

(3) Connect hydraulic cart and drain system.

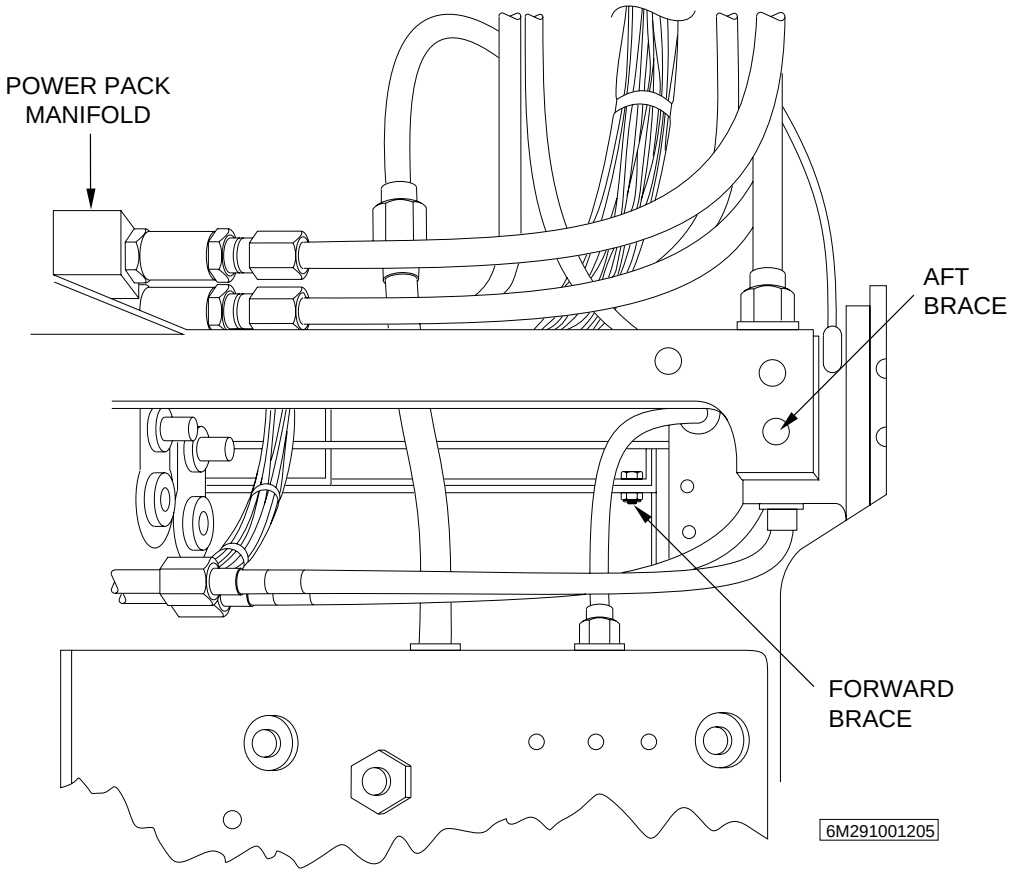
(4) Remove bolts securing aft brace to pack and remove brace.

(5) Disconnect hydraulic lines to pack.

(6) Remove bolts securing forward brace to aircraft.

(7) Lift and tilt pack aft into wheel well. Disconnect electrical connectors. Remove clevis pin securing emergency release cable to release arm. Remove pack.

(8) Cap or plug all open lines. Remove forward brace from pack.



Power Pack
Figure 205

J. Installation – Power Pack

(1) Remove shipping protectors and dust covers from new hydraulic power pack.

(2) Inspect new hydraulic power pack for evidence of damage. Actuate emergency release arm and check for free movement.

(3) Install forward brace to new hydraulic power pack.

(4) Remove dust covers from hydraulic lines.

(5) Position hydraulic power pack in place.

(6) Connect emergency release cable to release arm.

(7) Connect electrical connectors to hydraulic power pack and safety.

(8) Align hydraulic power pack into proper position. Secure forward brace.

(9) Install hydraulic lines to hydraulic power pack. Torque IAW Chapter 91.

(10) Install aft brace to hydraulic power pack.

(11) Perform Adjustment Test – Power Pack.

2. Adjustment/Test – Hydraulic Power Pack

A. Functional Test – Hydraulic Power Pack

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

CAUTION: USE ONLY A HYDRAULIC CART SERVICED WITH MIL-H-83282 HYDRAULIC FLUID. DO NOT SERVICE AIRCRAFT HYDRAULIC SYSTEM WITH CART THAT CONTAINS MIL-H-5606 HYDRAULIC FLUID.

(1) Fill system using hydraulic cart and operate the gear and flaps a minimum of six (6) times.

(a) If hydraulic power pack does not shut off properly, remove the round access panels on right side of wheel well and adjust turnbuckle in hydraulic power pack shutoff cable.

(b) Shortening the cable will cause hydraulic power pack to shut off earlier.

NOTE: When free falling landing gear, hydraulic fluid may be discharged through the pressurization relief vent on the inboard side of nacelle. This discharge indicates that the reservoir was slightly overfilled. To prevent this occurrence, ensure that hydraulic fluid level is no higher than FULL mark on sight unit.

(2) Perform hydraulic power pack integrity check as follows:

(a) With landing gear down (gear selector handle down) and locked and flaps up, normal hydraulic pressure applied, select electrical power off. There should be no movement in either landing gears or flaps. Check operation of landing gear free fall system.

WARNING: ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF UNNECESSARY PERSONNEL AND OBJECTS DURING HAND MOTORING ENGINES.

(3) Open both hydraulic shutoff valves. With hydraulic cart off, rotate each propeller by hand and check for pressure indication (L HYD PRESS or R HYD PRESS).

WARNING: HYDRAULIC SYSTEM IS UNDER PRESSURE. USE CAUTION WHEN LOOSENING OR DISCONNECTING HYDRAULIC LINES.

(a) Should hand propping engines fail to build pressure, it may be necessary to:

1 loosen respective pump pressure line at bottom of hydraulic power pack.

WARNING: ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF UNNECESSARY PERSONNEL AND OBJECTS DURING HAND MOTORING ENGINES.

2 Should hand propping engines still fail to produce pressure, it may be necessary to employ STARTER TEST SWITCH (not to exceed 30 seconds).

(b) If necessary, hydraulic power pack pressurization check valve may be removed and 30 psi (2.11 kg/cm^2) air pressure may be added to assist in priming the pumps.

(4) When both pumps are primed and landing gear and flap retraction and extension are satisfactory, wash down hydraulic power pack area (using varsol or equivalent) to remove fluid residue. Using compressed air blow hydraulic power pack area dry.

(5) Remove aircraft from jacks.

(6) Move aircraft to a run up area and operate both engines IAW AFM.

(7) Check hydraulic power pack area for leaks.

(8) Install all access covers and panels.

B. Adjustment Test – After O-Ring Replacement

NOTE: Failure of either the upper or lower body O-ring is rare and may be indicative of problems with the air pressure source to the pack or the pack pressurization relief valve. After replacing either pack body O-ring, Fairchild Aircraft recommends the following test:

(1) Remove hydraulic power pack pressurization relief valve.

(2) Using an air pressure regulator and gauge, slowly apply air pressure to valve in direction indicated by arrow on valve.

(3) Begin applying pressure to valve. The valve should open at 43 plus or minus one psi. Replace any valve found defective.

(4) Connect pressure gauge in place of relief valve. The gauge should be in the line coming from the pack.

(5) Operate right engine at 96% and observe that the gauge reads 31 plus or minus one psi.

(6) Install relief valve and operate both engines.

(7) Check pack area for leaks.

(8) Install all access panels and covers.

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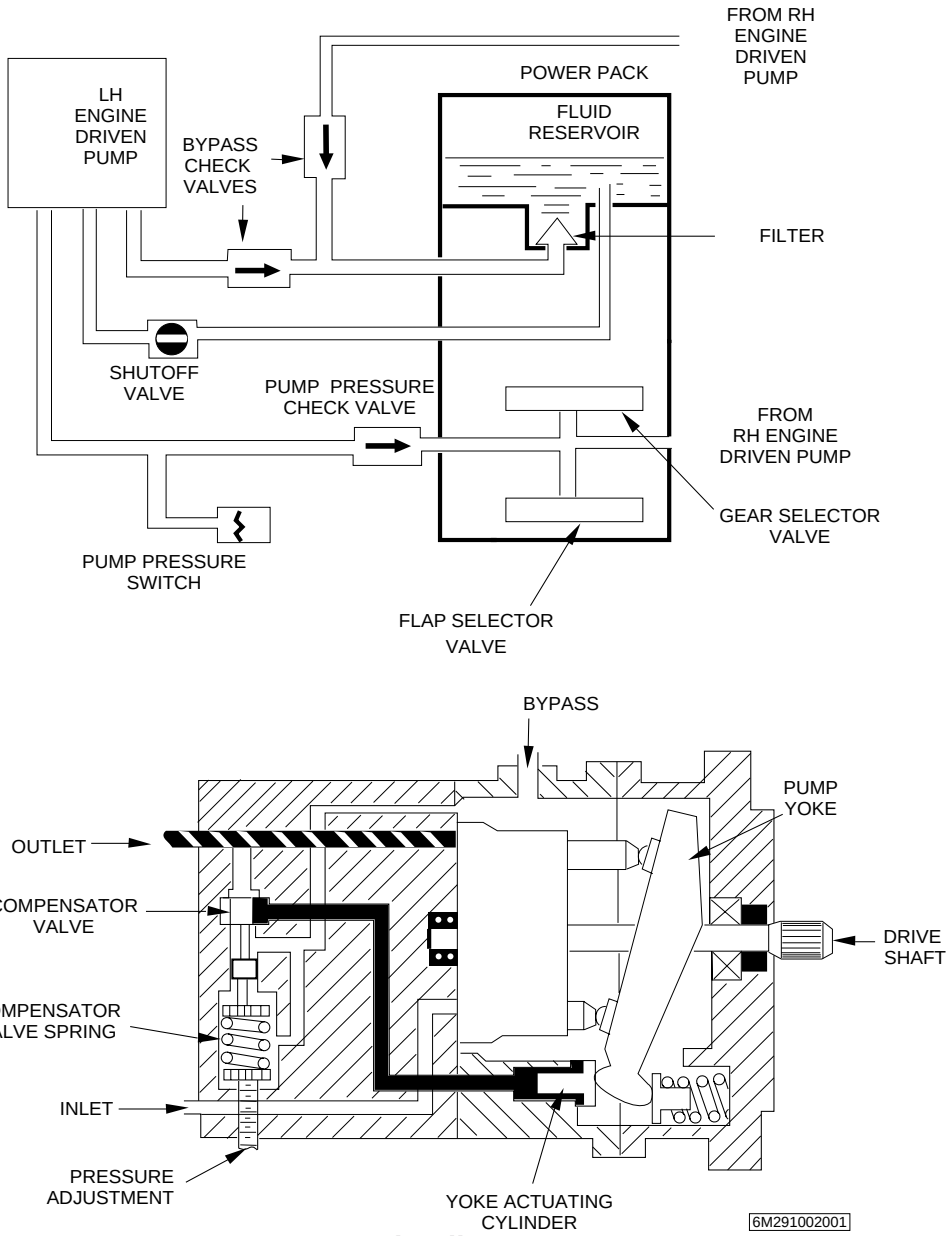
HYDRAULIC ENGINE DRIVEN PUMP – DESCRIPTION AND OPERATION

1. General

A. Hydraulic engine driven pumps, P/N PV3-044-26, are variable displacement, piston type, with an integral pressure compensator. Each pump maintains a pressure of 2,000 psi (140.6 kg/cm^2). At takeoff RPM each pump provides a flow rate of 6.20 gallons per minute. Earlier aircraft used a different pump, P/N PV3-022-018, with a lower flow capacity. Merlin III and IV aircraft may be operated with either pump, used interchangeably. Metro aircraft must be operated with the PV3-044-26 pumps.

B. With one pump inoperative, the other pump will produce 2,000 psi (140.6 kg/cm^2); however, the reduced flow rate will lengthen slightly the time it takes the gear to retract.



Hydraulic Pump
Figure 1

HYDRAULIC ENGINE DRIVEN PUMP – FAULT ISOLATION

PROBLEM:

ENGINE DRIVEN PUMP IS NOISY DURING START, CRUISE, OR SHUT-DOWN.

PREREQUISITES:

None.

START:

Verify air pressure level in hydraulic power pump. Disconnect pressure supply line from hydraulic power pack. Install direct reading pressure gauge in pressure line. With engines operating, check for 31 + 1 psi (2.18 + 0.0703 kgm/cm^2) head pressure. Is pressure correct?

–NO► Isolate area of obstruction or disconnected pressure supply line. Connect line, repair broken line, or clear obstruction.

YES▼

Check filler cap for positive seal. Does filler cap seal properly?

–NO► Replace defective filler cap.

YES▼

Check by-pass valve. Arrow should be pointing towards hydraulic power pack. Is valve properly installed?

–NO► Install valve with arrow pointing in direction of hydraulic power pack.

YES▼

If valve properly installed, remove by-pass valve and check for free flow in direction of arrow, and no flow in reverse direction. Does valve properly free flow?

–NO► Replace defective by-pass valve.

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END

HYDRAULIC ENGINE DRIVEN PUMP – MAINTENANCE PRACTICES

1. Removal/Installation – Hydraulic Engine Driven Pump

A. Removal – Hydraulic Pump

(1) Close hydraulic shutoff valve.

(2) Remove side and bottom cowling.

(3) Remove starter-generator.

(4) Disconnect case drain line.

(5) Disconnect inlet line.

(6) Disconnect outlet line.

(7) Cap all open lines.

(8) Remove nuts (MS21044C5) and washers (AN960-C516L) holding pump to accessory section.

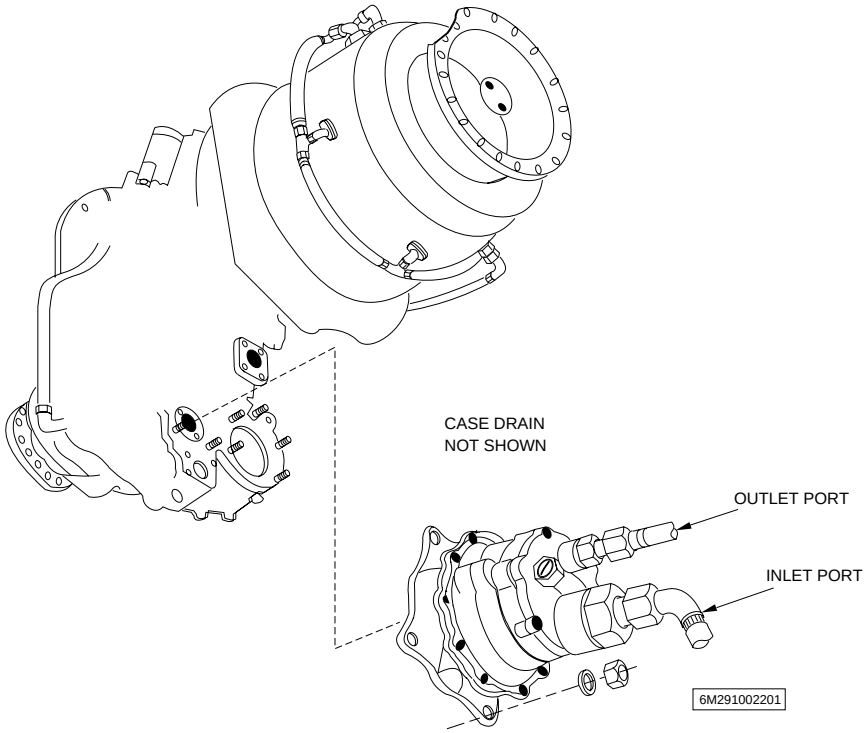
(9) Lift pump off of studs.

(10) Remove gasket.

B. Installation – Hydraulic Pump

(1) Fill pump housing with hydraulic fluid MIL-H-83282 using case drain port.

(2) Clean gasket residue from pump mating area on accessory section.



Hydraulic Pump
Figure 201

(3) Install new gasket, P/N MS9135-01.

(4) Lube pump spline with MIL-G-21164 (Avery Co., San Antonio, TX) grease.

(5) Set pump on studs and seat on gasket.

(6) Install washers and nuts; torque to 70 inch-pounds (7.91 N·m).

(7) Install reducers and packings at inlet, outlet, and drain ports. Torque inlet reducer to 500 to 700 inch-pounds (56.49 to 79.09 N·m); torque other reducers at 90 to 150 inch-pounds (10.17 to 16.95 N·m).

LOCATION	PACKING	REDUCER
Drain port	MS28778-6	AN919-6J
Inlet port	MS28778-16	AN919-29D
Outlet port	MS28778-6	AN919-6J

(8) Attach drain, inlet, and outlet lines to reducers and torque to 90-150 inch-pounds.

(9) Open shutoff valve.

(10) Rotate propeller by hand and see if pump provides pressure. If no pressure is produced, loosen outlet line and rotate propeller until fluid pressure is produced. Tighten outlet line.

(11) Install starter-generator.

(12) Install cowling.

(13) Start engine.

(14) Bleed pump pressure line at the power pack if necessary.

2. Adjustment/Test – Hydraulic Engine Driven Pump

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Pump Pressure Adjustment – Hydraulic Engine Driven Pump

(1) The pump manufacturer recommends that no pressure adjustments be attempted with the pump installed on the aircraft.

B. Pump Priming – Hydraulic Engine Driven Pump

(1) During maintenance operations involving the hydraulic system, one or both hydraulic pumps may have air introduced into their lines or “lose prime”. The symptoms may be a loud clattering noise during engine start and lack of hydraulic pressure when the engine is running. The following procedures may be followed to prime the pumps.

(2) Priming With Engine Not Operating

(a) Apply approximately 30 psi air pressure to the power pack. Air pressure may be applied through the pack pressurization relief valve.

CAUTION: USE A WRENCH TO HOLD THE FIREWALL FITTING WHEN LOOSENING THE SUCTION LINE. IF THE FIREWALL FITTING IS NOT SECURELY HELD, THE LINE ON THE AFT SIDE OF THE FITTING MAY LOOSEN AND LEAK. THE LINE ON THE AFT SIDE OF THE FIREWALL FITTING IS NOT READILY AVAILABLE FOR TIGHTENING.

(b) Loosen the pump suction line, either at the pump or at the firewall.

(c) Rotate propeller by hand until pressure is obtained, if necessary, the pressure line may be loosened until fluid begins to flow.

(3) Priming With The Engine Operating

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF PERSONNEL AND OBJECTS DURING ENGINE RUN.

(a) Start engines, briefly advance one engine to 96% to pressurize the power pack.

(b) Loosen the pump suction line (largest line) at the firewall.

CAUTION: SECURELY HOLD THE FIREWALL FITTING WHEN LOOSENING THE LINE. WHEN FLUID BEGINS TO FLOW, TIGHTEN THE LINE.

(c) Loosen the pump pressure line (intermediate size line) at the firewall.

CAUTION: SECURELY HOLD THE FIREWALL FITTING WHEN LOOSENING THE LINE. WHEN FLUID BEGINS TO FLOW, TIGHTEN THE LINE.

(d) If the hydraulic pump is still not producing pressure, loosen pump pressure line at the pressure manifold at bottom of power pack.

(e) If pressure has not been obtained within two minutes, shut down the engine. Check to see if the pump feels warm, indicating that pump is turning. Follow the procedures under Priming With Engine Not Operating.

(4) Hydraulic Pump Cavitation.

NOTE: Engine driven hydraulic pumps may operate for a maximum of 20 minutes with zero pressure or loss of fluid and continue in service if the following inspections are accomplished.

(a) Inspect main system filter located on power pack and nose wheel steering filter located in nose wheel well for metal contamination.

(b) Test each engine hydraulic pump for proper pressure (2,000 psi) [140.6 kg/cm^2] by running one engine at a time and operating the flaps and steering.

(c) if tests are satisfactory, no further action is required.

3. Inspection/Check – Hydraulic Engine Driven Pump

A. Inspection – Hydraulic Engine Driven Pump

(1) Inspect the engine driven pumps for security of mounting, signs of leakage, and general condition. Maximum permissible leakage at the mounting pad is 0.5 cc/hour.

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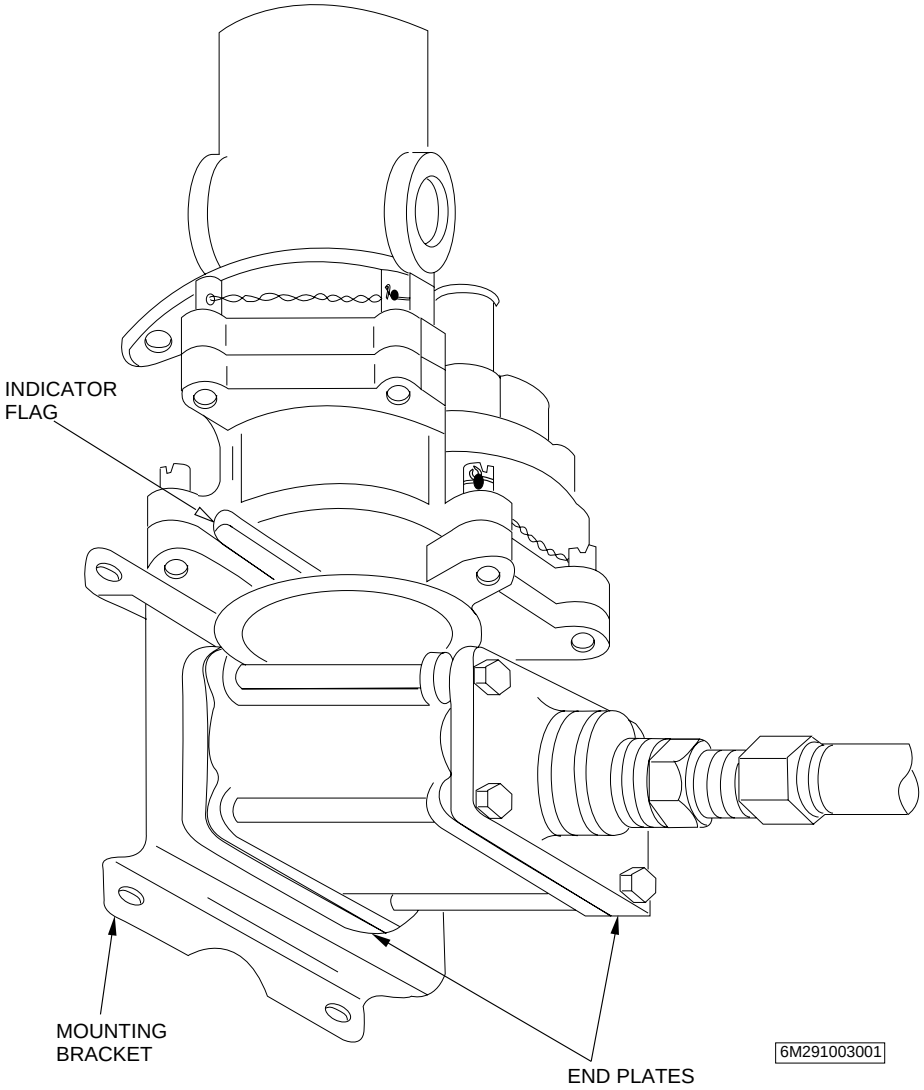
HYDRAULIC SHUTOFF VALVE – DESCRIPTION AND OPERATION

1. General

A. A hydraulic shutoff valve is installed aft of each engine firewall to control the supply of fluid to the respective hydraulic pump.

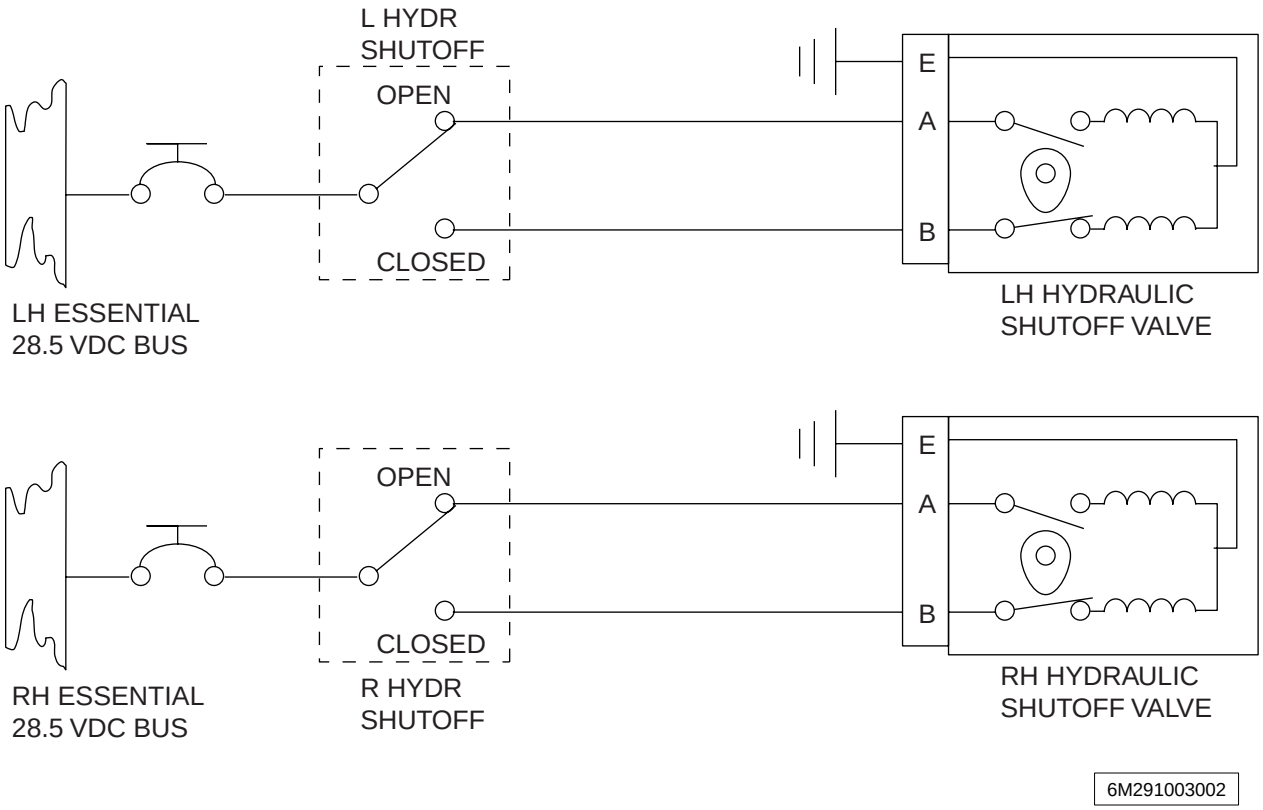
CAUTION: IF HYDRAULIC SHUTOFF VALVES ARE CLOSED WITH ENGINE DRIVEN PUMPS OPERATING, A RAPID TEMPERATURE RISE WILL OCCUR WITHIN ONE MINUTE AND THE PUMP TEMPERATURE LIMIT WILL BE EXCEEDED. HYDRAULIC SHUTOFF VALVES SHOULD NOT BE CLOSED UNLESS AN EMERGENCY EXISTS.

(1) The hydraulic shutoff valve, located between nacelle stations 99 and 109 on the left side of each nacelle is bolted to a bracket. The shutoff valve is an electric motor driven rotating ball type that remains in the last selected position when power is removed.



Hydraulic Shutoff Valve
Figure 1

(2) Two switches in the cockpit center pedestal control electrical power to each shutoff valve. An indicator flag is installed on each shutoff valve so that the valve position can be easily determined.



Hydraulic Shutoff Valve Schematic
Figure 2

HYDRAULIC SHUTOFF VALVE – FAULT ISOLATION

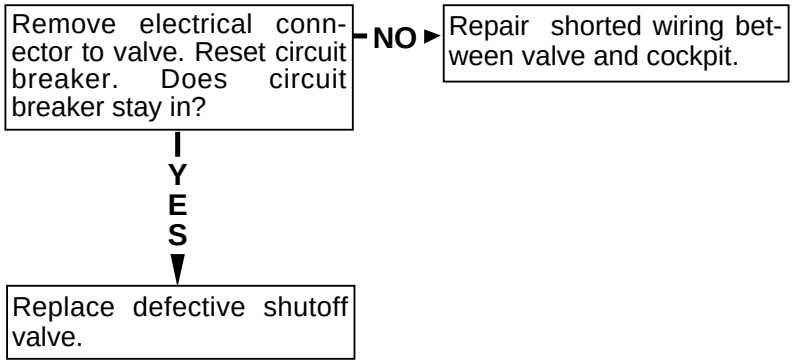
PROBLEM:

SHUTOFF VALVE CIRCUIT BREAKER TRIPS.

PREREQUISITES:

Electrical power ON.

START:



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HYDRAULIC SHUTOFF VALVE – FAULT ISOLATION

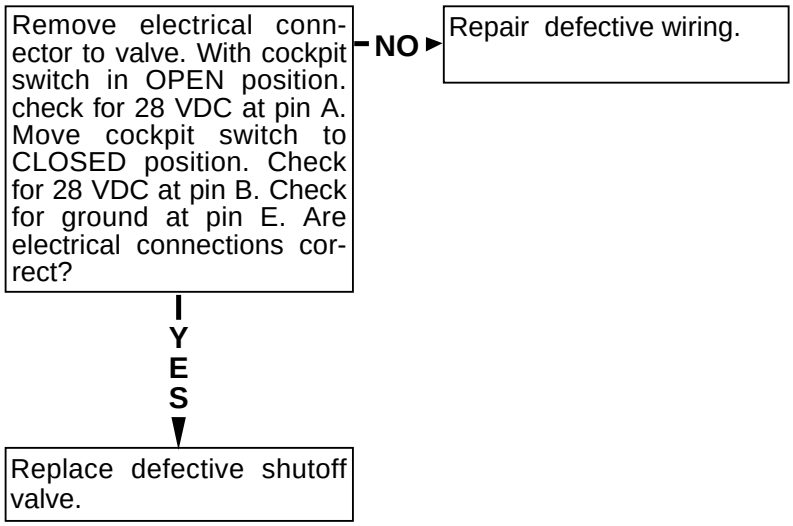
PROBLEM:

SHUTOFF VALVE CIRCUIT BREAKER TRIPS.

PREREQUISITES:

Electrical power ON.

START:



END

HYDRAULIC SHUTOFF VALVE – MAINTENANCE PRACTICES

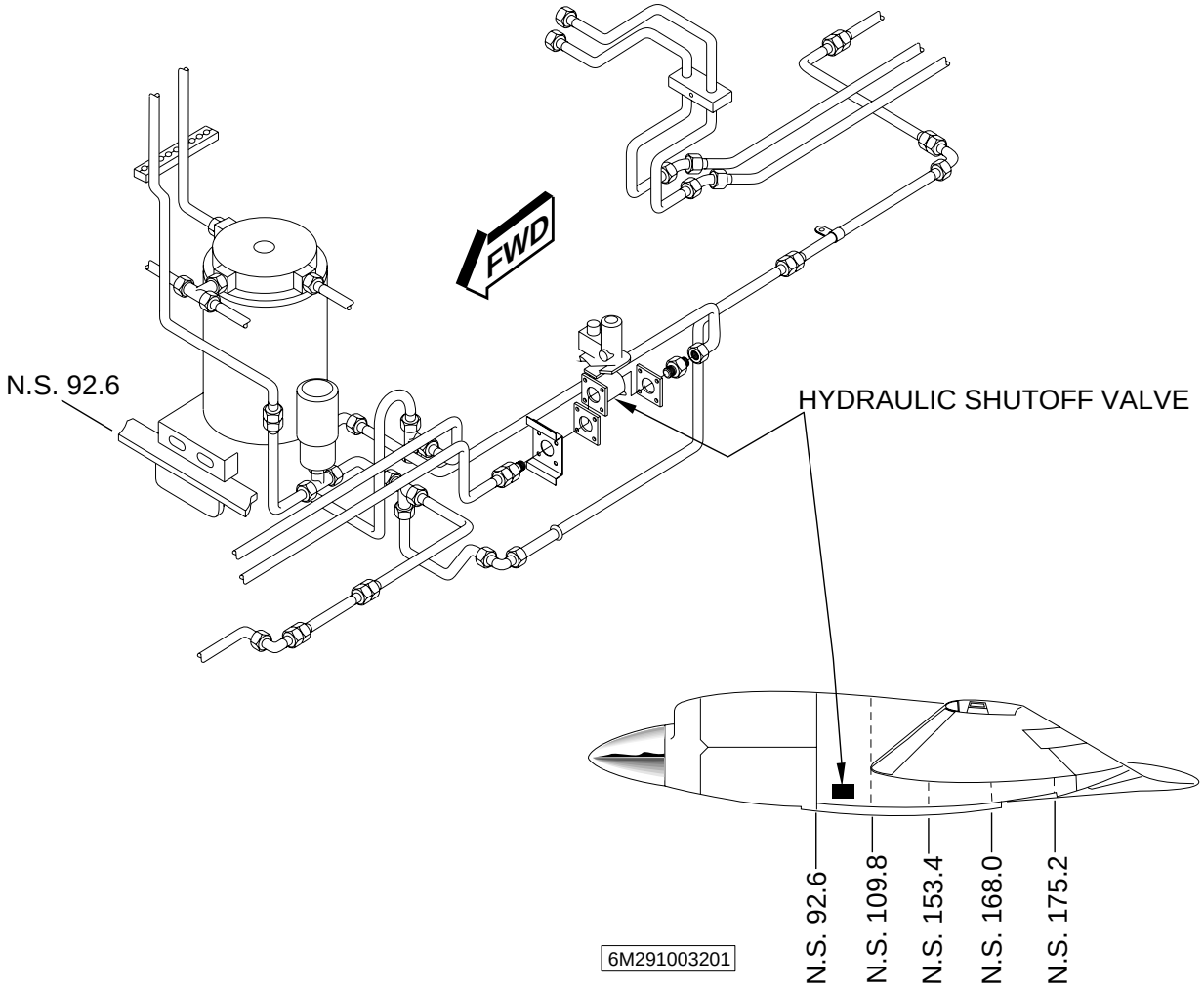
1. Removal/Installation – Shutoff Valve

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Removal – Hydraulic Shutoff Valve

WARNING: HYDRAULIC SYSTEM IS UNDER PRESSURE. USE CAUTION WHEN WORKING WITH THIS SYSTEM.

(1) Bleed off reservoir pressure.



Hydraulic Shutoff Valve
(Left Nacelle Shown)
Figure 201

(2) Remove second and third access panel aft of firewall.

CAUTION: WHEN REMOVING HYDRAULIC LINE BETWEEN SHUTOFF VALVE AND FIREWALL, MAKE SURE THE B-NUT ON FORWARD SIDE OF FIREWALL DOES NOT LOOSEN. SECURE FORWARD B-NUT BEFORE LOOSENING AFT B-NUT.

(3) Disconnect supply and outlet lines.

(4) Remove bolts and washers securing hydraulic shutoff valve to bracket.

(5) Withdraw hydraulic shutoff valve into wheel well.

(6) Disconnect electrical connector from hydraulic shutoff valve.

(7) Remove hydraulic shutoff valve.

(8) Retain end plates and connector.

B. Installation – Hydraulic Shutoff Valve

(1) Connect electrical connector to hydraulic shutoff valve.

CAUTION: DO NOT OVER TORQUE BOLTS AS DAMAGE TO END PLATES COULD RESULT.

(2) Position hydraulic shutoff valve to bracket and secure with bolts and washers. Torque bolts 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(6) Connect hydraulic supply and outlet lines to hydraulic shutoff valve. Torque each end fitting 150 to 250 inch-pounds (16.95 to 28.25 N·m).

(7) Apply hydraulic pressure to system during engine run.

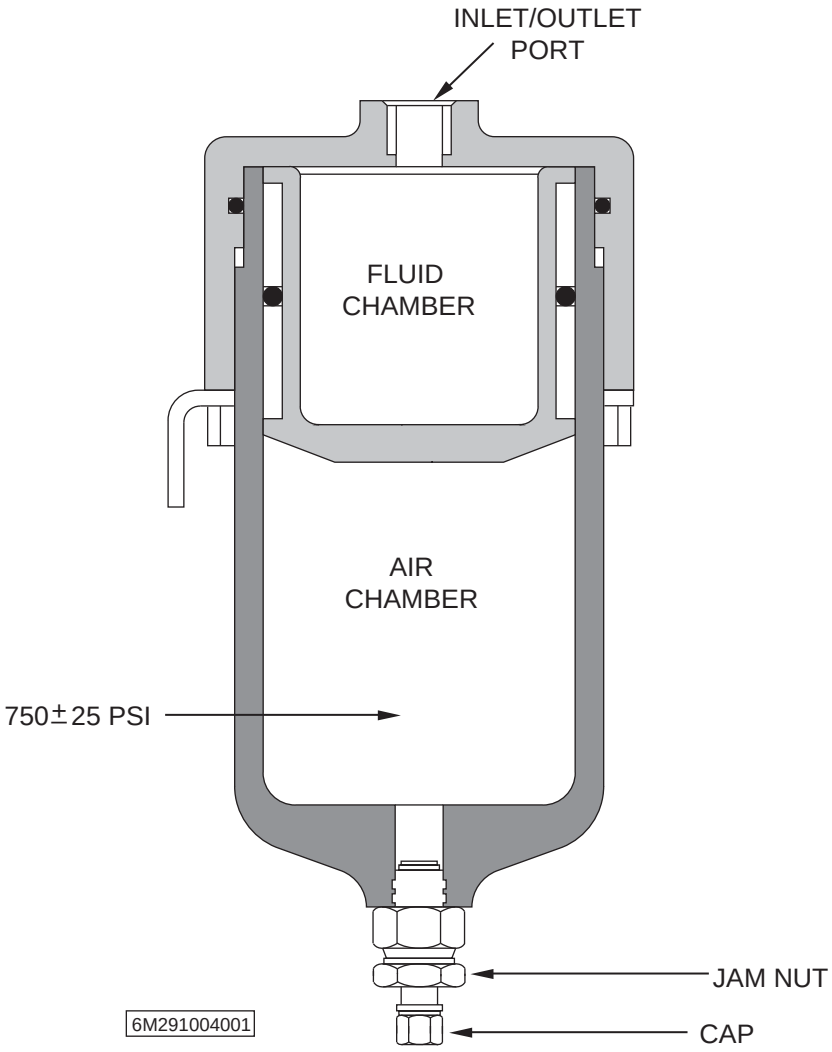
(8) Inspect for leaks.

(9) Install access panels.

HYDRAULIC ACCUMULATOR - DESCRIPTION AND OPERATION

1. General

A. A hydraulic accumulator is installed in the left side of the power pack area. The accumulator is used to smooth out the pressure fluctuations that occur in the hydraulic system. The accumulator is the piston type with dry nitrogen under pressure on one side and hydraulic fluid on the other side.



Hydraulic Accumulator
Figure 1

HYDRAULIC ACCUMULATOR - MAINTENANCE PRACTICES

1. Servicing – Hydraulic Accumulator

A. Servicing Procedures (Refer to Figure 201)

(1) Locate emergency landing gear release handle at copilot inboard seat track, rotate handle fully aft and secure.

(2) Apply external electrical power (Refer to 24-40-00).

(3) Clear all personnel and equipment beneath flaps.

(4) Bleed of any residual hydraulic pressure by cycling flaps until hydraulic pressure indicator reads zero.

(5) Remove external electrical power (Refer to 24-40-00).

WARNING: HYDRAULIC RESERVOIR IS UNDER 30 PSI PNEUMATIC HEAD PRESSURE. USE CAUTION WHEN LOOSENING OR REMOVING FILLER CAP.

(6) Locate and open hydraulic reservoir access door on outboard side of left nacelle.

(7) Lift spring lock on filler cap and slowly loosen hydraulic filler cap until head pressure is released.

(8) Tighten reservoir filler cap, insert spring lock and close access door.

(9) Remove hydraulic power pack access cover.

(10) Verify nitrogen service bottle main shutoff valve and regulator valve are closed.

(11) Remove dust cap and connect nitrogen service bottle supply hose to accumulator nitrogen filler valve.

(12) Open nitrogen service bottle main shutoff valve and verify pressure is over 800 psi.

(13) Turn accumulator nitrogen filler valve jam nut counterclockwise to unseat and fully open valve.

CAUTION: NITROGEN PRESSURE MUST BE INTRODUCED INTO ACCUMULATOR SLOWLY TO PREVENT ACCUMULATOR PISTON FROM SLAMMING AGAINST AND DAMAGING UPPER BODY.

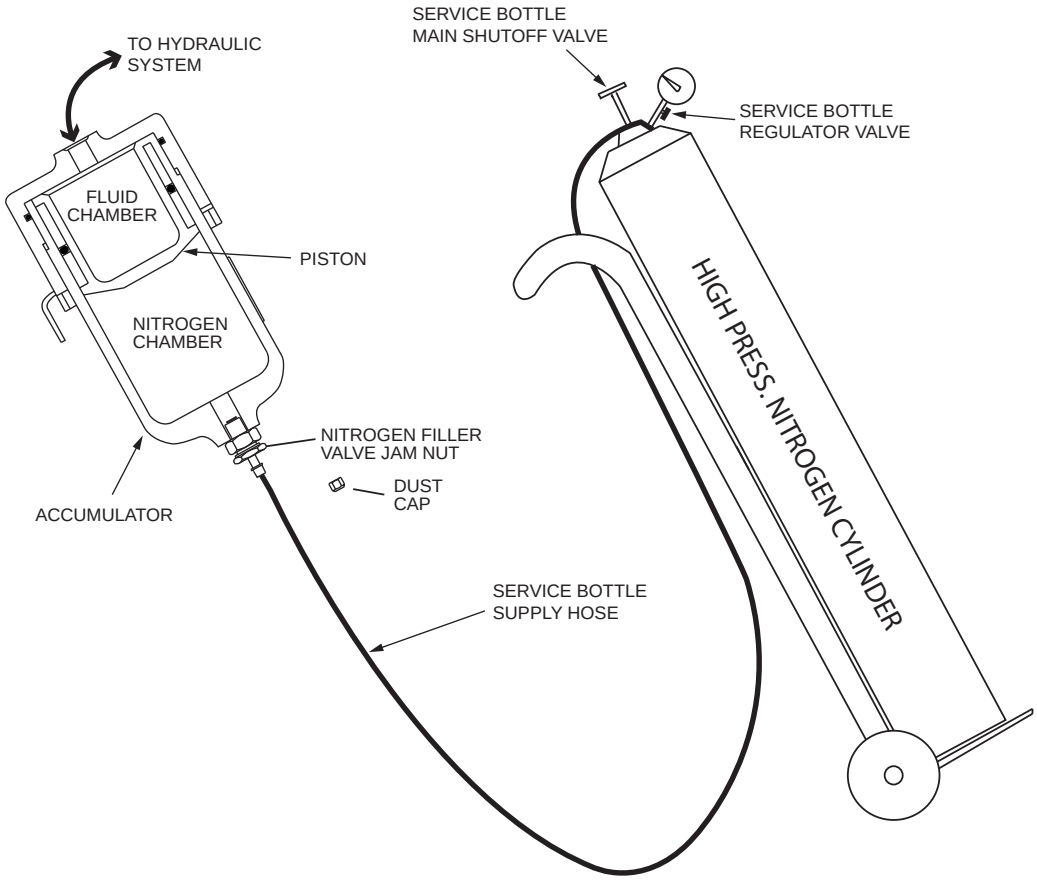
(14) Very slowly open service bottle (regulator), set to 25 psi (1.75 kgm/cm^2) initially. Increase pressure slowly to 50 psi (3.5 kgm/cm^2) and then on up to 750 ±25 psi (52.7 ± 1.75 kgm/cm^2) maximum pressure.

(15) Turn accumulator nitrogen filler valve jam nut clockwise to fully close and tighten valve.

(16) Close nitrogen service bottle main shutoff valve and regulator valve.

(17) Slowly loosen nitrogen service bottle hose from accumulator filler valve, bleeding off pressure in hose, then remove hose.

(18) Install accumulator filler valve dust cap.



Hydraulic Accumulator Servicing
Figure 201

(19) Leak check accumulator with a soap solution.

(20) Install hydraulic power pack access cover.

2. Removal/Installation – Hydraulic Accumulator

A. Removal – Hydraulic Accumulator

(1) Apply external electrical power (Refer to 24-40-00).

(2) Clear all personnel and equipment beneath flaps.

(3) Bleed of any residual hydraulic pressure by cycling flaps until hydraulic pressure indicator reads zero.

(4) Remove external electrical power (Refer to 24-40-00).

WARNING: HYDRAULIC RESERVOIR IS UNDER 30 PSI PNEUMATIC HEAD PRESSURE. USE CAUTION WHEN LOOSENING OR REMOVING FILLER CAP.

(5) Locate and open hydraulic reservoir access door on outboard side of left nacelle.

(6) Lift spring lock on filler cap and slowly loosen hydraulic filler cap until head pressure is released.

(7) Tighten reservoir filler cap, insert spring lock on cap and close access door.

(8) Remove hydraulic power pack access cover.

(9) Disconnect inlet/outlet line from accumulator. Install plug and cap on tube and fitting.

(10) Remove the two bolts and washers securing accumulator to keelson and remove accumulator.

B. Installation – Hydraulic Accumulator

(1) Service accumulator with nitrogen IAW 1.A Servicing - Hydraulic Accumulator procedure in this section.

(2) Remove protective cap and plug from accumulator inlet/outlet fitting and aircraft tubing.

(3) While holding accumulator upright, fill accumulator with MIL-H-83282 hydraulic fluid through inlet/outlet port at top of accumulator.

(4) Position accumulator in place and secure with two bolts and washers. Torque bolts 35 to 45 inch-pounds (3.95 to 5.08 Nm.)

(5) Connect aircraft tubing to inlet/outlet fitting at top of accumulator. Torque end fitting 50 to 90 inch-pounds (5.65 to 10.17 Nm.)

(6) Locate emergency landing gear release handle at copilot inboard seat track, rotate handle fully aft and secure.

(7) Apply external electrical power (Refer to 24-40-00).

(8) Clear all personnel and equipment beneath flaps.

(9) Apply external hydraulic power to aircraft (Refer to 12-10-20) and cycle flaps.

(10) Check accumulator installation for leaks.

(11) Disconnect external hydraulic power (Refer to 12-10-20).

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

(12) Disconnect external electrical power (Refer to 24-40-00).

(13) Rotate emergency landing gear release handle back to normal position.

(14) Install hydraulic power pack access cover.

AUXILIARY HYDRAULICS – DESCRIPTION AND OPERATION

1. General

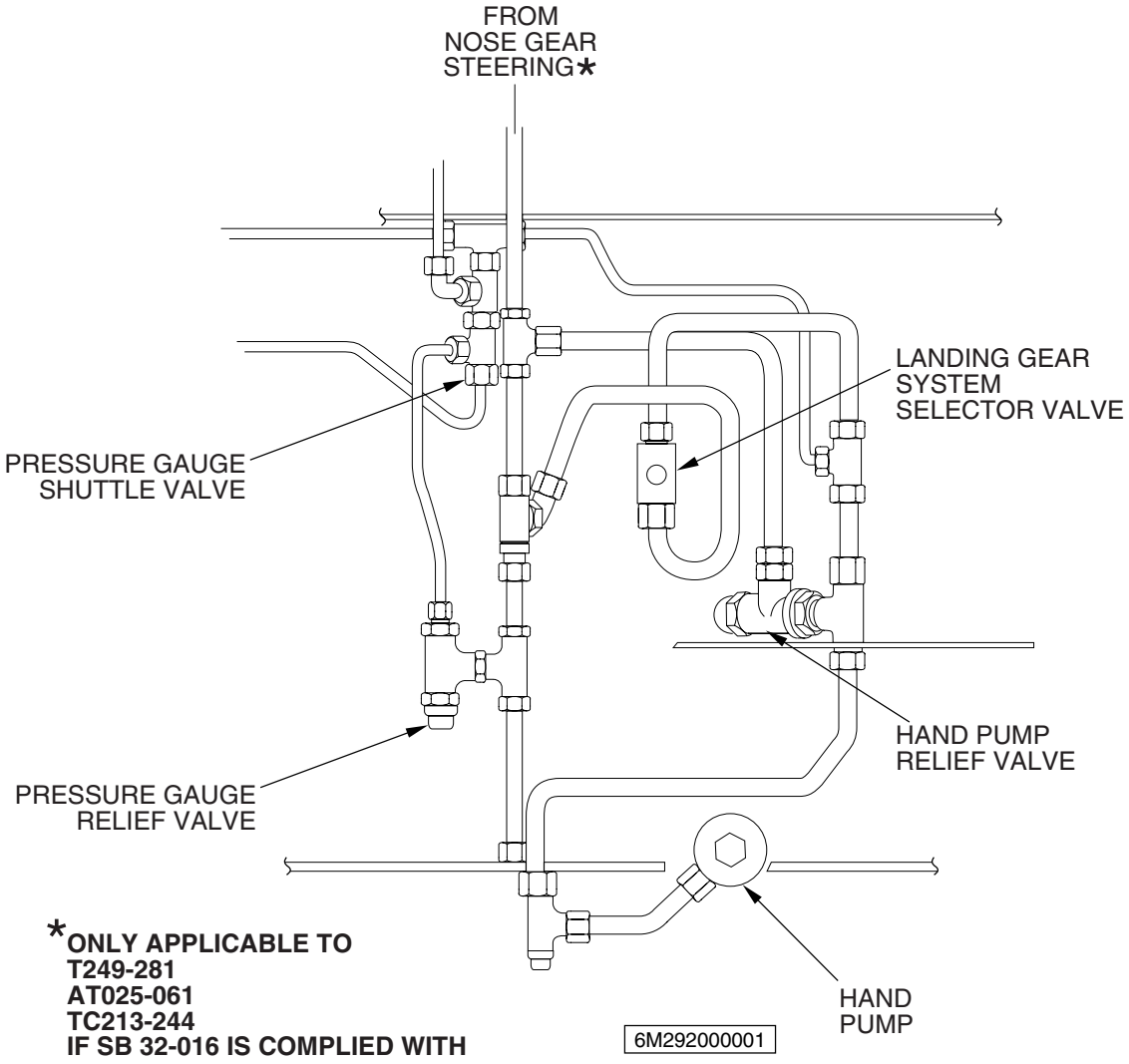
A. The auxiliary hydraulic system is designed to lower and lock landing gear should the main hydraulic system fail.

(1) A minimum of one U.S. quart (0.9463 liter) of hydraulic fluid is reserved in the reservoir, below the standpipe level, for use by the auxiliary hydraulic system.

(2) A landing gear emergency release handle in the cockpit unlocks landing gear for free fall.

(3) A hand pump in the cockpit provides sufficient pressure to lock landing gear in the down position. Hand pump hydraulic pressure is indicated on the hydraulic pressure gauge.

(4) There are no provisions to raise landing gear or lower or raise flaps, with the hand pump.



***ONLY APPLICABLE TO**
T249-281
AT025-061
TC213-244
IF SB 32-016 IS COMPLIED WITH

Auxiliary Hydraulic Components (Later Models)
Figure 1

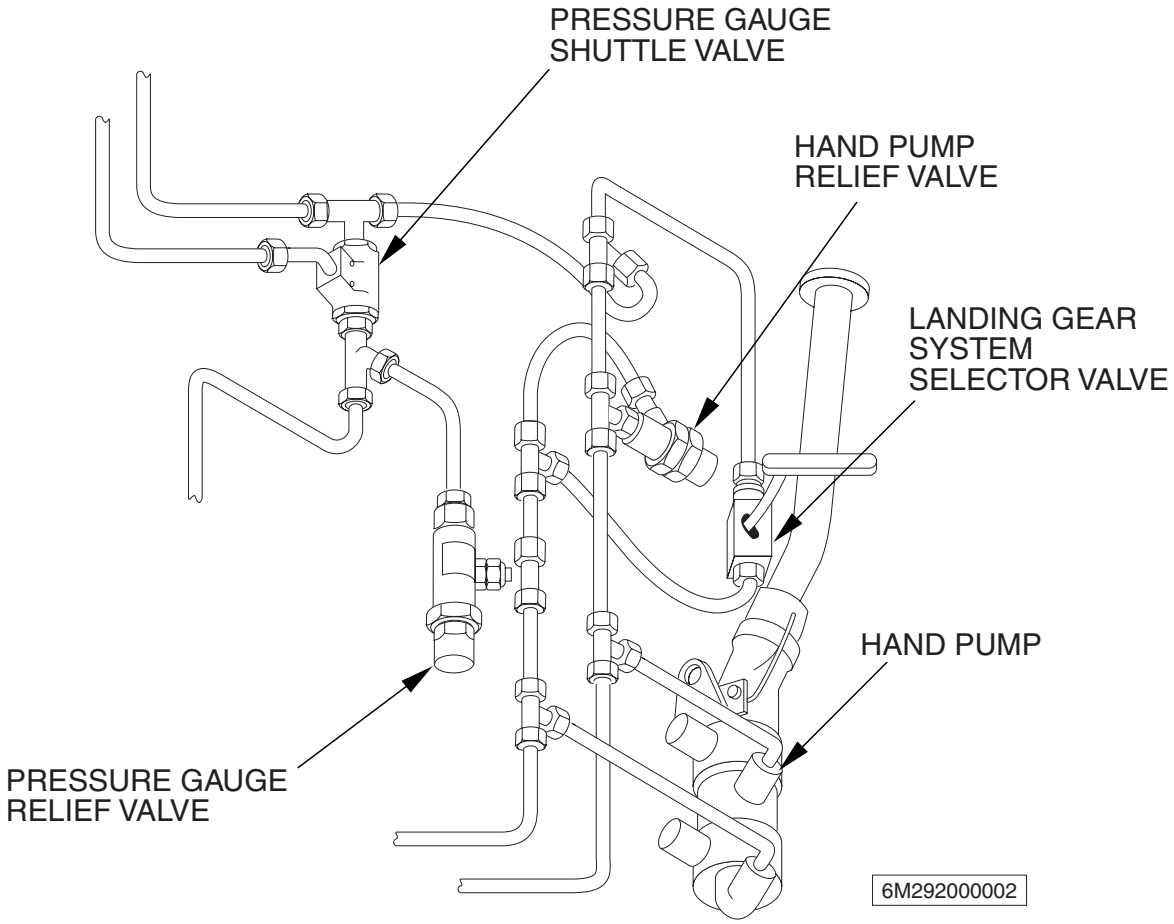
B. In the event of failure of the main hydraulic system, fluid is drawn from the sump in the reservoir by actuating the hand pump in the cockpit.

(1) The landing gear selector valve is moved forward to EMER GEAR and diverts fluid pressure to emergency gear down cylinders in the three wheel wells.

(2) A shuttle valve (located in LH nacelle power pack area) directs hand or engine driven pump hydraulic pressure to the hydraulic pressure gauge transducer.

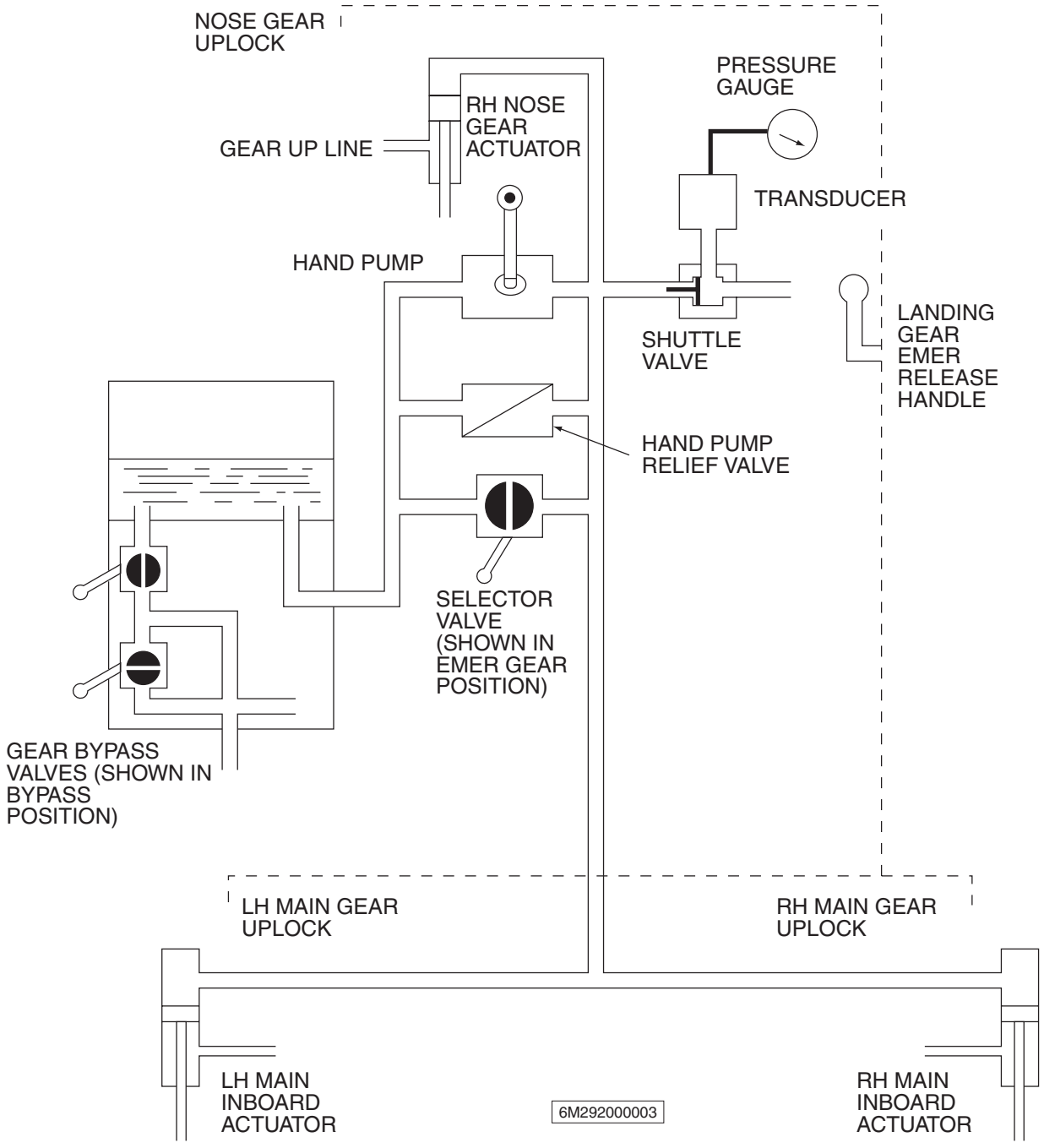
(3) A hand pump relief valve prevents excessive pressure. Approximately 800 psi (56.24 kg/cm^2) pressure is required to lower and lock landing gear.

(4) After lowering landing gear, hydraulic pressure should not bleed below 500 psi (35.15 kg/cm^2); however, should pressure bleed to zero, the overcenter down locks will hold landing gear in down and locked position.



Auxiliary Hydraulic Components (Early Models)
Figure 2

(5) A minimum of one U.S. quart (0.9463 liter) of hydraulic fluid is available in the reservoir for use by the auxiliary hydraulic system.



Auxiliary Hydraulics
Figure 3

1. Inspection/Check – Auxiliary Hydraulic System

A. Inspection

NOTE: During every retraction test, perform a functional check of the auxiliary system as follows:

(1) Place emergency release handle in the normal (forward position).

(2) Ensure that all gear are down and locked.

(3) Place landing gear system selector valve in the EMER GEAR position.

(4) Starting with zero system pressure, use hand pump to raise the hydraulic pressure to 1,000 psi (70.30 kg/cm^2).

(5) Time pressure bleed down rate. If pressure bleeds down to 700 psi (49.21 kg/cm^2) in less than five minutes, check system for leaks.

(6) Check emergency release handle in emergency position and landing gear system selector valve in EMER GEAR position.

(7) With 700 psi (49.21 kg/cm^2) indicating on the hydraulic pressure gauge, apply 2,000 psi (140.60 kg/cm^2) to the system with an auxiliary hydraulic cart.

(8) Place landing gear switch in UP position. The gear may unlock, but should not retract.

HYDRAULIC HAND PUMP – DESCRIPTION AND OPERATION

1. General

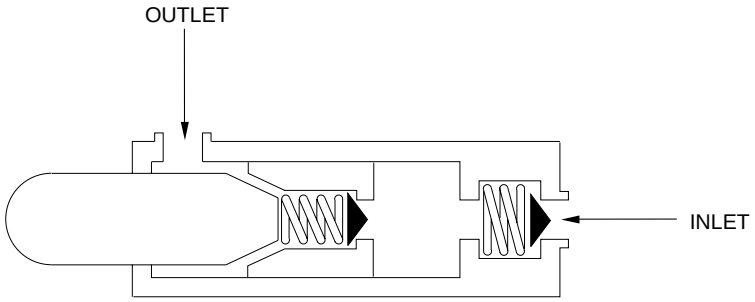
A. The hydraulic hand pump, located inboard of the pilot's seat on the floor, is a double action type and generates pressure to lock landing gear down.

(1) Hydraulic fluid from the sump of the reservoir is pumped to gear down cylinders.

(2) Output of the hand pump should be 800 to 1,500 psi (56.24 to 105.45 kg/cm^2), depending upon force applied. A 60 pound (27.27 kilogram) force applied at the end of the handle will generate approximately 1,500 psi (105.45 kg/cm^2) pressure.

(3) With landing gear down and locked, gauge pressure should indicate 500 to 800 psi (35.15 to 56.24 kg/cm^2.)

(4) There are no provisions made to raise landing gear, or lower or raise flaps with the auxiliary hydraulic system.



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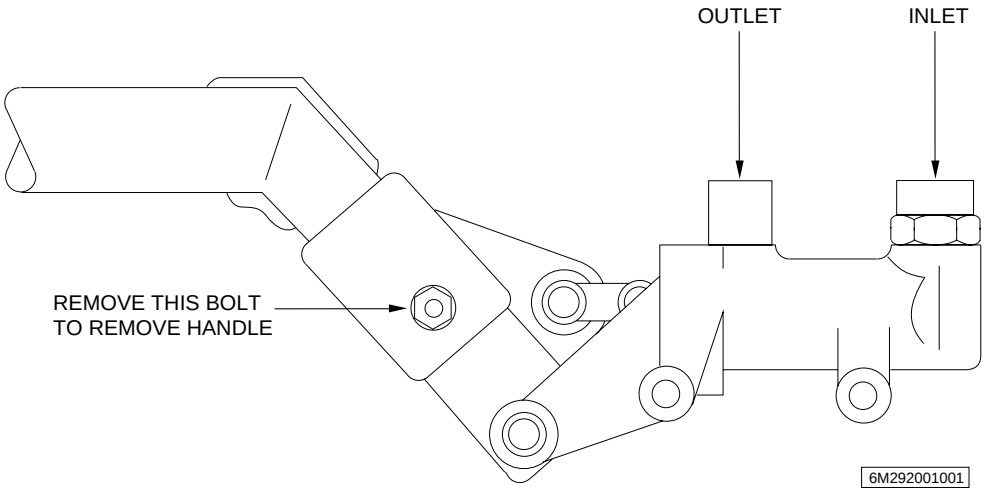

Hydraulic Hand Pump
Figure 1

HYDRAULIC HAND PUMP – FAULT ISOLATION

PROBLEM:

HYDRAULIC HAND PUMP WILL NOT INDICATE PRESSURE.

PREREQUISITES:

None.

START:

Check hydraulic fluid in sight gauge, outboard left nacelle. Is fluid visible?
- **NO** ► Service hydraulic reservoir.
- **YES** ▼

Verify air pressure level in hydraulic power pack. With engine operating, is pressure of 31 + 1 psi (2.18 + 0.703 kg/cm^2) avail-able?
- **NO** ► Check lines for obstructions or disconnects. Connect, repair, or clear obstructions. Is pressure available?
- **YES** ▼

Is landing gear emergency release handle properly positioned?
- **NO** ► Position handle properly.
- **YES** ▼

Is landing gear selector valve in correct position (EMER GEAR)?
- **NO** ► Position gear valve correctly. Does gear build pressure?
  - **NO** ► Landing gear selector valve is defective. Replace valve.
- **YES** ▼

Check hand pump relief valve. If relief valve is open, pressure will bleed back into suction line. Is relief valve closed?
- **NO** ► If relief valve will not close, valve is faulty. Replace valve.

CONTINUE

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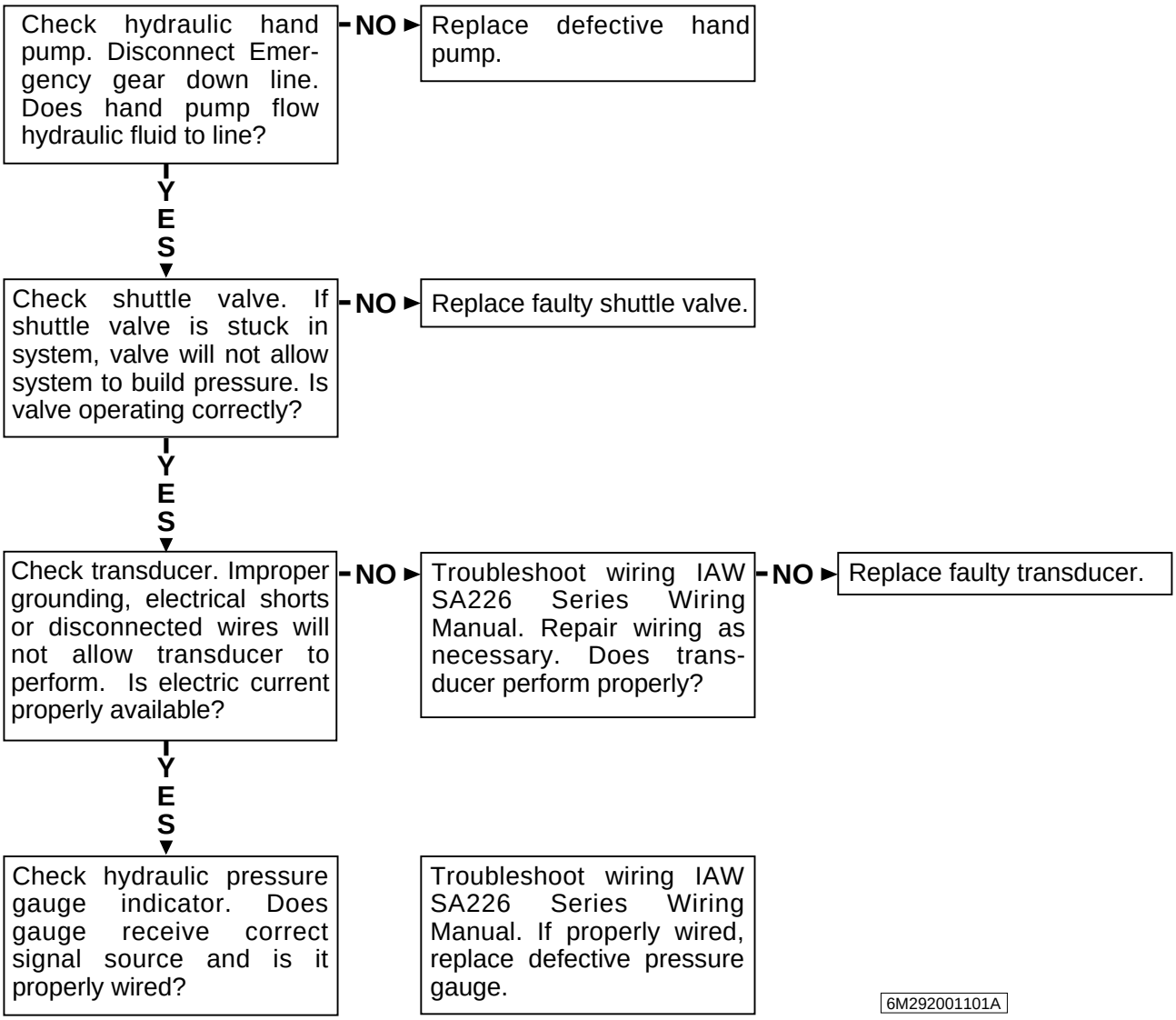
HYDRAULIC HAND PUMP – FAULT ISOLATION

Continued



END

HYDRAULIC HAND PUMP – MAINTENANCE PRACTICES

1. Removal/Installation – Hydraulic Hand Pump

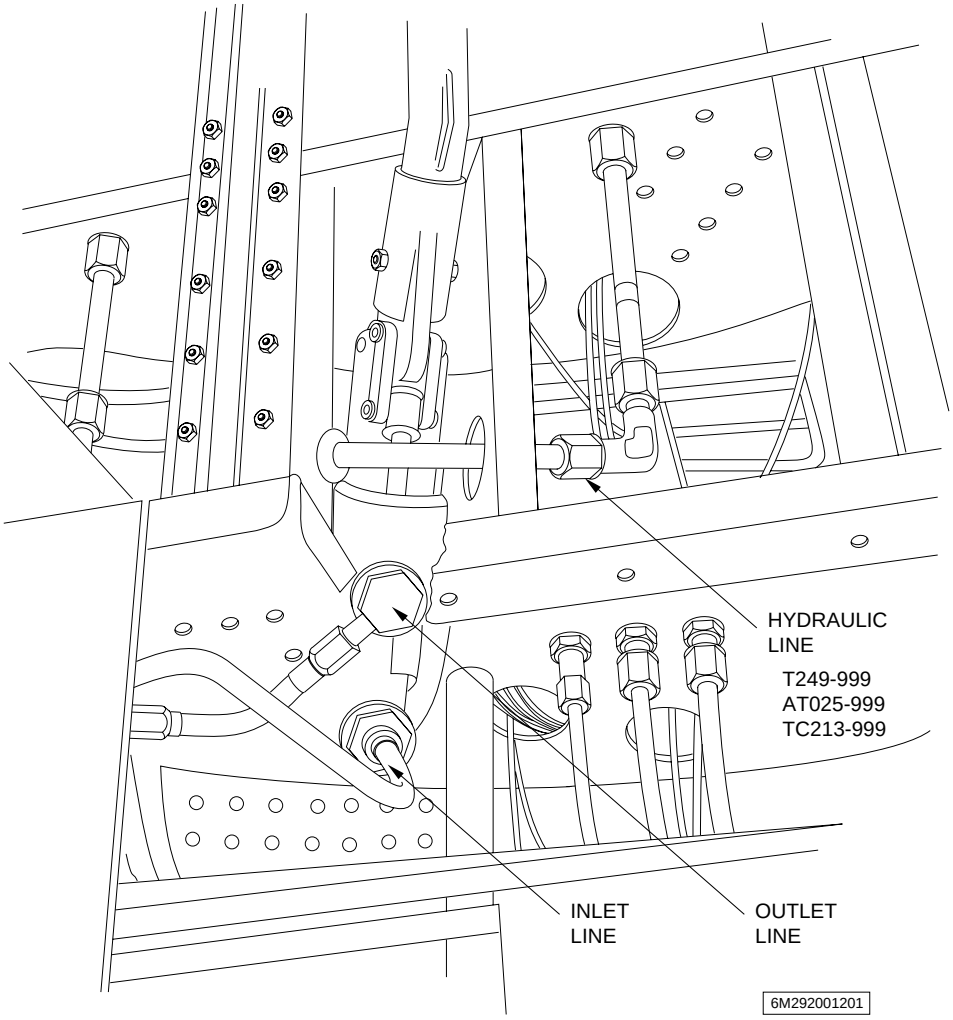
WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Removal – Hydraulic Hand Pump

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove center cockpit floorboard and pilot's floorboards IAW Chapter 53.

(3) For aircraft T249 and up, AT025 and up, and TC213 and up, remove hydraulic line as indicated in Figure 201.



Hydraulic Hand Pump
Figure 201

(4) Remove bolts washers, and nuts securing hand pump.

(5) Disconnect inlet and outlet lines from pump and remove pump. Cap open lines to prevent contamination.

(6) Remove bolt, washer, and nut securing handle to pump and remove handle.

(7) Remove fittings from inlet and outlet sides of pump.

B. Installation – Hydraulic Hand Pump

(1) Install handle into new pump and secure with bolt, washer, and nut.

(2) Install fittings into pump.

(3) Install AN919-8-6D reducer and MS28778-8-8 O-ring in inlet side of pump.

(4) Install AN775-6D bolt, AN776-6D elbow, and AN901-6A washers on outlet side.

NOTE: It may be necessary to reposition the AN776-6D elbow slightly during pump installation.

(5) Position pump in place. Secure pump with bolts, washers, and nuts.

(6) Reposition AN776-6D elbow as necessary to connect to outlet line. Remove cap and install outlet line. Torque end fitting 75 to 125 inch-pounds (8.47 to 14.12 N·m). Tighten AN775-6D bolt that secures AN776-6D elbow.

(7) Remove cap and install inlet line. Torque end fitting 75 to 125 inch-pounds (8.47 to 14.12 N·m).

(8) Install hydraulic line as shown in Figure 201. Torque end fitting 75 to 125 inch-pounds (8.47 to 14.12 N·m).

(9) Operate pump and check for leaks.

(10) Install floorboards IAW Chapter 53.

(11) Install pilot's seat IAW Chapter 25.

2. Inspection/Check – Hydraulic Hand Pump

A. Inspection – Hydraulic hand pump

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove center cockpit floorboard and pilot's floorboards IAW Chapter 53

(3) Inspect the hand pump for evidence of leakage, smooth operation, and a pressure indication when the hand pump is operated.

(4) Install floorboards IAW Chapter 53.

(5) Install pilot's seat IAW Chapter 25.

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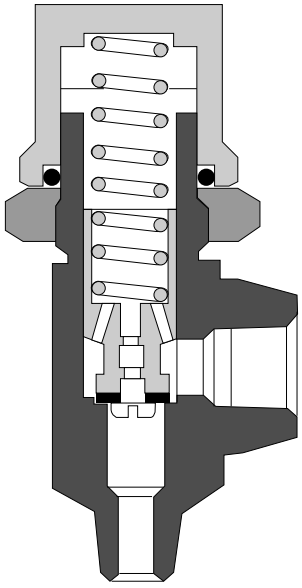
HYDRAULIC HAND PUMP RELIEF VALVE – DESCRIPTION AND OPERATION

1. General

A. The hydraulic hand pump relief valve is used to prevent overpressurization of the hydraulic system by the hydraulic hand pump. The valve is set to open at 2,200 + 25 psi (154.66 + 1.75 kgm/cm^2.) Relief setting for the valve is stamped on a tag attached to the valve. The relief valve is installed under the floorboard beneath the pilot's seat.



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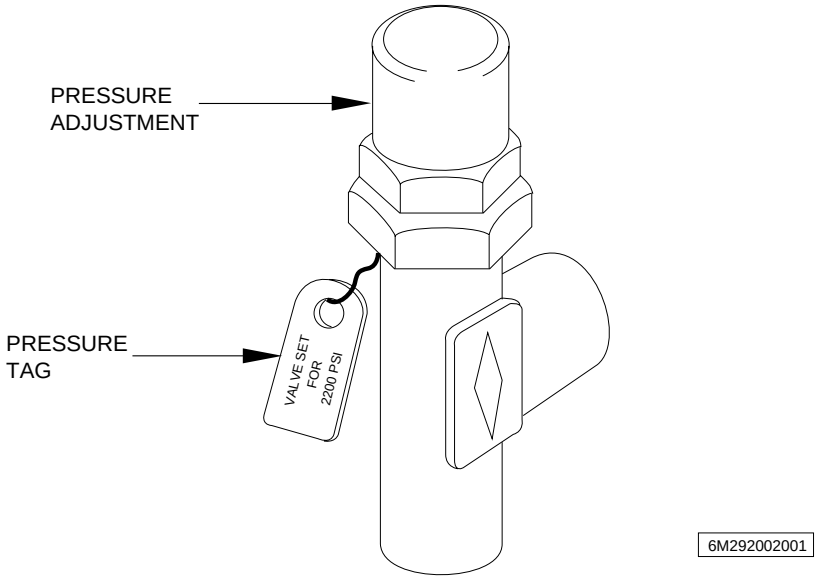


Hydraulic Hand Pump Relief Valve
Figure 1

HYDRAULIC HAND PUMP RELIEF VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – Hydraulic Hand Pump Relief Valve

A. Removal – Hydraulic Hand Pump Relief Valve

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove inboard seat track and floorboard IAW Chapter 53.

(3) Locate hydraulic hand pump relief valve (see section 29-20-00 Figures 1 and 2).

(4) Remove lines to relief valve and remove valve. Cap open lines.

(5) Remove and retain fittings installed in valve.

B. Installation – Hydraulic Hand Pump Relief Valve

(1) Inspect tag attached to new valve for a pressure setting of 2,200 psi (154.66 kg/cm^2).

(2) Install fittings in valve.

(3) Remove caps from lines.

(4) Position valve in place.

(5) Install hydraulic lines. Torque end fittings 75 to 125 inch-pounds (8.47 to 14.12 N·m).

(6) Operate hand pump and check for leaks.

(6) Install floorboard and inboard seat track IAW Chapter 53.

(7) Install pilot's seat IAW Chapter 25.

2. Inspection/Check – Hydraulic Hand Pump Relief Valve

A. Inspection – Hydraulic Hand Pump Relief valve

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove center cockpit floorboard and pilot's floorboards IAW Chapter 53

(3) inspect the valve for general condition and evidence of leakage.

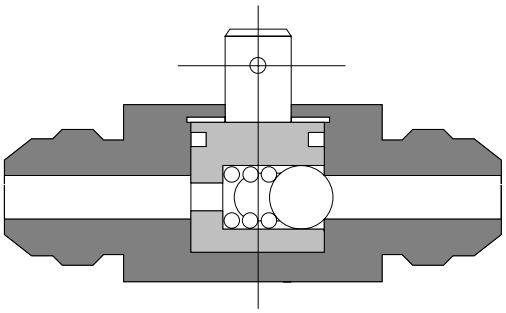
(4) Place the landing gear system selector valve in the EMER GEAR position and operate the hand pump. Inspect the relief valve for leaks.

(5) Install floorboards IAW Chapter 53.

(6) Install pilot's seat IAW Chapter 25.

LANDING GEAR SELECTOR VALVE – DESCRIPTION AND OPERATION

A. The landing gear system selector valve is located beneath the cockpit floorboard adjacent to the inboard seat track of the pilot's seat. The valve is actuated by a handle that also blocks movement of the hand pump unless the selector valve is in the EMER GEAR position. Moving the valve to the EMER GEAR (closed) position diverts hand pump pressure to the emergency gear down lines. A pip pin in the valve bracket assembly prevents accidental movement of the valve.



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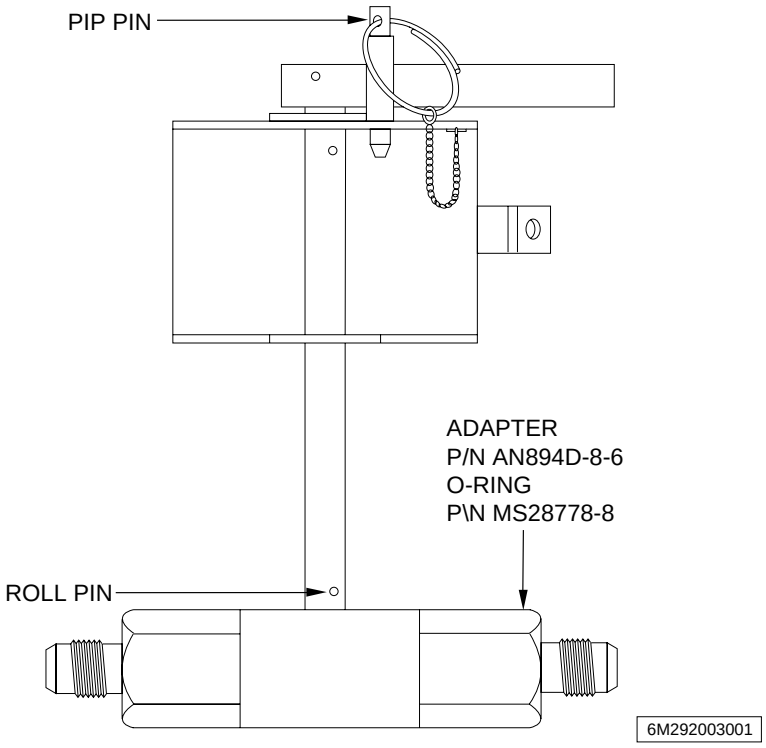


Landing Gear Selector Valve
Figure 1

LANDING GEAR SELECTOR VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – Landing Gear Selector Valve

A. Removal – Landing Gear Selector Valve

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove pilot's floorboard and seat track IAW Chapter 53.

(3) Disconnect lines to selector valve.

(4) Withdraw valve and bracket assembly.

(5) Cap open lines.

(6) Remove roll pin and separate valve from handle and bracket assembly.

(7) Remove adaptors and O-rings from valve.

B. Installation – Landing Gear Selector Valve

(1) Install handle and bracket assembly on new valve. Secure handle to valve using roll pin.

(2) Install adaptors and MS28778-8 O-rings to valve and tighten.

(2) Remove caps from hydraulic lines.

(3) Position valve in place and install hydraulic lines to valve. Torque end fittings 75 to 125 inch-pounds (8.47 to 14.12 N·m).

(3) Operate hand pump and check for leaks. Check valve for proper operation. With valve handle pointing forward (EMER GEAR), the hand pump should build pressure. With the valve handle pointing towards the copilot's seat (NORM GEAR), the hand pump should not build pressure.

(4) Install floorboards and seat track IAW Chapter 53.

(5) Install pilot's seat IAW Chapter 25.

2. Inspection/Check – Landing Gear Selector Valve

A. Inspection – Landing Gear selector valve

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove center cockpit floorboard and pilot's floorboards IAW Chapter 53

(3) Move the valve handle forward and operate the hand pump. Inspect for general condition of the valve and leakage when the hand pump is operated.

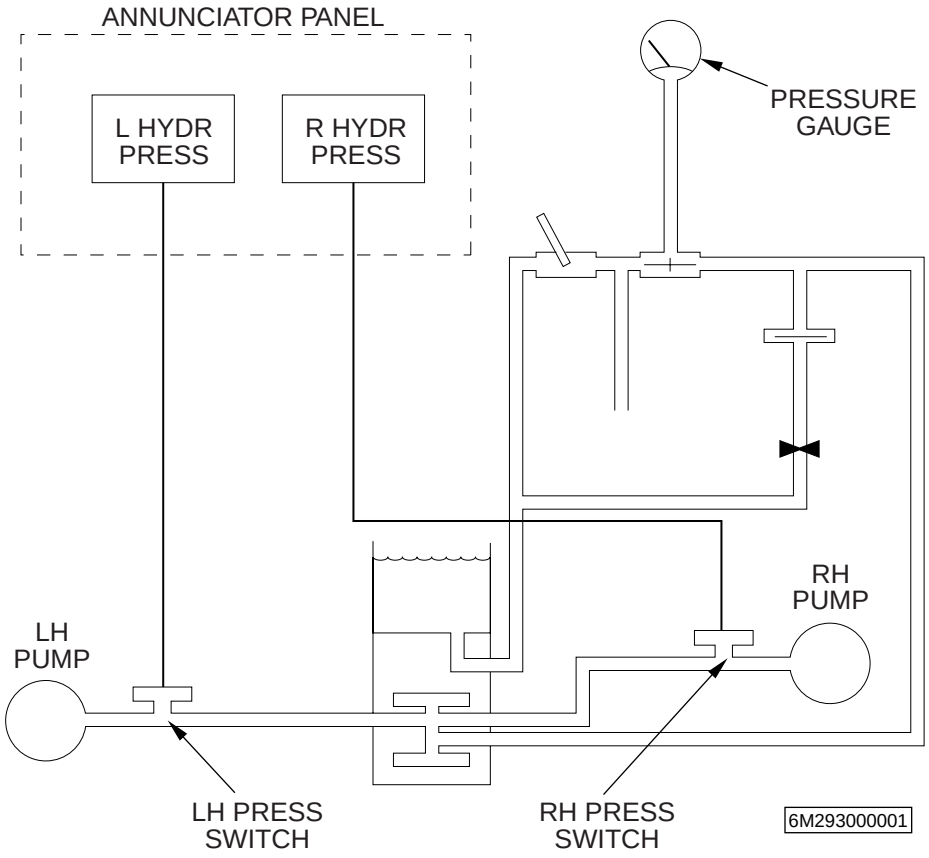
(4) Install floorboards IAW Chapter 53.

(5) Install pilot's seat IAW Chapter 25.

HYDRAULIC INDICATING – DESCRIPTION AND OPERATION

1. General

A. Hydraulic indicating consists of a direct reading pressure gauge installed in the instrument panel and two low pressure warning lights (L HYDR PRESS and R HYDR PRESS) installed in the annunciator panel. The pressure gauge will display either normal or auxiliary system pressure, whichever is higher. One low pressure warning light is connected to the output of each hydraulic pump. The warning light will be illuminated when the respective pump output pressure drops below 1,600 to 1,700 psi (112.48 to 119.51 kg/cm^2).



Hydraulic Indicating
Figure 1

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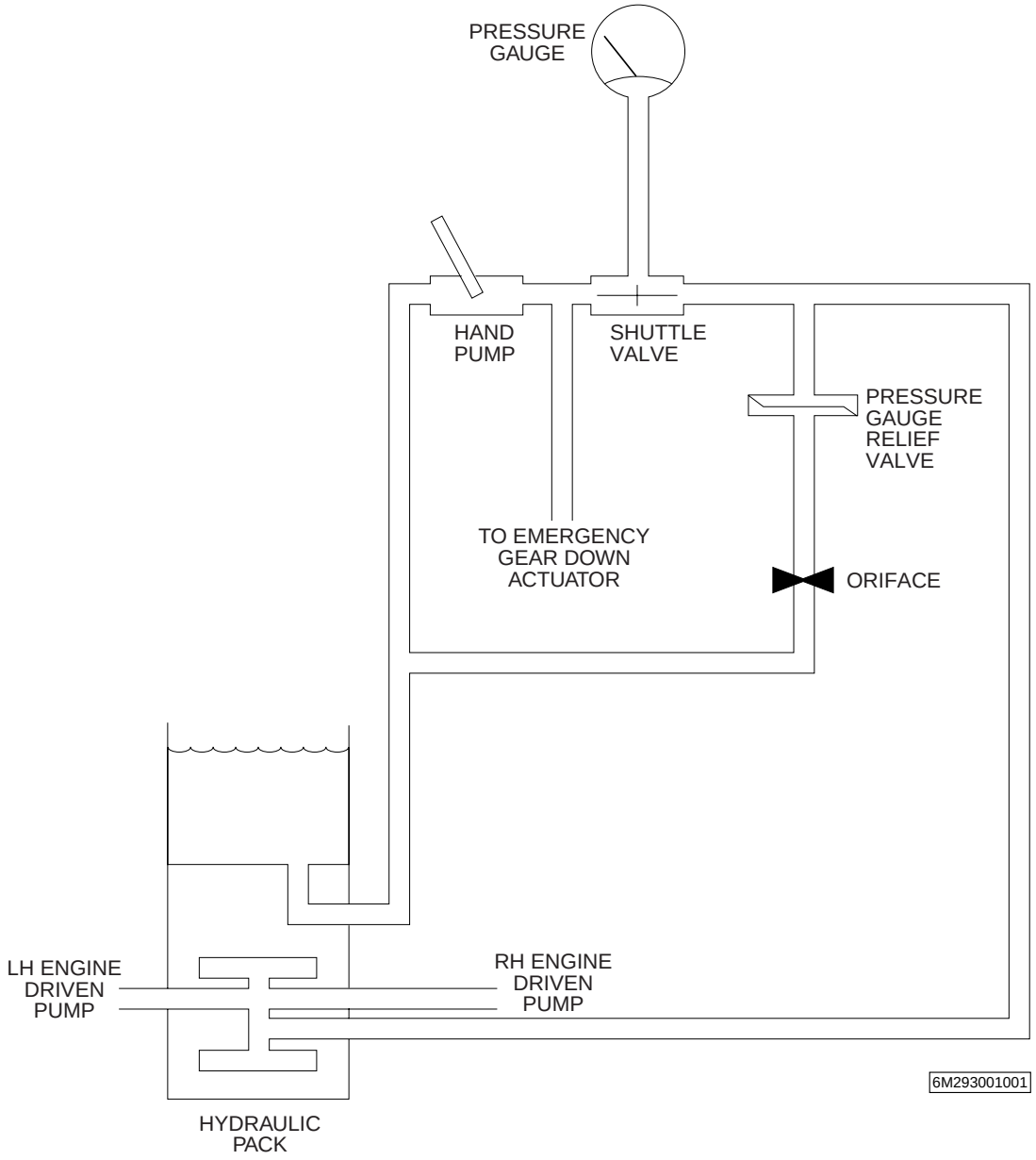
HYDRAULIC PRESSURE INDICATING – DESCRIPTION AND OPERATION

1. General

A. The hydraulic pressure indicator system consists of a direct pressure gauge, a pressure actuated shuttle valve, and a gauge pressure relief valve.

(1) The pressure gauge is usually installed in the lower right hand corner of the instrument panel, the exact location depending on the avionics installation.

(2) The shuttle valve and pressure relief valve are installed adjacent to the hydraulic pump beneath the pilot's seat (see Section 29-20-00, Figures 1 and 2).



Hydraulic Pressure Gauge
Figure 1

B. The shuttle valve is connected to both the normal system and the auxiliary system.

(1) The greater pressure will displace the shuttle and be displayed on the pressure gauge.

(2) The engine driven pump output contains small surges that occur with changes in engine RPM. These surges can become trapped in the small capillary gauge supply line and cause the gauge to indicate higher than actual system pressure.

(3) A relief valve is installed as a constant pressure bleed to provide better gauge accuracy. An orifice is installed in the relief valve return line to prevent excessive pressure drop should the relief valve stick open.

C. Aircraft prior to T220, AT009, and TC204 were not equipped with this relief valve. These aircraft were equipped with a earlier design hydraulic pack that was not as well internally sealed as later packs. The internal leakage of the earlier packs prevented entrapment of pump pressure surges.

HYDRAULIC PRESSURE INDICATING – FAULT ISOLATION

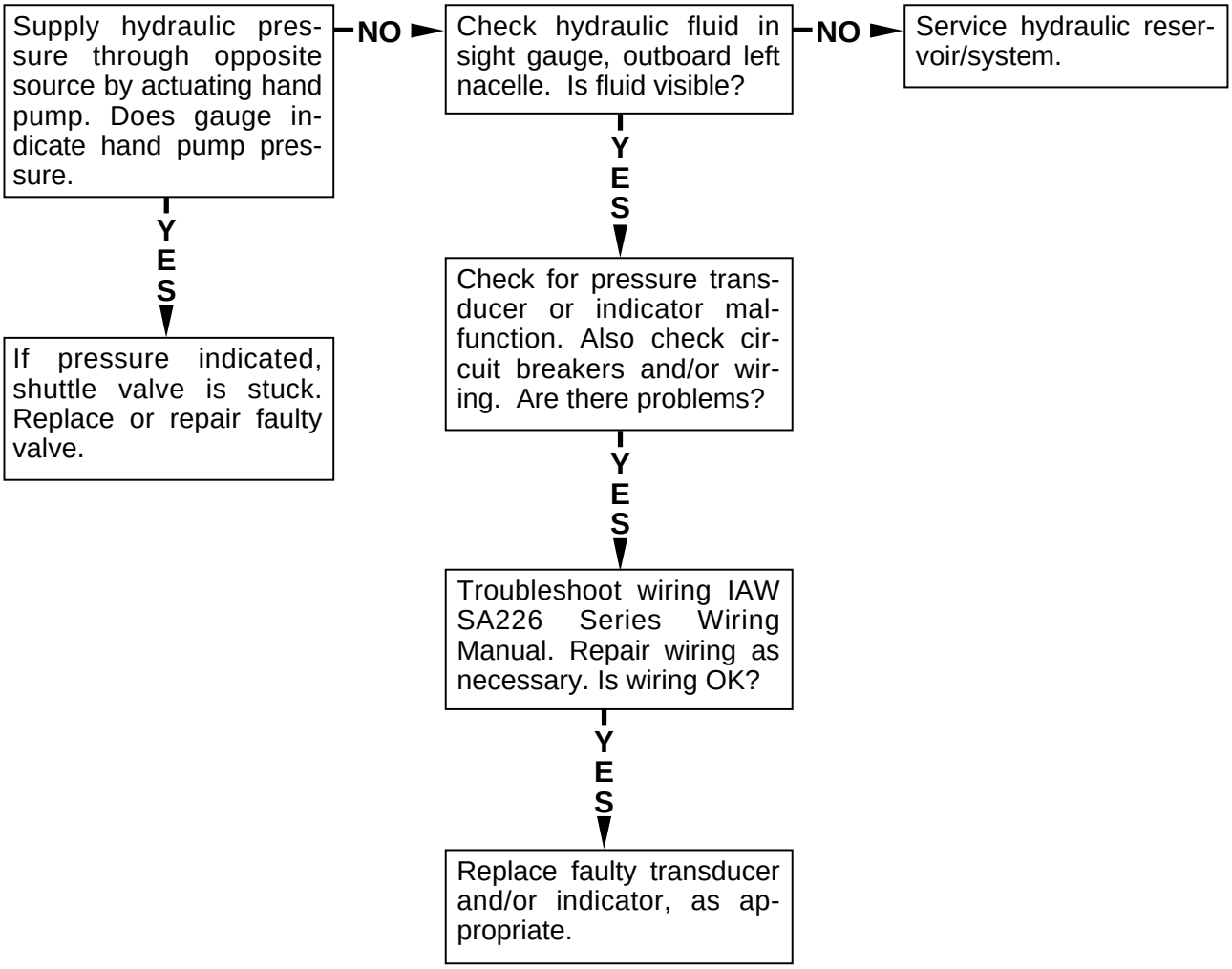
PROBLEM:

NO INDICATION ON GAUGE WHEN UNDER NORMAL SYSTEM PRESSURE.

PREREQUISITES:

None.

START:



END

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1. Removal/Installation – Hydraulic Pressure Indicating

A. Removal – Pressure Gauge Shuttle Valve

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove pilot's inboard seat track and floorboard IAW Chapter 53.

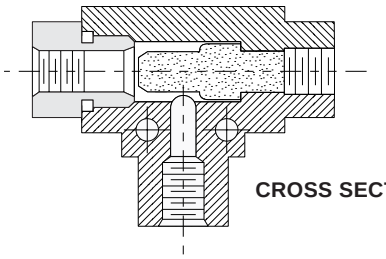
(3) Disconnect hydraulic lines from shuttle valve.

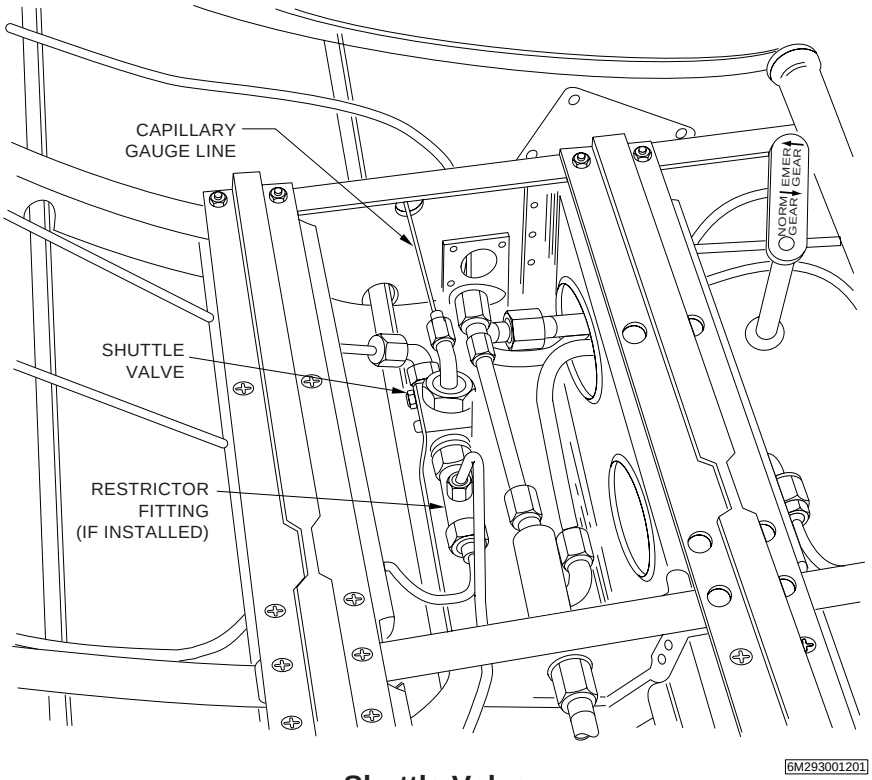
(4) Cap open hydraulic lines.

(5) Remove mounting bolts securing valve.

(6) Remove valve.

(7) Remove and retain fittings from valve. Note that the aft "T" fitting is a special restrictor fitting.





Shuttle Valve
Figure 201

B. Installation – Pressure Gauge Shuttle Valve

(1) Install original fittings in new valve. Ensure that fittings are oriented correctly. On applicable aircraft, ensure that the special restrictor fitting will be aft when valve is installed.

(2) Position shuttle valve and secure with bolts.

(3) Remove caps from hydraulic lines.

(4) Install hydraulic lines to shuttle valve. Torque 75 to 125 inch-pounds (8.47 to 14.12 N·m).

(5) Rotate system selector valve to EMER GEAR and actuate hand pump. Check valve and fittings for leaks. Move selector to NORM GEAR and apply hydraulic pressure to system (either by hydraulic cart or by rotating propeller). Recheck valve and lines for leaks. Check gauge for proper indication.

(6) Install and secure pilot's floorboard and seat track.

(7) Install and secure pilot's seat.

C. Removal – Pressure Gauge Relief Valve

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove pilot's inboard seat track and floorboard IAW Chapter 53.

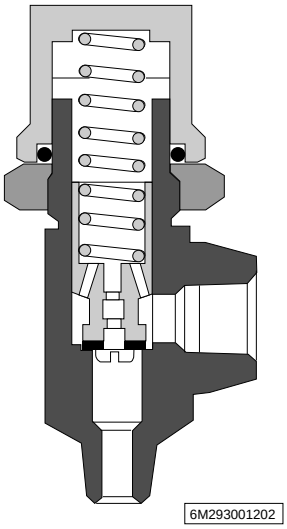
(3) Using Figures 1 and 2, Section 29-20-00, locate gauge pressure relief valve.

(4) Remove hydraulic lines from relief valve.

(5) Cap open lines.

(6) Remove relief valve.

(7) Remove and retain fittings installed in valve.



CROSS SECTION

Pressure Gauge Relief Valve
Figure 202

D. Installation – Pressure Gauge Relief Valve

(1) Inspect tag attached to new valve for a pressure setting of 2,050 psi (114.12 kg/cm^2).

(2) Install original fittings in new relief valve.

(3) Remove caps from hydraulic lines.

(4) Position relief valve in place.

(5) Install and tighten hydraulic lines to relief valve. Torque end fittings 75 to 125 inch-pounds (8.47 to 14.12 N·m).

(6) Apply hydraulic pressure to system, either by a hydraulic cart or by operating the engines. Check relief valve for leaks.

(7) Install floorboard and seat track IAW Chapter 53.

(8) Install pilot's seat IAW Chapter 25.

2. Inspection/Check – Hydraulic Pressure Indicating

A. Inspection – Hydraulic Pressure Indicating

(1) Remove pilot's seat IAW Chapter 25.

(2) Remove pilot's floorboards IAW Chapter 53

(3) Move landing gear system selector valve to the EMER GEAR position and actuate the hand pump.

(4) Inspect the pump cylinder, shuttle valve, gauge pressure relief valve, and transducer for leaks and security as well as the pressure gauge for pressure indication.

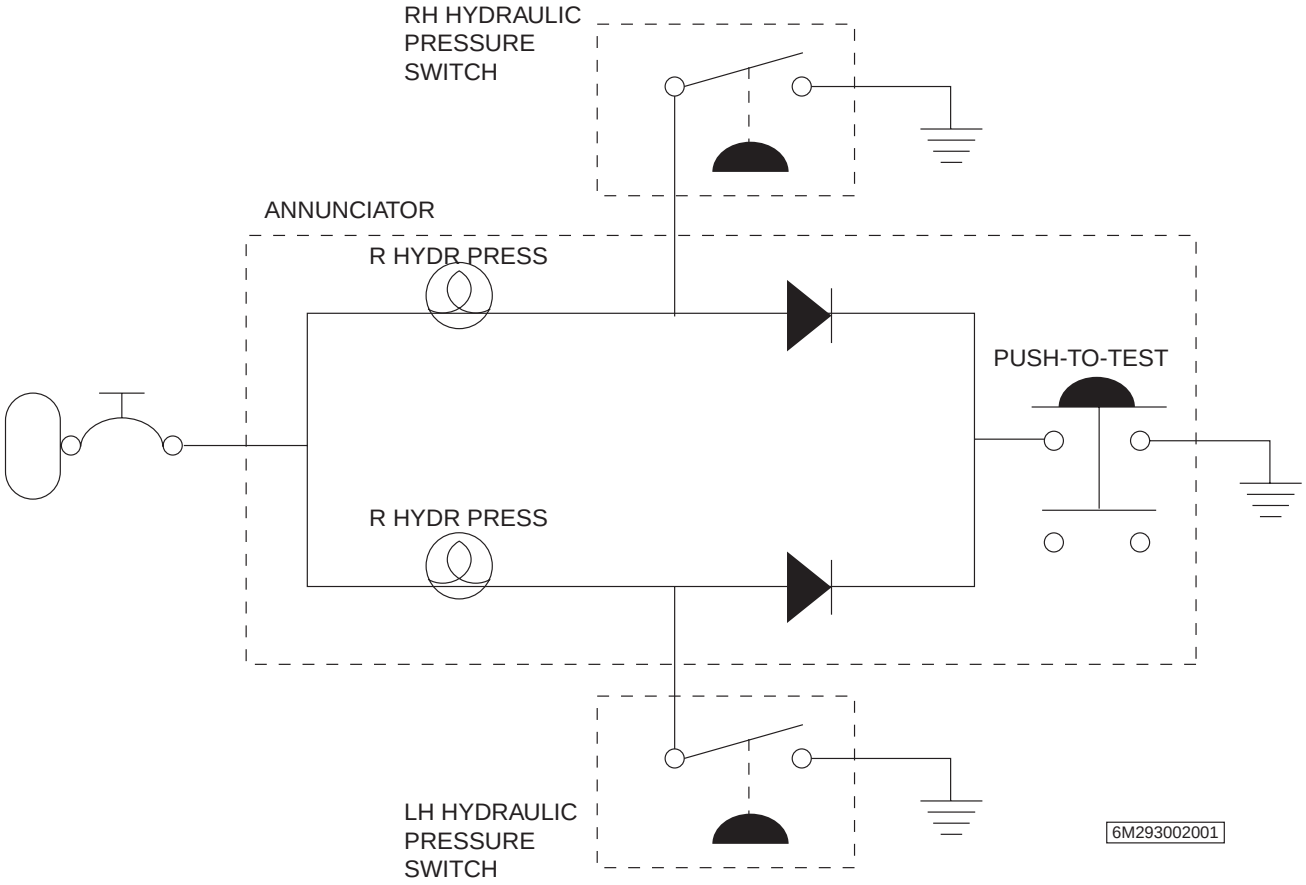
(5) Install floorboard and seat track IAW Chapter 53.

(6) Install pilot's seat IAW Chapter 25.

HYDRAULIC PRESSURE WARNING – DESCRIPTION AND OPERATION

1. General

A. Two lights in the annunciator panel (L HYDR PRESS and R HYDR PRESS) and two pressure switches in the left nacelle are used in the pressure warning system. The pressure switches close at 1,600 to 1.700 psi (112.48 to 119.51 kg/cm^2), providing a ground to the indicator lights (see Figure 1). Aircraft after T278, AT056, and TC229, and those aircraft having complied with Service Bulletin 29-003 (R) are equipped with two snubbers in the supply line to each switch. These snubbers smooth pump output fluctuations, thereby increasing the service life of the pressure switches.



Hydraulic Pressure Warning
Figure 1

HYDRAULIC PRESSURE WARNING – FAULT ISOLATION

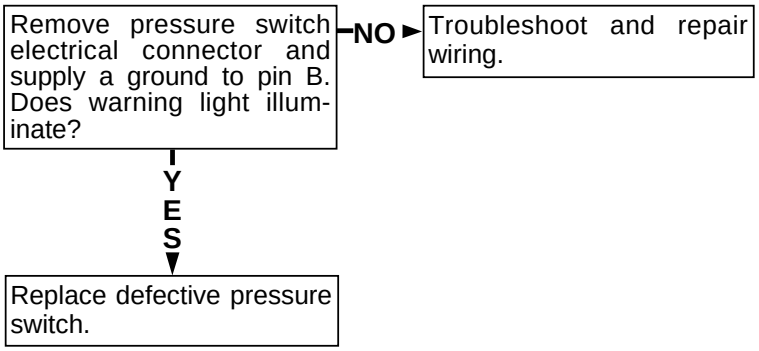
PROBLEM:

NO WARNING LIGHT (NO HYDRAULIC PRESSURE).

PREREQUISITES:

Electrical power ON.

START:



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END

HYDRAULIC PRESSURE WARNING – FAULT ISOLATION

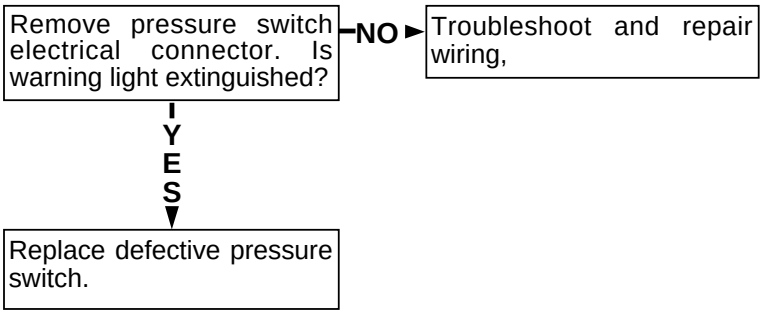
PROBLEM:

WARNING LIGHT IS ILLUMINATED AT ALL TIMES (HYDRAULIC PRESSURE IS NORMAL).

PREREQUISITES:

Electrical power ON.

START:



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END

HYDRAULIC PRESSURE WARNING – MAINTENANCE PRACTICES

1. Removal/Installation – Hydraulic Pressure Warning

A. Removal – Hydraulic Pressure Warning Switch

(1) Remove transmitter access panel on left nacelle.

(2) Locate pressure switch to be removed (the forward switch is the right and the aft switch is the left).

(3) Remove and plug hydraulic line to switch. Remove electrical connector from switch.

(4) Loosen bolt securing clamp around switch. Withdraw switch through clamp.

(5) If applicable, remove snubber (P/N 1KAC-1E) from switch.

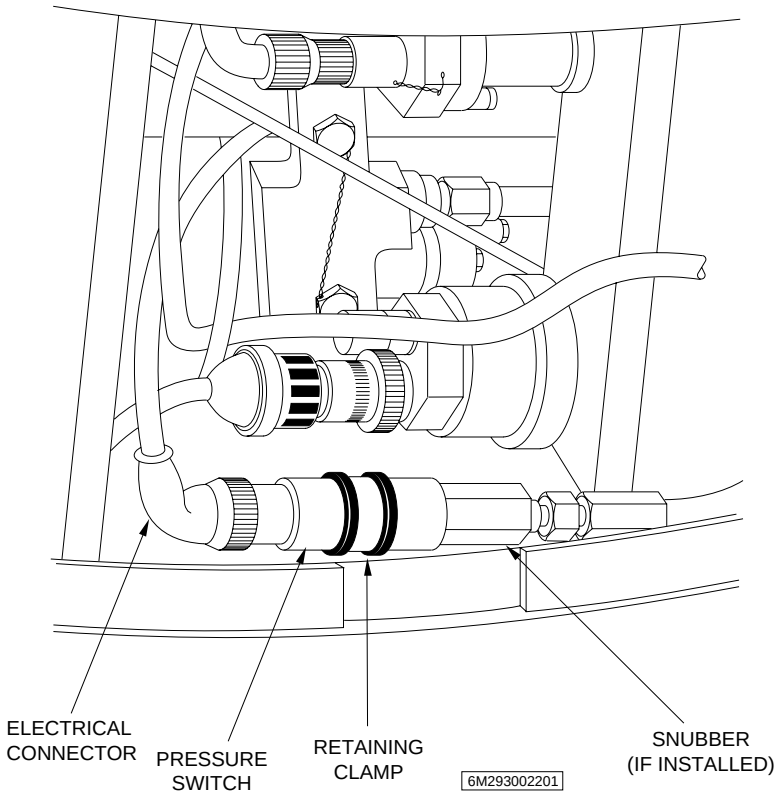
B. Installation – Hydraulic Pressure Warning Switch

(1) If applicable, install snubber (P/N 1KAC-1E) in the new switch. Position new switch within clamp and tighten bolt securing clamp. Remove plug and install hydraulic line to switch.

(2) Install and safety wire the electrical connector to the switch.

(3) Motor respective engine to build hydraulic pressure. Check switch for leaks. Check annunciator light for proper operation.

(4) Install transmitter access panel.



Hydraulic Pressure Switch
Figure 201

2. Inspection/Check – Hydraulic Pressure Warning

A. Inspection – Hydraulic Pressure switch

(1) Check the pressure switch for security of mounting, evidence of leakage, and secure electrical connections.

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CHAPTER 30 – ICE AND RAIN PROTECTION

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ICE AND RAIN PROTECTION – DESCRIPTION AND OPERATION

1. General

A. This chapter provides maintenance practices for ice and rain protection equipment installed on SA226 series aircraft.

(1) This equipment includes; aluminum and fiberglass fuselage ice guards, neoprene rubber and fabric guards used to protect the radome, vertical stabilizer, wing root fairings and landing light deflectors.

(2) Also included in this chapter are inflatable deice boots used to protect wing and horizontal stabilizer leading edges, and non-rotating deice components required to deliver electrical power to propeller deice boots.

2. Erosion Guard (30-00-10)

A. To prevent sand and dirt erosion of exposed components, neoprene rubber and/or fabric guards are attached to the leading edge of the vertical stabilizer, inboard wing, and radome nose.

3. Deice Boot (30-10-00)

A. Wing and horizontal stabilizer leading edges are protected from ice accumulation by inflatable deice boots.

4. Engine Intake Anti-Ice (30-20-00)

A. Engine bleed air is used to prevent ice from forming in the engine intake. Bleed air is controlled by an electrically operated deice valve.

5. Pitot Anti-Ice (30-30-00)

A. Electrically powered, thermostatically controlled heating elements are incorporated in the pitot mast to prevent ice accumulation.

6. SAS Vane Deice (30-30-10)

A. SAS vanes are electrically heated To prevent ice accumulation.

7. Windshield Wiper (30-40-00)

A. Electrically powered windshield wipers help remove rain and moisture from the windshield.

8. Windshield Deice (30-40-10)

A. Electrically powered thermostatically controlled windshield heating prevents ice forming on the windshield.

9. Propeller Deice (30-60-00)

A. Propeller deice consists of all non-rotating deice components required to deliver electrical power to the propeller deice boots.

10. Brush Block (30-60-10)

A. Brush blocks transfer electrical power from aircraft source to propeller blade deice boots.

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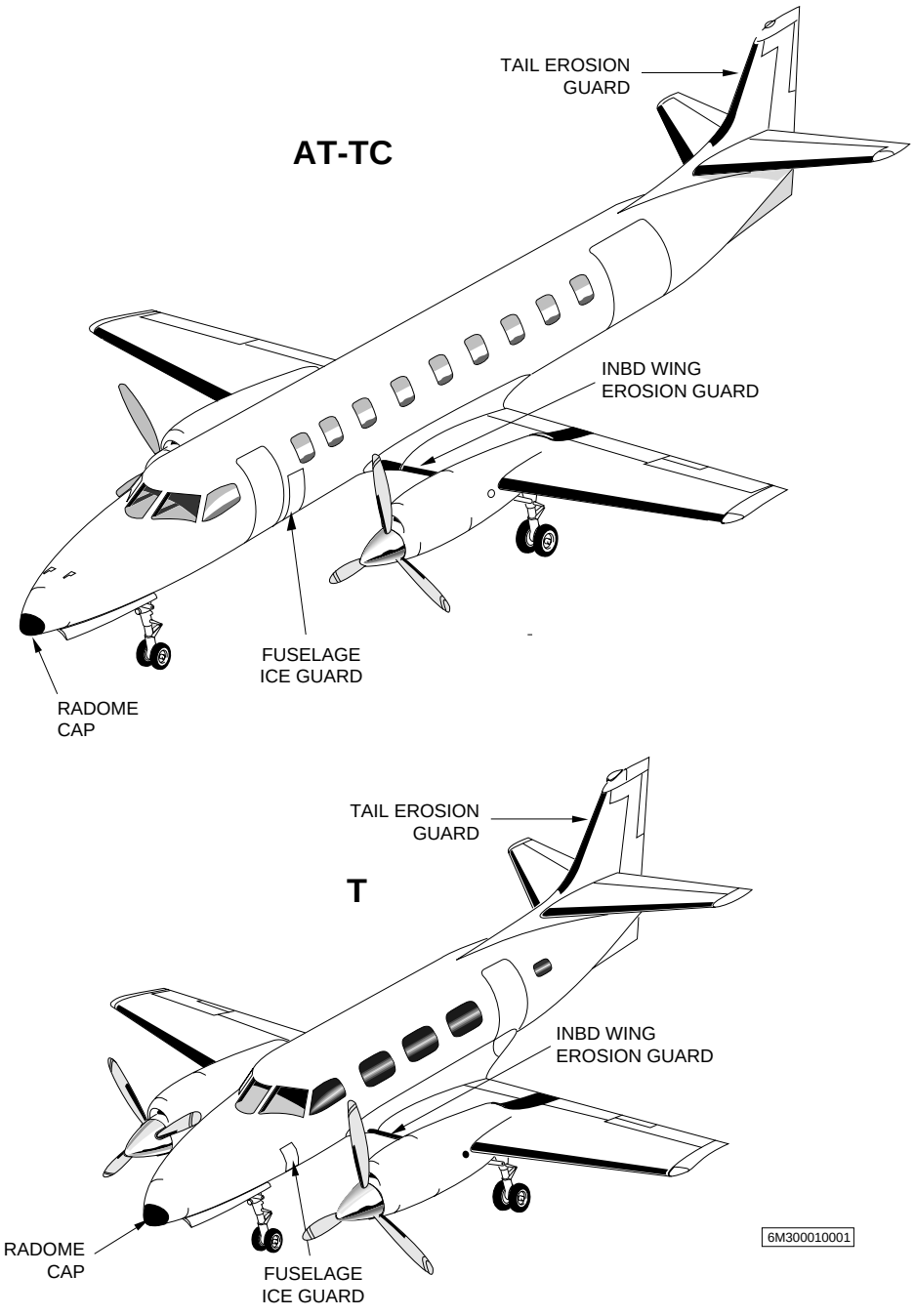
EROSION GUARD – DESCRIPTION AND OPERATION

1. General

A Protective guards are installed to prevent ice, rain, sand, and dirt erosion on airfoil leading edge surfaces and radome nose. Neoprene rubber and/or fabric guards are cemented to the radome, vertical stabilizer, and inboard wing section. Fiberglass or aluminum ice guards protect the fuselage in the propeller arc areas.



Erosion Guard – Ice and Rain
Figure 1

EROSION GUARD – DESCRIPTION AND OPERATION

2. Radome Cap

A A neoprene rubber and fabric radome cap is secured to the radome nose with contact cement.

3. Tail Erosion Guard

A A neoprene rubber and fabric tail erosion guard is secured to the leading edge of the vertical stabilizer with contact cement. The guard is 4.00 inches (10.16 cm) wide, 45 inches (114.3 cm) long, and 0.025 inches (0.0635 cm) thick.

4. Wing Root Fairing Erosion Shoe (LH, RH)

A A plexiglass wing root fairing erosion shoe is secured to the leading edge surface of the fiberglass wing root fairing with screws.

5. Inboard Wing Erosion Guard

A An inboard wing erosion guard is installed inboard of each nacelle on T & AT aircraft and optional on TC aircraft.

6. Fuselage Ice Guards

A The fuselage ice guards are riveted on the fuselage over sealer in the propeller arc area. Ice guards are customer option, made of either aluminum or fiberglass.

EROSION GUARD – MAINTENANCE PRACTICES

1. Removal/Installation – Erosion Guard

NOTE: The following removal/installation procedures apply to non-inflatable neoprene and fabric erosion guards installed on the vertical stabilizer, radome nose, inboard wing root, and aluminum and fiberglass fuselage ice guards.

WARNING: AVOID PROLONGED BREATHING OF VAPORS. IF DIZZINESS OR NAUSEA OCCUR, OBTAIN FRESH AIR IMMEDIATELY. AVOID CONTACT WITH SKIN OR EYES. USE SOLVENT RESISTANT GLOVES TO MINIMIZE SKIN EXPOSURE. WEAR SAFETY GLASSES TO MINIMIZE CHANCE OF EYE CONTACT. IF EYE CONTACT OCCURS, FLUSH EYES WITH WATER FOR 15 MINUTES AND SEE A PHYSICIAN. IF SKIN CONTACT OCCURS, WASH THOROUGHLY WITH SOAP AND WATER. IF SWALLOWED, DO NOT INDUCE VOMITING. GET IMMEDIATE MEDICAL ATTENTION.

ALL SOLVENTS AND CEMENTS USED ARE FLAMMABLE AND THEIR FUMES TOXIC. USE ONLY IN A WELL VENTILATED AREA AWAY FROM SPARKS OR FLAMES.

A. Materials Required – Rubber Erosion Guards

(1) Scotch-Grip 1300L Cement (3M Company).

(2) Methyl propyl ketone (MPK).

(3) Toluene.

(4) Paint Brushes, one inch and three inch.

(5) Rubber Roller, two inch (B.F. Goodrich, P/N 74-451-74) or equivalent.

(6) Metal Stitcher Roller, 1/4 inch B.F. Goodrich, P/N 74-451-89 or equivalent.

(7) Carpenter's Chalk Line and Tape Measure.

(8) Razor Blades and Holder.

(9) Cleaning Cloths (Lint Free).

(10) Masking Tape, One inch.

(11) Safety Pin or Needle.

(12) Isopropyl alcohol.

(13) 3M PA-1 plastic squeegee or equivalent.

(14) Liquid Detergent.

(15) Sponge or spray bottle.

(16) Marking Pen.

B. Removal – Rubber Erosion Guards

(1) Apply toluene or MPK to guard's upper corner line while applying tension to peel guard off.

NOTE: Apply a minimum amount of solvent. A trigger squirt bottle aids in applying solvent.

(2) Continue to soften adhesion line and pull away edge with uniform tension.

(3) Remove erosion guard.

NOTE: Clean dried cement off guard with Toluene or MPK. Clean cement off aircraft with KE9002 paint remover (B.F. Goodrich), Turco 388, or Kelite 2L.

C. Installation – Rubber Erosion Guards

(1) Mask off area around guard with one (1) inch tape. Allow 0.250 to 0.500 inch (0.635 to 1.27 cm) margin.

(2) Clean aircraft surface twice with MPK. On final cleaning, swab with clean solvent and wipe dry quickly (before solvent is dry) with a clean dry cloth to avoid leaving a film.

(3) Clean back side of guard with toluene. Change cloth often to prevent contamination of cleaned area.

(4) Thoroughly mix 1300L cement. If necessary, thin 1300L cement with toluene or MPK up to 5% volume.

NOTE: Do not apply 1300L cement if relative humidity is higher than 90%, or if temperature is below 50°F (10°C).

(5) Apply one (1) even brush coat to aircraft surface and back surface of guard.

(6) Allow cement to dry a minimum of one (1) hour at 50°F (10°C) or above when relative humidity is less than 75%. If relative humidity is 75-90%, allow additional drying time.

(7) Stir 1300L cement and apply a second coat to both surfaces. Allow to air dry for a minimum of one hour.

NOTE: Cement may be applied to aircraft and erosion guard up to 48 hours before installation providing parts are kept covered and clean.

(8) Mark leading edge centerline of wing or vertical stabilizer, using carpenter's chalk line, or straight edge and ball point pen.

(9) Mark erosion guard centerline on inside using carpenter's chalk line, or straight edge and ball point pen.

(10) Roll erosion guard with cemented side out, keeping edges even.

(11) To reactivate cement, apply toluene at one end of wing or tail, proceeding along centerline. Do not allow guard to touch reactivated cement until ready for permanent bonding.

(12) Reactivate cement along centerline of aircraft surface and guard a few inches at a time, pressing and rolling until guard is attached along entire centerline.

(13) When cement is tacky, press guard onto wing or stabilizer. Mate centerlines working out any air pockets with rubber roller. Keep bond line as straight as possible. Use rubber roller to roll down guard working along bond line toward trailing edge.

(14) If guard lifts and cement cobwebs after rolling, cement is too wet. Wait until cement is tacky, then repeat bonding procedures.

(15) If guard is attached off center (centerlines not coinciding), soften cement with MPK and reposition guard. Apply a minimum of solvent while applying sufficient tension to lift guard. Reapply any cement that pulls loose.

(16) Rubber roll entire guard applying pressure to ensure good bond.

(17) Roll trailing edges with metal stitcher roller.

(18) Pierce blisters with a common safety pin to remove entrapped air. Remove all blisters.

(19) Remove all masking tape and tape residue.

D. Removal – Radome Protective Cap

(1) Apply a minimum amount of solvent (toluene or MPK) to upper edge corner seam while applying tension to peel guard back. A trigger squirt bottle aids in applying solvent.

(2) Continue to soften adhesion line and pull trailing edge with uniform tension.

(3) Remove protective cap.

E. Installation – Radome Protective Cap

(1) Thoroughly wash refurbished, primed radome with isopropyl alcohol and wipe dry with lint free cloth.

(2) Place an orientation mark on top center of radome surface with a marking pencil.

(3) Position boot over radome with erosion liner still in place.

(4) Rotate boot to obtain optimum fit.

(5) Mark orientation line on top center of protective cap to match orientation mark at top center of radome surface. Make an additional mark at bottom center of radome and protective cap.

(6) Turn guard inside out and place over radome. Disregard orientation marks.

(7) Make a wetting solution of 25% isopropyl alcohol, and 75% water. Add one (1) teaspoon of liquid detergent per gallon of solution.

(8) Carefully remove transparent liner. Use a sponge or spray bottle filled with wetting solution and saturate exposed adhesive surface as liner is being removed.

(9) Ensure entire adhesive surface has been well saturated with wetting solution.

(10) Remove guard and completely saturate radome surface with wetting solution.

(11) Position the guard over the radome (adhesive side to radome) carefully aligning orientation marks. During this step, the boot will be right side in.

(12) After proper alignment has been made, squeegee out wetting solution, starting at top center and working outboard and down.

(13) Pierce blisters with a common safety pin to remove entrapped air or wetting solution and squeegee cap flat against the radome surface.

(14) At radome station 6.50, wrap one (1) inch masking tape around outside circumference of protective cap. Position tape so that the forward edge aligns with 6.50 inch line.

(15) Trim guard along forward edge of masking tape. Be careful not to cut into radome skin.

(16) Wash guard with isopropyl alcohol. Wipe dry and paint with polyurethane paint.

F. Removal – Fuselage Ice Guards

(1) Remove rivets attaching ice guard to fuselage.

(2) Loosen ice guard at edge with putty knife.

(3) Remove ice guard.

G. Installation – Fuselage Ice Guards

(1) Clean sealant residue from fuselage and ice guard.

(2) Trim ice guard as needed to fit location.

(3) Drill and countersink ice guard rivet holes with #30 drill bit. Use hole locator to match fuselage holes.

(4) Clean fuselage and back side of ice guard with MPK.

(5) Mix PR-1422B-2 sealing compound. Follow container mixing instructions, temperature limitations, and shelf life restrictions.

(6) Fay seal ice guard to fuselage by coating both with sealant. Install ice guard wet, forcing sealer out around entire edge during riveting.

(7) Rivet ice guard to fuselage.

(8) Tape 0.250 inch (0.635 cm) from edge of ice guard with one (1) inch masking tape.

(9) Fillet seal edge of ice guard with PR-1422B-2 sealant.

(10) Remove all masking tape.

2. Cleaning/Painting – Erosion Guards

A. Cleaning/Painting – Rubber Erosion Guards

CAUTION: DO NOT USE PETROLEUM PRODUCTS AS CLEANING AGENTS. THEY DAMAGE RUBBER.

(1) Clean neoprene and fabric erosion guards with mild soap and water using a soft scrub brush. In cold weather it may be necessary to warm the wash solution or use warm air from a portable ground heater. Temperature of cleaning solution should not exceed 140°F (60°C).

B. Surface Coating

(1) Use silicone based coatings (ICEX, B.F. Goodrich or equivalent) to lower ice adhesion strength. Follow product instructions.

(2) Use erosion coatings (Age-Master, B.F. Goodrich or equivalent) to slow weathering and ozone damage. Follow product instructions.

C. Surface Refurbishing

(1) Weathered erosion guards can be resurfaced using B.F. Goodrich kit 74-451-L. It will smooth roughened surfaces and fill small cracks when applied per directions furnished with kit.

D. Cleaning/Painting – Radome Cap

WARNING: OBSERVE MANUFACTURERS SAFETY PRECAUTIONS.

(1) Wipe the boot surface with a rag saturated with methyl propyl ketone (MPK) or equivalent before painting.

NOTE: Do not prime before painting.

(2) Allow MPK or equivalent to dry approximately 10-30 minutes. If MPK or equivalent use is not desirable, skuff lightly with #3774 Scotchbrite, followed by a cleanup with ethanol.

(3) Apply top coat directly to the cap.

3. Inspection/Check – Erosion Guard

A. During regularly scheduled inspections, check guard for cuts, blisters, perforations, and edge lifting. The guard should be replaced at the first sign of damage.

B. If the polyurethane radome cap is left unpainted, a paste wax should be periodically applied to the surface to prevent staining.

4. Approved/Repair – Erosion Guard

A. Approved Repair

(1) Minor surface damage can be repaired using B.F. Goodrich kit #74-451-AA. Follow instructions furnished with kit.

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DEICE BOOT – DESCRIPTION AND OPERATION

1. General

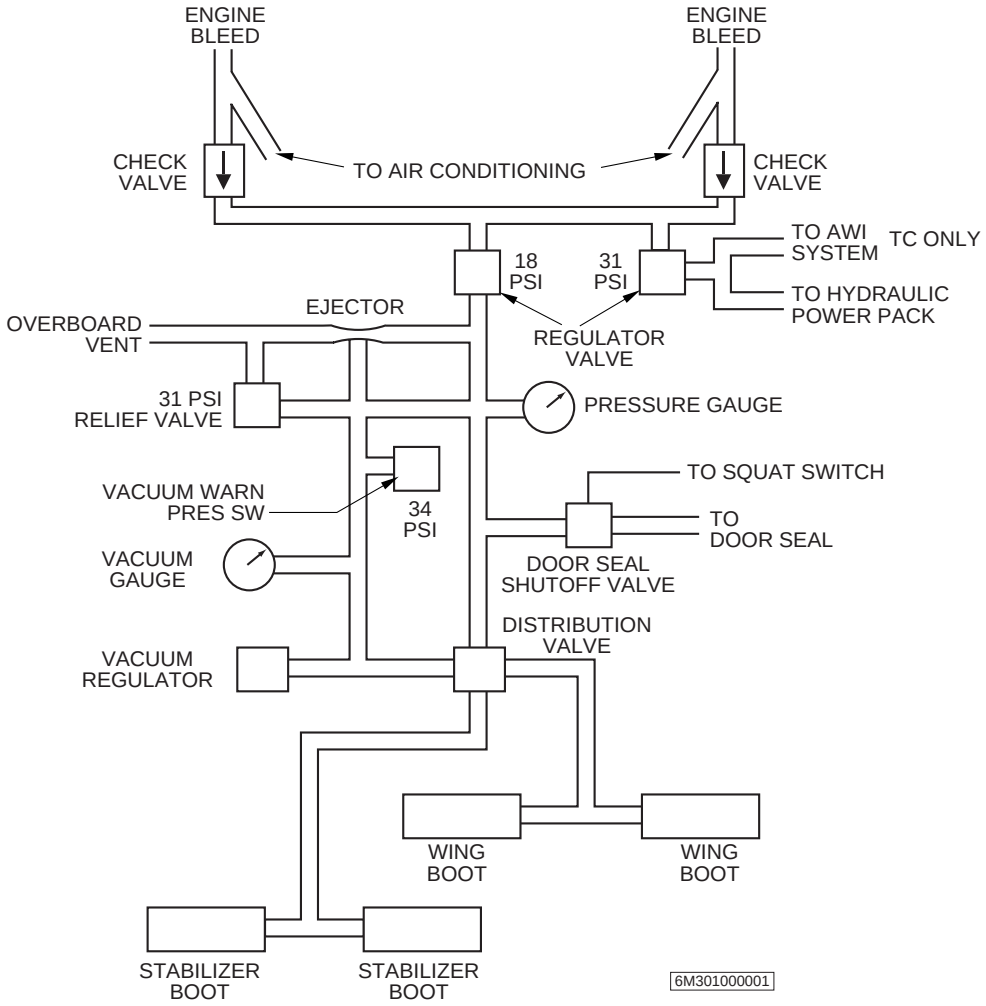
A. Pneumatic deice boots remove ice accumulation on wing and horizontal stabilizer leading edges.

B. Neoprene rubber and fabric deice boots are cemented to the wing and stabilizer leading edges and connected by plastic and aluminum plumbing to the deice boot distribution valve.

C. The valve connects to the pneumatic system (see Chapter 21) and the vacuum system (see Chapter 37) to inflate and deflate the deice boots as shown in Figure 2. This action loosens the ice allowing its removal by the slipstream.

D. The deice boot is controlled by a three position (AUTO, OFF, MANUAL) deice boots switch located on the pilot's switch panel. AUTO position activates the deice relay and deice boot timer for automatic sequencing of deice boot inflation. MANUAL position is spring loaded and inflates all deice boots together.

E. Deice boot power comes from circuit breakers on either LEFT or RIGHT ESSENTIAL 28 VDC BUS as selected by the DEICE BOOTS BUS TRANSFER switch.



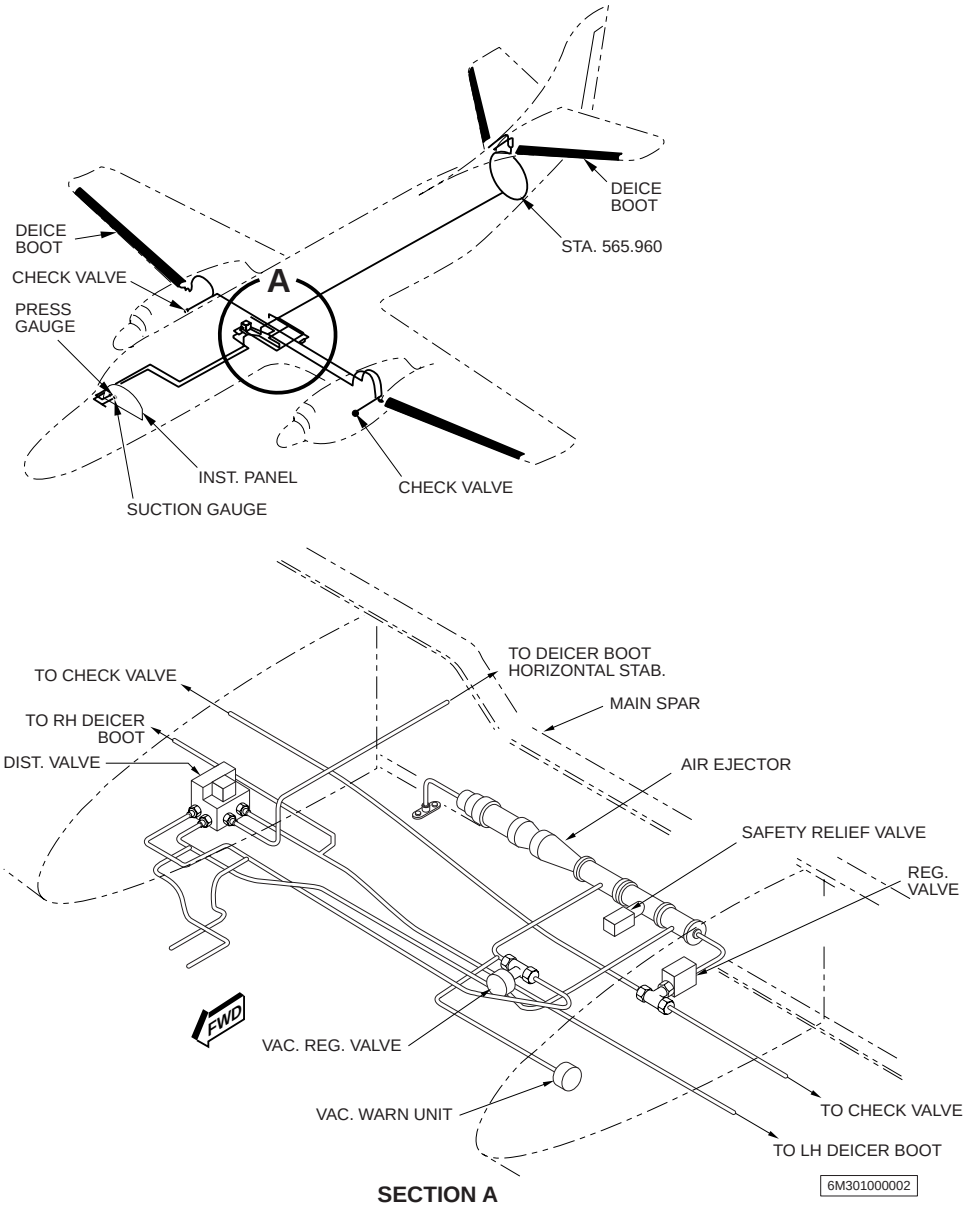
Deice Boot Pressure System
Figure 1

DEICE BOOT – DESCRIPTION AND OPERATION

2. Wing and Fuselage Plumbing

A. Wing and fuselage plumbing connects pneumatic pressure and vacuum from the deice boot distributor valve to the deice boot. Plumbing to the horizontal stabilizer deice boot enters the fuselage at a bulkhead union under the center section and exits the fuselage through a pressurization seal tube assembly under the forward vertical stabilizer fairing.

B. A "Y" shaped yoke divides the plumbing just forward of the horizontal stabilizer spar. Plumbing connects to the deice boot inside the stabilizer at inboard leading edge. Plumbing to the wing boot divides just left of the deice boot distributor valve and is routed over the exhaust duct to a fitting beneath the wing ice detection light. A tube and hose assembly connect to the main wing deice boot hose connection at that point.



SECTION A

Deice Boot System
Figure 2

DEICE BOOT – DESCRIPTION AND OPERATION

3. Deice Boot (Figures 1 and 2)

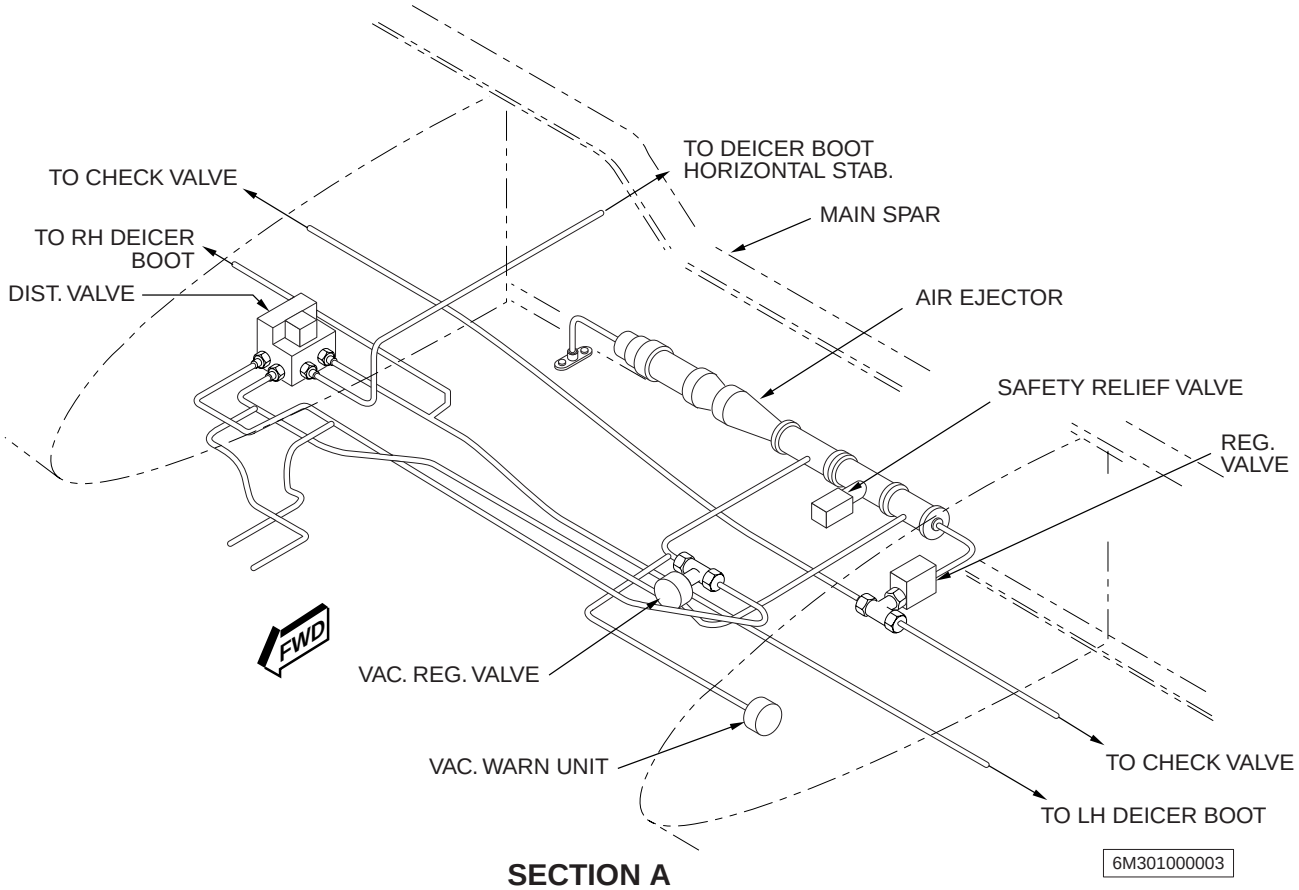
A. Deice boots, located on the leading edge of the wing and horizontal stabilizer are constructed of neoprene rubber and fabric layers incorporating simultaneous spanwise inflation tubes. A hose fitting on each deice boot connects to plumbing from the pneumatic pressure and vacuum systems. When inflated they break up accumulated ice. When not in use the deice boot is held tightly against the wing or stabilizer leading edge by vacuum.

B. Deice boots are cemented to the leading edge of the wing or stabilizer. A fuel resistant sealant is used to fair in the leading edge of the deice boot with the wing.

4. Deice Boot Distributor Valve (Figure 3)

A. A deice boot distributor valve is located in the wing center section of the right wing root rib, under the right forward wing to fuselage fairing. Two independent electric solenoid valves separately control inflation of the wing and horizontal stabilizer deice boot.

B. The valve opens to inflate the deice boot when 28 VDC is applied. When the voltage is terminated, a spring closes the valve and vacuum is applied to deflate the boot. The deice boot distributor valve continues to apply vacuum while the system is OFF to keep the deice boot held tightly against the leading edges.



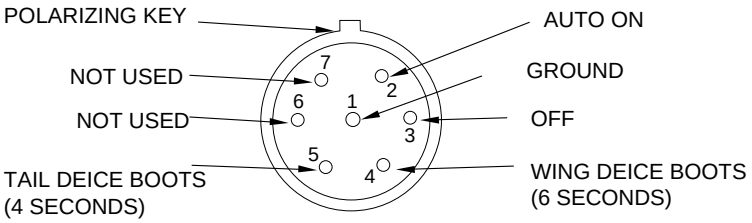
SECTION A

Deice Boot Distributor Valve and Center Section Plumbing
Figure 3

5. Deice Boot Timer (Figure 4)

A. A deice boot timer, located on electrical panel EP4 forward of the copilot's rudder pedals, is activated by a deice relay when DEICER BOOTS switch is in AUTO. The timer closes the solenoid valves in the deice boot distributor valve in a 3 minute repeating sequence. The sequence always starts with 6 seconds of wing deice boot inflation, followed by 4 seconds of horizontal stabilizer inflation, then completed by 170 seconds of system rest with all deice boots deflated.

B. When the system is turned OFF, the deice boot timer returns to recycle position automatically, so it will start with wing inflation, regardless of what part of sequence it is in when turned off.



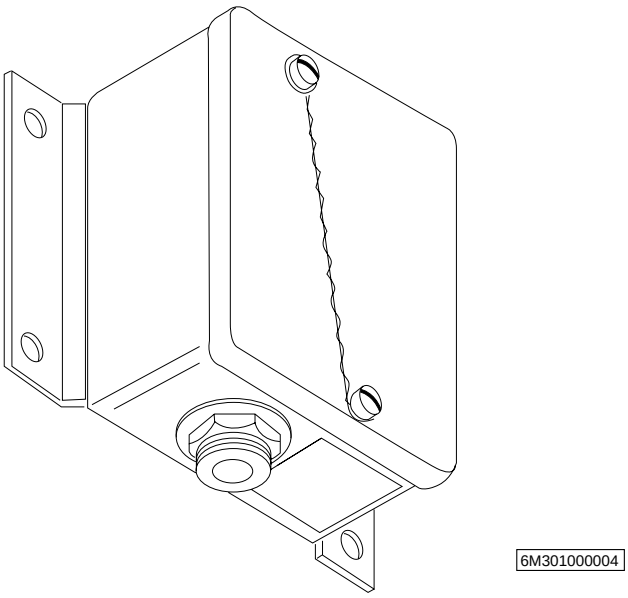
ALL TIMES ±10% AT 28 VDC

RECEPT. PIN	"TIMES" FOR ONE SEQUENCE (SECONDS)		
	6	4	170
4 (WING)	ON	OFF	OFF
5 (TAIL)	OFF	ON	OFF

3 MINUTES

recycle

TIMER SEQUENCE CHART

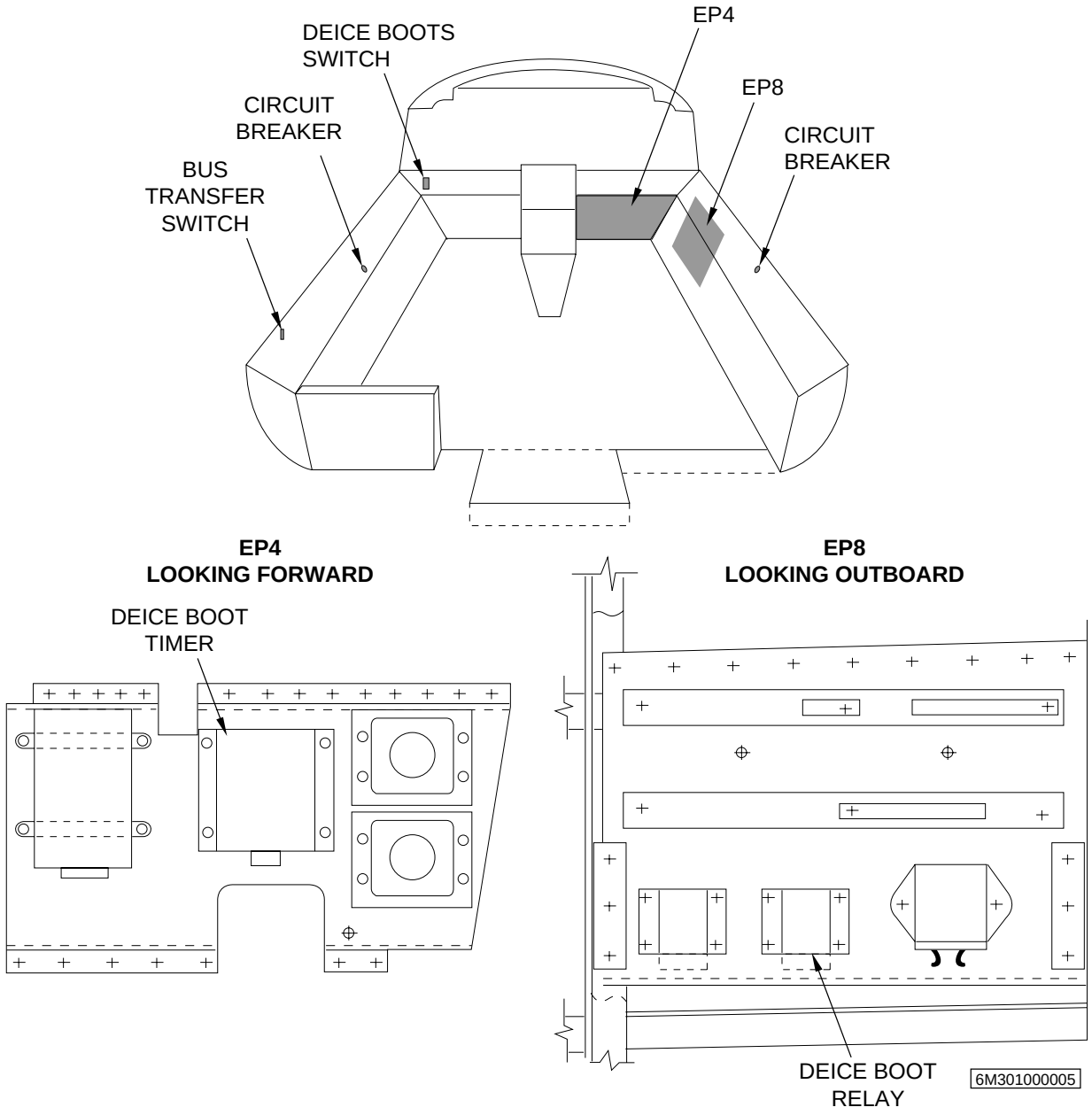


Deice Boot Timer
Figure 4

6. Deice Boot Relay (Figure 5)

A. A deice relay is located on electrical panel EP8 in the right hand console. When the DEICER BOOTS switch is in AUTO, the relay is closed by 28 VDC, connecting the deice boot timer to the deice boot distributor valve and activating the timed sequence.

B. When the DEICER BOOTS switch is moved to OFF or MANUAL, the deice relay terminates power to the deice boot timer return pin so the timer will return to recycle.



Deice Boot Electrical Components
Figure 5

7. Deice Boot Switch (Figure 5)

A. A three position (AUTO, OFF, MANUAL) deice boot switch is located on the pilot's switch panel EP5.

(1) AUTO position activates the deice relay and deice boot timer for automatic deice boot operation.

(2) MANUAL position is spring loaded to OFF. When the switch is held down, electrical power is directed to both solenoid valves of the deice boot distributor valve inflating all deice boots.

(3) OFF position stops deice boot functioning, but continues to power the deice boot timer recycle function.

8. Deice Boot Bus Transfer Switch (Figure 5)

A. A DEICE BOOT bus transfer switch, located on the left console ESSENTIAL BUS TRANSFER is used to select power for the deice boot from circuit breakers on either left or right essential buses. The switch is normally in the left position.

9. Deice Boot Circuit Breakers (Figure 5)

A. Circuit breakers are located on the LEFT and RIGHT ESSENTIAL BUSES. Deice boots may be powered from either bus.

10. Deice Pressure Transducer

A. A deice pressure transducer, located on the right side of the center section, on the wing root rib under the right forward wing root fairing, is powered by 28 VDC from the deice boot bus transfer switch and provides an electrical signal to operate the deice pressure indicator in the cockpit.

11. Deice Pressure Indicator

A. A deice pressure indicator, located on the instrument panel right side, allows the flight crew to monitor pneumatic pressure. An electrical signal from the deice pressure transducer powers the indicator. Lighting is internal, powered by the RH 5 volt light dimmer power supply and is controlled by the copilot's flight instrument dimmer knob on the RH console switch panel.

DEICE BOOT – FAULT ISOLATION

1. Troubleshooting – Deice Boot

A. To assist in the isolation of faults, the following troubleshooting block diagrams are provided.

B. For additional information and schematic diagrams refer to SA226 Wiring Manual.

C. The prerequisites at the beginning of each diagram require an Operational Test to determine which problem statement applies.

D. In the event of multiple problems, correct the faults in the order they appear on the following pages.

DEICE BOOT – FAULT ISOLATION

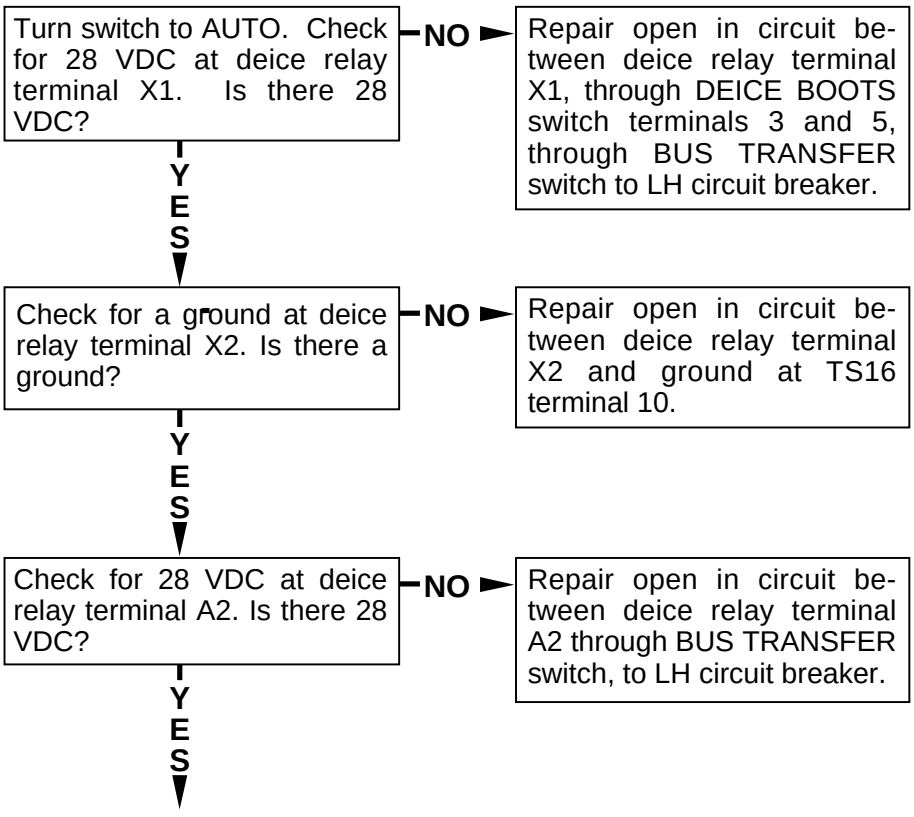
PROBLEM:

DEICE BOOTS DO NOT OPERATE IN AUTO. BOOTS INFLATE/DEFLATE NORMALLY IN MANUAL. DEICE BOOT TIMER DOES NOT RECYCLE.

PREREQUISITES:

Test deice boot. See Adjustment/Test – Deice Boot.

START:



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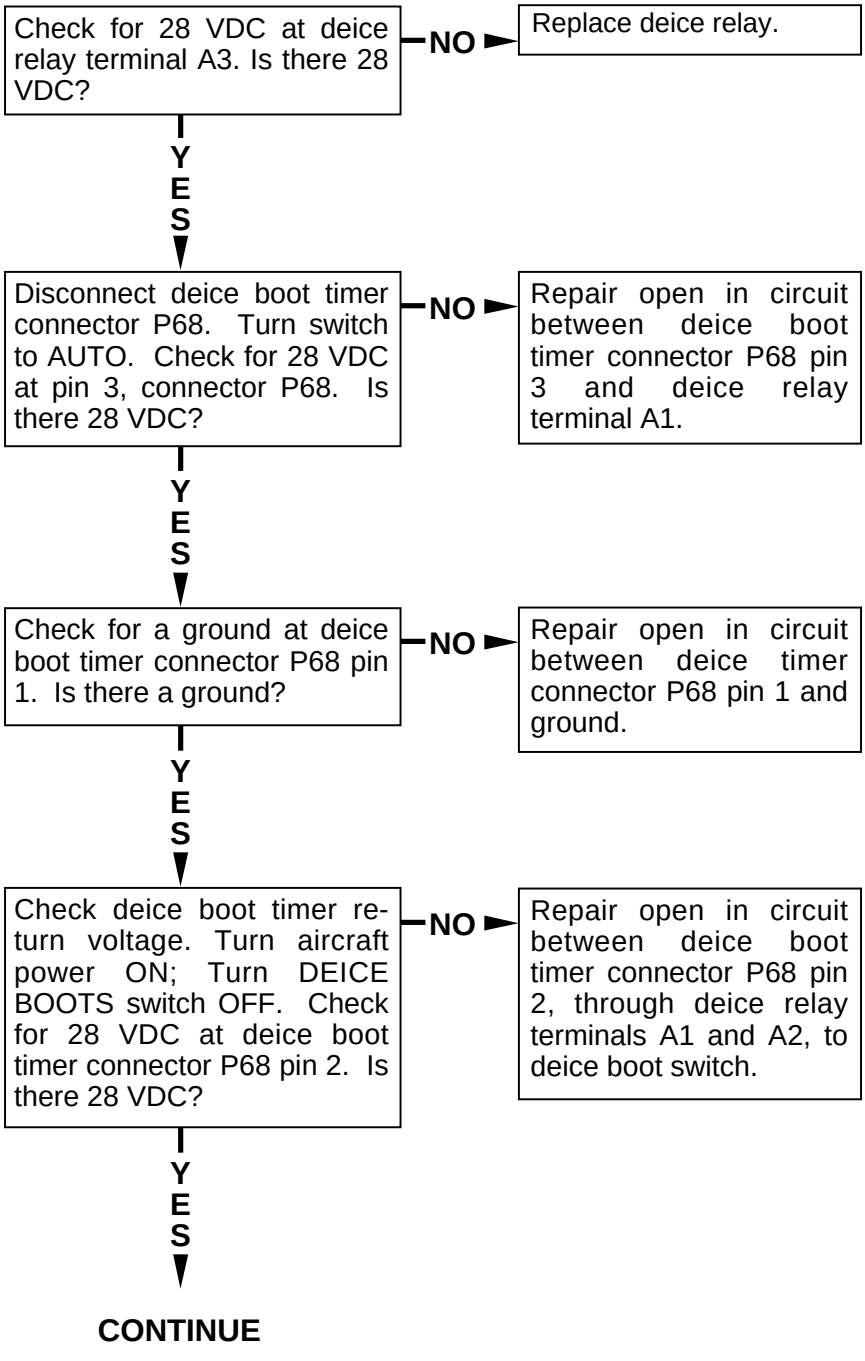

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DEICE BOOT – FAULT ISOLATION

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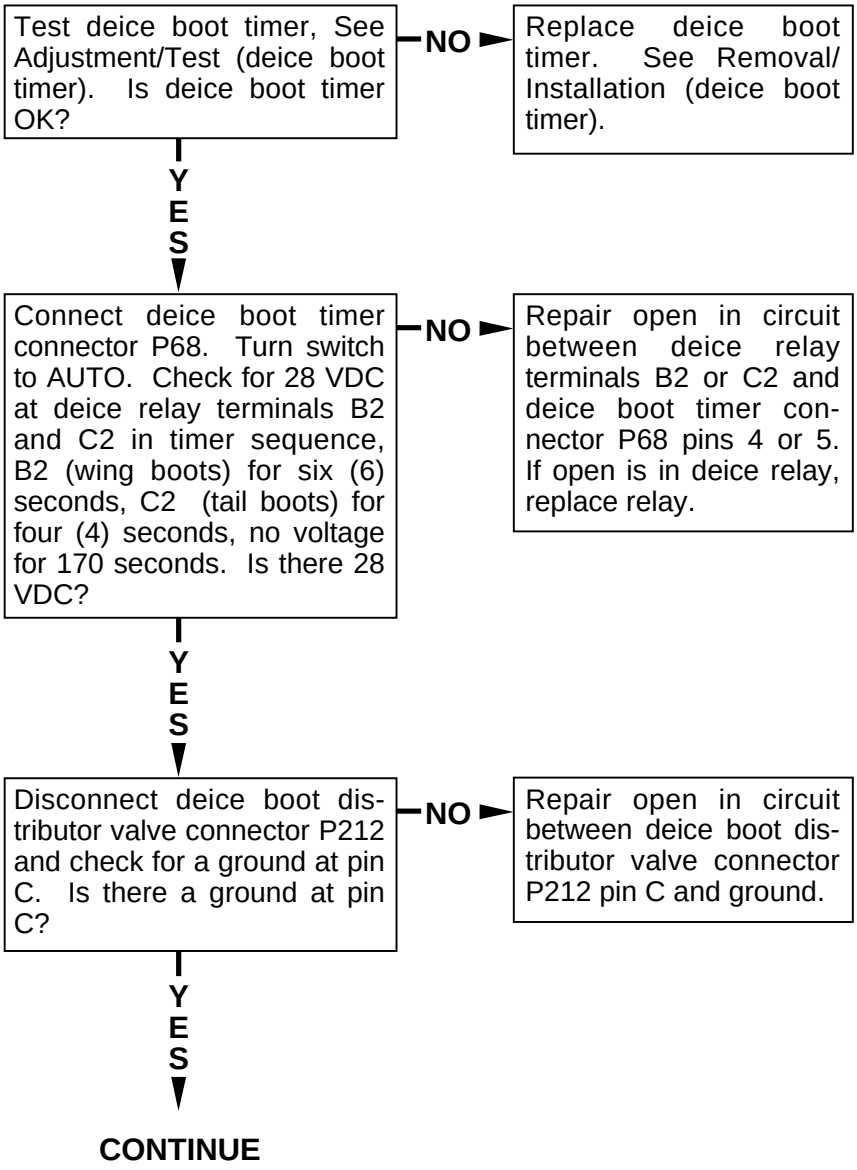

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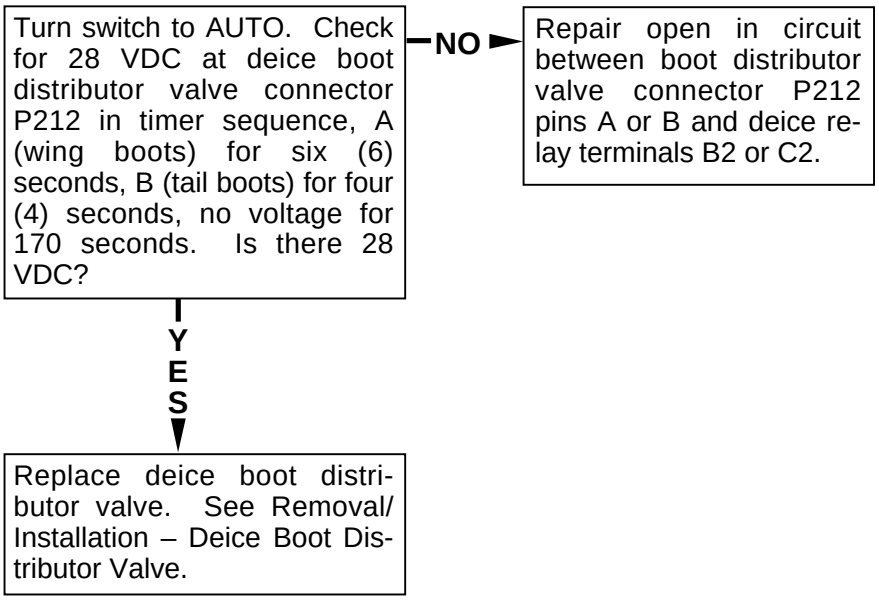

END

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DEICE BOOT – FAULT ISOLATION

PROBLEM:

DEICE BOOT DOES NOT INFLATE OR DEFLATE FULLY IN AUTO OR MANUAL.

PREREQUISITES:

Test SAS vane deice. See Adjustment/Test – Deice Boot.

START:

Check deice pressure indicator and suction indicator. Are pneumatic and vacuum pressure normal?

NO → Repair pneumatic system or vacuum system.

YES ↓

Disconnect deice boot plumbing line on left side of deice boot distribution valve (forward line tail boots, aft line wing boots). Connect pressure/ vacuum gauge and operate system. See Adjustment/Test Deice Boot. Does gauge indicate normal system pressure and vacuum?

NO → Replace deice boot distributor valve. See Removal/Installation (deice boot distributor valve).

YES ↓

Reconnect line at deice boot distributor valve. Disconnect line at deice boot hose connection. Connect pressure/vacuum gauge to hose and operate sys-tem. See Adjustment/Test Deice Boot. Does gauge indicate normal system pressure and vacuum?

NO → Repair leak in deice boot plumbing between deice boot hose connection and deice boot distributor valve.

YES ↓

Repair leak in deice boot. See Approved Repairs – Deice Boot, or replace deice boot.

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END

DEICE BOOT – MAINTENANCE PRACTICES

1. Removal/Installation – Deice Boot

A. Removal – Deice Boot Timer

(1) Remove copilot seat IAW chapter 25.

(2) Remove safety wire and disconnect deice boot timer connector.

(3) Remove screws securing timer to electrical panel EP4.

(4) Remove deice boot timer.

B. Installation – Deice Boot Timer

(1) Position deice boot timer on electrical panel EP4.

(2) Secure timer to panel with screws.

(3) Connect and safety wire deice boot timer connector.

(4) Install copilot seat IAW chapter 25.

(5) Functional Test deice boot, see Adjustment/Test – Deice Boot.

C. Removal – Deice Boot Distributor Valve

(1) Remove RH forward fuselage to wing fairing and RH forward wing center section access panel.

(2) Loosen clamps on rubber hoses connected to deice boot distributor valve ports. Slide rubber hoses off ports.

(3) Remove safety wire and disconnect deice boot distributor valve connector.

(4) Remove safety wire and screws securing valve to wing root brackets.

(5) Remove deice boot distributor valve.

D. Installation – Deice Boot Distributor Valve

(1) Position deice boot distributor valve on wing root brackets.

(2) Install and safety wire screws securing valve.

(3) Connect and safety wire deice boot distributor valve connector.

(4) Slide rubber hoses on valve ports, install clamps on hoses.

(5) Check deice boot distributor valve for leaks and proper function, see Adjustment/Test–Deice Boot.

(6) Install RH forward fuselage to wing fairing and RH forward center section access panel.

E. Materials Required – Deice Boot

(1) Scotch-Grip 1300L Cement (3M Company).

(2) Fuel Barrier Cement, A-851-B (B.F. Goodrich Co.).

(3) Methyl propyl ketone (MPK).

(4) Toluene.

(5) Scotch-Seal EC801 Filler Compound (3M Company) or alternate PR-1440 (PRC) Product Research and Chemical Corp.

(6) Masking Tape, 1.00 inch.

(7) Paint Brushes, 1.00 inch and 3.00 inch.

(8) Industrial Razor Blade Knife.

(9) Hook Knife (B.F. Goodrich, P/N 74-451-45 or equivalent).

(10) Rubber Roller, 2.00 inch (B.F. Goodrich, P/N 74-451-74 or equivalent).

(11) Metal Stitcher Roller, 0.250 inch (B.F. Goodrich, P/N 74-451-89 or equivalent).

(12) Carpenter's Chalk Line and Tape Measure.

(13) Cleaning Cloths (Lint Free).

(14) Hypodermic needle.

F. Removal – Wing and Horizontal Stabilizer Deice Boot

(1) Remove inspection panel at inboard end of wing to gain access to deice boot hose connection and remove hose.

WARNING: AVOID PROLONGED BREATHING OF VAPORS. IF DIZZINESS OR NAUSEA OCCURS, OBTAIN FRESH AIR IMMEDIATELY. AVOID CONTACT WITH SKIN OR EYES. USE SOLVENT RESISTANT GLOVES TO MINIMIZE SKIN EXPOSURE. WEAR SAFETY GLASSES TO MINIMIZE CHANCE OF EYE CONTACT. IF EYE CONTACT OCCURS, FLUSH EYES WITH WATER FOR FIFTEEN (15) MINUTES AND SEE A PHYSICIAN. IF SKIN CONTACT OCCURS, WASH THOROUGHLY WITH SOAP AND WATER. IF SWALLOWED, DO NOT INDUCE VOMITING. GET IMMEDIATE MEDICAL ATTENTION.

ALL SOLVENTS AND CEMENTS USED ARE FLAMMABLE AND THEIR FUMES TOXIC. USE ONLY IN A WELL VENTILATED AREA AWAY FROM SPARKS OR FLAMES.

(2) Apply minimum amount of methylpropyl ketone (MPK) to upper corner of deice boot while applying tension to peel corner back. A trigger squirt bottle aids in applying solvent.

(3) Separate deice boot from wing for a distance of four (4) inches (10.16 cm) from trailing edge along entire length.

(4) Continue to soften adhesion line with MPK. Apply uniform tension to pull deice boot off. If deice boot is to be scrapped, it may be removed in spanwise sections parallel to inflation tubes.

(5) Remove old cement with paint remover and clean airfoil leading edge thoroughly with MPK.

G. Installation – Wing and Horizontal Stabilizer Deice Boot

NOTE: The following boot installation methods are recommended by M7 Aerospace. For B.F. Goodrich installation procedures, refer to B.F. Goodrich Reports and Service Letters.

WARNING: VERIFY AIRCRAFT IS ELECTRICALLY GROUNDED TO PREVENT SPARKS WHICH COULD IGNITE SOLVENT VAPORS.

(1) Mask area around deice boot with 1.00 inch tape, allowing 0.250 to 0.500 inch (0.635 to 1.27 cm) margin.

(2) Clean leading edge surface twice with MPK. On final cleaning, swab with clean solvent and wipe dry before solvent dries.

(3) For each installation, prepare a 1.00 inch by 8.00 inch (2.54 by 20.32 cm) adhesion test specimen, from excess material trimmed off deice boot(s). Cement test strip adjacent to deice boot(s). Install test specimen using identical procedures as those used to bond deice boot(s). Leave 1.00 inch (2.54 cm) of strip uncemented for test purposes.

CAUTION: A-851-B FUEL BARRIER CEMENT IS REQUIRED ON WING DEICE BOOTS OVER INTEGRAL FUEL TANKS TO AVOID FUEL DEGRADATION OF 1300L INSTALLATION CEMENT.

(4) On wing surface sections over integral fuel tanks, apply A-851-B Fuel Barrier Cement.

(5) Thoroughly mix A-851-B cement (if necessary, thin to brushing consistency). Brush on one (1) even coat, allow one half (1/2) hour to dry, and brush on a second coat. Allow second coat two (2) hours minimum drying time before applying 1300L installation cement.

(6) Clean back surface of deice boot twice with MPK.

NOTE: Change cloth frequently to avoid contamination of cleaned areas. Do not saturate back surface too heavily with solvent nor scrub repeatedly. Allow to dry thoroughly before cementing.

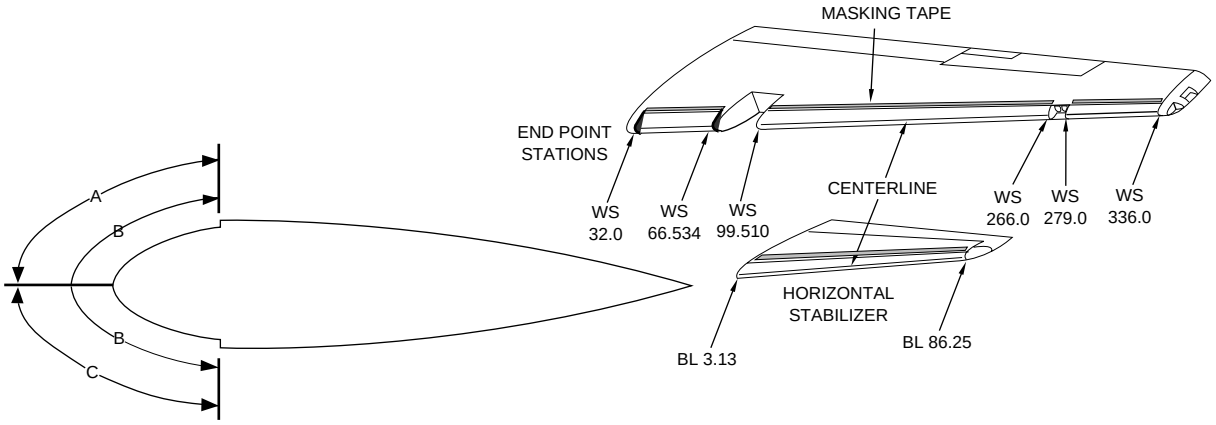
(7) Thoroughly mix 1300L installation cement. If necessary, thin cement with Toluene or MPK up to 5% by volume.

(8) Do not apply 1300L cement if relative humidity is higher than 90% or if temperature is below 50°F (10°C).

(9) Apply one (1) even brush coat of 1300L cement to wing leading edge and back surface of deice boot.

(10) Allow cement to dry minimum of one (1) hour at 50°F (10°C) or when relative humidity is less than 75%. If relative humidity is 75% to 90%, allow additional drying time.

(11) Determine end points to place chalk line on leading edge of airfoil (Figure 201). Snap chalk line or mark straight edge with a ballpoint pen.



WINGS - MEASURE DIMENSION "A" OR "C" TO CENTERLINE

TAIL - MEASURE DIMENSION "B" FROM TOP TO BOTTOM

LOCATION	STATION	DIMENSION	
WING, INBD, INBD	WS 32.0	A = 5.875	C =10.500
WING, INBD, OUTBD	WS 66.534	A = 5.875	C =10.500
WING, CENTER INBD	WS 99.510	A = 5.250	C = 7.250
WING, CENTER, OUTBD	WS 266.0	A = 3.250	C = 4.500
WING, EXTNSN, INBD	WS 279.0	A = 4.500	C = 4.625
WING, EXTNSN, OUTBD	WS 336.0	A = 4.500	C = 4.625
STABILIZER, INBD	BL 3.13	B = Measure top to bottom	
STABILIZER, OUTBD	BL 86.25	B = Measure top to bottom	
NOTE: Dimensions are in inches.			

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Measuring Chalk Line End Points
Figure 201

(12) Stir 1300L cement and apply a second coat to both surfaces. Allow to air dry for a minimum of one (1) hour.

NOTE: Cement may be applied to aircraft and deice boots up to 48 hours before installation providing parts are covered and kept clean.

(13) Trim inboard end of wing deice boot to fit into nacelle. Use template made as shown in Figure 202.

(14) Beginning with outboard end, roll deice boot toward end, with cemented side out.

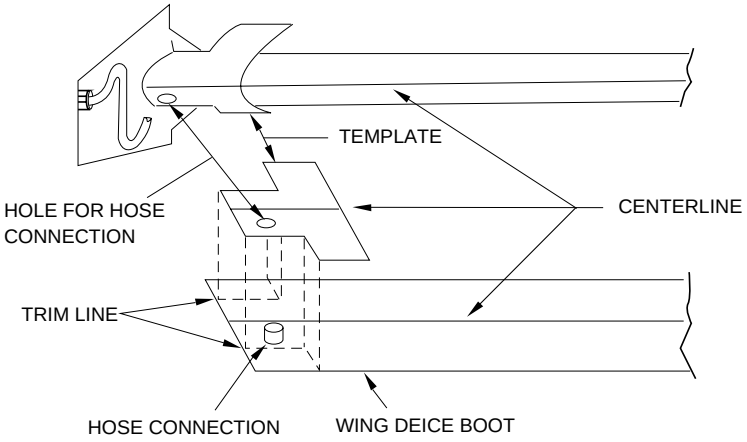
(15) To reactivate cement, apply toluene to cement with a clean lint free cloth. Do not allow deice boot to contact reactivated cement until ready for permanent bonding.

(16) Reactivate cement strip along centerline of deice boot 3.00 inches wide and 18.00 inches (45.72 cm) long. Ensure that centerline of boot aligns with centerline of airfoil.

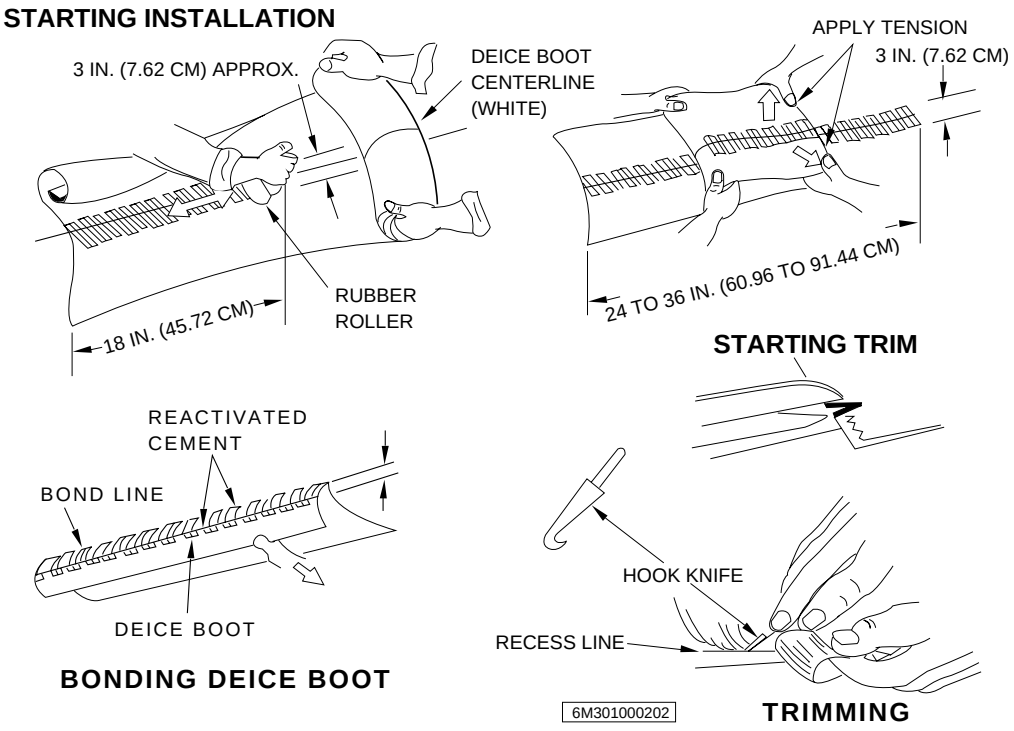
(17) Reactivate cement on matching section of airfoil leading edge.

(18) Unroll deice boot against airfoil leading edge while applying light tension.

(19) When cement is tacky, press deice boot to airfoil making sure centerlines are lined up (Figure 202). Keep bond line as straight as possible.



ATTACHING DEICE BOOT CENTERLINE



Installation Deice Boot
Figure 202

(20) On horizontal stabilizer or outboard wing deice boot, ensure that hose connections will mate with hole in leading edge.

(21) Use 2.00 inch rubber roller to press deice boot firmly to airfoil along centerline in area where cement has been reactivated.

(22) Reactivate cement strip on boot and airfoil centerline 3.00 inches (7.62 cm) wide for additional 24.00 to 36.00 inches (60.96 to 91.44 cm) (Figure 202).

CAUTION: AVOID USING EXCESS SOLVENT. AVOID TWISTING, BENDING, OR JERKING LOOSE FROM BONDED AREA.

(23) If deice boot should attach off center (centerlines not lined up), apply MPK with a small brush or trigger squirt bottle to soften bonded line. Apply a minimum of solvent to bond line while applying sufficient tension to peel back deice boot. Reapply any cement that pulls loose.

(24) Continue to reactivate cement along deice boot centerline until entire length is attached. Roll centerline area firmly with rubber roller.

(25) Starting at inboard edge, hold deice boot away from airfoil to reveal bond line. Reactivate cement strip 3.00 inches (7.62 cm) wide by 60.00 inches (152.4 cm) long, working from bond line toward trailing edge (Figure 202).

(26) Keep toluene moistened cloth tight in fold of bond line, wiping first along airfoil leading edge, then along corresponding surface of deice boot.

(27) Roll deice boot down with rubber roller starting at bond line and rolling spanwise while working toward trailing edge. Hold deice boot back while roller does work of mating surfaces.

(28) If deice boot lifts after rolling and cement cobwebs, cement is too wet. Wait until cement is tacky and repeat bonding procedures.

(29) Continue to reactivate cement along bond line until deice boot reaches end of wing. Use knife to trim deice boot after starting cut with scissors or razor blade (Figure 202).

(30) Rubber roll spanwise over entire surface of deice boot applying pressure to ensure a good bond. Roll edges outside of inflatable area with narrow stitcher roller.

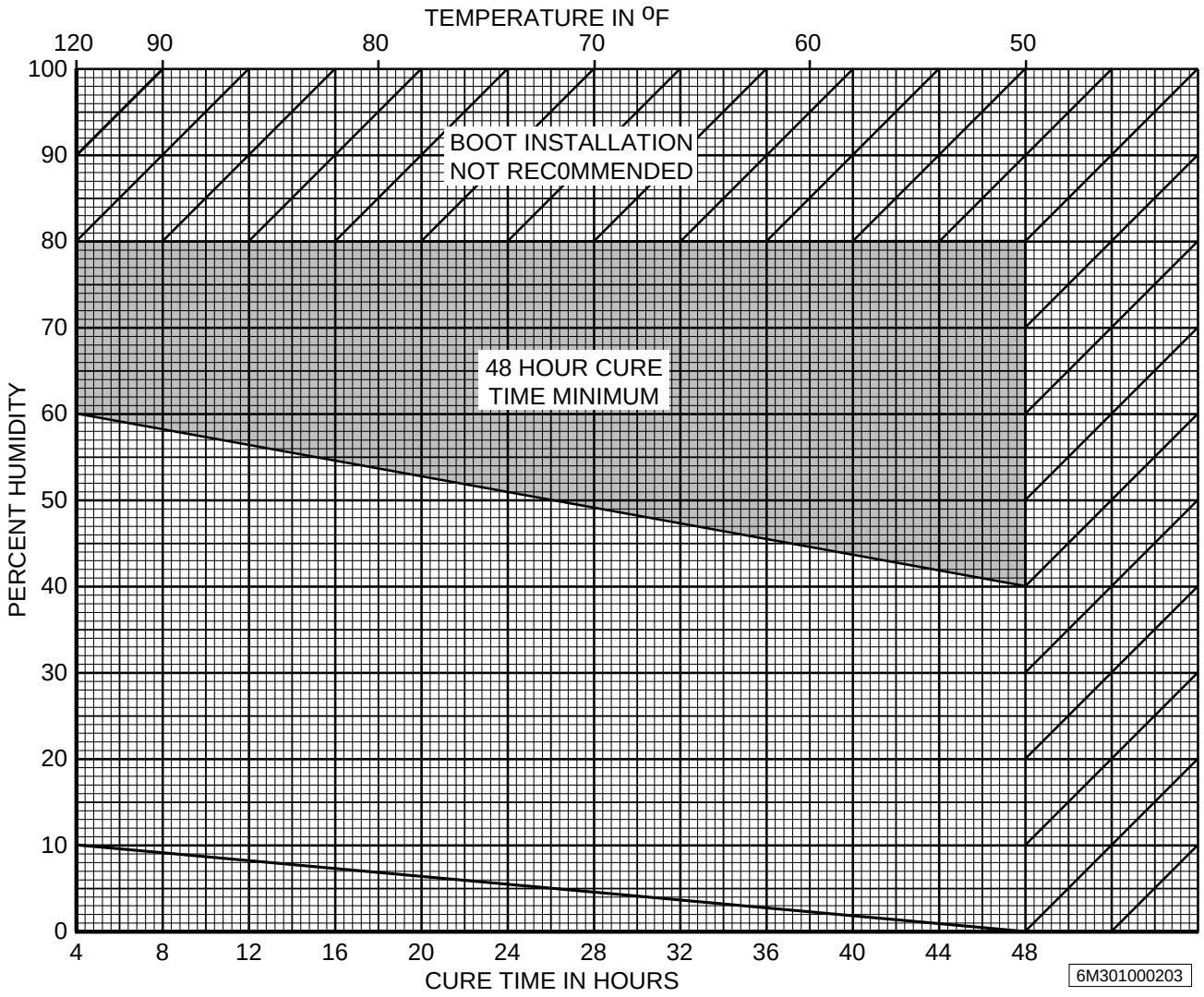
(31) Apply masking tape to deice boot and aircraft leading edge surface 0.125 to 0.250 inch (0.3175 to 0.635 cm) from edge of deice boot.

(32) Mix EC-801 or PR-1440 filler compound according to instructions on container. Apply filler with small brush around edges of deice boot to fair it to adjacent surface

NOTE: Conductive edge sealer may be used to prevent static discharge. Apply A56B for neoprene Di-Icers or apply 74-451-K on Estane De-Icers.

(33) Remove all masking tape and tape residue.

(34) Refer to Figure 203 for proper cure time of deice boots. Verify this cure time with an adhesion test using a test strip attached to the aircraft skin adjacent to installed boot.



Deice Boot Cure Time
Figure 203

H. Installation – Wing and Horizontal Stabilizer Deice Boot (Alternate Procedure)

NOTE: This alternate procedure is applicable to deice boot installations in which the 48 hour dry time prior to deice boot inflation required by 1300L cement is not available. Use of Bostik cement shortens the dry time prior to deice boot inflation to 4 hours. The following boot installation methods are recommended by M7 Aerospace. For B.F. Goodrich installation procedures, refer to B.F. Goodrich Reports and Service Letters.

Instructions must be followed exactly, to obtain shortened dry time.

Do not attempt installation when temperature in below 50°F (10°C), or relative humidity is above 90%.

DEICE BOOT – MAINTENANCE PRACTICES

WARNING: VERIFY AIRCRAFT IS ELECTRICALLY GROUNDED TO PREVENT SPARKS WHICH COULD IGNITE SOLVENT VAPORS.

AVOID PROLONGED BREATHING OF VAPORS. IF DIZZINESS OR NAUSEA OCCUR, OBTAIN FRESH AIR IMMEDIATELY. AVOID CONTACT WITH SKIN OR EYES. USE SOLVENT RESISTANT GLOVES TO MINIMIZE SKIN EXPOSURE. WEAR SAFETY GLASSES TO MINIMIZE CHANCE OF EYE CONTACT. IF EYE CONTACT OCCURS, FLUSH EYES WITH WATER FOR FIFTEEN (15) MINUTES AND SEE A PHYSICIAN. IF SKIN CONTACT OCCURS, WASH THOROUGHLY WITH SOAP AND WATER. IF SWALLOWED, DO NOT INDUCE VOMITING. GET IMMEDIATE MEDICAL ATTENTION.

ALL SOLVENTS AND CEMENTS USED ARE FLAMMABLE AND THEIR FUMES TOXIC. USE ONLY IN A WELL VENTILATED AREA AWAY FROM SPARKS OR FLAMES.

(1) Mask area around deice boot with 1.00 inch tape, allowing 0.250 to 0.500 inch (6.35 to 12.70 mm) margin.

(2) Clean leading edge surface twice with MPK. On final cleaning, swab with clean solvent and wipe dry before solvent dries.

(3) For each installation, prepare a 1.00 by 8.00 inch (2.54 by 20.32 cm) adhesion test specimen, from excess material trimmed off deice boot(s). Cement test strip adjacent to deice boot(s). Install test specimen using identical procedures as those used to bond deice boot(s). Leave 1.00 inch (2.54 cm) of strip uncemented for test purposes.

CAUTION: A-851-B FUEL BARRIER CEMENT IS REQUIRED ON WING DEICE BOOTS OVER INTEGRAL FUEL TANKS TO AVOID FUEL DEGRADATION OF BOSTIK INSTALLATION CEMENT.

(4) On wing surface sections over integral fuel tanks, apply A-851-B Fuel Barrier Cement.

(5) Thoroughly mix A-851-B cement. If necessary, thin to brushing consistency. Brush on one (1) even coat, allow one half (1/2) hour to dry, then brush on a second coat. Allow second coat two (2) hours minimum drying time before applying Bostik 1007 primer.

(6) Clean back surface of deice boot twice with MPK.

NOTE: Change cloth frequently to avoid contamination of cleaned areas. Do not saturate back surface too heavily with solvent or scrub repeatedly. Allow to dry thoroughly before cementing.

(7) Thoroughly stir Bostik 1007 primer.

(8) Apply one (1) even brush coat of Bostik 1007 primer to masked area of wing or stabilizer leading edge.

(9) Allow primer to air dry a minimum of thirty (30) minutes.

(10) Determine end points to place chalk line on leading edge of airfoil (Figure 201). Snap chalk line or mark straight edge with a ballpoint pen.

(11) Thoroughly stir Bostik 1096M cement.

(12) Thoroughly stir Boscodur 9R accelerator.

(13) Mix eight (8) parts of Bostik 1096M cement to one (1) part of Boscodur 9R accelerator. Prepare only enough for immediate use as mixture gels in 6 to 8 hours.

(14) Apply one (1) even brush coat of cement to back surface of deice boot and to leading edge of wing and stabilizer. Allow to air dry a minimum of thirty (30) minutes at relative humidity below 75% or a minimum of one (1) hour at relative humidity between 75% and 90%.

(15) Trim inboard end of wing deice boot to fit into nacelle. Use template made as shown in Figure 202.

(16) Apply a second even brush coat of cement to back surface of deice boot and leading edge of wing and stabilizer. Allow to air dry a minimum of thirty (30) minutes at relative humidity below 75% or a minimum of one (1) hour at relative humidity between 75% and 90%.

NOTE: Install deice boots within 4 hours to avoid curing of cement prior to installation. If cement cures prior to installing deice boot, cement and primer must be removed from the deice boots and leading edges. The deice boot installation must be reaccomplished.

(17) Beginning with outboard end, roll deice boot toward inboard end, with cemented side out.

(18) To reactivate cement, apply toluene to cement with a clean lint free cloth. Do not allow deice boot to contact reactivated cement until ready for permanent bonding.

(19) Reactivate cement strip along centerline of deice boot, 3.00 inches (7.62 cm) wide and 18.00 inches (45.72 cm) long. Ensure centerline of boot aligns with centerline of airfoil.

(20) Reactivate cement on matching section of airfoil leading edge.

(21) Unroll deice boot against airfoil leading edge while applying light tension.

(22) When cement is tacky, press deice boot to airfoil ensuring centerlines are lined up (Figure 202). Keep bond line as straight as possible.

(23) On horizontal stabilizer or outboard wing deice boot, ensure hose connections will mate with hole in leading edge.

(24) Use 2.00 inch rubber roller to press deice boot firmly to airfoil along centerline in area where cement has been reactivated.

(25) Reactivate cement strip on boot and airfoil centerline 3.00 inches (7.62 cm) wide for additional 24.00 to 36.00 inches (60.96 to 91.44 cm) (Figure 202).

CAUTION: AVOID USING EXCESS SOLVENT. AVOID TWISTING, BENDING OR JERKING LOOSE FROM BONDED AREA.

(26) If deice boot should attach off center (centerlines not lined up), apply MPK with a small brush or trigger squirt bottle to soften bond line. Apply a minimum of solvent to bond line while applying sufficient tension to peel back deice boot. Reapply any cement that pulls loose.

(27) Continue to reactivate cement along deice boot centerline until entire length is attached. Roll centerline area firmly with rubber roller.

(28) Starting at inboard edge, hold deice boot away from airfoil to reveal bond line. Reactivate cement strip 3.00 inches (7.62 cm) wide by 60.00 inches (152.4 cm) long, working from bond line toward trailing edge (Figure 202).

(29) Keep toluene moistened cloth tight in fold of bond line, wiping first along airfoil leading edge, then along corresponding surface of deice boot.

(30) Roll deice boot down with rubber roller starting at bond line and rolling spanwise while working toward trailing edge. Hold deice boot back while roller does work of mating surfaces.

(31) If deice boot lifts after rolling and cement cobwebs, cement is too wet. Wait until cement is tacky and repeat bonding procedures.

(32) Continue to reactivate cement along bond line until deice boot reaches end of wing or end of stabilizer. Use knife to trim deice boot after starting cut with scissors or razor blade (Figure 202).

(33) Rubber roll spanwise over entire surface of deice boot applying pressure to ensure a good bond. Roll edges outside of inflatable area with narrow stitcher roller.

(34) Apply masking tape to deice boot and aircraft leading edge surface 0.125 to 0.250 inch (3.175 to 6.35 cm) from edge of deice boot.

(35) Mix EC-801 filler compound according to instructions on container. Apply filler with small brush around edges of deice boot to fair it to adjacent surface.

NOTE: Conductive edge sealer may be used to prevent static discharge. Apply A56B for Neoprene De-Icers or apply 74-451-K on Estane De-Icers.

(36) Remove all masking tape and tape residue.

(37) Verify this cure time with an adhesion test using a test strip attached to the aircraft skin adjacent to installed boot.

NOTE: Allow 4 hours dry time after installation is completed before flying aircraft or inflating newly installed deice boots.

Inspect newly installed deice boots after initial periods of operation for signs of lifting from aircraft.

I. Installation – Deice Boot Stall Strip (Figure 204)

NOTE: Stall strips are fitted to aircraft having serial numbers T201-275, and 277-291, TC201-336, and AT001-074. For part numbers, refer to the Illustrated Parts Catalog.

(1) Use 3M 1300L cement or SC2000 Tip Top Cement with RF-SC hardener available from REMA-TECH, 200 Paris Avenue, North Vale, NJ.

(2) Mark area on deice boot where stall strip is to be attached (see Figure 204).

(3) Clean stall strip and deice boot using MPK or equivalent.

(4) Buff deice boot and stall strip bonding areas with sandpaper to remove glossy spots.

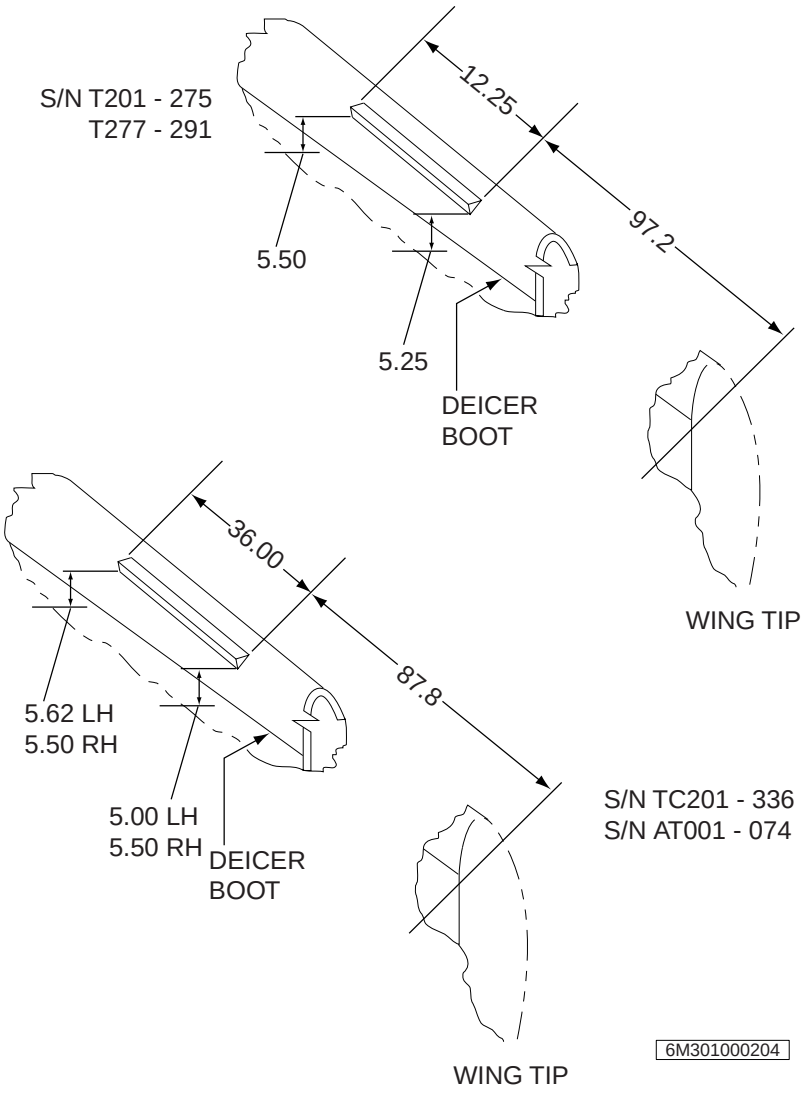
(5) Mix SC2000 cement with 10% RF-SC Hardener. Mixture of cement and hardener must be used within two hours.

(6) Apply mixture to both surfaces and allow to dry well.

(7) Apply second coat to both surfaces and allow to dry only until slightly tacky.

(8) Position stall strip on deice boot and press down firmly. Use masking tape to hold down strip while drying.

(9) Allow to cure for six (6) hours before inflating deice boot or flying aircraft.



Wing Deice Boot
Stall Strip Location
Figure 204

J. Removal – Deice Tubes and Fittings

(1) To remove individual tubes, disconnect tube at each end, loosen or remove any clamps and tie straps supporting tube and remove tube.

(2) Protect open tubes and fittings from contamination with dust covers or place in a clean polyethylene bag and seal.

K. Installation – Deice Tubes and Fittings

CAUTION: ENSURE ALL TUBES AND FITTINGS ARE CLEAN AND FREE OF CONTAMINATION PRIOR TO INSTALLING ON AIRCRAFT. TUBES MAY BE FLUSHED WITH SOLVENT, DRAINED AND THOROUGHLY DRIED.

PROTECT ALL TUBING (RIGID, PLASTIC AND FLEXIBLE HOSES) FROM CHAFING AT EACH CUTOUT BY GROMMETS OR CLAMPS. PROVIDE SUFFICIENT CLEARANCE BETWEEN THE TUBE AND PROJECTIONS SUCH AS BOLTS OR NUTS TO PREVENT CONTACT DUE TO VIBRATION.

(1) Install plastic tubing so that each separate piece of tubing will have 0.500 inch (1.27 cm) slack per foot (30.48 cm) to allow for expansion and contraction. When clamping plastic tubing, be careful not to deform tube.

CAUTION: ENSURE NO THREAD COMPOUND GETS IN THE BORE OF THE FITTING OR ON THE FITTING END OR CONE SURFACE. NO COMPOUND SHOULD BE ALLOWED ON THE TUBING INSIDE FLARE SURFACE, INTERIOR OF THE TUBE OR INSIDE ANY SYSTEM COMPONENTS. ALWAYS USE THREAD COMPOUND SPARINGLY.

(2) To prevent galling and seizure of threads, MIL-T-5544 thread compound should be applied to male threads of fittings as follows:

(a) Align tube and press flare firmly against fitting end.

(b) Tighten nut one-half (1/2) to one (1) turn (this ensures compound will not get on first one [1] to one and one-half [1 1/2] threads) then sparingly apply MIL-T-5544 thread compound to male threads.

(3) Procedure for installing and tightening fittings is as follows:

NOTE: Nylo-Seal and Swagelok tube fittings are received completely assembled with the nut finger tight. Do not disassemble before use as this may result in contamination of the fitting.

(a) Insert tube into fitting. Make sure tube rests firmly on shoulder of fitting and nut is finger-tight.

(b) Scribe nut at 6 o'clock position. This mark is used for reference when tightening nut.

(c) On Nylo-Seal fittings, tighten nut two (2) to two and one-half (2 1/2) turns.

(d) On Swagelok fittings, tighten nut one and one-quarter (1 1/4) turn.

2. Adjustment/Test – Deice Boot

CAUTION: DO NOT TEST DEICE BOOT UNTIL PROPER CURE TIME HAS ELAPSED.

A. Functional Test – Deice Boot.

NOTE: Deice boot functional test can be done by operating engines for pressure source as an alternate to connecting an external air pressure source. See Maintenance Manual, Chapter 71, Engine Ground Operation.

DEICE BOOT – MAINTENANCE PRACTICES

WARNING: WHEN OPERATING ENGINES FOR PRESSURE SOURCE, BLEED AIR TUBING FROM NACELLES TO REGULATOR WILL BECOME VERY HOT. USE CARE TO PREVENT BURNS.

(1) Disconnect high temperature bleed air tubing from air conditioning bleed air on left side of wheel well, see Maintenance Manual, Chapter 36, Pneumatic, Removal/Installation – High Temperature Bleed Air Tubing.

(2) Connect regulated air pressure source to high temperature bleed air tubing. Set pressure source 31 to 33 psi (2.179 to 2.320 kg/cm^2).

(3) Adjust deice pressure regulator to produce a pressure reading of 16 to 17 psi (1.125 to 1.195 kg/cm^2).

(4) Reduce pressurization source output pressure to 10 to 11 psi (0.703 to 0.773 kg/cm^2).

(5) Adjust vacuum regulator 4.5 inches HG.

(6) Increase air pressurization source pressure to 75 psi (5.27 kg/cm^2.) Vacuum pressure and deice pressure should remain in green arc range.

(7) Turn aircraft electrical power ON. Move DEICE BOOTS switch to AUTO.

(8) Verify immediate wing deice boot inflation for six (6) second period followed by immediate deflation.

(9) Verify inflation of horizontal stabilizer deice boots for four (4) second period followed by deflation. Deice boot inflation and deflation should be quick and complete.

(10) Verify 170 second rest period with deice boots deflated.

(11) Verify sequence of wing deice boot operation, horizontal stabilizer deice boot operation and rest period repeating every three (3) minutes.

(12) Move DEICE BOOTS switch from AUTO to OFF midway through sequence (after wing deice boots deflate).

(13) Wait for deice boot timer to recycle (three [3] minutes).

(14) Move DEICE BOOTS switch to AUTO.

(15) Verify Deice boot begins sequence with wing deice boot inflation. If sequence begins at another point, deice boot timer is not returning to recycle position.

(16) Move DEICE BOOTS switch to MAN.

(17) Verify all deice boots inflate together and remaining inflated as long as DEICE BOOTS switch is held in MAN.

(18) Release DEICE BOOTS switch. Switch will spring back to OFF.

(19) Verify all deice boots deflate together.

(20) Disconnect external air pressure source and connect high temperature bleed air tubing to air conditioning bleed air, see Maintenance Manual, Chapter 36, Pneumatic, Removal/Installation – High Temperature Bleed Air Tubing.

B. Electrical Test – Deice Boot Timer

(1) Remove copilot seat IAW Chapter 25.

(2) Remove safety wire and disconnect deice boot timer connector.

(3) Remove screws securing timer to electrical panel EP4.

(4) Remove deice boot timer.

(5) Use a power supply with a range of 0-15 amps, 0-32 VDC, 0.75% or less ripple (Electro Products Labs Model NFB or equivalent) and use three (3) ampere maximum load on each output (pins 4 and 5).

(6) Connect deice boot timer pin 1 (ground) to power supply ground.

(7) Connect pins 4 and 5 to voltmeter.

(8) Connect 28 VDC to deice boot timer pin 3 (timer start).

(9) Verify 28 VDC (± 2.8 VDC) immediately at deice boot timer pin 4 (wing deice boots) for six (6) seconds, followed by 28 VDC (± 2.8 VDC) immediately at pin 5 (horizontal stabilizer deice boots) for four (4) seconds, then no voltage for 170 seconds.

(10) Verify sequence is repeated every three (3) minutes.

(11) Disconnect 28 VDC from deice boot timer pin 3 during mid point of sequence (after voltage switches from pin 4).

(12) Connect 28 VDC to deice boot timer pin 2 (timer return to recycle).

(13) Wait for timer to recycle (three [3] minutes).

(14) Disconnect 28 VDC from deice boot timer pin 3.

(15) Repeat automatic timer test.

NOTE: Deice boot timer must always begin with voltage at pin 4 (wing deice boots) or it is not returning to recycle.

(16) Position deice boot timer on electrical panel EP4.

(17) Install screws securing timer to panel.

(18) Connect and safety deice boot timer connector.

(19) Install copilot seat IAW Chapter 25.

(20) Functional test deice boots.

C. Adhesion Test – Deice Boot

(1) Perform the following adhesion test after replacing wing boot.

(2) Allow test strip to dry four (4) hours or more before performing adhesion test.

(3) Using excess boot material trimmed from the ends of the boot, prepare one (1) test specimen for each deice boot installed. The specimen should be 1.00 by 8.00 inches full thickness strip of boot.

(4) Bond specimen to skin near boot installation. Leave 1.00 inch of material loose.

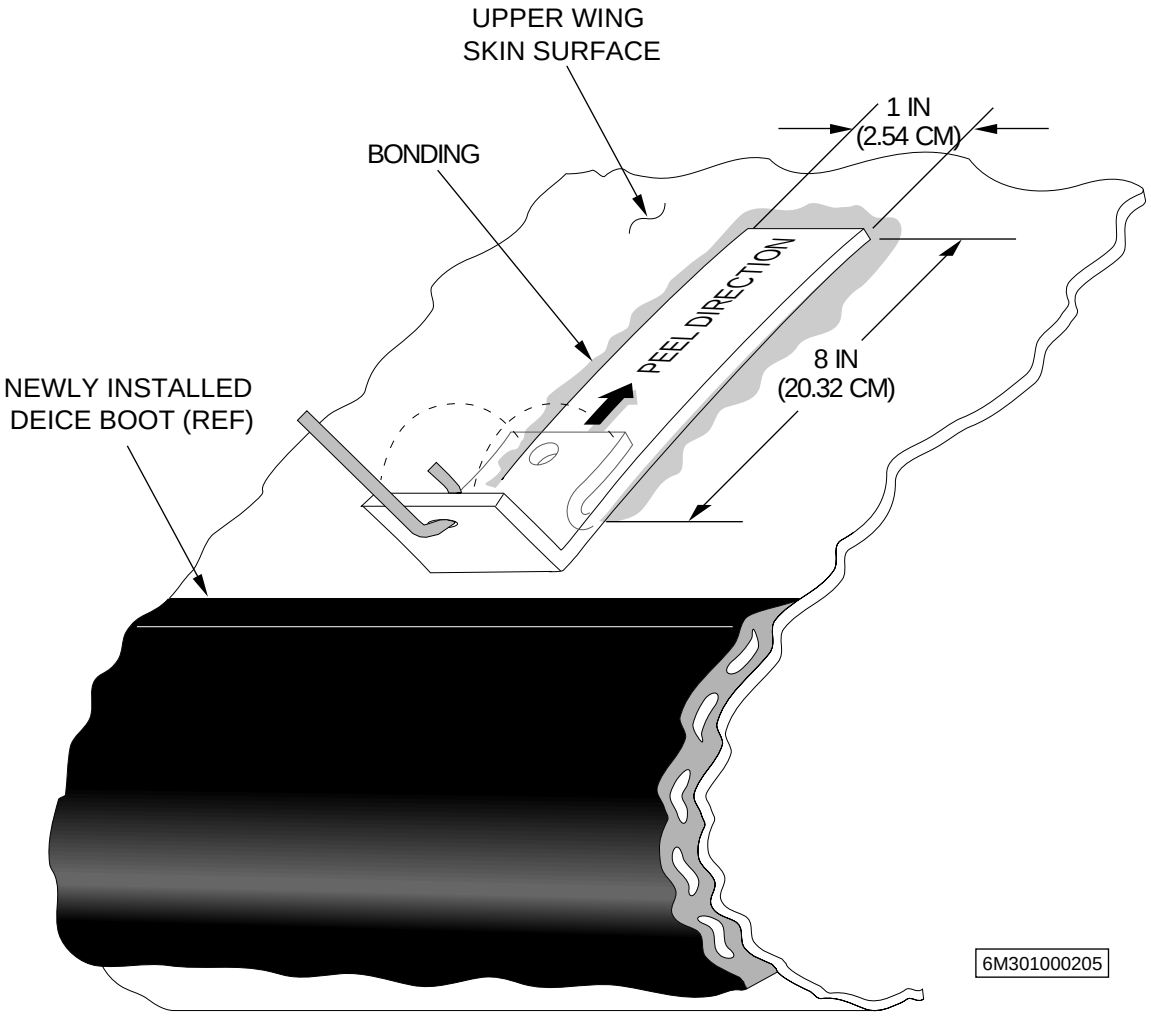
(5) Attach spring scale to uncemented end of test strip.

(6) Measure force required to remove test strip at a rate of 1.00 inch per minute. Apply force 180° to surface (strip doubled back on itself).

(7) A minimum of 5 pounds tension (pull) shall be required to remove test strip. If less than 5 pounds is found, remove deice boot and repeat bonding procedure.

(8) Completely remove test strip and adhesive from wing or horizontal stabilizer.

(9) Connect hose to deice boot and install access panel.



Deice Boot Adhesion Test
Figure 205

3. Inspection/Check – Deice Boot

A. Inspection – Deice Boot

(1) At scheduled intervals, inspect deice boot for holes, severe scuffing, abrasion and deterioration (cracking or crazing).

(2) Inspect sharp radius areas closely. Use a strong light to provide adequate illumination.

(3) Inflate deicer boots briefly. Deflate and inspect surface. Repair leaks and cracks or resurface. Use a solution of mild soap and water to check for air leaks.

4. Cleaning/Painting – Deice Boot

A. Cleaning – Neoprene and Fabric Deice Boot.

CAUTION: PETROLEUM PRODUCTS DETERIORATE RUBBER AND SHOULD NOT BE USED AS A CLEANING AGENT.

(1) Clean neoprene and fabric deice boot with mild soap and water, using a soft scrub brush.

NOTE: In cold weather it may be necessary to warm the wash solution or use warm air from a portable ground heater. Temperature of cleaning solution shall not exceed 140°F (60°C).

B. Surface Coatings – Neoprene and Fabric Deice Boot

(1) Treat installed deicers twice a year with Age-Master No. 1, (B.F. Goodrich or equivalent), IAW directions on the label.

(2) During icing season, treat deicers with Icex IAW directions on the label.

NOTE: Icex does not prevent ice formation or remove ice but does improve deicing efficiency of the deicer. It may be applied directly over AgeMaster treated surfaces.

5. Approved Repair – Deice Boot

A. Surface refurbishing, cold patch repair, and pin hole repair are the types of repairs that can be made on installed pneumatic deice boots.

(1) Refurbishing procedures may be repeated if deemed necessary by the operator; however, it is not generally recommended that a deice boot be refurbished more than two (2) times.

(2) Cold patch repairs, if made correctly, can generally be considered permanent for the life of the deice boot.

(3) Pin hole repairs may be used in place of patches for small repairs. These repairs eliminate the visibility of repairs that result after repairing multiple smaller holes with a patch kit.

B. Surface Refurbishing

(1) Weathered deice boots can be refurbished by using B. F. Goodrich Protective Coating kit 74-451-L. This coating has been designed as a resurfacing material for scuffed deice boots in need of surface repair or those that show signs of deterioration. The coating will repair small cracks and improve the appearance of the deice boot surface.

NOTE: Protective coating material is not intended to repair severely damaged deice boots or for repairing deice boots with air leaks. Before applying the 74-451-L kit, the deice boot should be checked for leaks with a solution of mild soap and water.

C. Cold Patch Repair

(1) Scuff damage.

(a) This type of damage will be most commonly encountered and in most cases need not be repaired with patches. Scuff damage that need not be repaired with patches are abrasions of the deice boot surface that do not penetrate through the surface layer (approx. 0.010 in) and do not leak when the deice boot is inflated.

CAUTION: IF THE DEICE BOOT WAS PREVIOUSLY COATED WITH THE #74-451-L KIT, THE COATING MUST BE REMOVED BEFORE A COLD PATCH CAN BE APPLIED.

(b) If the scuff is severe and has caused the removal of the entire thickness of the surface ply, the damage should be repaired with cold patch IAW procedures and materials from Repair Kit #74-451-AA.

(2) Damage to inflatable tube area.

(a) Cuts, tears, or ruptures that do not cross a stitch line.

NOTE: A stitch is the line that forms the edges of the inflatable tubes.

1 Replace any deice boot that has a cut, tear, or rupture that exceeds 0.750 inch in length or that is within 0.125 in of a stitch line.

2 Cuts, tears, or ruptures less than 0.750 inch in length and are not within 0.125 in of a stitch line can be repaired with cold patch IAW procedures and materials from Repair Kit #74-451-AA.

(b) Cuts, tears, or ruptures that cross a stitch line.

NOTE: If a stitch line is cut, torn, or ruptured do not repair; replace the deice boot.

(c) Broken or cut threads.

1 Broken or cut threads can be detected by inflating the deice boot. A broken or cut stitch will cause the deice boot to balloon (inflate) between the adjacent tubes where the stitch is cut. Replace the deice boot if such a condition occurs.

(d) Separation of outer layer.

1 Replace the deice boot if the area of looseness exceeds 1.00 sq in.

2 For areas less than 1.00 sq in repair by trimming away all loose surface material, then applying cold patch IAW procedures and materials from Repair Kit #74-451-AA.

(e) Damage to non-inflatable area.

1 Cuts, tears, or ruptures.

a Replace deice boot if cut, tear, or rupture exceeds 3.00 inch in length.

b For cuts, tears, or ruptures less than 3.00 inch in length apply cold patch IAW procedures and materials from Repair Kit #74-451-AA.

2 Separation of outer layer.

a Replace deice boot if area of looseness exceeds 3.00 sq inches.

b For cuts, tears, or ruptures less than 3.00 sq inch, repair by trimming away all loose surface material, then applying cold patch IAW procedures and materials from Repair Kit #74-451-AA.

(f) Repair limits.

1 In order to maintain the maximum functional efficiency of the pneumatic deice boot, the following limits are recommended by B. F. Goodrich:

Allowable Number and Concentration of Patches		
Patches	1 1/4 X 2 1/2	3 per 12 inch square
Patch	2 1/2 X 5	2 per 12 inch square
Patch	5 X 10	1 per 12 inch square
Two Patches plus one patch	1 1/4 X 2 1/2 2 1/2 X 5	per 12 inch square

NOTE: The above limits are not absolute. The deice boot will still function satisfactorily with a higher concentration of patches. However, the operator should cycle the deice boots to ensure the inflation of the deice boot has not been impaired.

Patch Limitations
Table 201

D. Pinhole Repair

(1) Pinhole repair kit No. 74-451-AE, designed specifically for the effective and longer-lasting repair of small pinholes up to 1/16 inch, can be used in place of patch repairing.

(2) Apply a small amount of liquid Estane (polyurethane) directly to the surface of the repair area. This will create a strong and virtually invisible seal over the damaged spot.

(3) With the 74-451-AE repair kit it is permissible to repair up to 20 separate pinholes in any 12 inch square area on the deice boot.

NOTE: Because the elastomeric material used to repair the pinholes (Liquid Estane) is applied directly to seal the pinhole, the repaired area will seamlessly expand and contract along with the expansion and contraction of the deice boot, making for a durable, permanent repair.

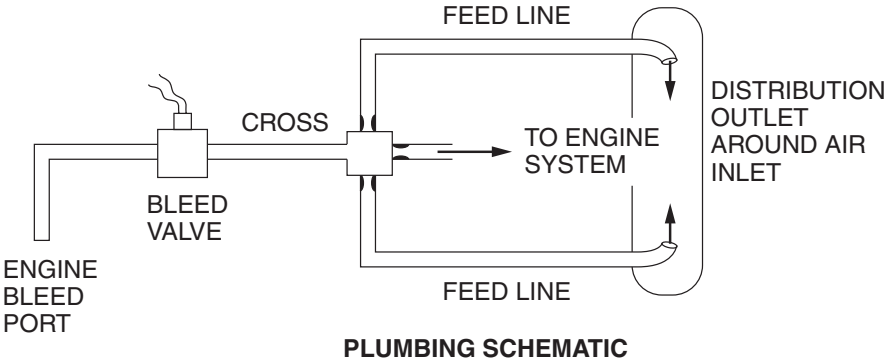
ENGINE INTAKE ANTI ICE – DESCRIPTION AND OPERATION

1. General

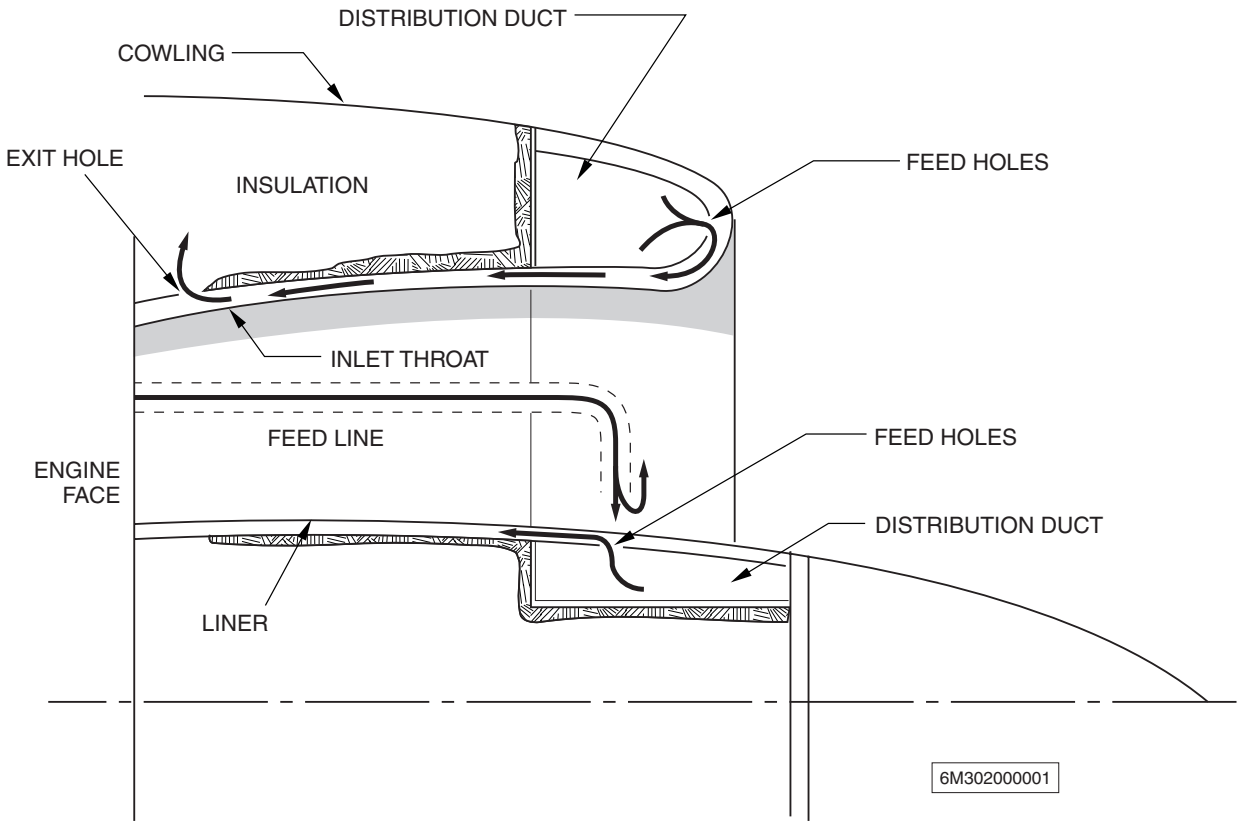
A. The engine inlets are anti-iced with engine compressor bleed air. Anti-icing air is obtained from two sources:

(1) The main anti-icing compressor bleed port.

(2) Cabin pressurization compressor bleed port



PLUMBING SCHEMATIC



Engine Air Inlet Deicing System
Figure 1

ENGINE INTAKE ANTI ICE – DESCRIPTION AND OPERATION

B. The flow from the main anti-icing bleed port runs through an anti-icing valve to a tee; the tee has an orifice or 0.234 inch diameter. At the tee the bleed air splits; one side going to the engine anti-ice shield, and the other side to the right hand side of the distributor duct. The left hand side of the distributor duct is supplied by a supplementary line which has an anti-icing valve identical to the main anti-icing valve. The line supplying the left hand side of the distributor duct has a 0.204 inch diameter orifice.

C. The engine furnished air-inlet anti-icing shield is contoured around the lower portion of the casting that forms the engine air passage and extends from the rear face of the inlet flange aft to the compressor cavity.

D. The distributor duct encircles the engine air inlet within the cowling. Air flows from the distributor duct through a series of holes across the top and bottom skin. The double skin is insulated to minimize heat loss and has a 0.04 inch gap.

E. Both anti-ice valves incorporate a limit switch to limit the position of the valve. The valves are controlled by a three-position toggle switch for each engine. The three switch positions are ON, OFF, and MOMENTARY OFF. When the switch is positioned ON the valves open and an annunciator panel light is illuminated. If either valve fails to open the annunciator light does not illuminate. When the switch is positioned OFF the valve closes and the light goes out. When the switch is positioned to MOMENTARY OFF the annunciator panel light will illuminate if both valves are closed. If the annunciator panel light does not illuminate, with the switch in MOMENTARY OFF, one or both valves are open.

F. The entire upper portion of the compressor air inlet is integral with the engine gearbox, and is kept free of ice formation by heat transfer from the engine oil to the inlet surface.

G. The lip of the external oil cooler scoop is heated for ice protection by engine scavenge oil. Oil pumped from the engine passes through an aluminum tube which is an integral part of the scoop assembly and forms the leading edge of the scoop. Heat transferred from the oil to the tube keeps the tube surface above freezing, preventing the formation of ice.

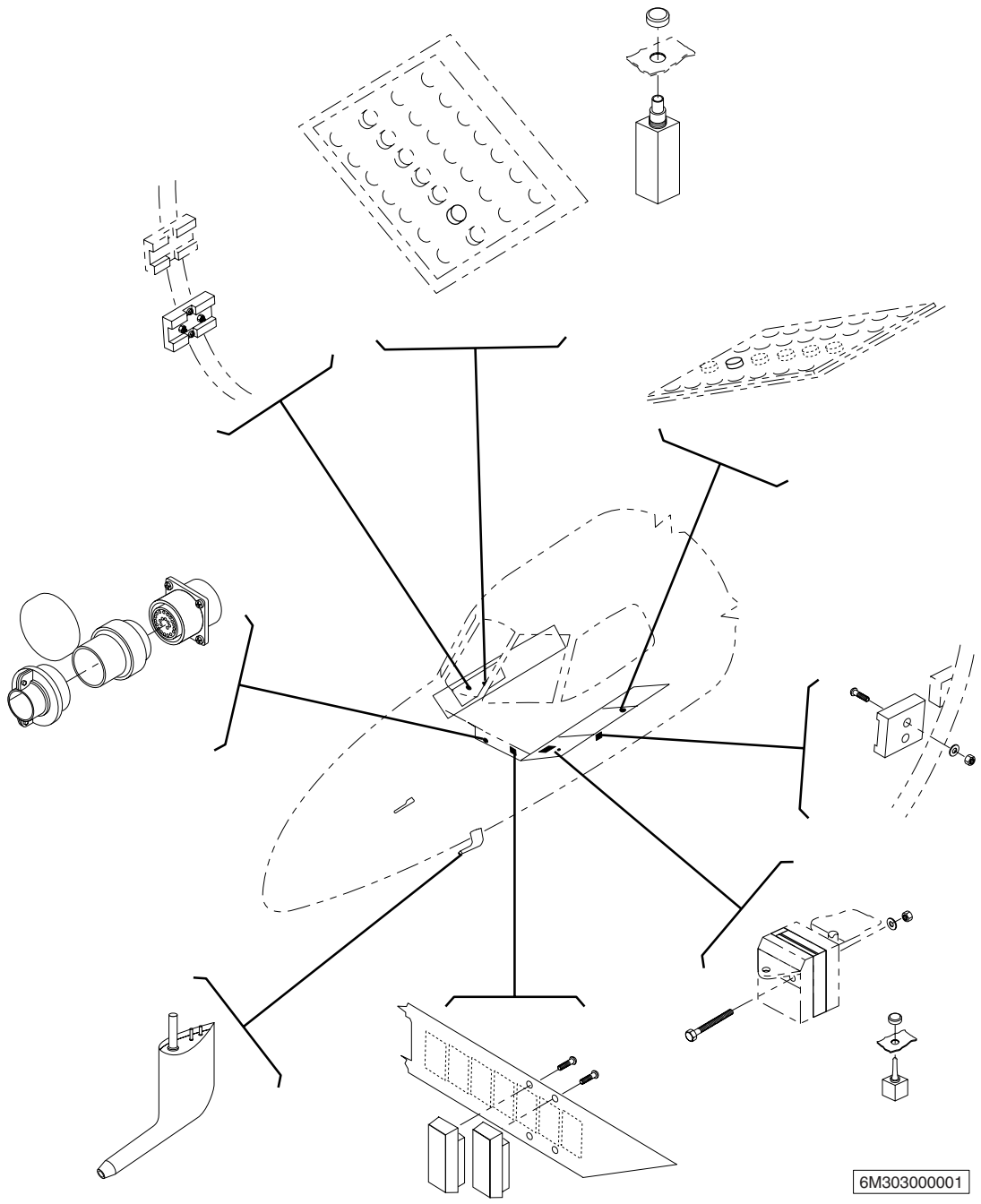
H. The oil is then routed to the air-oil heat exchanger before being returned to the oil tank. The scavenge oil used to anti-ice the oil cooler scoop is at high temperature, and passes through the oil cooler lip tube before any cooling of the oil.

I. The oil cooler contains a thermostat which controls oil temperature by routing all flow to the core at the high oil temperature limit and bypassing the core at the low limit. The thermostat maintains the oil temperature within the prescribed limits of 55° to 110°C.

PITOT ANTI-ICE – DESCRIPTION AND OPERATION

1. General

A. Electrical heating prevents formation of ice on aircraft pitot tubes (see Figure 1). The left and right pitot tube heating elements have separate power switches and circuit breakers. A selector switch allows the pitot to monitor either system on a single ammeter, which is located on the left hand console.



Pitot Heat Control
Figure 1

PITOT ANTI-ICE – DESCRIPTION AND OPERATION

2. Pitot Tube

A. Pitot tubes are located on the left and right lower outside of the nose section. Each pitot tube has an internal heating element that operates on 28 VDC. A two pin push on plug connects the heating element to the aircraft electrical wiring.

3. Pitot and SAS Heat Switch

A. The three position left and right PITOT & SAS HEAT switches are located on the pilot's switch panel EP5.

(1) Switch positions are:

(a) PITOT HEAT ..... Upper Position.

(b) OFF ..... Center Position.

(c) PITOT and SAS HEAT ..... Lower Position.

(2) When PITOT HEAT (upper position) is selected, only pitot heat is operating.

(3) When PITOT & SAS HEAT (lower position) is selected, both pitot heat and SAS heat operate.

(4) A jumper wire between the switches energizes SAS anti-ice if either the left or right PITOT & SAS HEAT is turned ON.

4. Circuit Breaker

A. A circuit breaker located on the LEFT ESSENTIAL BUS panel powers the left pitot anti-ice. Another circuit breaker located on the RIGHT ESSENTIAL BUS panel powers the right pitot anti-ice.

5. Pitot Ammeter Shunt

A. Pitot ammeter shunts are located under the forward baggage compartment floor on each side above the pitot tubes.

B. Electric current from the pitot heating element passes through the shunt. The current produces a small voltage that varies with the amount of current in the system. The voltage causes the pitot ammeter to deflect and indicate the amount of current flow in the system.

6. Pitot Ammeter Switch

A. A Pitot Ammeter Switch is located on the left console to the left of the Tri-Unit Meter Assembly. The pitot ammeter is contained inside the unit.

B. A green arc from four to ten amps indicates normal operating current range of the system.

PITOT ANTI ICE – FAULT ISOLATION

1. Troubleshooting – Pitot Anti-Ice System

A. To assist in the isolation of system faults, the following troubleshooting block diagrams are provided.

B. For additional information and schematic diagrams refer to the SA226 Wiring Manual.

C. The prerequisites at the beginning of each diagram require an Operational Test of the system to determine which problem statement applies.

D. In the event of multiple problems correct the faults in the order they appear below.

PITOT ANTI ICE – FAULT ISOLATION

PROBLEM:

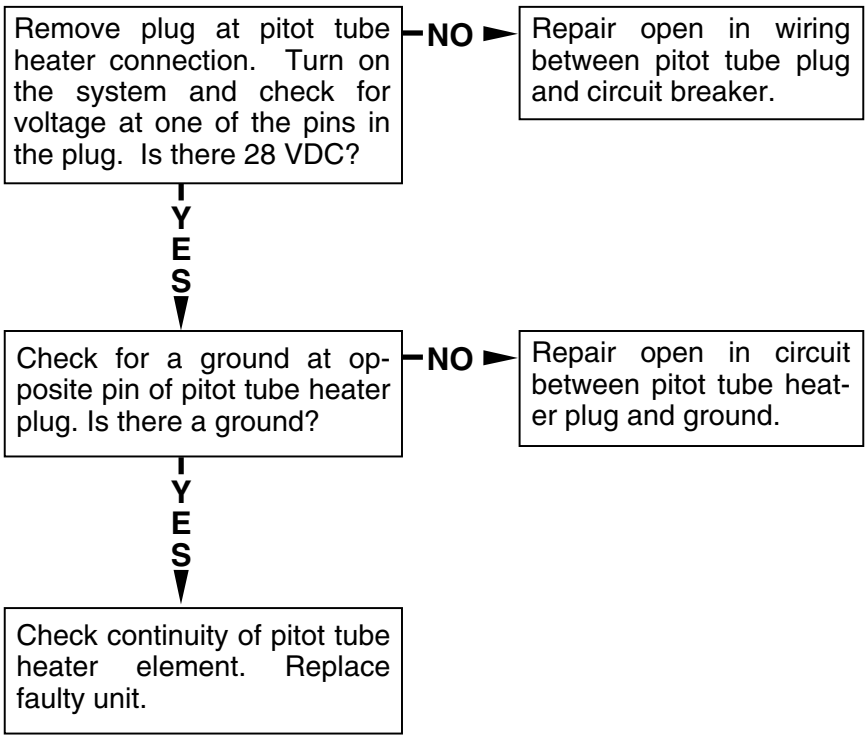
PITOT TUBE DOES NOT HEAT.

WARNING: PITOT TUBE WILL BECOME EXTREMELY HOT. DO NOT TOUCH.

PREREQUISITES:

Test pitot anti-ice. See Adjustment Test – Pitot Anti-Ice.

START:



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PITOT ANTI-ICE – FAULT ISOLATION

PROBLEM:

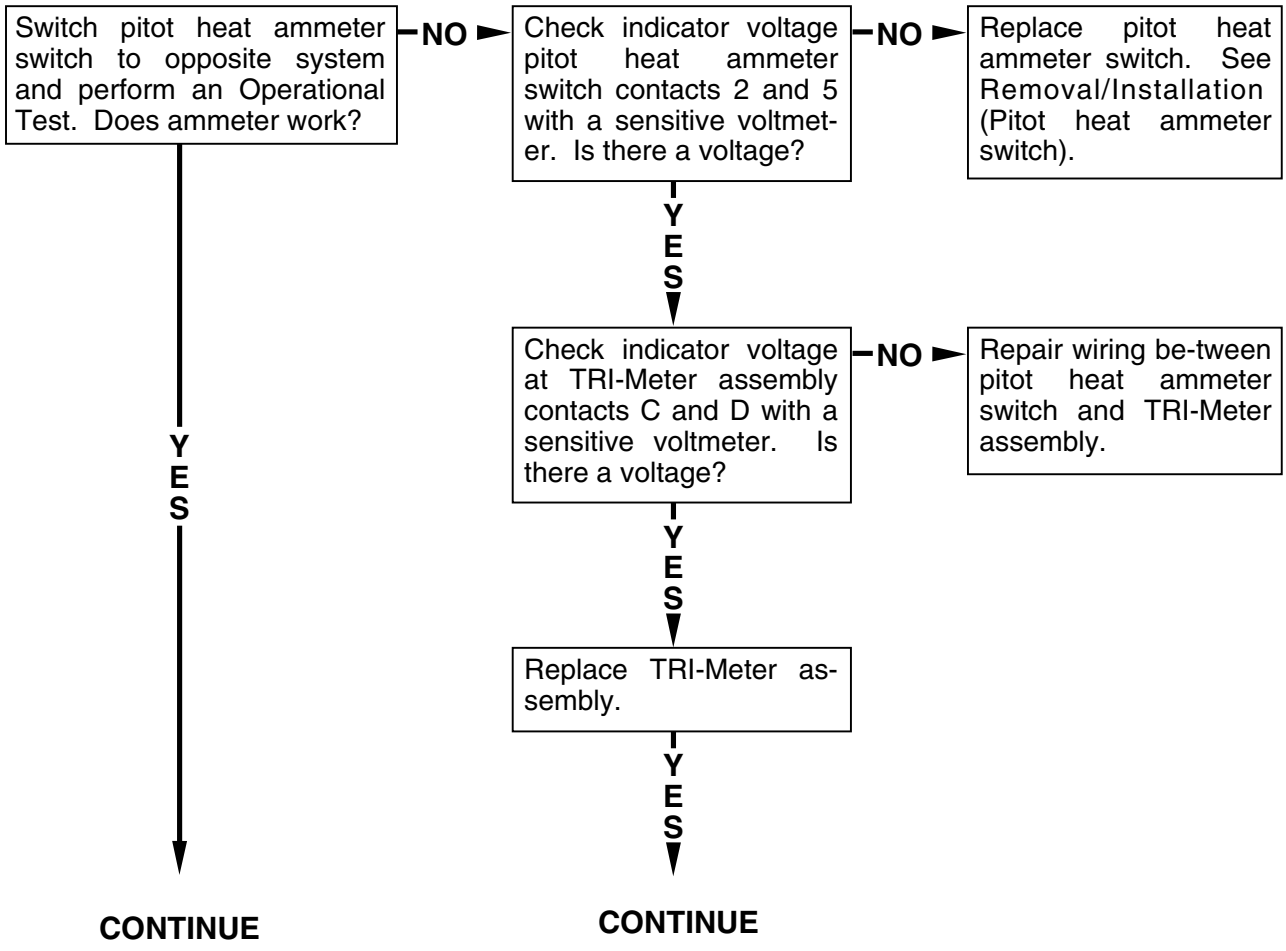
PITOT TUBE HEATS NORMALLY. PITOT AMMETER DOES NOT OPERATE.

WARNING: SAS VANE WILL BECOME EXTREMELY HOT. DO NOT TOUCH.

PREREQUISITES:

Test pitot anti-ice. See Adjustment Test – Pitot Anti-Ice.

START:



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CONTINUED

Check indicator voltage at pins 1 and 3 (LH), or 4 and 6 (RH) of pitot ammeter switch with a sensitive voltmeter. Is there voltage?

NO ►

Repair open wiring between pitot ammeter switch contacts 1 and 3 (LH) or 4 and 6 (RH) and pitot heat shunt.

YES ▼

Replace pitot heat ammeter switch. See Removal/Installation – Pitot heat ammeter switch.

CONTINUED

END

4M303000102A

PITOT ANTI-ICE – MAINTENANCE PRACTICES

1. Removal/Installation – Pitot Anti-Ice

A. Removal – Pitot Tube

(1) Gain access to area inside forward baggage compartment where pitot tube connections are located.

(2) Pitot tubes for the aircraft are mounted to the forward baggage compartment skin, below aft floorboard, between frames at F.S. 51.31 and pressure bulkhead at F.S. 69.31. Access to left pitot tube is gained by removing flexible duct on cockpit fresh air blower outlet.

(3) Disconnect flexible tubing from pitot tube hose connection. Hold pitot tube hose connection with a wrench while loosening fitting with second wrench.

(4) Disconnect pitot tube heater plug. Remove pitot mast before disconnecting plug on aircraft.

(5) Remove screws securing pitot tube to mount.

(6) Remove pitot tube.

B. Installation – Pitot Tube

(1) Connect pitot tube heater plug before mounting pitot mast.

(2) Position pitot mast on mount.

(3) Install screws securing pitot mast.

(4) Connect flexible tubing to pitot tube hose connection. Hold pitot tube hose connection with wrench while tightening fitting with second wrench.

(5) On the aircraft, install clip covering right pitot tube mount. Seal area with fillet of Pro-Seal 890 or equivalent. Follow mixing instructions on container.

(6) Install flexible duct on cockpit fresh air blower outlet on left side. Install forward baggage compartment aft floorboard.

(7) Functional test pitot anti-ice, see Adjustment/Test – Pitot Anti-Ice.

C. Removal – Pitot Ammeter Shunt

(1) Remove interior components from baggage compartment as necessary to gain access to pitot shunt.

(a) On the aircraft the shunt is located in the left and right rear bay, below aft baggage compartment floorboards, between frames at F.S. 51.31 and pressure bulkhead at F.S. 69.31.

(2) Tag and disconnect wires on pitot ammeter shunt.

(3) Remove screws and nuts securing pitot ammeter shunt.

(4) Remove pitot ammeter shunt.

D. Installation – Pitot Ammeter Shunt

(1) Position pitot ammeter shunt on mount.

(2) Install screws and nuts securing pitot ammeter shunt.

(3) Connect wires to pitot ammeter shunt.

(4) Install forward baggage compartment interior sections.

(5) Functional test pitot anti-ice, see Adjustment/Test – Pitot Anti-Ice.

2. Adjustment/Test – Pitot Anti-Ice

A. Functional Test – Pitot Heater

WARNING: PITOT TUBES AND SAS VANE WILL BECOME EXTREMELY HOT. DO NOT TOUCH.

(1) Turn aircraft electrical power ON.

(2) Select either LH or RH position of PITOT HEAT switch.

(3) Position PITOT HEAT switch to either PITOT HEAT or PITOT & SAS HEAT.

(4) Verify PITOT HEAT AMPS indication. PITOT HEAT AMPS will indicate six to eight amps initially, then decrease slowly to three to five amps.

(5) Check pitot tube for heating. A drop of water will boil immediately on contact with pitot tube after operating temperature is reached.

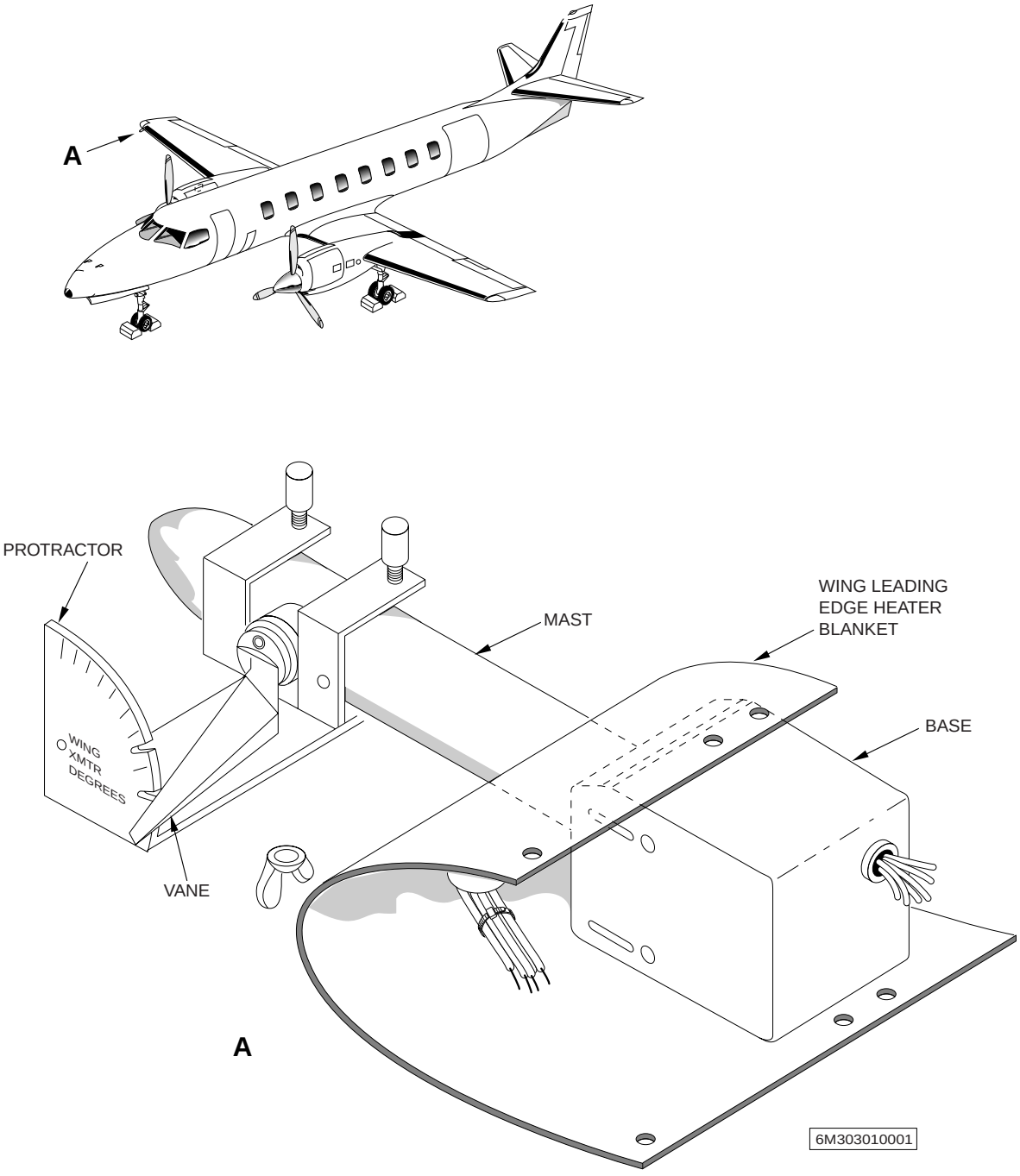
(6) Position PITOT HEAT switch to OFF position.

(7) Turn aircraft electrical power OFF.

SAS VANE DEICE – DESCRIPTION AND OPERATION

1. General

A. The SAS vane and probe of the SAS mast are equipped with heating elements for anti-icing. A thermostat is imbedded in the SAS vane mast to prevent overheating if the anti-ice circuit is energized while on the ground or in a high ambient temperature condition.



SAS Vane Deice
Figure 1

SAS VANE DEICE – DESCRIPTION AND OPERATION

B. The SAS anti-ice system receives 28 VDC electrical power from the LH ESSENTIAL BUS when either LH or RH Pitot and SAS switch is placed in the ON position. By turning on either of these switches, a SAS heat relay is energized to direct electrical power from the LH ESSENTIAL BUS to the SAS anti-ice system master control unit located in the RH wing tip. From the master control unit, electrical power is applied to the three heating elements within the system.

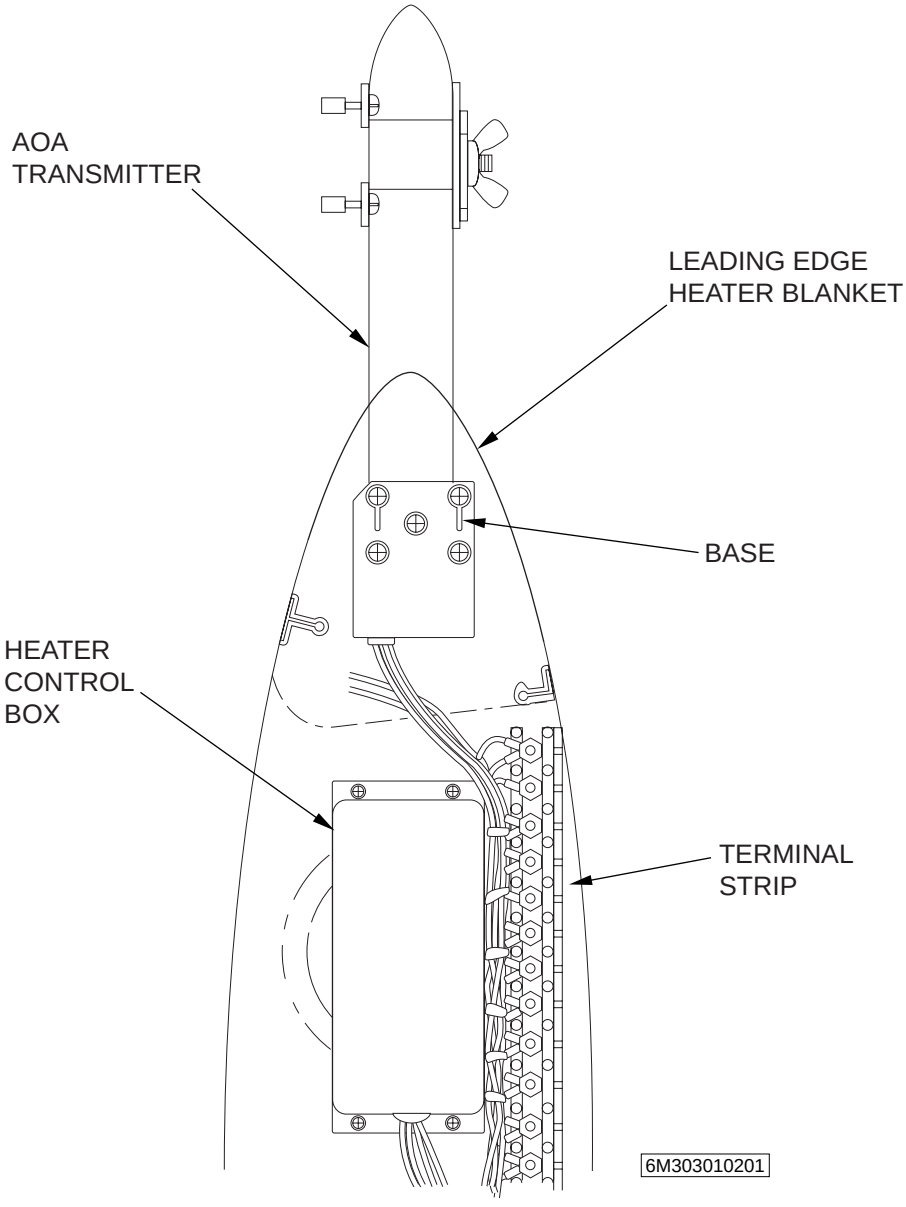
C. An auxiliary heater blanket provided for the wing leading edge in the area of the SAS vane mast is also thermostatically controlled.

D. The vane pivots on a shaft that extends into the body of the mast. The vane aligns itself with the airflow over the wing. The pivot shaft is linked mechanically to the pot. The electrical signal from this pot is used by the SAS computer for angle of attack information. The vane has a range of 90° (approximately 80 above zero degrees and approximately 10 below). Each aircraft is provided with a vane protractor, marked in degrees, to be used for calibration of the system. (see Maintenance Manual, chapter 27 for more information on SAS vane deice.

SAS VANE DEICE– MAINTENANCE PRACTICES

1. Removal/Installation – SAS Vane Deice

A. For removal/Installation of SAS Vane Deice components, refer to procedures for AOA transmitter, heater blanket, heater control box, see chapter 27.



SAS Vane Deice
Figure 201

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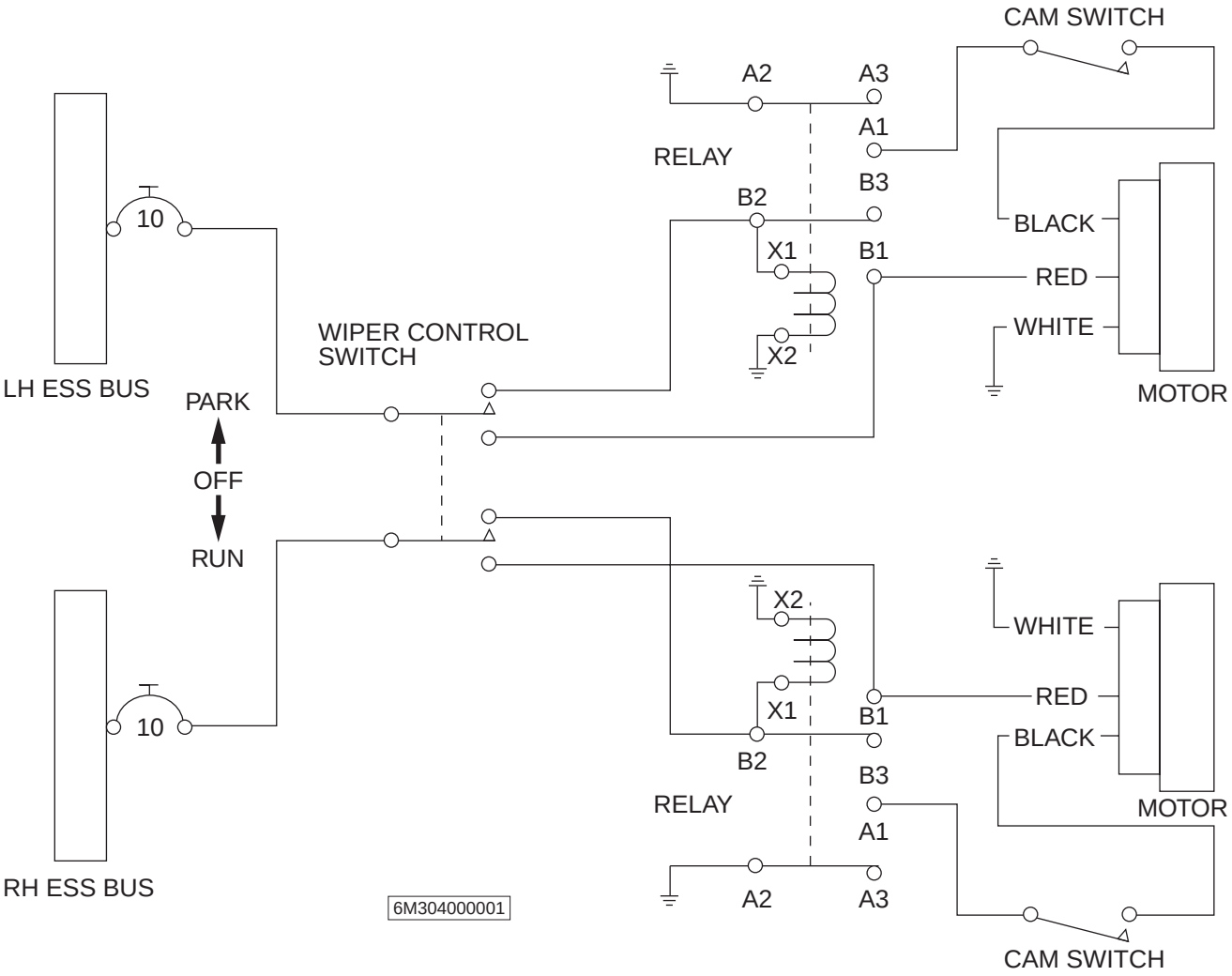
WINDSHIELD WIPER – DESCRIPTION AND OPERATION

1. General

A. The windshield wiper system installed in the aircraft is a dual system, controlled by one centrally located control switch.

B. Electrical power is supplied to the pilot wiper system from the LEFT ESSENTIAL BUS. Power for the copilot wiper system is supplied by the RIGHT ESSENTIAL BUS. A single three position (ON-OFF-PARK) control switch, located on the pilot switch panel controls operation of both systems, while maintaining positive system separation (electrical schematic is shown in Figure 1).

C. The dual windshield wiper system is placed into operation by placing the W/S wiper switch in the ON position. The wipers run at the maximum speed of approximately 250 strokes per minute. To PARK, depress the control switch to the PARK position and hold until both wiper arms reach their parked position. The control switch is spring loaded and will return to the OFF position automatically, when released.



Windshield Wiper Circuit
Figure 1

D. Wiper Motor

(1) Two wiper motors, one for each system, are used in the aircraft. The motors are rated at 1/2 HP, each drawing approximately 2.5 amps.

E. Flexible Drive Cable

(1) Flexible drive cables are connected between the motor and converter to transmit power to the converter.

F. Converter

(1) Two converters, one for each wiper system, converts the rotating force of the motors to a reciprocating motion to drive the wiper blades. A worm gear in the converter provides a 100:1 reduction of motor rpm; each revolution produces two strokes, one in each direction.

(2) Automatic parking is provided by a micro-switch attached to the rear of the converter. The micro-switch is actuated by a cam shut-off when the control switch is placed in the PARK position.

G. Drive Arm Assembly

(1) The drive arm assembly is attached to the shaft of the converter. A spring incorporated in the drive arm provides the correct blade pressure on the windshield.

H. Blade Assembly

(1) The blade assembly is attached to the drive arm by a spring clip. The blade assembly is flexible to follow the contour of the windshield.

WINDSHIELD WIPER – MAINTENANCE PRACTICES

1. Removal/Installation – Windshield Wiper

A. Removal – Wiper Motor

(1) Remove glareshield.

(2) Remove necessary instruments allowing access to windshield wiper components.

(3) Disconnect electrical connector.

(4) Remove flex cable.

(5) Remove screws attaching motor to mounting bracket.

(6) Remove motor by lifting vertically.

B. Installation – Wiper Motor

(1) Position motor and secure to bracket with screws.

(2) Connect flex cable and safety wire.

(3) Connect electrical connector.

(4) Install removed instruments.

(5) Install glareshield.

C. Removal – Wiper Motor Relay

(1) Remove glareshield.

(2) Remove necessary instruments to allow access to windshield wiper components.

(3) Through removed instrument holes remove electrical connections.

(4) Remove screws attaching relay to mounting bracket.

(5) Remove relay vertically.

D. Installation – Wiper Motor Relay

(1) Position relay and attach to bracket with screws.

(2) Attach electrical leads to relay.

(3) Install removed instruments.

(4) Install glareshield.

E. Removal – Wiper Converter

(1) Remove glareshield.

(2) Remove necessary instruments allowing access to converter.

(3) Remove windshield wiper motor.

(4) Remove sealant around converter drive shaft at aircraft structure.

(5) Disconnect flex drive cable.

(6) Remove bolts and spacers attaching converter to mounting bracket.

(7) Remove converter by lifting vertically.

F. Installation – Wiper Converter

(1) Position converter and secure to bracket with bolts and spacers.

(2) Connect flex drive cable.

(3) Reseal converter drive shaft at aircraft structure with PR1221.

(4) Install windshield wiper motor.

(5) Install removed instruments.

(6) Install glareshield.

2. Adjustment/Test – Windshield Wiper

A. Adjust Parking of Wiper Blades

CAUTION: DO NOT OPERATE WIPER ON DRY SURFACE.

(1) Operate wiper and stop blades at the extreme left or right side of the windshield, depending on the side most suitable for parking blades.

(2) Rotate the cam shut-off slowly until contact with micro switch is made. At this point, hold cam to keep from turning and tighten cap screw with an allen wrench. Turn wiper ON, then to PARK, and observe that blades stop in the same position each time.

WINDSHIELD ANTI-ICE – DESCRIPTION AND OPERATION

1. General

A. The windshield incorporates an electrical heating grid to provide an anti-ice capability in the critical viewing area of the windshield. The heating grid is a series of very fine wires embedded in the interlayer adjacent to the outer glass ply and is electrically connected to upper and lower busbars. The busbars are electrically connected to terminal block in the lower, inboard corner on the inside surface of the windshield.

As part of the anti-ice control system, two Resistance Temperature Detectors (RTD) are incorporated into the windshield. These RTDs are located in the interlayer immediately adjacent to the glass outer ply and are electrically connected to the terminal block located on the inboard, lower, inside surface of the windshield. They measure electrical resistance which provides a direct correlation to the temperature of the windshield. Only one of the two RTDs is utilized in the windshield heating system at any time with the other RTD serving as a spare.

The electric terminal block is located on the inside surface of the windshield in the lower inboard corner. The terminal block incorporates a total of five electrical posts, two of which have 10-32 UNF threads (G and P) and three of which have 6-32 UNC threads (S1, S2, and SG) with each post on the terminal block given a designation.

Each windshield system consists of a temperature controller, electrical heat power relay, windshield heat circuit breakers, temperature controller circuit breakers, and **"L"** or **"R"** green **"WS HEAT"** annunciation located on the main annunciation panel. A single three-position windshield heat **"HI/LO/OFF"** selection switch turns both heating systems on. In the **"HIGH"** and **"LOW"** position, the controllers energize the heating element relays and simutaneously the **"L"** and **"R"** green **"W/S HEAT"** annunciation located on the main annunciation panel. The electrical resistance in the windshield RTD sensors changes as a function of temperature and they are calibrated such that when the windshield is less than 95°F (35°C), the heat relay is closed to apply electrical power to the windshield heating grid. When the windshield temperature is greater than 104°F (40°C), the heat relay is opened to remove electrical power to the windshield heating grid. The **"L"** and **"R"** green **"W/S HEAT"** annunciation will cycle on and off as this occurs.

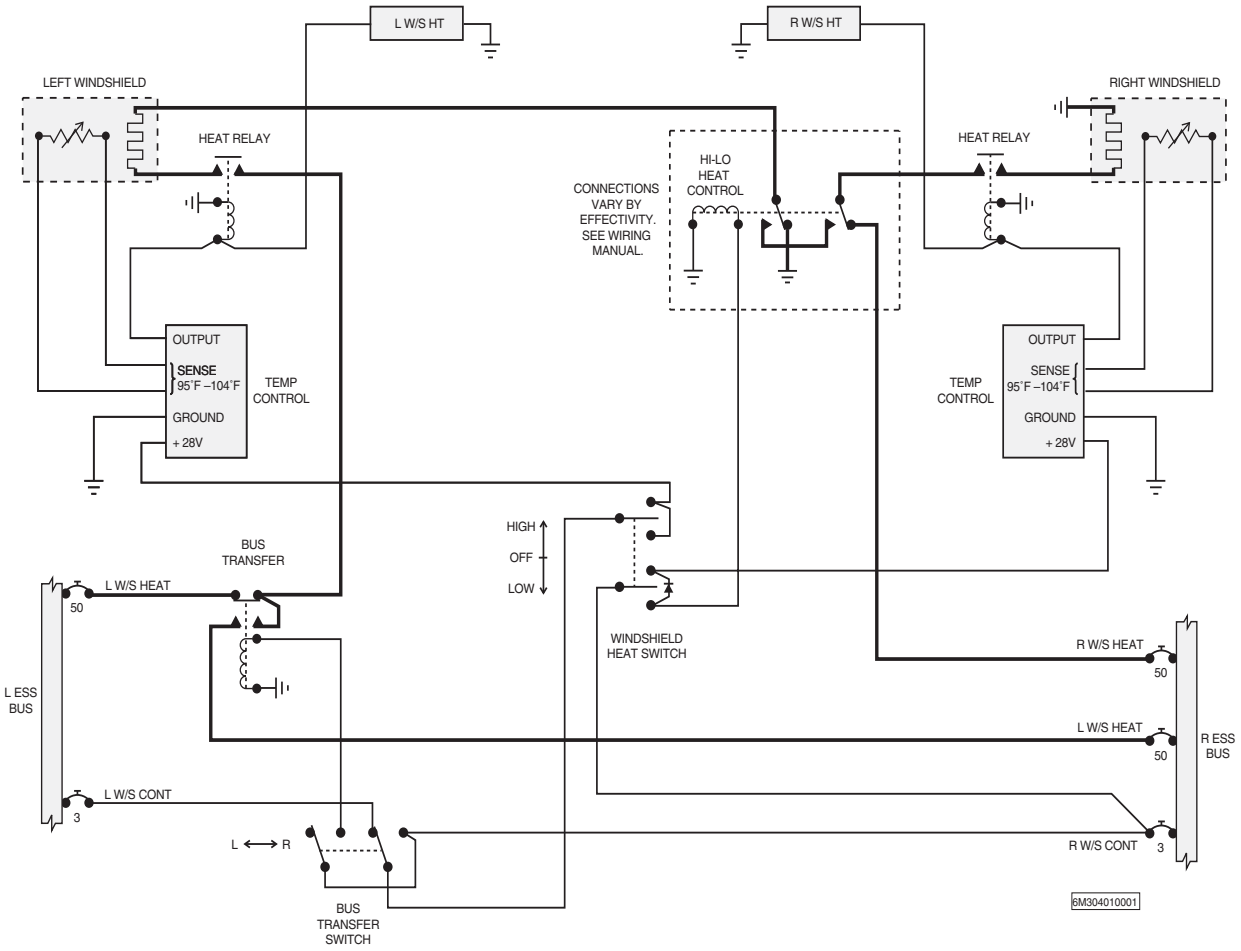
28VDC power for the operation of the windshield heat is provided by the left and right essential buses. With **"HIGH"** heat selected, electrical system voltage is applied to each windshield from its respective bus. With **"LOW"** heat selected, both windshields are connected in series to the Left Essential Bus. Selecting **"OFF"** de-energizes the windshield controllers and windshields.

The windshield heating system also incorporates a bus transfer switch which when selected to **"BUS TRANSFER RIGHT"** provides an alternate power source for the left windshield and controller from the R Essential Bus circuit breakers.

B. Windshield (see Figure 1)

(1) The windshield assembly is composed of an inner and outer thermally toughened pane of glass, with a polyurethane and polyvinyl butyral layer sandwiched between. The power to produce heat energy is dissipated over the windshield by means of a transparent conductive grid applied to the inner surface of the outer pane. Electrical connections to the conductive grid are made by busbars along the upper and lower edges of the glass. A temperature sensing element (i.e. RTD) is imbedded in the windshield and regulates the electrical energy to the windshield.

WINDSHIELD ANTI-ICE – DESCRIPTION AND OPERATION



Windshield Heat Electrical Schematic
Figure 1

C. Heat Relays (see Figure 1)

(1) Heat relays, located under the cockpit floor at FS 88.0, control electrical power to each windshield. The relays are energized by the output of the temperature controller. When the relay is closed voltage is applied to the windshield heating grid.

D. Hi-Lo Heat Control Relay (see Figure 1)

(1) The hi-lo heat control relay, located under the cockpit floor on the aft side of the frame at FS 79.0, is controlled by the **"HI/LO/OFF"** switch on the left hand switch panel. When the switch is in the **"HI"** position, the hi-lo heat control relay is de-energized. The left and right heat relays and heating elements are isolated to their respective sides. 28.5 VDC is applied to each heater element. With the switch in the **"LO"** position the hi-lo transfer relay is energized. The heat relays and heating elements are connected in series. 28.5 VDC is applied to the heater elements connected in series.

E. Temperature Control Unit

(1) The temperature control units, one for each pilot's windshield, mounted on the aft side of the forward pressure bulkhead, control the application of voltage to the windshield heat relays. When the windshield temperature is greater than 104°F (40°C), the heat relays de-energize, removing the voltage from the heater grid. As the temperature of the windshield drops below 95°F (35°C), the temperature controller energizes the heat relays, applying voltage to the heater grid.

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WINDSHIELD ANTI-ICE – MAINTENANCE PRACTICES

1. Removal/Installation – Windshield Anti-ice

A. Removal – Heat and Hi - Lo Relays

(1) Pull and tag windshield heat control and power circuit breakers.

(2) Remove pilot's seat in accordance with Chapter 25, this manual.

(3) Remove floorboards.

(4) Disconnect wiring to hi-lo relay, located aft side of the frame at FS 79, or heat relay, located forward side of frame at FS 88.

(5) Remove screws securing relay(s) to frame.

B. Installation – Heat and Hi - Lo Relays

(1) Attach relay(s) to frame with screws.

(2) Connect electrical wiring to relay.

(3) Install floorboards.

(4) Install pilot's seat in accordance with Chapter 25, this manual.

(5) Remove tag and reset windshield heat control and power circuit breakers.

C. Removal – Temperature Control Unit

(1) Pull and tag windshield heat control and power circuit breakers.

(2) Remove pilot's seat in accordance with Chapter 25, this manual.

NOTE: Temp/Control/Units are attached to a bracket on the forward pressure bulkhead.

(3) Disconnect electrical connector.

(4) Remove screws attaching bracket to structure.

(5) Remove screws attaching units to bracket.

D. Installation – Temperature Control Unit

(1) Position control units on mounting bracket and secure with screws.

(2) Connect electrical connector.

(3) Attach bracket to structure with screws.

(4) Install pilot's seat in accordance with Chapter 25, this manual.

(5) Remove tag and reset windshield heat control and power circuit breakers.

2. Adjustment/Test – Windshield Anti-ice

A. Functional Test – Temperature Control unit

(1) Tools Required

(a) Decade variable resistor, 0 to 400 ohms, of laboratory accuracy. (A common 500 ohm potentiometer may be used with less accurate results).

(b) D.C. Voltmeter.

(2) Test Procedures

(a) Disconnect leads from the left or right temperature control unit, located on panel EP3 forward of the pilot's rudder pedals. Terminals 1 and 2 (sensing).

(b) Connect the variable resistor to terminals 1 and 2 (sensing), of the temperature control unit.

(c) Set the variable resistor to less than 300 ohms.

(d) Pull the windshield heat circuit breakers to prevent windshield overheating.

(e) Close the battery switch and place windshield heat switch to high.

(f) Observe W/S heat cycle light illuminates on the annunciator panel.

(g) Check temperature control unit output voltage (terminal 8 to ground). This should be battery voltage.

(h) Increase resistance of the variable resistor until W/S heat cycle light extinguishes and heat relay opens with an audible click.

(i) Check temperature control unit output voltage (terminal 8 to ground). This should be 0 volts.

(j) Check resistance of the decade variable resistor or disconnect potentiometer and measure resistance then reconnect.

NOTE: Normal resistance necessary to turn off the temperature control unit is 335 +/- 10 ohms.

(k) Decrease resistance of the variable resistor until W/S heat cycle illuminates and heat relay closes with a audible click.

(l) Check resistance of the decade variable resistor or disconnect potentiometer and measure resistance.

NOTE: Normal resistance necessary to turn off the temperature control unit is 332 +/- 10 ohms.

(m) Remove the variable resistor from the temperature control unit.

(n) Reconnect leads to terminals 1 and 2 of the temperature control unit.

(o) Reset the windshield heat circuit breakers pulled in step **2**., A., (2), (d).

(p) Perform an operational test of the windshield heat system.

B. Operational Test – Low Operation

(1) Switch aircraft electrical power ON.

(2) Position WSHLD HT switch to LOW.

(3) Verify L W/S HT and RT W/S HT annunciator lights illuminate and listen for audible click from heat relay and heat control relay. An audible click, louder than high operation will be heard due to third relay closing.

NOTE: W/S heat cycle lights may illuminate briefly or not at all on warm days because windshield panels are already at normal operating temperature 90°F to 110°F (32°C to 43°C.) Cooling windshield panels with water may be required during operational test.

(4) Verify evidence of heating as a slight distortion of vision through windshield panels (less than high operation).

(5) Verify either L W/S HT or R W/S HT light extinguishes, with an audible click from heat relay, as windshield reaches operating temperature.

(6) Verify operation of heat control relay by moving WSHLD HT switch to HIGH until both L W/S HT and R W/S HT lights extinguish.

(7) Quickly move WSHLD HT switch between OFF and LOW before windshield cools enough allowing illumination of L W/S HT and R W/S HT lights and closing of heat relays. Verify audible click of heat control relay.

NOTE: One W/S heat cycle light may remain illuminated since power is removed from both heating elements if either heat relay opens during low (series) operation.

(8) Position WSHLD HT switch to OFF.

(9) Switch aircraft electrical power OFF.

C. Operational Test – Bus Transfer Operation

(1) Pull LEFT ESSENTIAL BUS 50 Amp circuit breaker (L WINDSHIELD; HEAT) and 3 Amp circuit breaker (L WINDSHIELD; CONT) to simulate loss of electrical power.

(2) Switch aircraft electrical power ON.

(3) Position WINDSHIELD HEAT bus transfer switch to RIGHT BUS. Verify Bus transfer relay closes with an audible click.

NOTE: Windshield heat operates normally in both HIGH and LOW with ESSENTIAL BUS TRANSFER switch in RIGHT BUS.

(4) Switch aircraft electrical power OFF.

(5) Position WINDSHIELD BUS TRANSFER switch to LEFT BUS.

(6) Reset LEFT ESSENTIAL BUS circuit breakers.

D. Functional Test – Windshield Panel

(1) Remove glareshield and instrument panel equipment necessary to gain access to windshield panel wire terminals.

(2) Remove two wires attached to windshield panel terminals labeled S (Sensor).

(3) Verify resistance between terminals is 330 to 340 ohms. Resistance is higher with increasing windshield panel temperature.

(4) Connect two wires to windshield panel terminals labeled S.

(5) Remove two wires attached to windshield panel terminals labeled P (Positive) and G (Ground.)

(6) Verify resistance between terminals is less than one (1) ohm.

(7) Connect wires to windshield panel terminals labeled P and G.

(8) Install instrument panel equipment and glareshield.

E. Functional Test – Temperature Control

CAUTION: TO PREVENT OVERHEATING OF WINDSHIELD PANELS, PULL L WINDSHIELD; HEAT, 50 AMP CIRCUIT BREAKER AND R WINDSHIELD; HEAT, 50 AMP CIRCUIT BREAKER BEFORE PERFORMING TEMPERATURE CONTROL TEST.

(1) Remove wires from temperature control terminals 1 and 2.

(2) Connect a decade variable resistor, 0 to 400 ohms, of laboratory accuracy, between temperature control terminals 1 and 2. Set resistor to less than 300 ohms. A 500 ohm common potentiometer may be used with less accurate results.

(3) Switch aircraft electrical power ON.

(4) Verify L W/S HT and R W/S HT lights illuminate and listen for heat relay closing with audible click.

(5) Increase resistance until L W/S HT and R W/S HT lights extinguish and heat relay opens with audible click.

(6) Check resistance of decade variable resistor or disconnect potentiometer and measure resistance; then reconnect. Normal resistance necessary to turn temperature control OFF is 335 ohms.

(7) Decrease resistance until L W/S HT and R W/S HT light illuminates and heat relay closes with audible click.

(8) Check resistance of decade variable resistor or disconnect potentiometer and measure resistance; then reconnect. Normal resistance necessary to turn temperature control ON is 332 ohms.

(9) Connect wires to temperature control terminals 1 and 2.

(10) Reset L and R Windshield heat circuit breakers.

F. Required Equipment

(1) Voltmeter 0-30 VDC accuracy 1% F.S.

(2) Power supply 28 ± 0.5VDC 1 amp min. rating.

(3) Helipot (TR1) 100 ohms ±1 ohm, Beckman lab model T10-A 0.22 AMP, 5 watt.

(4) Switch (S1, S2) SPST 1 amp min. contact rating.

(5) Resistor (R1) 300 ohms 0.1% 1 watt.

(6) Resistor (R2, R3) 125 ohms 5% 10 watts.

(7) Lamp (DS1) 40 MA at 28VDC.

G. Functional Test – Windshield Heat Controller (Figure 201)

(1) Set S1 and S2 in open position.

(2) Connect controller to test fixture (see Figure 201).

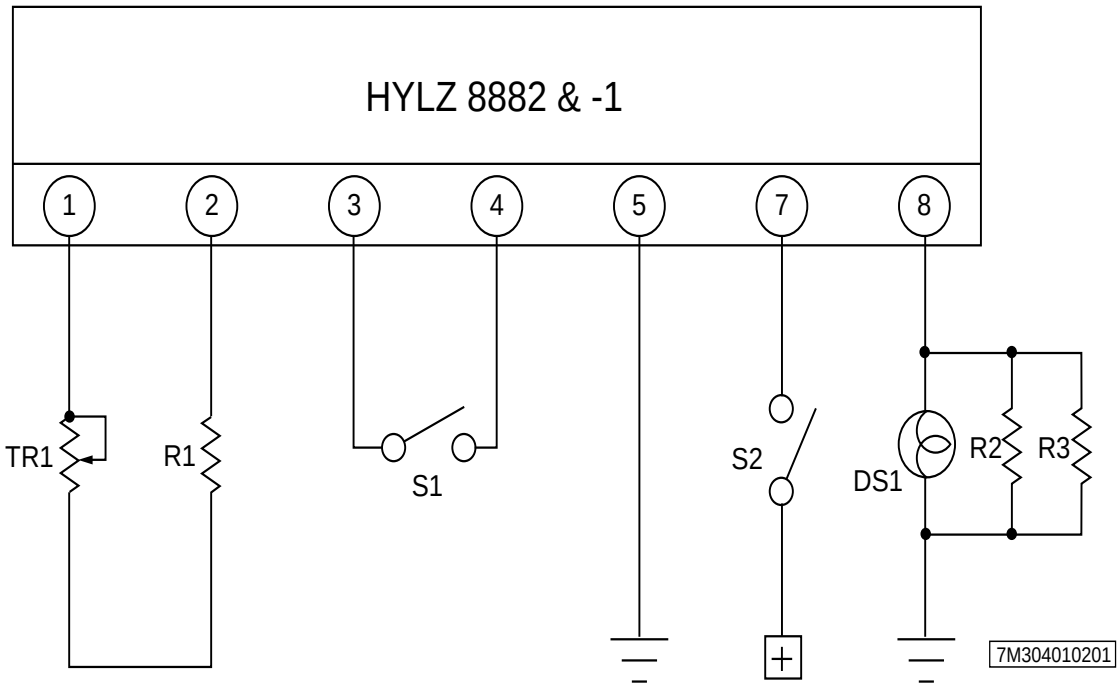
(3) Close S2.

(4) Set power supply voltage to 28 ± 0.5VDC.

(5) Set TR1 to 46.0 Ohms (118.5°F). Ensure DS1 is OFF.

(6) Slowly decrease TR1 until DS1 turns ON. Record TR1 resistance.

(7) Slowly increase TR1 until DS1 turns OFF. Check that TR1 resistance is between 33.5 Ohms (102°F) and 36.5 Ohms (106°F). Record TR1 resistance.



Windshield Heat Controller Test Fixture
Figure 201

(8) Subtract resistance recorded in **2.**, F., (6) from **2.**, F., (7). Check that the difference is between 3.0 Ohms (4.45°F) (Minimum) and 8.5 Ohms (13.6°F) (Maximum).

(9) Close S1.

(10) Set TR1 to 55 Ohms (132.9°F). Ensure DS1 is OFF.

(11) Slowly decrease TR1 until DS1 turns ON. Record TR1 resistance.

(12) Slowly increase TR1 until DS1 turns OFF. Check that the resistance is between 45.0 Ohms (118°F) and 49.0 Ohms (125°F). Record TR1 resistance.

(13) Subtract resistance recorded in **2.**, F., (11) from **2.**, F., (12). Check that the difference is between 3.0 Ohms (4.45°F) (Minimum) and 8.5 Ohms (13.6°F) (Maximum).

(14) Open S2.

(15) Remove controller from fixture.

WINDSHIELD HEAT MONITORING SYSTEM – DESCRIPTION AND OPERATION

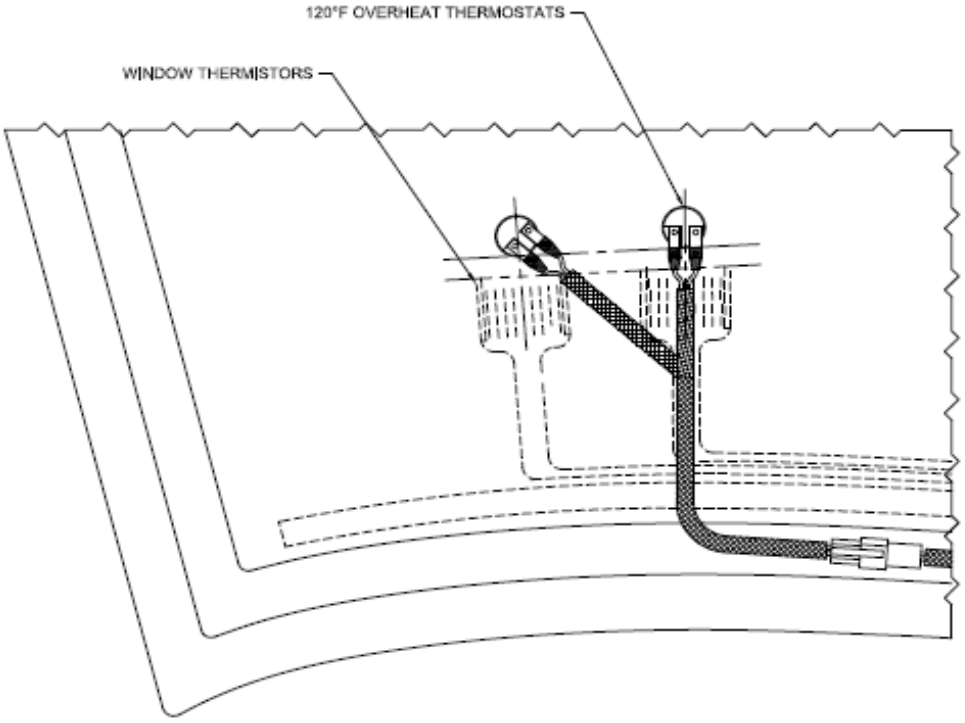
1. General

A. A windshield heat monitoring system has been incorporated to provide a "RED" annunciation, warning the pilot to take corrective action in the event of uncontrolled heating of the left or right windshield heating grid. The thermostatic switches attached to the window will activate the red "L W/S HT" or "R W/S HT" windshield overheat warning light in the annunciators located on the pilot and copilots glareshield. The lights will illuminate when the inner glass temperature of the windshield reaches nominal 120°F (48.9° C).

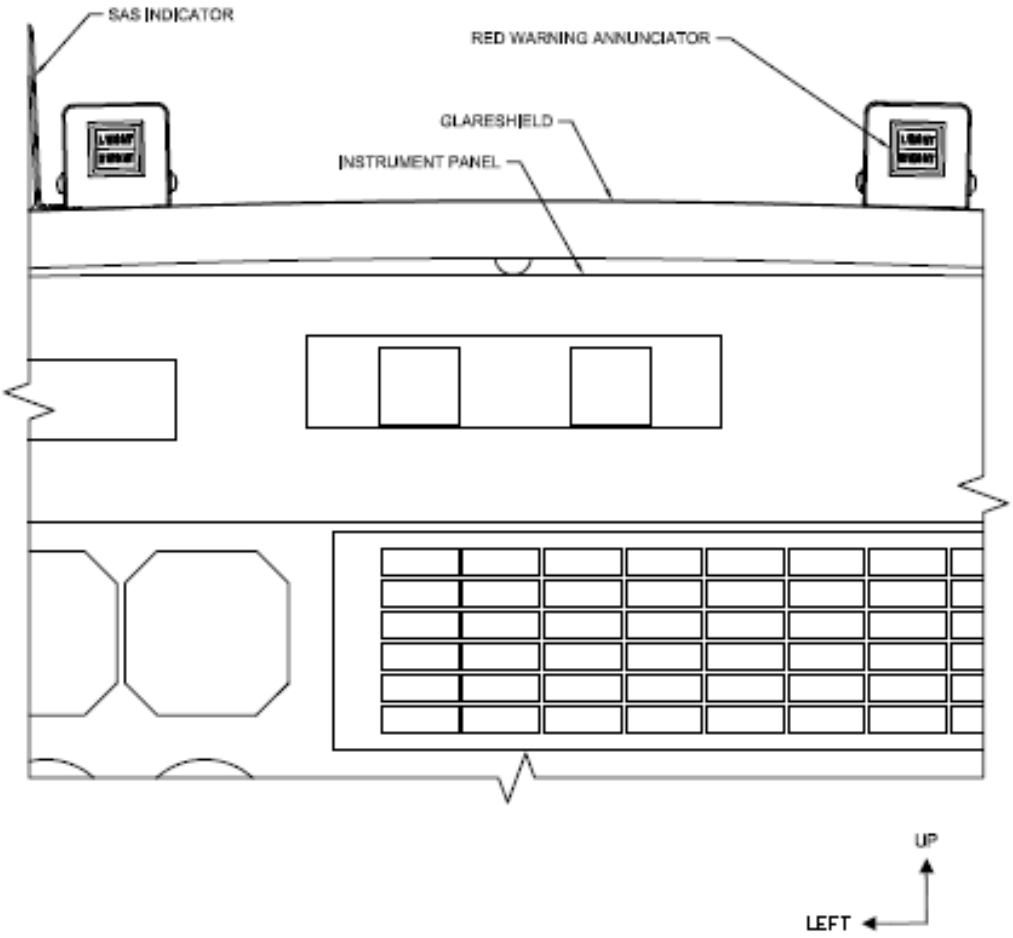
NOTE: The windshield heat monitoring annunciation is operational when "HIGH HT" is selected only. This prevents eroneous annunciation during hot day operation when only "LOW HT" is required.

NOTE: This system is only applicable to SA226 series aircraft with 27-19442 or 27-19482 windshields having dual lower temp sensors along the bottom edge of the windshield.

The system consistes of LH and RH annunciator assemblies, LH and RH thermostat switches, and associated wiring. The two button thermostatic switches are bonded to the inner glass surface of the LH and RH windshields using P/N 08752 3M mirror adhesive. The thermostatic switches are located just above the windshield RTD (Resistance Temperature Detectors) sensors (see Figure 1).Pilot and copilot annunciator assemblies are mounted on top of the glareshield just forward of the glareshield lip in a central location and provides the crew with a PTT (Push-to-Test) function (see Figure 2). The light(s) will indicate a red "L W/S HT" or "R W/S HT" if either windshield has reached the 120°F (48.9° C) temperature.



Thermostatic Switches
Figure 1



Annunciator Light Location
Figure 2

WINDSHIELD HEAT MONITORING SYSTEM – MAINTENANCE PRACTICES

1. Removal/Installation - Windshield Heat Monitor

   A. Removal - Thermostat Switch Assembly

      (1) Pull L and R WINDSHIELD CONT circuit breakers.

      (2) Disconnect windshield thermostat switch connector P9001 and/or P9002 as applicable.

      (3) Apply masking tape to outer surface of windshield and mark position of thermostats.

      (4) Grasp body of thermostat firmly with pliers and twist thermostat to break bond. Apply twisting motion parallel to windshield glass surface to prevent damage to windshield glass pane.

      (5) Repeat step 1.A.(4) above to remove remaining thermostat.

      (6) Remove adhesive residue from windshield using a sharp razor blade.

   B. Installation - Thermostat Switch Assembly

      NOTE: Due to possible damage to thermostat switches during removal, thermostat switches **MAY NOT** be reused.

      NOTE: Windshield Heat Monitor thermostat switches must be adhered to windshield with P/N 08752 Mirror Adhesive available from 3M Company (CAGE code 34360).

      (1) Ensure no adhesive residue is present in thermostat switch mounting area. Clean inner surface of windshield around mounting area of thermostat switches IAW Chapter 12-20-15, this manual.

      (2) If switch position is not already marked per step 1.A.(3) above, apply masking tape to outer surface of windshield and mark thermostat switch position per Figure 201.

      (3) Prepare glass surface at final position mounting and thermostat switch mounting surfaces per adhesive kit instructions. **DO NOT** use razor blade to clean thermostat switch housing.

      (4) Position and adhere thermostat switches to windshield per adhesive kit instructions.

      (5) Allow adhesive to cure per adhesive kit instructions.

      (6) Connect windshield thermostat switch connector P9001 and/or P9002 as applicable.

      (7) Reset L and R WINDSHIELD CONT circuit breakers.

      (8) Perform operational test of Winshield Heat Monitor System per step 2.A. of the section.

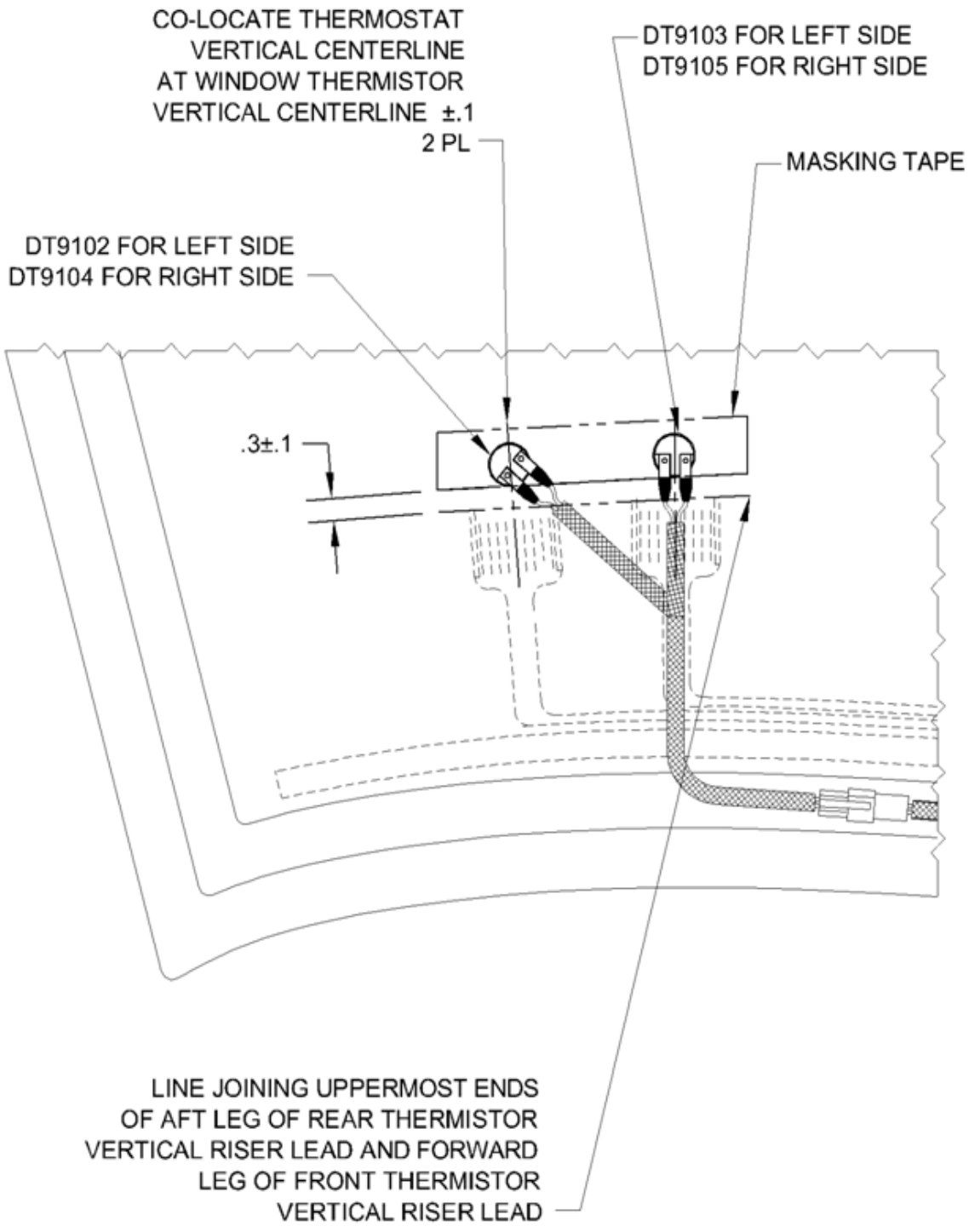
      (9) Seal any exposed portion of thermostat switch electrical terminals IAW SA226 Wiring Manual.

   C. Removal - Annunciator Assembly

      (1) Pull L and R WINDSHIELD CONT circuit breakers.

      (2) Loosen and lift instrument panel glareshield. Retain hardware for reinstallation.

      (3) Disconnect windshield annunciator assembly connector P9003 and/or P9004 as applicable. Extract wires from connector using 11-03-0002 extractor (Molex Incorporated) (CAGE code 1UX99).



VIEW LOOKING AT INNER SURFACE OF LEFT WINDSHIELD
RIGHT SIDE OPPOSITE
FIGURE 201

(4) Remove screws securing annuciator assembly to glareshield. Remove tie straps securing annuciator harness to glareshield.

(5) Lift annuciator assembly from glareshield. Guide wire harness through grommet and out of glareshield pass-through.

D. Installation - Annunciator Assembly

(1) Guide wire harness through grommet in glareshield. Secure annunciator to glareshield and tie strap wire harness to tie mount.

(2) If new annunciator is being installed, cut harness to length and terminate connector pins using 63819-1300 crimper (Molex Incorporated) (CAGE code 1UX99).

(3) Insert pins in connector P9003 and/or P9004 as applicable using 11-02-0001 insertion tool (Molex Incorporated) (CAGE code 1UX99).

(4) Connect P9003 and/or P9004 as applicable.

(5) Reinstall glareshield using hardware retained in step 1.C.(2).

(6) Reset L and R WINDSHIELD CONT circuit breakers.

(7) Perform operational test of Winshield Heat Monitor System per step 2.A. of the section.

(8) Seal any exposed portion of thermostat switch electrical terminals IAW SA226 Wiring Manual.

2. Adjustment/Test - Windshield Heat Monitor

A. Operational Test - Windshield Heat Monitor

(1) Pull L and R WINDSHIELD HEAT circuit breakers.

(2) Apply electrical power to the aircraft.

(3) Set Windshield Heat switch to HIGH.

(4) Press and hold switch cap on Pilot's Windshield Overheat Indicator. Verify upper and lower sectors illuminate. Release switch cap. Verify upper and lower sectors extinguish. Repeat for Copilot's Windshield Overheat Indicator.

(5) Individually short terminals of each thermostat switch and verify results per Table 201. Indicator segment must illuminate with red background and black text characters. Only the indicator segment listed should illuminate, all others must remain extinguished.

(6) Select Windshield Heat to LOW. Repeat tests per Table 201. No indicator sectors may illuminate with LOW selected.

(7) Select Windshield Heat to OFF. Remove electrical power from aircraft and reset L and R WINDSHIELD HEAT circuit breakers.

(8) Seal any exposed portion of thermostat switch electrical terminals disturbed during operational test IAW SA226 Wiring Manual.

THERMOSTAT	INDICATOR	SECTOR	TEXT	PASS/FAIL
DT9102	PILOT	UPPER	L W/S HT	
DT9103	COPILOT	UPPER	L W/S HT	
DT9104	PILOT	LOWER	R W/S HT	
DT9105	COPILOT	LOWER	R W/S HT	

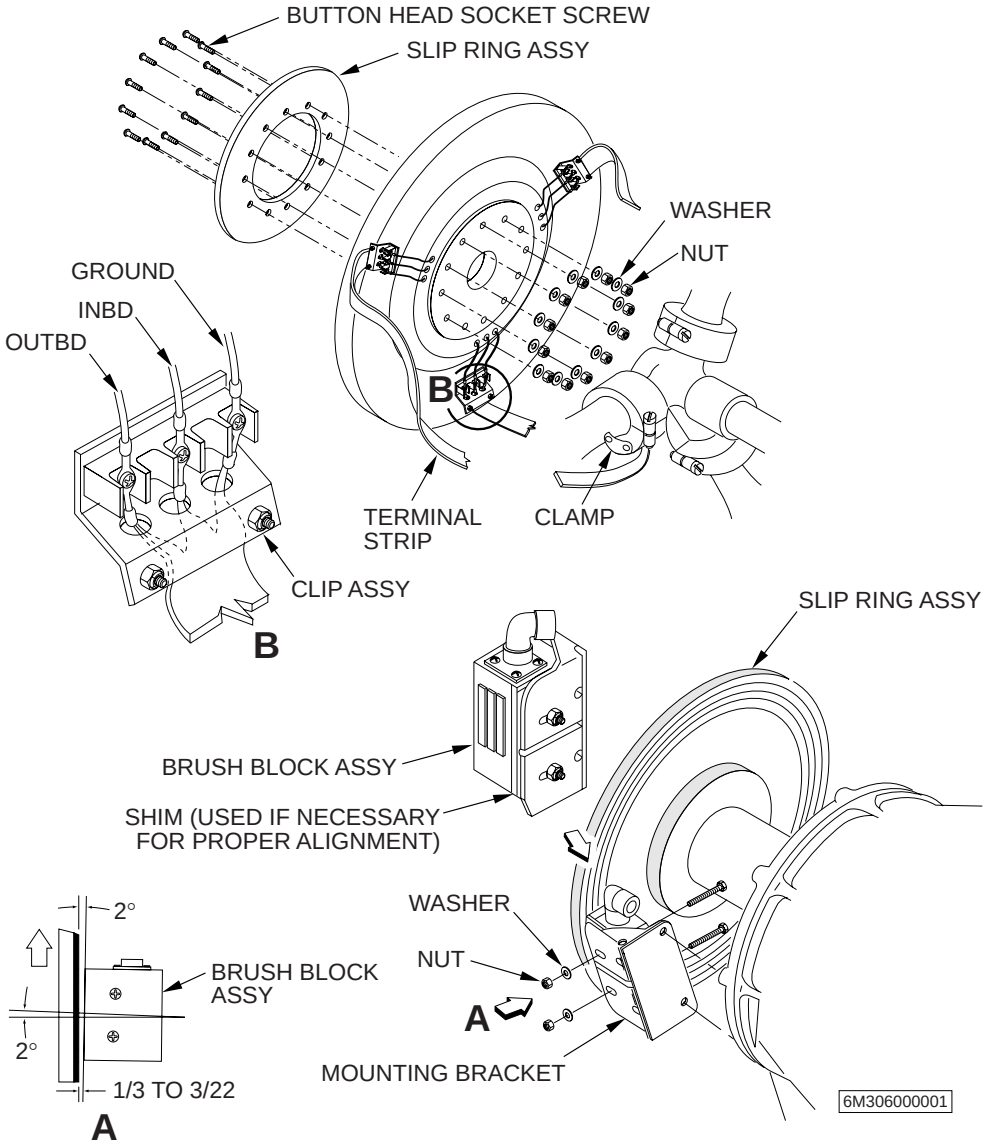
Windshield Heat Monitor Operational Test
Table 201

PROPELLER DEICE – DESCRIPTION AND OPERATION

1. General

A. The propeller deicing system is comprised of electrically heated deice boots bonded to each propeller blade, a slip ring and brush-block assembly to transfer electrical current to the heating elements in the deice boots, two electronic timers, an ammeter, and two control switches (see Figure 1 for components).

B. During system operation, the amount of current applied to each boot is monitored by means of a toggle switch located on the left side of the ammeter. The up position monitors the current flowing through the left propeller deice boots, while the down position monitors the current flowing through the boots on the right propeller.

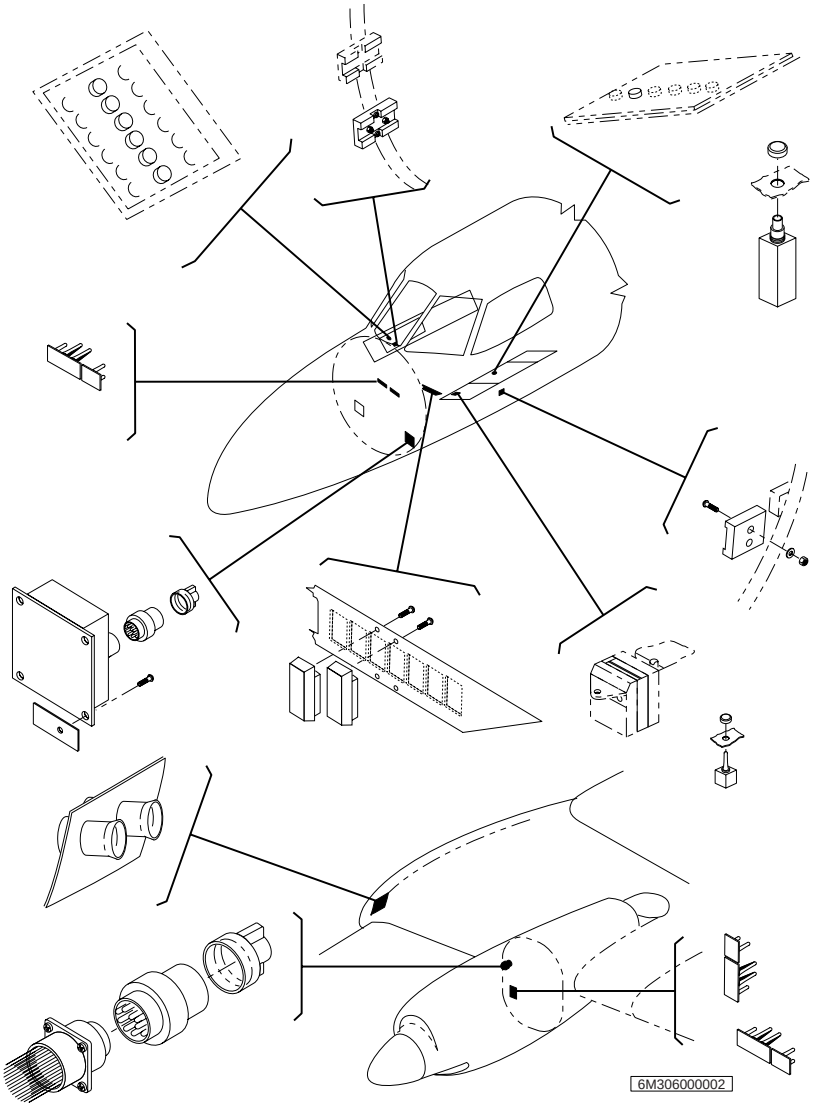


Propeller Deice Components
Figure 1

C. Current to each propeller deice boot flows from the respective timer to the brush assembly mounted on the front of each engine case and is conducted by the brush assembly to the slip rings installed on the spinner backing plate. The slip rings distribute current to the deice boots on the propeller blades. Heat from the boots reduce the grip of the ice, which is then removed by the centrifugal force of propeller rotation and the blast of the air load.

D. Power to the two heating elements in each boot on each blade is cycled by the timer to the outboard and inboard heating elements in the following sequence: RH outboard, RH inboard, LH outboard, and LH inboard. Since each of these phases is 30 seconds in duration, the timer makes a complete cycle every two minutes. When the timer switches phases of operation, the ammeter will register a momentary deflection.

E. Normal current flow to the boots is 14 to 18 amperes, registered on the ammeter. If the current rises beyond the switch limit, an integral circuit breaker will open, removing power from the system.



Propeller Deice Controls
Figure 2

PROPELLER DEICE – DESCRIPTION AND OPERATION

2. Components

A. Timer

(1) A sealed timer is installed for each propeller. The left hand timer is located on the left hand aft side of the forward pressure bulkhead and the right hand timer is located under the pilot's seat. The timer delivers current through the slip ring and brush block arrangement to the outer heaters and then the inner heaters.

B. Deicer

(1) The deicer is bonded to each propeller blade. It consists of special heater wires protected by fabric plies and by oil and abrasion-resistant rubber. Each deicer has a separate lead for the inboard and outboard heater and a third lead which is a common ground.

C. Slip Rings – Brushes and Brush Block

(1) A brush block assembly is attached to the engine to transfer electrical power to the rotating deicers. The brushes are spring loaded to press against the revolving slip rings, which are connected to the deice boots.

PROPELLER DEICE – FAULT ISOLATION

1. Troubleshooting – Propeller Deice Boots

A. To assist in the isolation of system faults, the following troubleshooting block diagrams are provided.

B. For additional information and schematic diagrams refer to the SA226 Wiring Manual.

C. The prerequisites at the beginning of each diagram require an Operational Test of the system to determine which problem statement applies.

D. In the event of multiple problems correct the faults in the order they appear below.

PROPELLER DEICE – FAULT ISOLATION

PROBLEM:

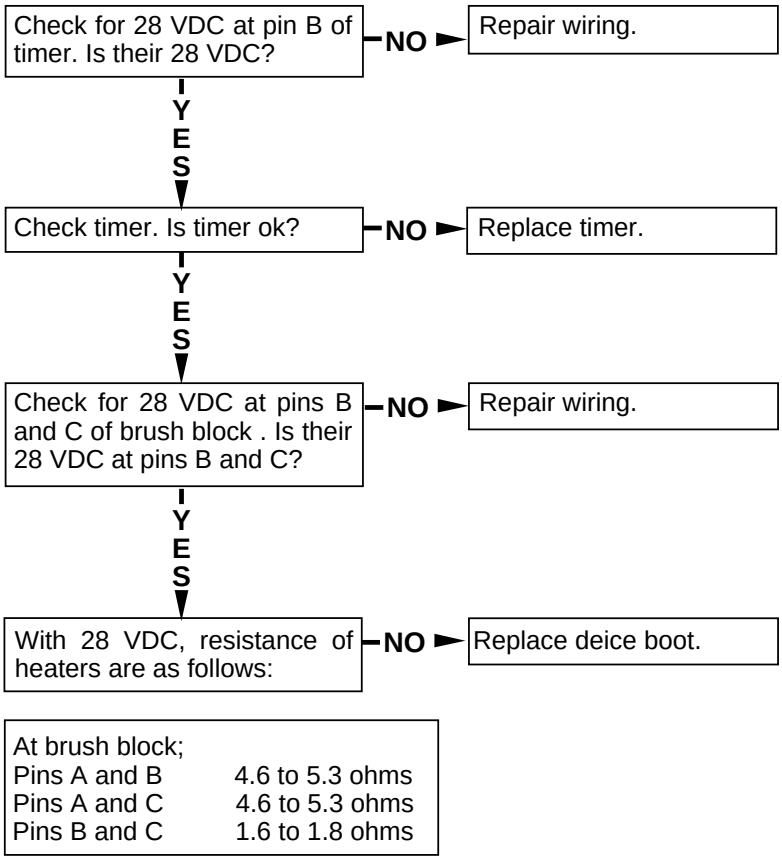
DEICING ELECTRICAL LOAD IN-OPERATIVE.

CAUTION: DO NOT OPERATE PROPELLER DEICE CONTINUOUSLY FOR MORE THAN FIVE MINUTES UNDER HIGH AMBIENT TEMPERATURE CONDITIONS (ABOVE 100°F [37.8°C]) WITHOUT ALLOWING THEM TO COOL DOWN.

PREREQUISITES:

Test propeller deice. See Adjustment/Test – Propeller Deice.

START:



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PROPELLER DEICE – MAINTENANCE PRACTICES

1. Removal/Installation – Propeller Deice

A. Special Tools and Materials for removal and installation of propeller deice boot:

(1) Toluol.

(2) Methyl propyl ketone (MPK) – Eastman Chemical Co.

(3) Cement EC-1403 – Minnesota Mining and Manufacturing Co.

(4) Cement A-56-B – B.F. Goodrich Co.

(5) Rubber Roller.

(6) 1/4 Inch Steel Roller.

B. Removal – Propeller Deice Boots

NOTE: Propellers are normally shipped from the factory with blade deice boots installed.

(1) Remove screws securing propeller spinner and remove spinner.

(2) Disconnect blade deice electrical leads from terminal block on face of spinner bulkhead.

(3) Disconnect electrical leads from terminal block on blade butt and hub.

(4) Remove clamp holding strap to propeller hub.

WARNING: TOLUOL OR METHYL PROPYL KETONE (MPK) ARE TOXIC, EXTREMELY FLAMMABLE, AND CAUSTIC. USE ONLY IN A WELL VENTILATED AREA. OBSERVE SAFETY PRECAUTIONS. AVOID EYE AND SKIN CONTACT, BREATHING OF VAPORS. AND INGESTION.

(5) Use toluol or (MPK) to soften adhesion line between deice boot and propeller blade.

(6) Starting at one corner of deice boot, loosen enough to grasp boot in jaw of vise-grip-plier or similar tool.

(7) Apply a steady pull on boot to remove from propeller blade. Continue using Toluol or MPK to soften adhesion. Remove boot slowly and carefully.

(8) Remove residual cement from propeller blade. Using solvent as necessary.

C. Installation – Propeller Deice Boots

CAUTION: FOLLOW MANUFACTURERS RECOMMENDATIONS ON USE OF PRODUCTS. CEMENT WILL SET IN ABOUT ONE HALF HOUR DEPENDING ON HUMIDITY. ALLOW 12 HOURS FOR THE CEMENT TO CURE BEFORE STARTING THE ENGINE. ALLOW 24 HOURS FOR THE CEMENT TO CURE BEFORE OPERATING THE DEICER BOOT.

(1) Position deice boot on propeller blade so that centerline at inboard is aligned with split in propeller blade clamp. Ensure that lead straps are in proper position to be clamped to blade retaining clamp.

(2) Mask off an area approximately 1/2 inch from outer end and each side of boot.

CAUTION: METAL AND RUBBER SURFACES MUST BE THOROUGHLY CLEAN TO OBTAIN MAXIMUM ADHESION.

(3) Thoroughly mix EC-1403 cement (3, Special Materials) and apply one even brush coat to blade.

(4) Allow cement to dry for at least one hour at 40°F (4.44°C) or above, when relative humidity is less than 75%. Allow two hours drying time if relative humidity is between 75 and 90%. Do not apply cement if relative humidity is above 90%.

(5) Apply a second even brush coat of EC-1403 cement to blade and one coat of cement to unglazed surface of deice boot.

NOTE: It is not necessary to cement more than 1/2 inch of deice strap.

(6) Position deice boot on propeller blade, starting one inch from blade retaining clamp.

(7) Ensure lead strap is in position to clamp to blade retaining clamp.

(8) Moisten cement lightly with Toluol or MPK and tack boot centerline on blade leading edge.

(9) If deice boot should be misaligned, pull it up immediately with a quick motion and reposition.

(10) Roll firmly along centerline with a rubber roller.

CAUTION: NEVER USE A METAL OR WOODEN ROLLER FOR THIS PURPOSE. DAMAGE TO HEATING ELEMENTS MAY RESULT.

(11) Gradually tilting roller, work boot carefully over each side of the blade contour. Avoid trapping air under boot.

(12) Roll outwardly from centerline to edges of boot. If excess material at edges tend to form puckers, work them out smoothly and carefully with fingers.

(13) Roll tapered edges of boot with narrow steel stitcher roller.

(14) Clean blade with a clean cloth moistened with Toluol or MPK. Use care so solvent does not run under edge of deice boot.

(15) Apply one even brush coat of A-56B conductive cement around edges of boot, allowing 1/16 to 1/8 inch overlap on boot, but not extending to masking tape.

(16) Remove masking tape after applying cement to obtain a near border.

(17) Install clamp securing lead strap to propeller blade retaining clamps.

(18) Connect lead terminals and install clamp on spinner bulkhead.

NOTE: There must be no slack between terminal and clip. Ensure enough slack between clip and clamp on blade to allow propeller feathering.

(19) Install propeller spinner and secure with screws.

2. Approved Repairs – Propeller Deice

CAUTION: WRINKLED OR LOOSE PATCHES MUST BE REATTACHED. LOOSEN THE BOND AN ADDITIONAL 1/4 INCH BEYOND THE WRINKLED OR LOOSE AREA WITH MPK OR ACETONE, THEN REATTACH THE LOOSE PORTION OF THE PATCH WITH EC-1403 CEMENT IN ACCORDANCE WITH THE PRECEDING STEPS.

ALLOW 12 HOURS FOR THE CEMENT TO CURE BEFORE STARTING THE ENGINE, AND 24 HOURS FOR THE CEMENT TO CURE BEFORE OPERATING THE PROPELLER DEICE SYSTEM.

A. Minor damage to the deice boots may be repaired with the rubber patch material provided in the manufacturer's Field Repair Kit No. 77-802. The following information describes the proper procedure for repair of a damaged propeller deice boot.

(1) Clean the rubber area being patched with MPK or acetone to remove all grease and dirt.

(2) Cut a patch large enough to overlap the damaged area 1/4 inch on all sides. Since this patch will be exposed to the airstream, the edges should be cut clean (without fringes) and beveled.

NOTE: If any of the heating element wires are exposed (but not broken) in the damaged area, cut a second patch large enough to extend 1/4 inch beyond all sides of the first patch.

(3) Apply an even coat of EC-1403 cement to the damaged area of the boot and to the patch. Allow the coated surfaces to air dry for approximately one hour.

(4) Apply a second brush coat of EC-1403 cement to the damaged area of the boot and the patch. Allow cement to become tacky.

(5) Stick either the center, or one edge of the patch lightly in place over the damaged area, then work the remainder of the patch down carefully to avoid trapping air between the boot and the patch.

CAUTION: NEVER USE A METAL OR WOODEN ROLLER TO SEAT THE PATCH. DAMAGE TO THE HEATING ELEMENTS IN THE BOOTS MAY RESULT.

(6) After allowing one hour drying time, rub the edges and center of the patch to assure proper sealing.

NOTE: If this patch covers a heating element, the second patch should be installed at this time.

BRUSH BLOCK – MAINTENANCE PRACTICES

1. Removal/Installation – Brush Block

A. Removal – Brush Block

(1) Remove forward engine cowling in accordance with instructions in Chapter 71.

(2) Disconnect electrical connector to brush block.

(3) Remove bolts, washers, and nuts attaching brush block to slip ring assembly.

(4) Remove brush block.

B. Installation – Brush Block

(1) Position brush block on slip ring and secure with bolts, washers, and nuts.

(2) Connect electrical connector.

(3) Install forward engine cowling in accordance with instructions in Chapter 71.

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MAINTENANCE MANUAL

CHAPTER 31 – INDICATING/RECORDING

LIST OF EFFECTIVE PAGES

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31-EFF PGS	-	2	NOV 01/03	31-10-00	MP	202	NOV 01/03
31-CONTENTS	-	1	NOV 01/03	31-10-10	DO	1	NOV 01/03
31-CONTENTS	-	2	NOV 01/03	31-10-10	DO	2	NOV 01/03
31-00-00	DO	1	NOV 01/03	31-10-20	DO	1	NOV 01/03
31-10-00	DO	1	NOV 01/03	31-10-30	DO	1	NOV 01/03
31-10-00	DO	2	NOV 01/03	31-50-00	DO	1	NOV 01/03
31-10-00	DO	3	NOV 01/03	31-50-00	MP	201	NOV 01/03

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MAINTENANCE MANUAL

CHAPTER 31 – INDICATING/RECORDING

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CONSOLE PANELS DESCRIPTION AND OPERATION	31-10-10	1	T AT TC	201-999 001-999 201-999
SWITCH PANELS DESCRIPTION AND OPERATION	31-10-20	1	T AT TC	201-999 001-999 201-999
PEDESTAL DESCRIPTION AND OPERATION	31-10-30	1	T AT TC	201-999 001-999 201-999
ANNUNCIATOR PANEL DESCRIPTION AND OPERATION	31-50-00	1	T AT TC	201-999 001-999 201-999
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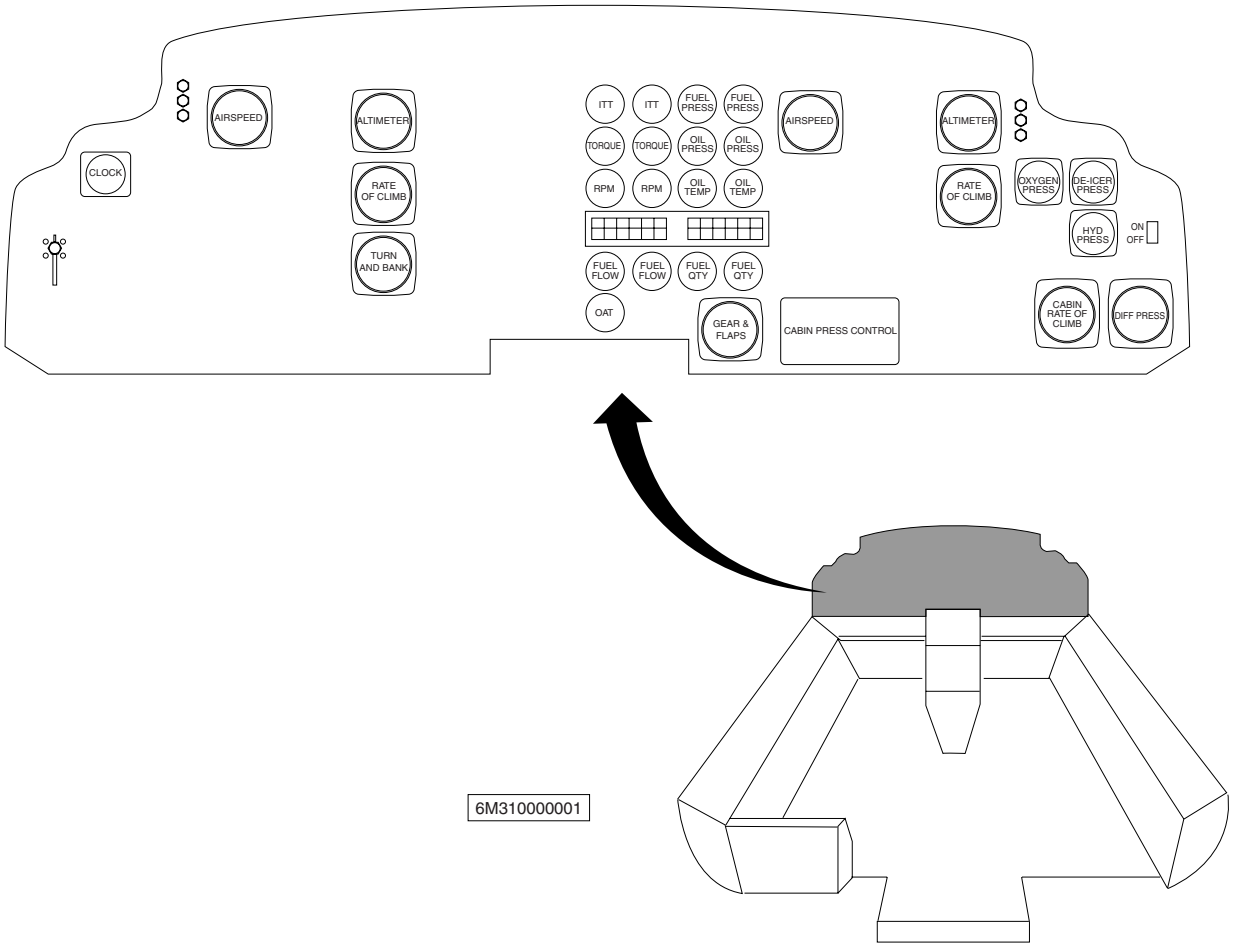
INDICATING/RECORDING – DESCRIPTION AND OPERATION

1. General

A. The basic indicating/recording system utilized on SA226 series aircraft provides the pilot and copilot with a visual indication of conditions in unrelated systems. The system consists of an instrument panel, left and right console panels, left and right switch panels and an annunciator panel that provides a lighted indication of systems operation and/or failure.

INSTRUMENT PANEL – DESCRIPTION AND OPERATION

A. The instrument panel is divided into three sections; pilot's instruments, engine instruments, and copilot's instruments. The panel is held in place by four bolts across the bottom of the panel and two bolts through shock mounts at the top of the panel. All instruments are front mounted on the panel, and the electrical leads to each instrument are of sufficient length to allow the instrument to be pulled out far enough from the panel to facilitate removal of the electrical connector.



Instrument Panel
Figure 1

B. The following is a listing of instruments and range markings, installed on the instrument panel, (see Figure 1).

(1) Interstage Turbine Temperature Indicator:

(a) Green Arc: 0° to 923°C.

(b) Red Radial: 923°C (max. takeoff and max, continuous power).

(c) Dashed Red Radial: 944°C (WET Power SA226-TC).

(d) Red Dot: 1149°C (max. allowable during starting up to 50% RPM).

(2) Engine Torquemeter:

(a) Green Arc: 0 to 2,206 foot-pounds.

(b) Yellow Arc: 2,206 to 2,468 foot-pounds (WET Power SA226-TC).

(c) Red Radial: 2,363 foot-pounds (max. allowable due to ram rise).

(d) Dashed Red Radial: 2,468 foot-pounds (WET Power – five min. limit SA226-TC).

(3) Percent RPM Indicator:

(a) Green Arc: 96% to 100%.

(b) Red Radial: 101%.

(4) Fuel Pressure:

(a) Green Arc: 20 to 80 psi.

(b) Yellow Arc: 15 to 20 psi.

(c) Red Radial: 15 psi minimum at 65% RPM.

(d) Red Radial: 80 psi maximum.

(5) Oil Pressure Indicator:

(a) Green Arc: 70 to 120 psi.

(b) Yellow Arc: 50 to 70 psi (at 65% RPM).

(c) Red Radial: 40 psi minimum at 65% RPM.

(d) Red Radial: 120 psi.

(6) Oil Temperature:

(a) Green Arc: 55° to 110°C.

(b) Yellow Arc: minus 40° to plus 55°C and 110° to 127°C.

(c) Red Radial: minus 40°C minimum.

(d) Red Radial: 127°C maximum.

(7) Fuel Quantity Indicator – 0 to 2,500 pounds.

(8) Hydraulic Pressure gauge:

(a) Green Arc: 1,700 to 2,100 psi.

(b) Yellow Arc: 1,400 to 1,700 psi.

(c) Red Radial: 2,100 psi.

(9) Outside Air Temperature – minus 50° to plus 50°C.

(10) Deicing Pressure Gauge:

(a) Green Arc: 12 to 19 psi.

(b) Yellow Arc: 10 to 12 psi and 19 to 21 psi.

(c) Red Radial: 10 psi minimum.

(d) Red Radial: 21 psi maximum.

(11) Oxygen Pressure Gauge:

(a) Green Arc: 1,600 to 1.850 psi.

(b) White Arc: 0 to 300 psi.

(c) Red Arc: 1,850 to 2,000 psi.

(12) Differential Pressure Gauge:

(a) Green Arc: 0 to 7.0 psi.

(b) Red Radial: 7 psi.

(13) Vertical Speed Indicator – 0 to plus 6,000 foot/minute.

(14) Cabin Rate of Climb – 0 to plus 2,000 foot/minute.

INSTRUMENT PANEL – MAINTENANCE PRACTICES

1. Removal/Installation – Instrument Panel

A. Removal – Instrument Panel

(1) Ensure electrical power is OFF.

(2) Remove glareshield mounting screws on left and right side.

(3) Lift glareshield up until mount clips are clear of molding.

(4) Disconnect glareshield electrical connector wires.

(5) Remove glareshield.

(6) Remove screws at corners of instrument (typical).

(7) Pull instrument from panel just far enough to allow removal of electrical connector and/or air connections.

CAUTION: IMMEDIATELY INSTALL CAPS OR PLUGS ON ALL OPEN LINES TO PREVENT CONTAMINATION IN THE SYSTEM.

(8) Disconnect pitot and static hoses from pilot and copilot indicators.

(9) Disconnect electrical connectors from pilot and copilot indicators.

(10) Remove indicators.

(11) Disconnect electrical connectors from panel lights located behind and at top of instrument panel.

(12) Remove four vertically installed bolts across bottom and behind front of instrument panel.

(13) Remove two screws at top of panel.

(14) Remove instrument panel.

B. Installation – Instrument Panel

(1) Ensure electrical power is OFF.

(2) Position instrument panel in cockpit.

(3) Install two screws at top of panel.

(4) Install four vertically mounted bolts at bottom of panel.

CAUTION: ENSURE THREAD COMPOUND DOES NOT GET INTO BORE OF FITTING, FITTING END, OR ON CONE SURFACE. COMPOUND SHOULD NOT BE ALLOWED ON INSIDE FLARE SURFACE, INTERIOR OF TUBE, OR INSIDE ANY SYSTEM(S) COMPONENTS. ALWAYS USE THREAD COMPOUND SPARINGLY.

(5) Apply a light coat of anti-seize compound (MIL-T-5544) to male fittings on static system hoses.

(a) To prevent galling and seizure of threads, MIL-T-5544 thread compound should be applied to threads of fittings as follows:

1 Align tubing and press the flare against the fitting end.

2 Tighten fitting one-half to one turn (this ensures compound will not get on to first one to one and one-half threads). Then sparingly apply MIL-T-5544 thread compound to the exposed threads. Continue tightening the fitting.

(6) Remove caps from lines as plumbing is connected to instruments.

CAUTION: ENSURE ALL FITTINGS AND HOSES ARE CLEAN AND FREE OF CONTAMINATION PRIOR TO INSTALLATION.

ALIGN FITTINGS PROPERLY DURING THREAD ENGAGEMENT TO PREVENT CROSS THREADING.

(7) Connect pitot and static hoses to pilot and copilot indicators. Torque end fitting 50 to 90 inch-pounds (5.65 to 10.17 N·m).

(8) Connect electrical connectors to pilot and copilot indicators

(9) Slide instrument into panel and secure with screws.

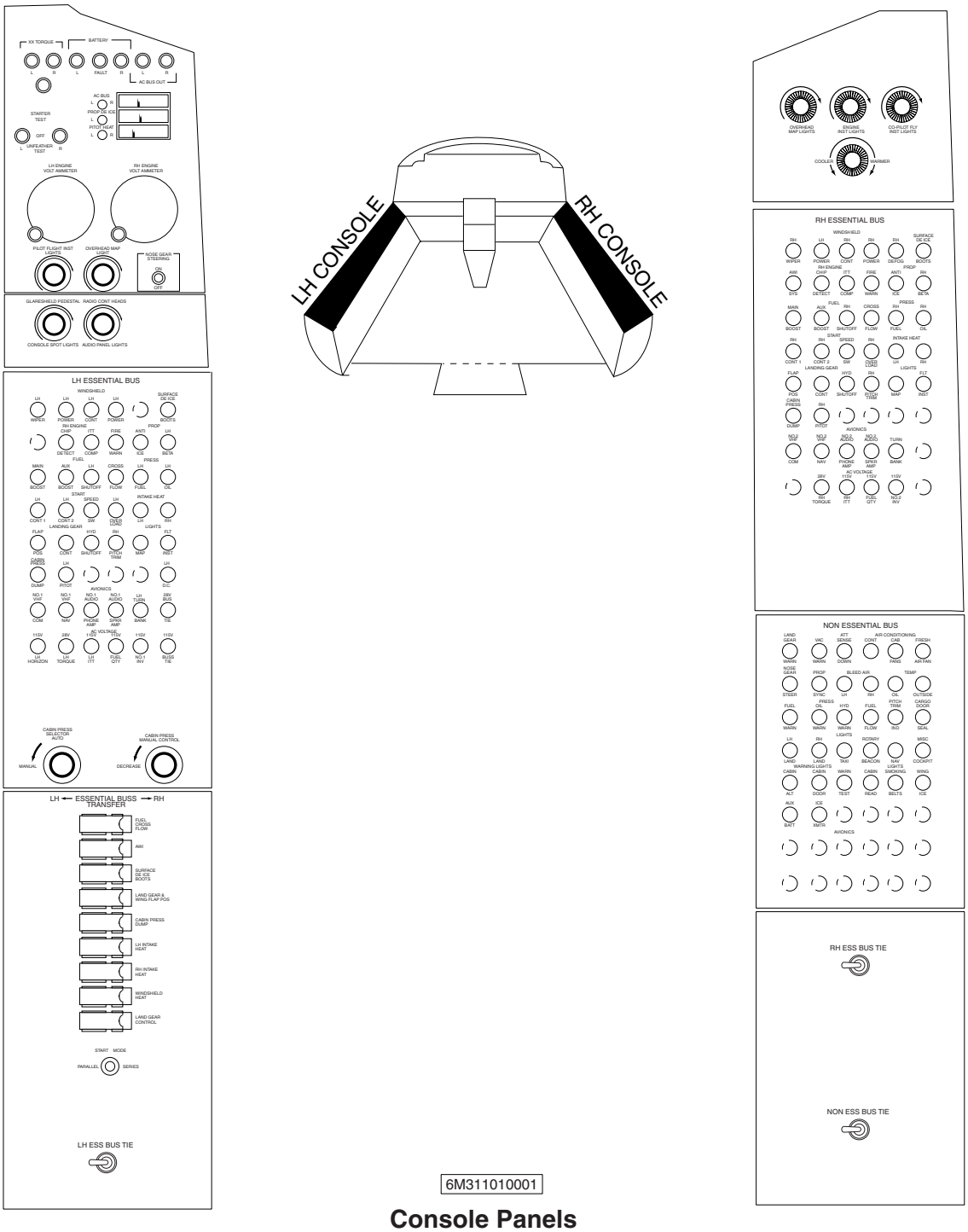
(10) Position glareshield over instrument panel and connect electrical wires

(11) Secure glareshield with screws on left and right side.

CONSOLE PANELS – DESCRIPTION AND OPERATION

1. General

A. Console panels are installed on the left and right side of the cockpit between F.S. 88 and F.S. 126. Console panels provide a mounting place for circuit breakers, switches and indicators



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Console Panels
Figure 1

2. Left Console Panel

A. The Left Console Panel consists of; left and right engine start/unfeather test, left and right AC bus warning lights, battery bus out fault lights, pitot heat, prop deice ammeters and left and right selector switches, AC bus voltmeter and left and right bus selector switches, nose gear steering switch, left and right generator volt-ammeters, pilot flight instrument lights rheostat, overhead map light rheostat, glareshield, pedestal, console, and spot lights rheostat, left essential bus circuit breaker panel, cabin pressure manual controls, fuel crossflow, turn and bank indicator, surface deice boots, landing gear and wing flap position, cabin pressurization dump, left intake heat, right intake heat, windshield heat, landing gear control, phone and mike jacks, and left essential bus tie.

3. Right Console Panel

A. The Right Console Panel consists of; overhead map light rheostat, engine instrument lights rheostat, copilot's flight instrument lights rheostat, temperature selector rheostat, right essential bus circuit breaker panel, nonessential bus circuit breaker panel, right essential bus tie, and nonessential bus tie.

SWITCH PANELS – DESCRIPTION AND OPERATION

1. General

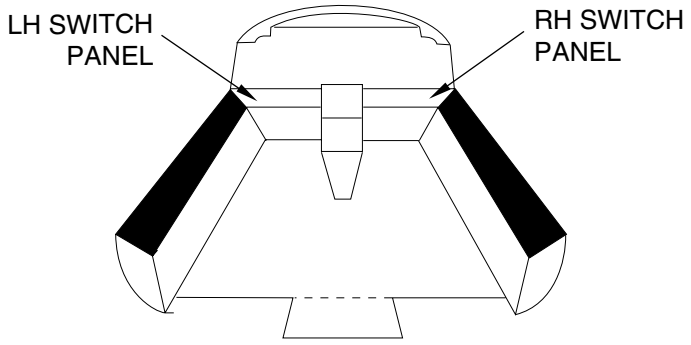
A. Left and right switch panels are installed below the instrument panel. Switch panels provide a mounting point for various control switches.

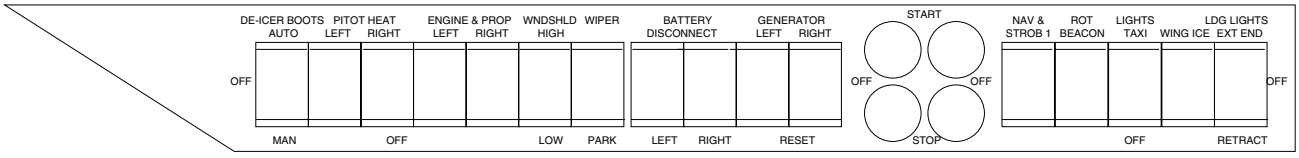
2. Left Hand (LH) Switch Panel

A. The left switch panel contains; deicer boots heat, left and right pitot heat, left and right engine and propeller anti-ice, windshield heat, windshield wiper control, left and right battery, left and right generator, left and right engine start and stop, and exterior lights.

3. Right Hand (RH) Switch Panel

A. The right switch panel contains; cabin temperature control, fresh air fan, bleed air right max. flow, bleed air left max. flow, bleed air right shut-off, bleed air left shut-off, cabin pressurization, fasten seat belts, no smoking, aisle lights, instrument inverters, and alternators.





LH SWITCH PANEL

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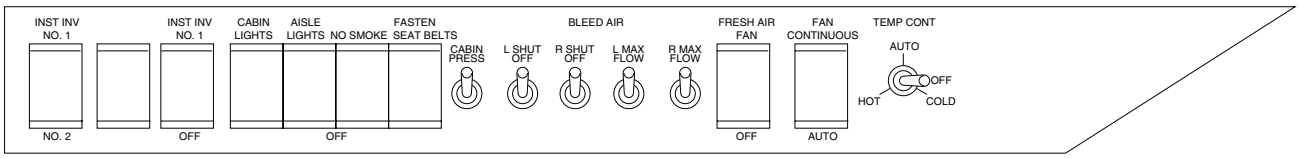


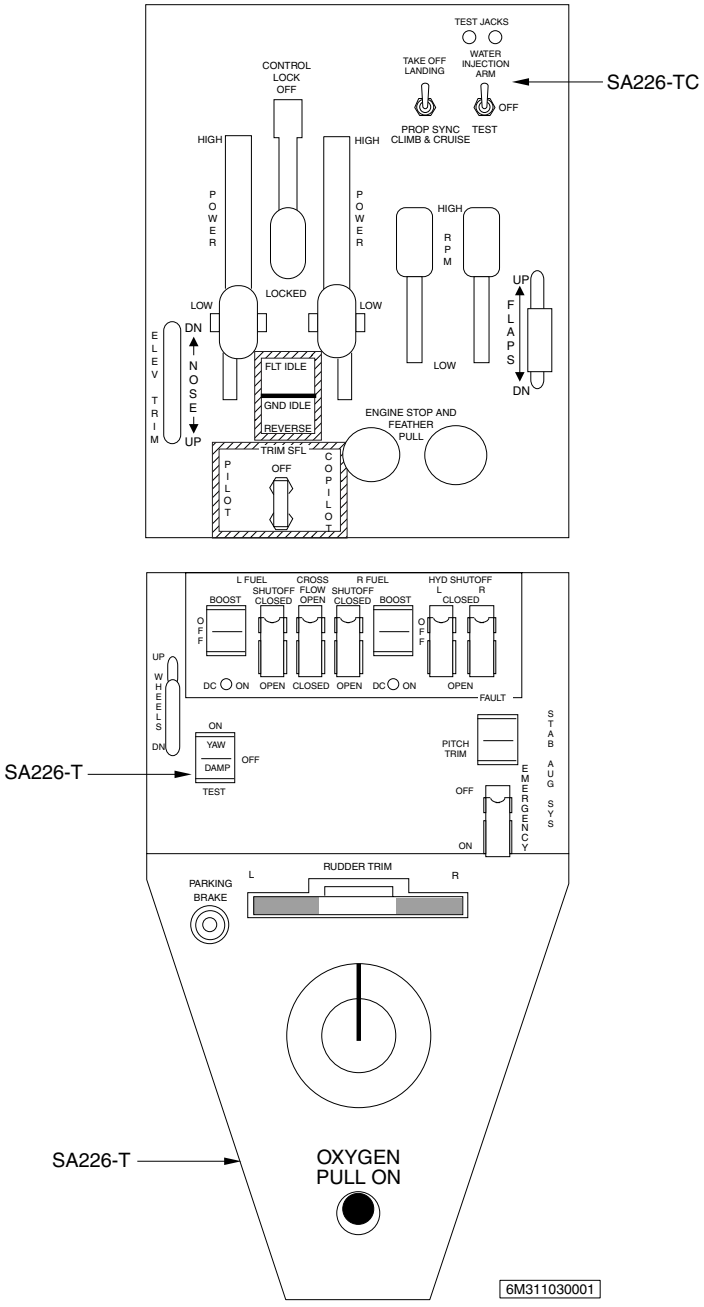
RH SWITCH PANEL

Switch Panels
Figure 1

PEDESTAL – DESCRIPTION AND OPERATION

1. Pedestal

A. The pedestal, located between the pilot and copilot's seats contains; engine power and RPM levers, engine controls lock, prop sync, flap control, manual engine stop and feather handles, left fuel boost and shut-off switches, fuel crossflow switch, right fuel boost and shut-off switches, left and right hydraulic shut-off switches, landing gear control, stick pusher emergency shut-off, master trim switch, parking brake, rudder and aileron trim, AWI master switch (SA226-TC Only), JATO fire switch, auxiliary trim switch.



Pedestal
Figure 1

ANNUNCIATOR PANEL – DESCRIPTION AND OPERATION

1. General

A. The annunciator panel, located in the upper center section of the instrument panel, is interconnected to numerous aircraft systems and is utilized to monitor systems operation.

(1) The annunciator panel contains:

(a) Red warning lights to indicate system failure.

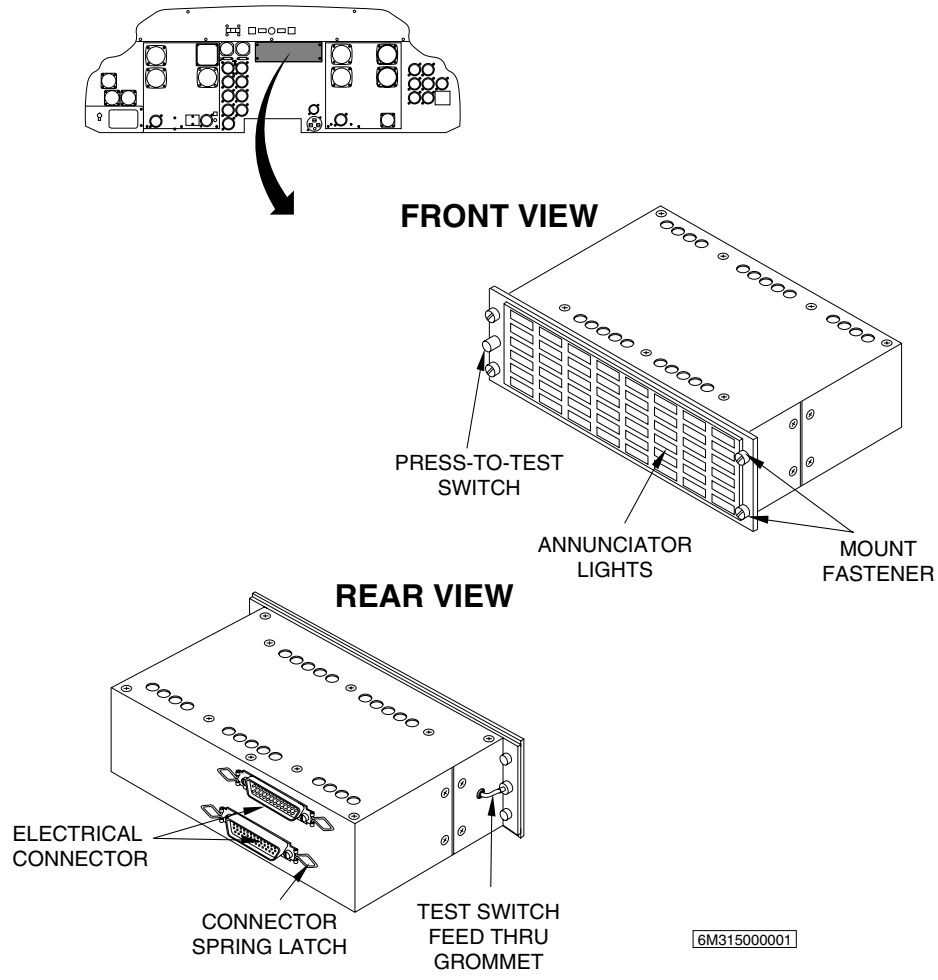
(b) Amber caution lights to indicate system failure of a less serious nature.

(c) Green lights to indicate systems on and/or operating.

(2) The annunciator panel has a press-to-test switch located on the left side of the annunciator panel.

(a) Pressing the press-to-test switch permits the crew to check continuity of annunciator panel light bulbs.

(3) The green system lights, when on, are automatically dimmed when the navigation lights are turned on. The red warning and amber caution lights cannot be dimmed.



Annunciator Panel
Figure 1

ANNUNCIATOR PANEL – MAINTENANCE PRACTICES

1. Removal/Installation – Annunciator Panel

A. Removal – Annunciator Panel (Figure 1)

(1) Ensure electrical power is OFF.

(2) Disengage Airloc mounting fasteners.

(3) Withdraw annunciator panel from the instrument panel.

(4) Release spring latches from each electrical connector, and disconnect electrical connectors from annunciator panel.

B. Installation – Annunciator Panel (Figure 1)

(1) Ensure electrical power is OFF.

(2) Connect electrical connectors and secure with spring latches on each connector.

(3) Position annunciator panel in instrument panel.

(4) Secure annunciator panel to instrument panel with Airloc mounting fasteners.

(5) Apply electrical power and push the press-to-test switch to ensure all light bulbs illuminate.

(6) Turn electrical power OFF.

C. Removal – Light Bulb – Annunciator Panel (Figure 1)

(1) Ensure electrical power is OFF.

(2) Depress left side of the light bulb module in annunciator panel.

(3) The right side of light bulb module will spring out, providing access to bulbs.

(4) Replace defective bulb(s) with new light bulb(s).

D. Installation – Light Bulb – Annunciator Panel (Figure 1)

(1) Reseat light bulb module in annunciator panel.

(2) Apply electrical power and push the press-to-test switch to ensure all light bulbs illuminate.

(3) Turn electrical power OFF.

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32-52-35	MP	216	SEP 15/09
32-52-35	MP	217	SEP 15/09
32-52-35	MP	218	SEP 15/09
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32-52-40	MP	201	OCT 01/06
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32-60-00	DO	1	OCT 01/06
32-60-00	DO	2	OCT 01/06
32-60-00	FI	101	OCT 01/06
32-60-00	FI	102	OCT 01/06
32-60-00	FI	103	OCT 01/06
32-60-00	FI	104	OCT 01/06
32-60-00	FI	105	OCT 01/06
32-60-00	FI	106	OCT 01/06
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32-60-00	MP	201	OCT 01/06
32-60-00	MP	202	OCT 01/06
32-60-10	DO	1	OCT 01/06
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32-60-10	MP	202	OCT 01/06
32-60-10	MP	203	OCT 01/06
32-60-10	MP	204	OCT 01/06
32-60-20	DO	1	OCT 01/06
32-60-20	DO	2	OCT 01/06
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32-60-20	MP	202	OCT 01/06
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32-60-30	MP	202	OCT 01/06	32-SERVC	SI	12	OCT 01/06
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LANDING GEAR – SERVICE INFORMATION

1. Introduction

A. The following information is supplied as a supplement to this chapter. Subject matter is included which is not applicable to an ATA 100 format, but is still of interest to Merlin and Metro owner/operators. The information in this section is not part of the basic manual, and may be removed or retained at the discretion of the owner/operator.

B. Revisions and additions to this section will be made as additional information is available. Input from individual owner/operators is encouraged, and may be submitted to:

M7 Aerospace, LP
Post Office Box 790490
San Antonio, Texas 78279-0490
ATTN: Technical Publications Department

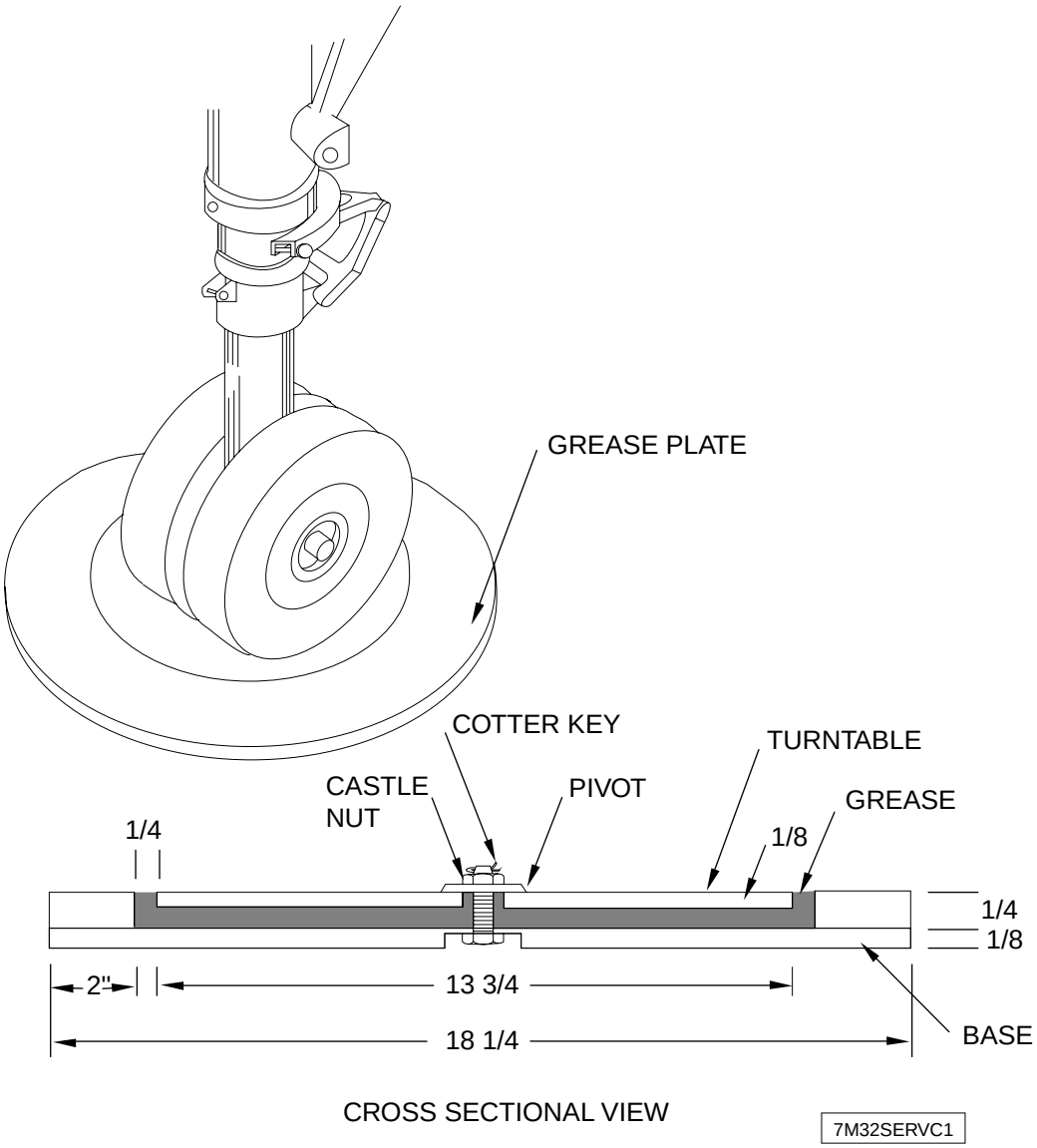
2. 32-01 – Landing Gear Test Equipment

A. Included on the following pages are several test sets, test fixtures, and test leads that are useful in performing troubleshooting and maintenance functions on the landing gear system. Most of this equipment is for use in testing or adjusting the nose gear steering system. None of this equipment is mandatory, and in most cases can be replaced by a simple voltmeter; however, substantial savings in manhours can be realized by having the proper test equipment.

B. Materials for this equipment were chosen primarily on the basis of their availability. Specific brand names were, wherever possible, deliberately deleted. Deviations from the listed materials is completely acceptable provided the end result performs the intended function.

(1) 32-02 – Grease Plate (TF32-01).

(a) A grease plate is useful during nose gear steering system test. Without the plate, depending on the loading and configuration of the aircraft the nose steering actuator may not be capable of turning nose wheels from stop to stop. The plate shown in Figure 1 is constructed of .125 aluminum plate. Two pieces of aluminum, .063 or thicker, with a liberal amount of grease placed in between them, will also suffice.



Grease Plate
Test Fixture 32-01
Figure 1

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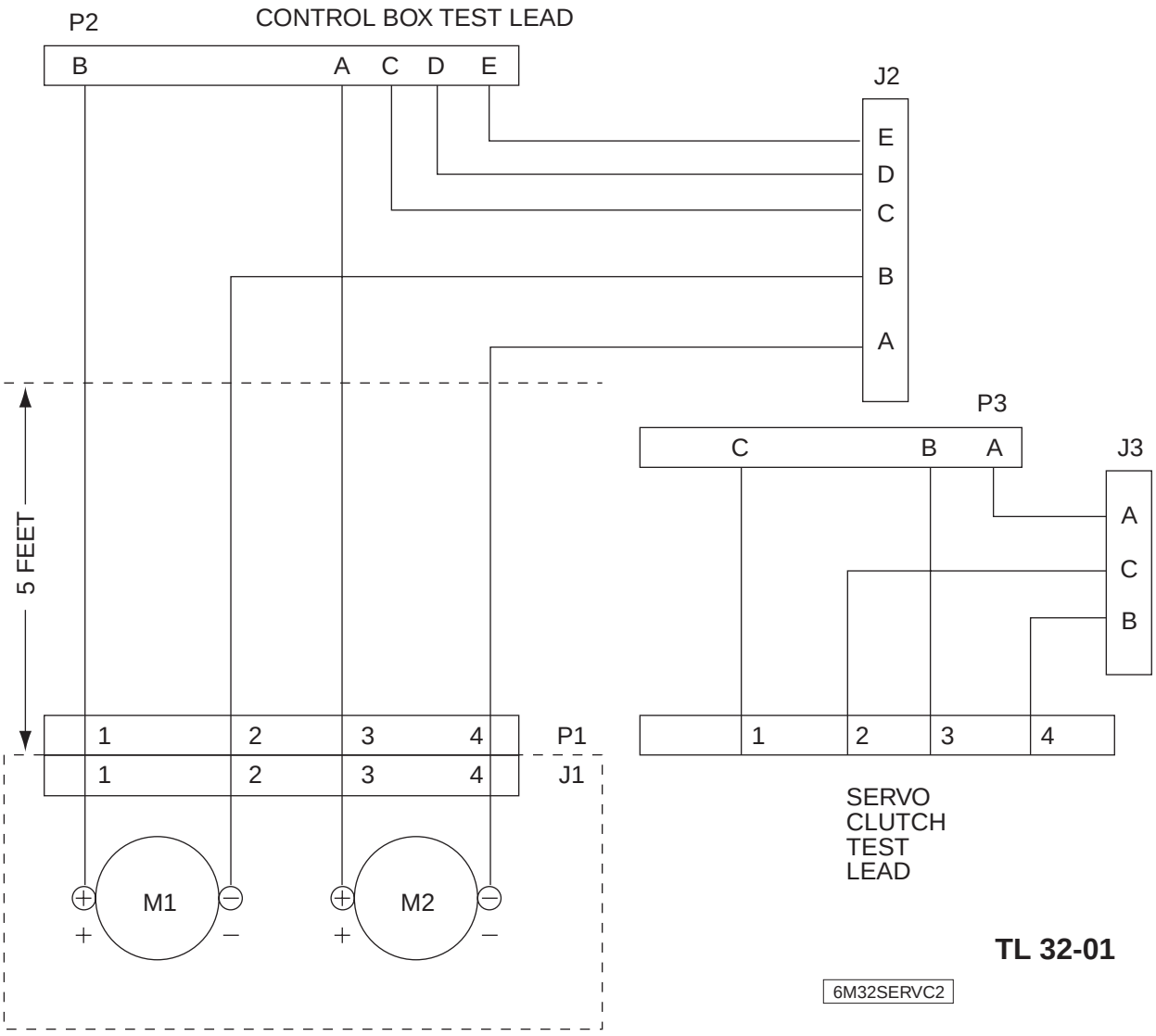
C. 32-03 – Steering Tester (TS 32-01)

(1) The steering tester, as shown in Figure 2, may be used to set the quiescent current for T201 through 248 or to check the switch assembly and servo current for AT001-024 and TC201-211EEE. With TL 32-01 the tester may be used at the servo clutch connector to check the system.

(2) Construction should include a small metal box for the meters and connector. Optionally, some operators have constructed TL 32-01 of sufficient length to connect to the nose steering servo and still have the tester located in the cockpit.

MATERIALS

All wire 16 AWG
M1 and M2 – 0-200 milliammeter matched.
P1 & J1 – optional for use with different test leads.
P2 – MS3106A-145-5S
J2 – MS3100A-14-5P
P3 – MS3106A-1056-3P
J3 – MS3106A-125L-3S



Steering Tester
Test Set 32-01
Figure 2

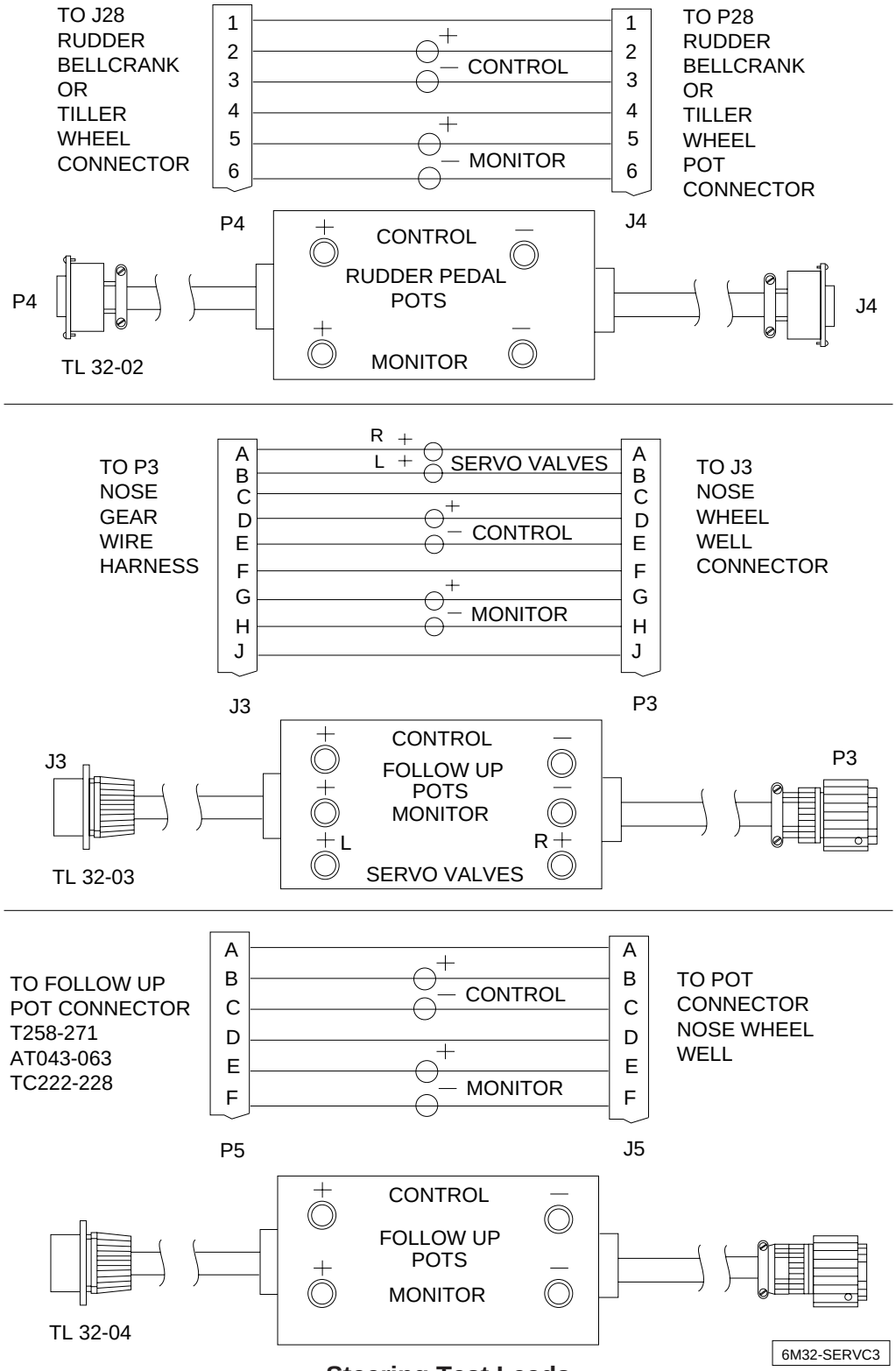
D. 32-04 – Steering Test Leads (TL 32-02, 32-03, 32-04)

(1) Test leads 32-02, 32-03, and 32-04 are primarily breakout devices that allow easy voltage measurements with the steering system fully installed and operable. They are particularly useful when trying to troubleshoot an intermitten problem. On aircraft T249-257, AT024-042, and TC211EEEE-221, a voltmeter must be connected directly to the follow-up pot wipers and ground.

(2) They are all constructed using a small metal box, banana type or pin type test jacks, and the appropriate connectors. Using a voltmeter connected to the appropriate test-jack, the following reading should be indicated.

	Centered	Left	Right
TL 32-02			
Rudder pedal pots			
Control	12V	5V	19V
Monitor	12V	5V	19V
Tiller wheel pots			
Control	12V	0V	24V
Monitor	12V	0V	24V
TL 32-03 – 32-04			
Follow-up pots			
Control	12V	16V	8V
Monitor	12V	16V	8V
Servo valves clutches			
L	1V	20V	1V
R	1V	1V	20V

Materials
P3 – PT06E-14-12P or MS3120E-14-12P
J3 – SP00SE-14-12S or MS3120E-14-12S
P4 – DE9P, DE24697 SHELL
J4 – DE9S, DE24697 SHELL
P5 – MS3120E10-6S
J5 – MS3120E10-6S



Steering Test Leads
Figure 3

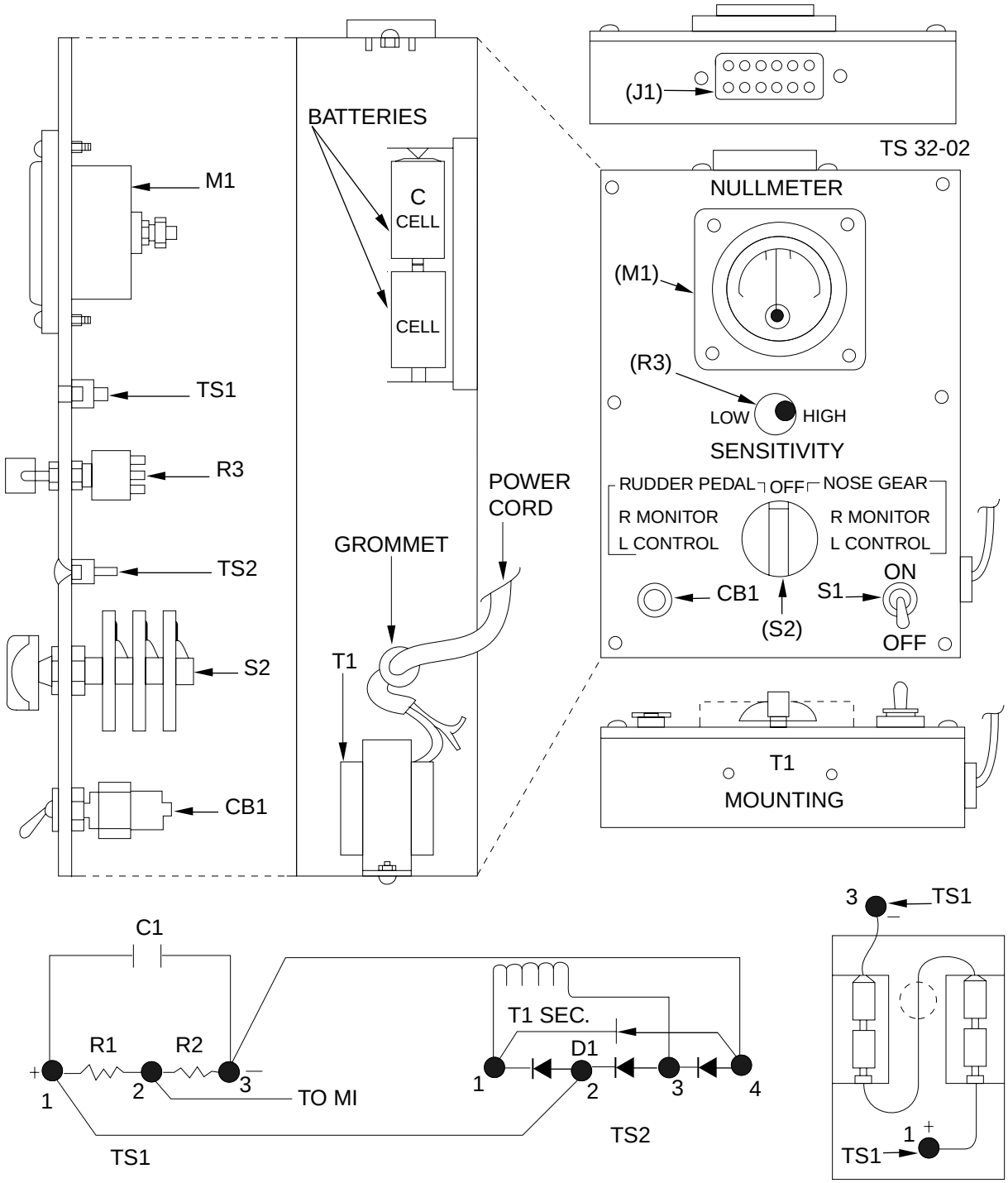
E. 32-05 – Nullmeter

(1) Plans to build a nullmeter tester are shown in Figures 4, 5, and 6. The tester is built into a metal box 10 X 6 X 2. The tester uses a rotary switch to select the pot to be nulled. A power supply can be used to replace the batteries (see Figure 5). Several leads can be made to connect the tester at the amplifier connector or directly at the pots to be nulled (see Figure 6).

(2) Materials

BAT – six volts, four C cells or equivalent.
R1 and R2 – 1K ohm .1% matched.
R3 – 100K ohm potentiometer.
M1 – zero center plus and minus 50 microamps.
S2 – five position, three stack, rotary switch.
J1 – DB15S or any 12 or more pin connectors.
P1 – DB15P or mating connector to J1.

All wire 22 AWG.
Power Supply
CB1 – one amp circuit breaker or fuse.
S1 – SPST – on-off switch.
T1 – transformer 115 volt primary 6.3 volt, one amp secondary type F-1052.
C1 – 150 MFD, 15 VDC
D1 – VS14B – bridge or four one amp rated diodes.

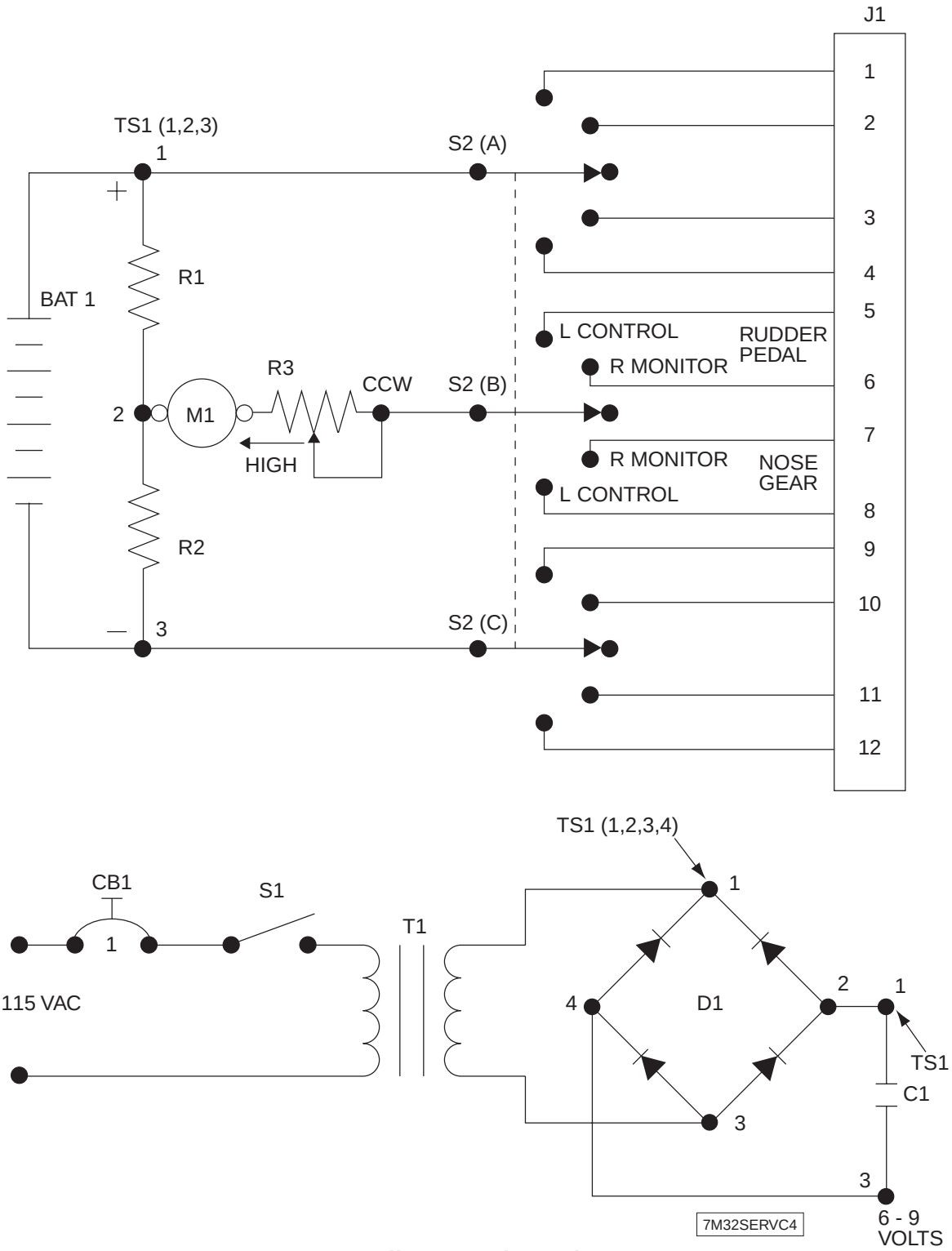


Remove Batteries for Power Supply Option

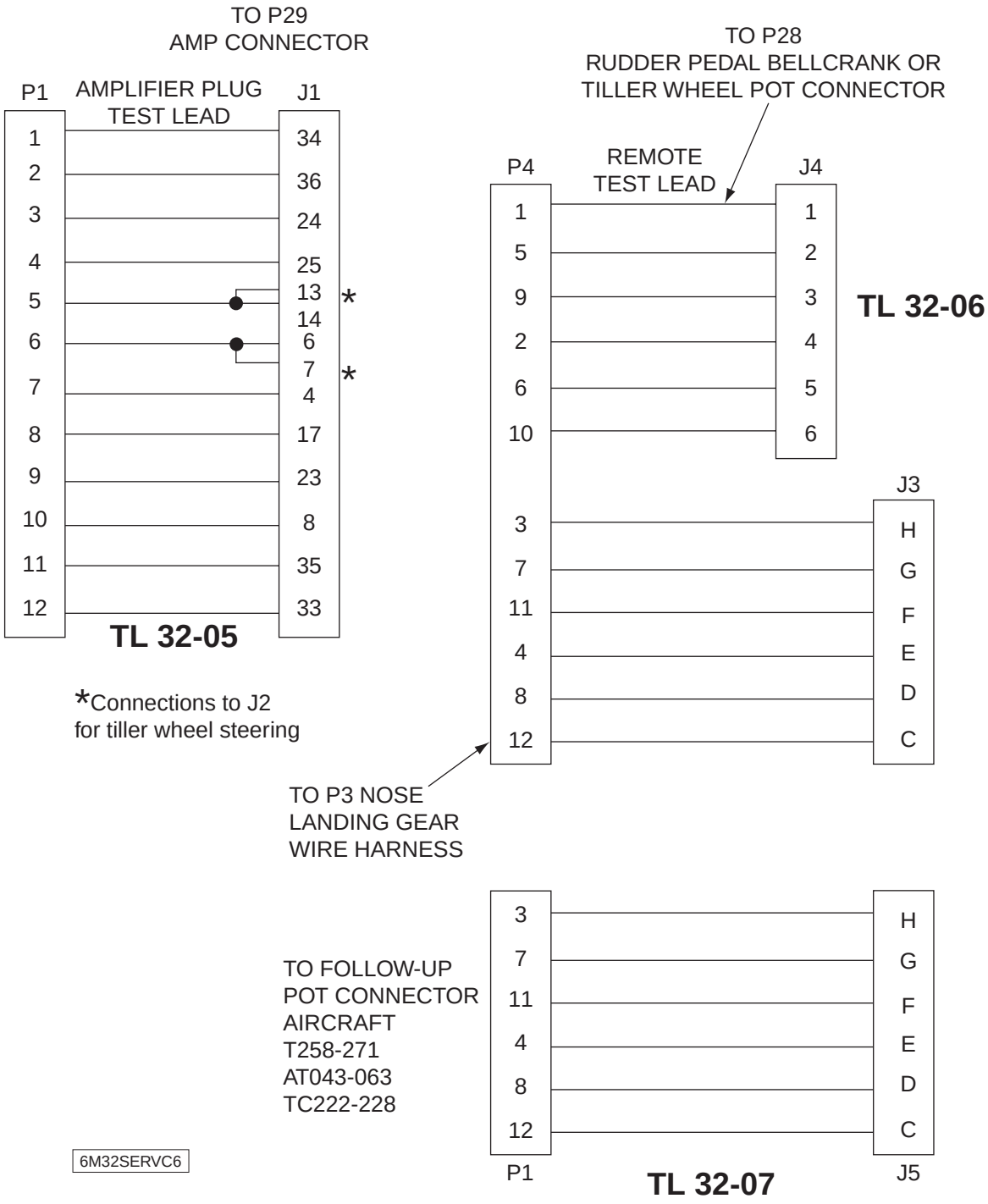
4C cells Batteries mounted in houlders.

7M32SERVC3

Nullmeter Test Set 32-02
Figure 4



Nullmeter Schematic
Test Set 32-02
Figure 5



Nullmeter Test Leads
Figure 6

F. Test Leads (TL 32-05, 32-06, 32-07)

(1) Test leads 32-05, 32-06, and 32-07 are used to connect the nullmeter into various parts of the nose gear steering system. Due to the type connector used in the steering system, the use of a specific test harness not only saves time, but also prevents damage to the pins of the connectors. TL 32-05 must be used to check follow-up pots on aircraft T249-257, AT024-042, and TC211EEEE-227.

(2) Test Lead lengths

TL 32-05 Approximately five feet between P1 and P2.
TL 32-06 Approximately two feet between P1 and J4; six feet between P1 and J3.
TL 32-07 Approximately six feet between P1 and J5.

All wire 22 AWG
P1 – BD15P or mating connector to J1 of nullmeter.
J2 – DC37P, DC2466D shell
J4 – DE9S, DE24697 shell
J3 – SP00SE-14-12S, MS3120E-14-12S
J5 – MS3120E10-6S

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3. 32-06 – Steering Tester (TS 32-03)

A. This tester and instructions are intended for use with the variable authority steering system with hydraulically powered actuator. The tester uses two voltmeters, four indicator lights, and several test jacks to monitor the various inputs and outputs to the amplifier and fault protection box. A test lead is also provided to measure the centering drive voltage at the actuator. The tester is intended to be connected between the amplifier and fault protection box and its electrical connector.

B. The tester can also be used to check the aircraft wiring with an ohmmeter by removing the test lead from the amplifier. The light and test jacks are numbered to correspond to the amplifier connector pins which they monitor. The following is a list of pin numbers and the function to which they connect:

(1) Fault control – provides a ground for the coil of the power control relay. Ground is lost during a fault.

(2) Annunciator panel light – provides ground for NOSE STEER light.

(3) First stage amplifier output – monitor pots.

(4) Follow-up monitor pot wiper.

(5) Not connected.

(6) Command monitor pot wiper – rudder pedal steering.

(7) Monitor pot summing junction – command monitor pot wiper – tiller wheel steering.

(8) Command monitor pot CCW – ground.

(9) Not connected.

(10) Not connected.

(11) Park light – park switch NC – ground.

(12) Not connected.

(13) Control pot summing junction – command control pot wiper – tiller wheel steering.

(14) Command control pot wiper – rudder pedal steering.

(15) Not connected.

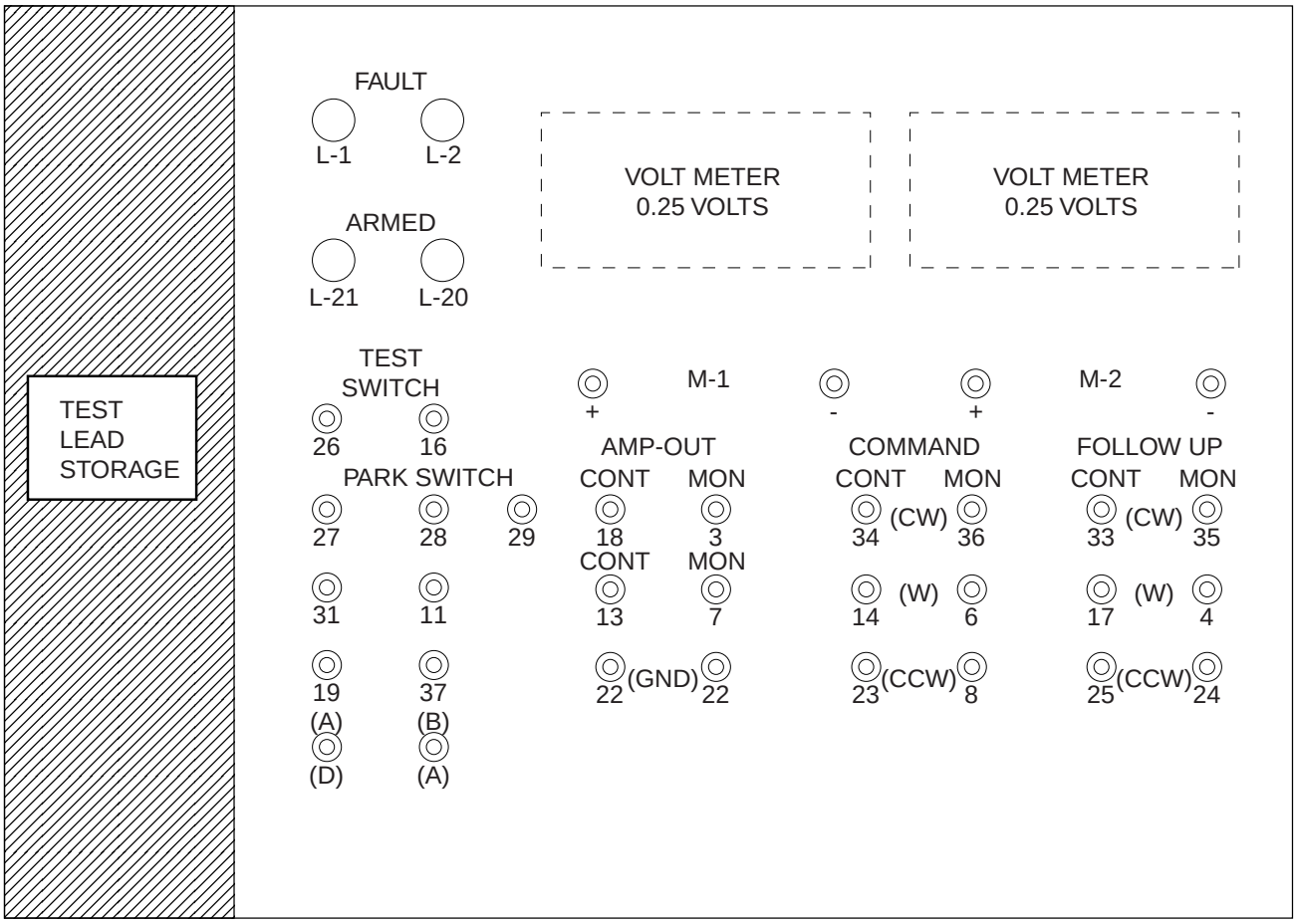
(16) Test switch input.

(17) Follow-up control pot wiper.

(18) First stage amplifier output – control pots.

(19) Left steering drive to servo valve pin C.

(20) System power for voltage regulator and max authority circuitry – 28 VDC from ARMED switch through left scissor switch.



Steering Tester
Test Set 32-03
Figure 7

(21) Turns on annunciator panel light circuit – 28 VDC directly from ARMED switch.

(22) System ground.

(23) Command control pot CCW – ground.

(24) Follow-up monitor pot CCW – ground.

(25) Follow-up control pot CCW – ground.

(26) Ground – test switch R.

(27) Park light power.

(28) Park switch power – 28 VDC.

(29) Park switch C – max authority control.

(30) Not connected.

(31) 24 VDC – test switch L.

(32) Not connected.

(33) Follow-up control pot CW – 24 VDC regulated.

(34) Command control pot CW – 24 VDC regulated.

(35) Follow-up monitor pot CW – 24 VDC regulated.

(36) Command monitor pot CW – 24 VDC regulated.

(37) Right steering drive to servo valve pin B.

4. System Checks

A. Most system checks are more easily done without hydraulic power on the actuator. A tow bar can be used to turn the nose gear. These checks are designed to be used in troubleshooting the steering system and for adjusting and checking potentiometer null (centering) voltage. Two jacks numbered 22 are provided to connect the negative jacks of M1 and M2: move the jumpers connected to the positive jacks to monitor the various other jacks.

(1) Remove amplifier connector and hook-up tester.

(2) Turn battery switches on and place nose steering switch to ARMED.

(a) Rudder pedal steering L-1, L-2, L-21, L-22 should illuminate (with right RPM lever high).

(b) Tiller wheel steering, no lights will illuminate.

(3) Rudder pedal steering – depress power lever switch, L-1 should extinguish. Power lever switch must be depressed or right RPM lever must be to LOW during remainder of checks.

(a) Tiller wheel steering – depress tiller wheel switch. L-2, L-21, L-22 should illuminate.

Tiller wheel switch must be depressed during remainder of checks.

(4) L-1 should illuminate during a fault or if power is not connected to the actuator. L-2 should blink during a fault.

(5) Command potentiometers:

(a) Rudder pedal pots – center nose gear. Center rudder pedals. Connect M-1 to J-14 and connect M-2 to J-6; both meters should indicate approximately 12 volts. Full left rudder pedal voltage should decrease to about five volts. Full right rudder pedal voltage should increase to about 19 volts. Both meters should read identical voltages for any rudder pedal position.

(b) Tiller wheel pots – center nose gear. Center tiller wheel. Connect M-1 to J-13 and connect M-2 to J-7; both meters should indicate approximately 12 volts. Full left tiller wheel voltage should decrease to about ten volts. Full right tiller wheel voltage should increase to about 14 volts. Both meters should read identical voltages for any tiller wheel position. This indication on the tiller wheel system is relatively insensitive because of the resistors in the tiller steering control box. A test lead connected at the tiller wheel pot connector (wiper to ground of each pot) should give the indication of center 12 volts, full left 0 volts and full right 24 volts.

(6) Follow-up pots:

(a) Center nose gear. Connect M-1 to J-17 and M-2 to J-4. Both meters should indicate approximately 12 volts. Do not force nose gear at the end of its movement. Turn the nose gear full left, voltage should increase to about 16 volts. Turn nose gear full right, voltage should decrease to about eight volts. Both meters should read identical voltages for any nose gear position.

(7) Troubleshooting potentiometers – jack numbers 34, 33, 36, and 35 should read 24 volts. Jack numbers 23, 25, 8, and 24 are all pot grounds. Jack numbers 14 or 13, 17, 6 or 7, and 4 are all pot wipers and should read approximately 12 volts. Breaks in the pot wiring will only be indicated by abnormal readings at the wipers. A wire carrying the excitation voltage to a pot wire to break the voltage at the wiper would be lower than normal. If a break exists in the pot ground wire the voltage at the wiper would increase. A break in the wire to the wiper would show no change of voltage when the pot was rotated. A defective winding in a pot would indicate the meter needle jumping when the pot is rotated. By disconnecting the tester from the amplifier the wires and pots can be checked individually with an ohmmeter. If a problem with pot tracking is suspected, connect M-1 positive to J-14 and negative to J-6. M-1 should read 0 volts for any position of the rudder pedals. For tiller steering, connect M-1 positive to J-13 and negative to J-7. M-1 should read 0 when tiller wheel is turned. Connect M-1 positive to J-17 and negative to J-4. M-1 should read 0 volts when nose gear is turned.

(8) Amplifier first stage outputs:

(a) Center rudder pedals or tiller wheel and nose gear. Connect M-1 to J-18 and M-2 to J-3. Both meters should read similar (but not necessarily identical) at about 11 volts. Voltage should increase for right rudder pedal or tiller wheel and left nose gear movement and decrease to the opposite condition.

(9) Test switch:

(a) Center nose gear and rudder pedals or tiller wheel. Connect M-1 to J-16. Voltage should be approximately 11 volts. Test left voltage should be 24 volts and test right 0 volts. L-1 should illuminate, and L-2 and NOSE STEER light on annunciator panel should blink. A slight movement of the rudder pedals or tiller wheel may be necessary.

(10) Max authority control voltages (rudder pedal steering only):

(a) Connect M-1 to J-27 and connect M-2 to J-29. Both meters should indicate 0 volts. Depress and hold the PARK button. M-2 should immediately indicate 25 volts, and M-1 should show a slowly increasing voltage and may require ten seconds to reach 24 volts. Releasing the PARK button reverses the operation with M-2 immediately indicating 0 volts and M-1 slowly decreasing. PARK button should slowly illuminate immediately indicating 0 volts and M-1 slowly decreasing. PARK button should slowly illuminate when depressed.

(11) Servo valve drive voltages:

(a) Center nose gear and rudder pedals or tiller wheel. Connect M-1 to J-19 and connect M-2 to J-37. Both meters should indicate approximately one volt. Left steering command will result in increasing voltage on M-1 to a maximum of approximately 20 volts. Right steering command results in a similar response on M-2. A remote lead is provided with the tester for use when adjustment of the centering voltage is required. Place the tester next to the nose gear, the meters can be easily viewed while making the adjustment. When using the remote lead, the tester is not connected at the amplifier. Remove the servo valve connector (smaller four pin connector) and connect remote lead to wire harness. Connect M-1 positive to J-19 (C) and connect negative to J (D). Connect M-2 positive to J-37 (B) and connect negative to J (A). Aircraft power must be on and the steering system armed. Power lever switch, RPM lever switch, or tiller wheel switch must be actuated. Adjust follow-up pots to obtain a reading of one volt or less on meters.

5. Steering Tester Construction (TS 32-03)

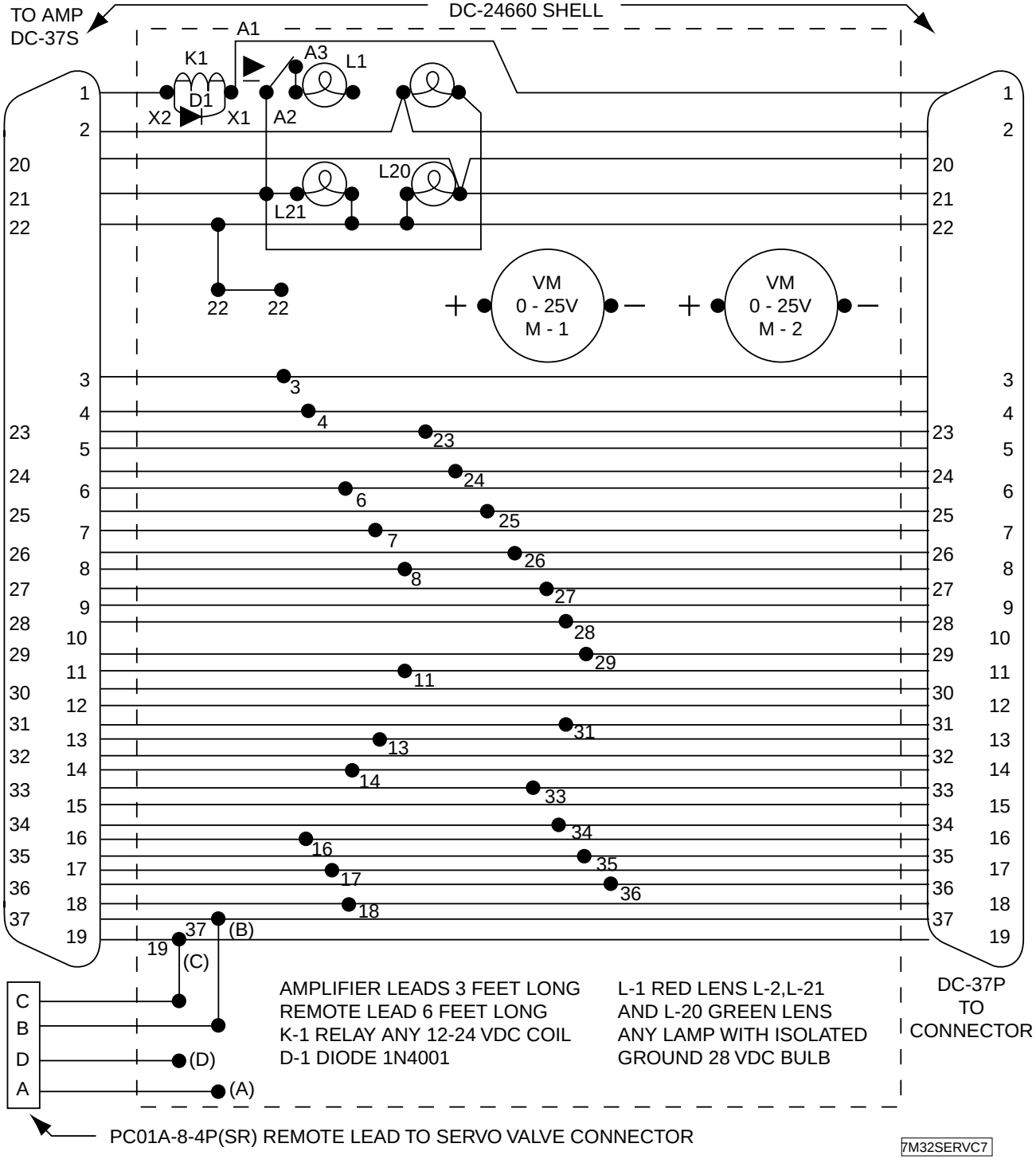
A. Build the tester into a metal box with a closeable top and a handle. A small metal tool box or fishing tackle box will serve. Test leads can be stored in the lid.

B. The tester should have two matched 0 to 25 volts DC voltmeters; however, two matched digital voltmeters with a two place decimal will provide more accuracy than the d'Arsonval type meters.

C. The four lights should be 28 VDC and have replaceable bulbs. Sockets must not be case grounded and must have two terminals. Light lens L-1 should have a red lens; and L-2, L-20, and L-21 should be green.

D. Use test jacks that are the banana type or the smaller pin type. Meter jacks should be the same type. Use four jumper leads, two red and two black, with male leads to connect the meters to the various test jacks.

E. All wires should be 22 AWG. Clamp all test leads inside the case. Connect all pins in the amplifier test plug; wires run through the tester and to the other plug.


TO AMP
DC-37S
DC-24660 SHELL
A1
K1
D1
X2
X1
A2
A3
L1
L20
L21
22
22
VM
0 - 25V
M - 1
VM
0 - 25V
M - 2
3
4
23
24
6
25
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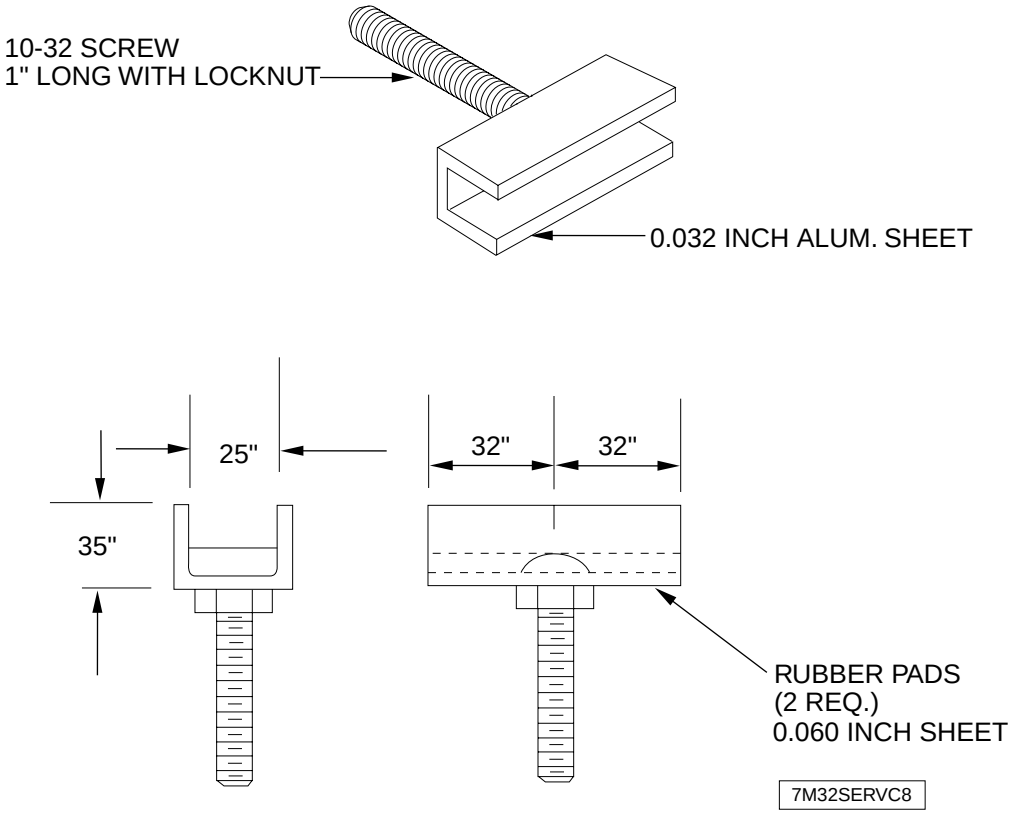
26
8
27
28
29
11
31
13
14
33
34
16
35
17
36
18
37
19
(B)
(C)
(D)
(A)
C
B
D
A
AMPLIFIER LEADS 3 FEET LONG
REMOTE LEAD 6 FEET LONG
K-1 RELAY ANY 12-24 VDC COIL
D-1 DIODE 1N4001
L-1 RED LENS L-2,L-21
AND L-20 GREEN LENS
ANY LAMP WITH ISOLATED
GROUND 28 VDC BULB
DC-37P
TO
CONNECTOR
PC01A-8-4P(SR) REMOTE LEAD TO SERVO VALVE CONNECTOR
7M32SERVC7

Figure 8

6. 32-07 – Wheel Speed Generator Drive Clip (TF 32-02)

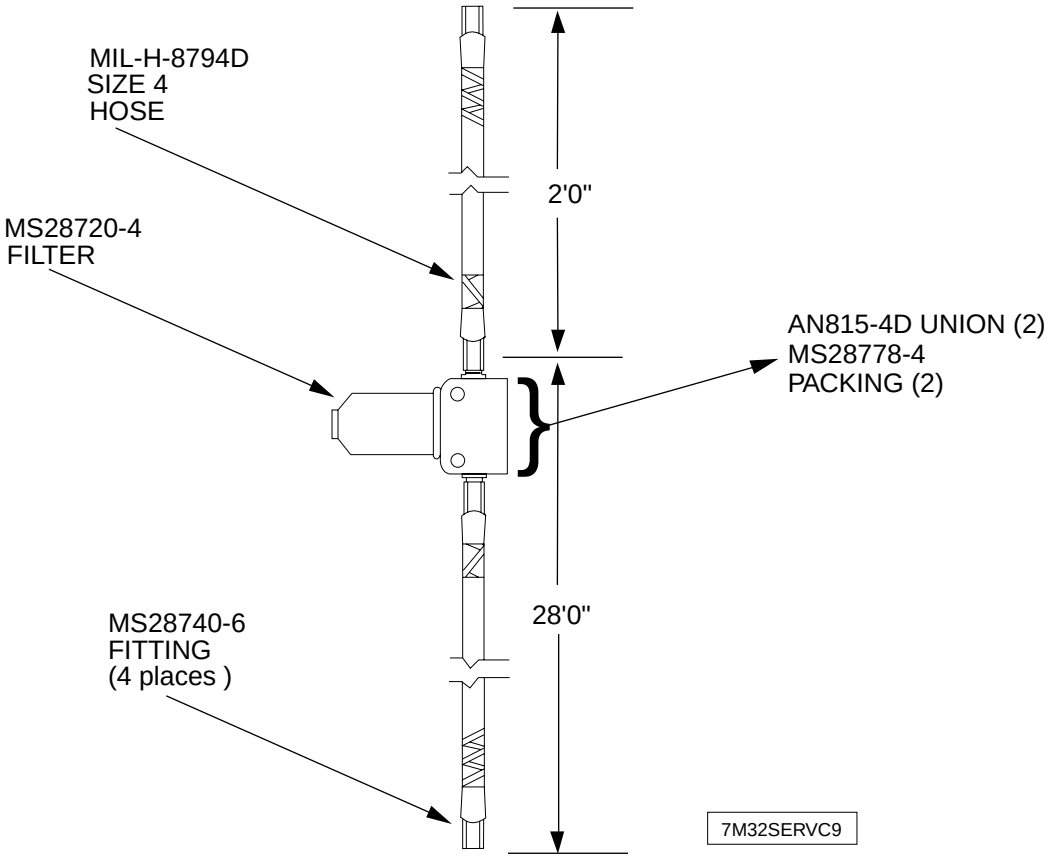
A. In order to drive the wheel speed generator, (Figure 9) clip is recommended.



Test Fixture 32-02
Figure 9

7. 32-08 – Brake Bleeding Fixture for Anti-Skid Brakes

A. The fixture shown in Figure 10 provides a means of circulating hydraulic fluid during bleeding procedures.



Test Fixture 32-03
Figure 10

8. 32-09 – Replacement of Landing gear Control Handle

A. Two types of landing gear handles are used on SA226 aircraft. A double microswitch type on aircraft T201-282, 298-999; AT01-061; TC201-246, 280-999 and a single toggle switch type on aircraft T283-207, AT062-070, TC247-279. When replacement of the single toggle switch control is required, it should be replaced by the double microswitch control (see Illustrated Parts Catalog for part numbers). Replacement will require drilling two mounting holes in the left side of the pedestal and a minor electrical wiring change. See Figure 11 for dimensions and wiring diagram.

B. Removal – Landing Gear Control Handle

(1) Ensure battery switches are off.

(2) Remove knob from landing gear control handle.

(3) Remove center pedestal edge lighted panel/overlay.

(4) Remove center pedestal cover plate, if necessary.

(5) Remove two screws attaching landing gear control handle assembly to left side of pedestal.

(6) Cut string ties or remove clamps, as necessary, to allow assembly to be brought out front of pedestal for easy access to wire terminals.

(7) Slide vinyl tubing back and remove wire terminals from switch terminals.

C. Installation – Landing Gear Control Handle

(1) Install wires on switch terminals of new assembly (see wiring schematic, Figure 11). If replacing single toggle switch control with double microswitch control:

(a) Replace #6 stud size wire terminals with #4 stud size wire terminals.

(b) Cut a length of 22 AWG (gauge) wire approximately four inches long and attach #4 stud size wire terminals.

(c) Cut four two inches of 1/4 inch vinyl tubing and slide over wire terminals.

(d) Connect wires to microswitches as shown in Figure 11, Wiring Schematic.

(2) Slide vinyl tubing over terminals and tie. Service Bulletin 24-011 detailed adding Vinyl tubing to landing gear control and flap control switches (see Service Bulletin if necessary).

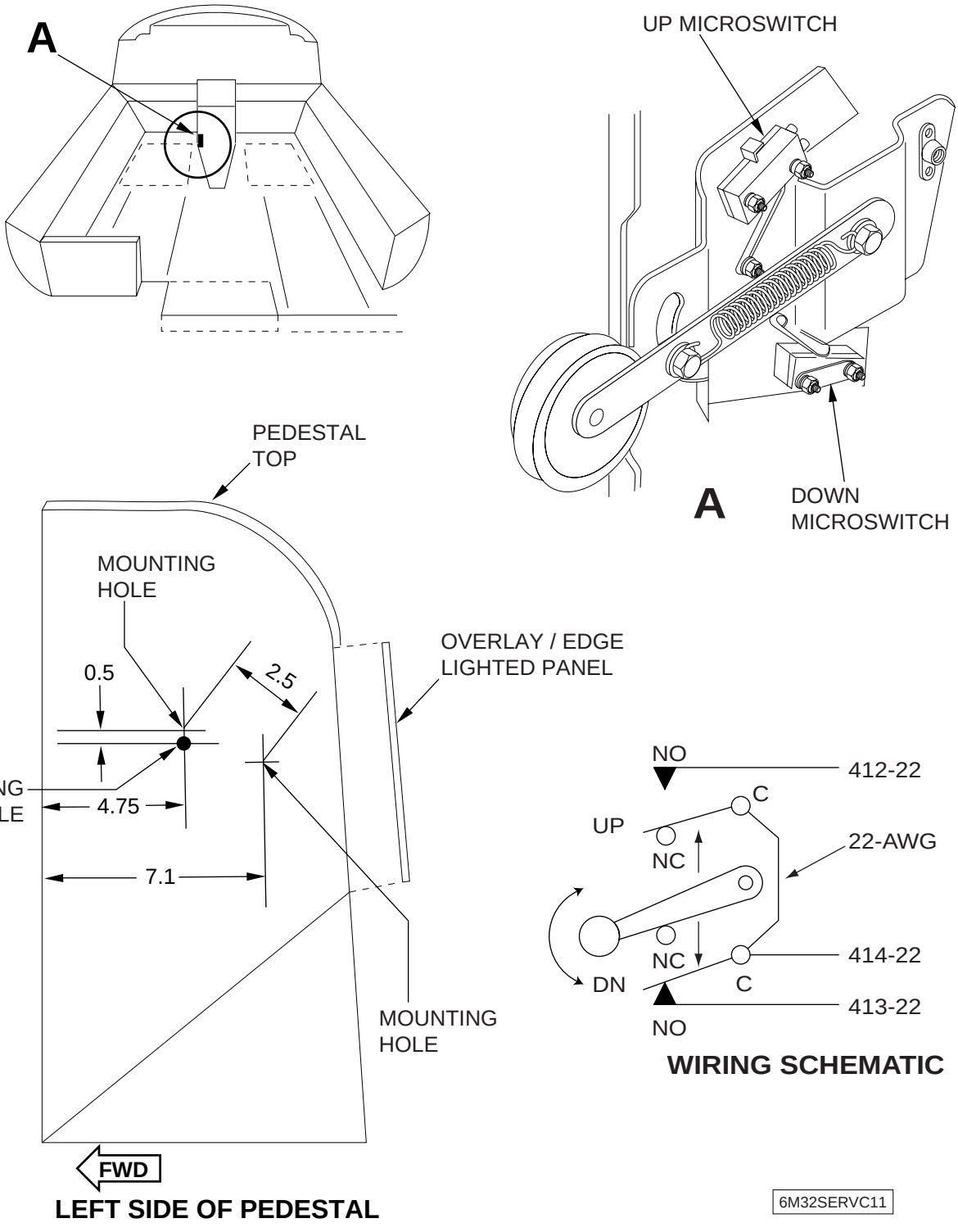
(3) Install landing gear control assembly with two screws. If replacing single toggle switch control with double micoswitch control;

(a) Drill and countersink two holes to accommodate #8 screws.

(b) Locate holes to dimensions shown in Figure 11.

(c) Install new landing gear control assembly with two screw previously used to attach old assembly to left side of pedestal.

(d) Tie and clamp wires as necessary.



Landing Gear Control Handle Figure 11

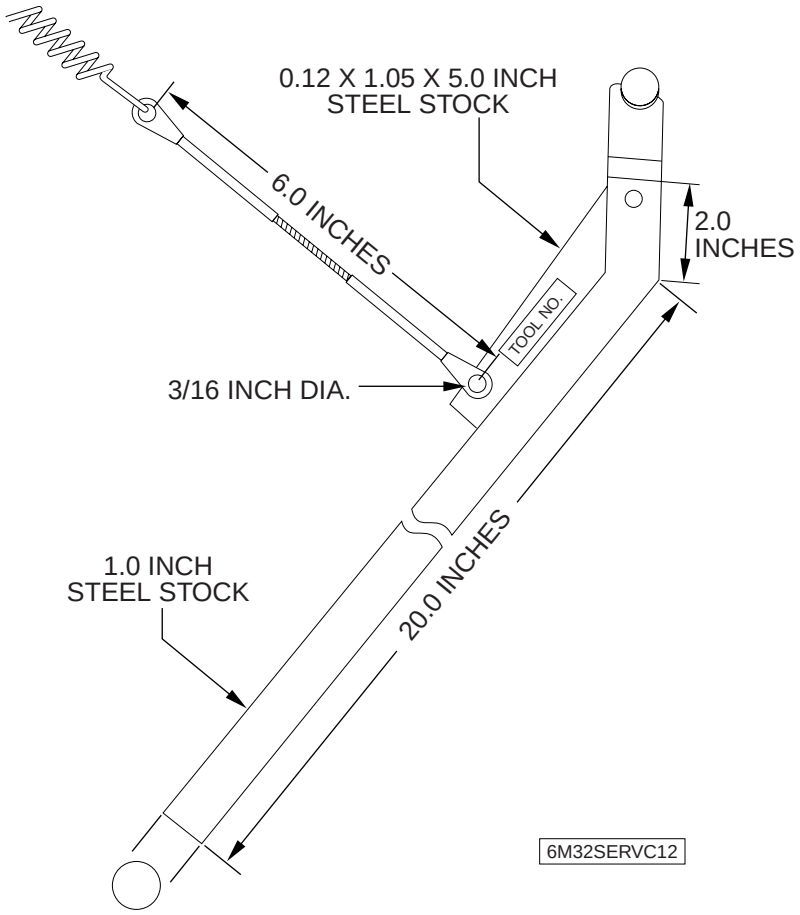
(4) Install pedestal center cover and edge lighted panel/overlay.

(5) Install knob on landing gear control handle.

(6) Perform landing gear extension and retraction check (see Section 32-30-00).

8. Spring Extender Tool

A. The tool shown is designed to help extend the bellcrank spring during removal, installation, or rigging.



Spring Extender Tool
Figure 12

LANDING GEAR – DESCRIPTION AND OPERATION

1. General

A. Arrangement of Data

(1) This chapter provides in detail the maintenance practices for the SA226 series aircraft landing gear system. Section one (32-00-00) provides a general description and operation of the landing gear and major components, operational function, strut servicing, gear lubrication, and maintenance practices.

(2) Section two and three (32-10-00 and 32-20-00) cover the main and nose landing gear respectively, including a separate description for each type gear, and the maintenance practices of disassembly, repacking, resealing, and installation of both main and nose landing gear. Instructions for the removal and installation of landing gear components, including drag brace, uplock and positioning cams, doors, and door actuators are also provided. Section four (32-30-00) provides a separate description and operation for landing gear retraction, normal extension, and emergency extension. This section also covers landing gear troubleshooting and rigging procedures for the gear, gear doors, and emergency release system.

(3) Section five (32-40-00) provides detailed maintenance practices for the main landing gear wheels, tires, and brakes. Section six (32-50-00) provides a description and operation of the steering system and the electrical and hydraulic variable authority system, and the optional tiller wheel system. It covers component maintenance practices for the steering control box, both carbon pile and switch control, the amplifier and fault protection box, the control system including steering wheel and tiller wheel, the rudder pedal control system, the electric and the hydraulic steering actuator, and the shimmy damper.

(4) Section seven (32-60-00) provides maintenance procedures for the position and warning system including indication, switches, control box, and warning circuitry.

B. Description

(1) The SA226 airplanes are equipped with fully retractable tricycle landing gear which incorporates air-oil shock struts and dual wheels. The main gear is attached to the wing nacelle structure and is retracted forward into the wheel well. The nose landing gear is mounted immediately forward of the forward pressure bulkhead and retracts forward.

(2) Components of the landing gear system include two hydraulic actuators installed on each gear, bellcrank and roller assemblies attached to the cylinder end of each actuator, a drag brace assembly attached to the aircraft structure and each gear shock strut. The landing gear also has two positioning cams mounted on each drag brace assembly, an uplock hook assembly attached to the upper portion of each drag brace, uplock rollers located on each gear strut, and flush fitted landing gear doors which when closed, completely enclose the landing gear.

C. Steering

(1) All SA226 series aircraft utilize a nose wheel steering system. Early aircraft used an electro-mechanically actuated system actuated either by a steering wheel and control cables to command an electric servo on the nose gear or by use of rudder pedals, or by use of an optional tiller wheel as a means to command an actuator on the nose gear. Later aircraft utilize a hydraulic actuator attached to the nose gear.

D. Brakes

(1) The standard brake system installed on SA226 series aircraft consists of a brake for each of the four main gear wheels. They are hydraulically actuated and contain automatic brake adjusters. As a customer option, some SA226 series aircraft have anti-skid/power brakes installed. These brake assemblies are identical to the standard brake, except for the installation of different brake adjusters, a wheel speed transducer, and different system operation.

E. Gear Indication

(1) The landing gear indication system consists of a gear indicator, gear position switches, gear warning horn and associated circuitry, flap position switch, and power lever position switches and associated power circuitry.

F. Operation

(1) Retraction, normal extension, and emergency extension.

(a) Landing gear retraction and normal extension is electrically controlled and hydraulically actuated. The landing gear handle, located on the cockpit pedestal, is used to direct 28.5 VDC electrical power from either the left essential bus or the right essential bus, as selected, to the landing gear selector valve installed on the underside of the power pack. The selector valve, when actuated, directs hydraulic pressure to the landing gear actuators for either retraction or extension of the gear.

(2) Landing gear actuators:

(a) Four identical double acting actuators are used for the main landing gear (two for each gear), and two identical actuators (different from the main landing gear actuators) are utilized for the nose landing gear.

(b) Several actuators have been used on SA226 series aircraft which have been provided by different manufacturers. Accordingly, actuators by different manufacturers or different part numbers are not mixed in any installation. Both actuators on each gear are used for gear retraction. During normal extension, only the left actuator of the nose landing gear and only the outboard actuator of each main landing gear are used. The right actuator of the nose landing gear and inboard actuator of each main landing gear are used for emergency extension of the gear, by use of a hand pump located in the cockpit.

(3) Uplock hook assembly:

(a) As each gear is fully retracted it engages a mechanical uplock hook. The uplock hook assembly consists of a spring-loaded uplock hook and mounting rod. The assembly, attached to the upper portion of the drag brace on each gear, functions as a mechanical uplock when the gear is fully retracted.

(b) When the last of the three gear fully retracts, electrical power is disconnected from the selector valve beneath the power pack. The selector valve returns to the neutral (off) position, and hydraulic pressure bleeds off rapidly, leaving the gear mechanically locked in the up position.

(c) During normal extension of the gear, each uplock hook is released by the initial movement of the actuator cylinder bellcrank and roller assemblies. During emergency extension of the gear, each uplock hook is released mechanically, via a cable, by movement of the emergency release lever, located on the cockpit floor adjacent to the left side of the copilot's seat.

(4) Bellcrank and roller assemblies:

(a) Bellcrank and roller assemblies are attached to the cylinder end of each gear actuator. The assembly engages a positioning cam and provides a means of mechanically locking the drag brace in the overcenter position when the gear is extended. When the gear is extended, initial movement of the actuating cylinder and bellcrank assembly releases the landing gear from the uplock hook.

(5) Drag brace assembly:

(a) The drag braces are attached to the aircraft structure and the landing gear shock struts. Positioning of the drag strut assemblies in an overcenter condition locks the gear in the extended position. The drag brace is held in the overcenter condition by the spring assisted bellcrank and roller assemblies and hydraulic pressure trapped in the normal gear-down actuators.

(6) Positioning cams:

(a) There are two positioning cams mounted on each of the drag brace assemblies. When the gear is extended, the bellcrank rollers engage a notch in the cam to lock the drag brace in the overcenter position.

(7) Uplock roller:

(a) An uplock roller is located on the strut of each gear. The roller engages the spring-loaded uplock hook when the gear moves into the fully retracted position, thereby mechanically locking the gear in the retracted position.

(b) During normal extension, hydraulic pressure is directed to the left actuator of the nose landing gear and the outboard actuator of each main landing gear. A one-way restrictor is installed on the extend side of the left actuator of the nose landing gear and the outboard actuators of each main landing gear to provide controlled rate and simultaneous extension of the gear.

(c) As each gear moves to the full extended (down) position, its dual drag brace unfolds, and the drag brace joints (or elbows) move to the overcenter position. This overcenter position of the extended drag brace is locked by the mechanical interference between the bellcrank and the roller assembly and the cam attached to each side of the drag brace.

(d) When the landing gear is down and locked, the down solenoid of the selector valve remains energized and pressure is constantly applied as long as electrical power and hydraulic pressure are available.

(e) Provisions are installed for emergency free fall extension of the landing gear. The free fall can be supplemented by hand pumped hydraulic pressure. Emergency hand pump pressure is not routed through the selector valve. When the emergency hand pump is used, hydraulic pressure is applied to the down side of the right nose gear actuator and inboard actuators of each main gear. Initial movement of the emergency extension system is transmitted to unlock the drag brace and release the mechanical uplock hook. The gear free falls of their own weight and the force of the airstream carries the gear to the down and locked position. The hand pump can be actuated to provide pressure to the actuators to further ensure locking of the gear. There is no provision for emergency retraction of the gear.

(8) Gear doors:

(a) The main gear and nose gear doors consists of a left and right door, hinge mounted to each side of the wheel nacelle forward of each gear assembly. Additionally, a spade door is hinge mounted aft of each of the main gear strut. The left and right doors are spring-loaded overcenter to the open position when the gear is extended. They are mechanically closed by linkage attached from the door to the door linkage hook when the gear retracts. The door actuator hook is actuated by a roller mounted on the drag brace which engages or disengages the hook of the door actuator as the gear retracts or extends. The spade door, attached to the gear strut by linkage, mechanically opens and closes with extension and retraction of the gear.

(9) Steering:

(a) All SA226 series aircraft steering systems turn a shaft (inner Cylinder) which extends through the nose gear strut assembly to the wheels. These systems, when used in conjunction with thrust from the engines, provide directional control of the aircraft while taxiing or parking.

(10) Brakes:

(a) Except for the anti-skid brake system, the standard brake system is operated manually by depressing the toe of the brake pedals mounted on the pilot's and copilot's rudder pedals. Fluid for the standard brakes is supplied from a small brake system reservoir mounted in the nose baggage compartment. The hydraulic system for the standard brakes is completely independent of the main 2.000 psi hydraulic system used for landing gear and wing flap actuation; whereas, the hydraulic system for the anti-skid/power brakes is provided by the main hydraulic system.

(b) Fluid is supplied by the reservoir to the four brake master cylinders through hydraulic tubing to the wheel brakes. Hydraulic pressure to the brake assembly forces the brake pistons to press the three carrier, lining, and button assemblies against the brake rotor disc and in turn against the three carrier, lining, and button assemblies installed on the torque plate.

(c) With the anti-skid/power brake system, the hydraulic pressure is metered proportionately toe pedal pressure via a power brake relay valve. Should the anti-skid system sense an impending skid condition, the system cycles to release or increase automatically metered brake pressure in order to prevent a wheel skid condition.

(11) Gear indication:

(a) The landing gear warning horn operates in conjunction with the power lever switches or the flap position switch to provide an aural warning of an unsafe gear condition. If the landing gear are not fully extended and either: (1) the flaps are extended more than 20° or, (2) power lever is retracted to the FLIGHT IDLE position, the warning horn will actuate.

NOTE: Aircraft T201-208 and AT001-003 do not incorporate the 20° warning function.

1. Servicing Landing Gear

A. Service nitrogen/air pressure IAW Chapter 12.

B. Fluid Servicing IAW Chapter 12.

C. Lubrication IAW Chapter 12.

2. Inspection/Check – Landing Gear

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

NOTE: This check must be performed after any maintenance is done on the landing gear such as rigging or adjustment of the landing gear, landing gear doors, or landing gear switches. It must also be done after replacement of any part of the landing gear, actuators, landing gear control handle, landing gear indicator, landing gear switches, landing gear emergency control valve, landing gear doors, landing gear emergency hand pump, hydraulic reservoir in the left wheel well, or any other components or part that might prevent the landing gear from extending to the down and locked position or cause the indication system to show a false down and locked indication.

A. Extension and Retraction Check

(1) Jack aircraft.

(2) Remove screws in lower panel of the left nacelle just aft of engine compartment.

(3) Remove red and orange caps on quick disconnect connections.

(4) Clean quick disconnect connections on aircraft and hydraulic cart with clean rag.

(5) Connect hydraulic cart to aircraft.

(6) Connect GPU to aircraft.

(7) Check landing gear handle in DOWN position and turn battery switches ON.

(8) Energize hydraulic cart and fill hydraulic reservoir to full mark with flaps up.

(9) Clear all personnel and equipment from landing gear wheel wells and landing gear arc.

(10) Check landing gear indicator shows all three gear down and locked.

(11) Check hydraulic pressure gauge shows approximately 2,000 psi.

(12) Place both power levers at the flight idle gate.

(13) Place landing gear handle to UP.

(14) All three gears must immediately start to retract, landing gear indicator must show all three gear in transit, and landing gear warning horn must sound.

(15) As landing gear are retracting, push both power levers forward approximately 1/8 inch. Check landing gear warning horn stops sounding.

(16) When all three landing gears are retracted, landing gear indicator must show all three landing gear up and locked. Check all landing gear doors fully closed and fared.

(17) Run flaps DOWN. As flaps pass 20° (about 5/8 on indicator). landing gear warning horn must sound.

(18) Run flaps UP. Before flaps pass 20° (1/2 on indicator), horn must stop sounding.

(19) Pull landing gear free fall handle completely aft. The landing gear uplocks should release before 45° of travel of the free fall handle. The hydraulic power pack should shut off prior to or simultaneously with the gear lock release. Both main gears should be rigged to release simultaneously, allowing symmetrical extension. The free fall handle should go over center and remain in the maximum travel position.

NOTE: If during free-fall, any of the three gears do not fall into the down and locked condition, simulate an airload by applying not more than 25 pounds of force at the lower end of strut (between the wheels) pulling rearward. If more than 25 pounds of force is required to get the gear down and locked, problems exist which must be identified and eliminated.

(20) Push landing gear emergency free-fall handle forward. All three landing gear must retract to up and lock position.

(21) Turn hydraulic cart OFF and bleed hydraulic pressure to ZERO.

(22) Pull LAND GEAR CONT circuit breaker on left essential bus.

(23) Pull pip pin and place landing gear selector valve handle forward to EMERGENCY position.

(24) Pull landing gear free-fall handle completely aft. Handle must stay aft and not spring forward. All three landing gears must extend. toward down position and nose gear should down and lock.

(25) Place landing gear handle to DN position.

(26) Pump emergency hand pump until hydraulic pressure gauge reads approximately 1,000 psi. All three landing gear must be down and locked. Hydraulic pressure must not bleed down below 700 psi in less than five minutes.

(27) Place LDG GEAR CONT bus transfer switch to RIGHT.

(28) Place free-fall handle in the EMERGENCY position. With 700 psi hand pump pressure showing on the gauge, apply 2,000 psi with hydraulic cart. Move the pedestal landing gear control handle to the UP position. Gauge pressure will increase to 2,000 psi. The Landing gear may or may not unlock, but in no case should it retract. Move landing gear control handle in the DN position and move landing gear selector valve to NORM GEAR and install pip pin. Note all three landing gears down and locked.

(29) Pull GEAR & FLAP POS circuit breaker. Landing gear indicator must show power OFF (three green lights extinguished or three crosshatches).

(30) Place LDG GEAR & WING POS transfer switch to RIGHT; Landing gear indicator must show all three gears down and locked.

(31) Energize hydraulic unit and cycle gear two times.

(32) With landing gear down and locked and flaps up, fill hydraulic reservoir to full mark.

(33) Disconnect hydraulic cart. Replace caps and close access panel.

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

(34) Turn battery switches OFF.

(35) Reset circuit breakers and place transfer switches to LEFT.

(36) Disconnect APU.

(37) Down jack aircraft.

B. Leak Check – Landing Gear Strut

(1) Check main and nose landing gear struts for leaks as follows:

NOTE: Refer to Chapter 5 for leak check time limit.

(a) Position a can or similar container to catch fluid from nitrogen service valve.

(b) Slowly open nitrogen service valve.

(c) Collect expelled hydraulic fluid.

(d) Use measuring cup to determine amount of fluid expelled.

(e) Replace floating piston seals if more than one (1) ounce of fluid is expelled from strut.

(f) Service strut(s) IAW Chapter (12).

LANDING GEAR SAFETY SWITCHES – DESCRIPTION AND OPERATION

1. General

A. Two landing gear safety switches are installed on each of the main landing gear scissors (see Figure 201) to sense whether the aircraft is on the ground or airborne and to control or override certain system functions accordingly. The functions are:

B. Left Main Landing Gear 8-Wire Switch

(1) Left engine air/ground start circuit selection.

(2) Nose wheel steering circuit is disabled in flight (when variable authority nose wheel steering is installed). On aircraft that have complied with SB32-037, steering is disabled through the downlock relays when gear starts to retract.

(3) Landing gear anti-retract safety switch. Disables up circuit of gear selector valve solenoid when aircraft is on the ground.

C. Left Main Landing Gear 6-Wire Switch

(1) Pitch trim out-of-trim-for-takeoff warning circuit (circuit is disabled when the aircraft is in flight).

(2) Stall avoidance system stick pusher; circuit is disabled when aircraft is on the ground.

(3) Stability augmentation system; augmentation is disabled when the aircraft is on the ground (SA226-T only).

(4) Yaw damper; system is disabled when the aircraft is on the ground (SA226-T only).

D. Right Main Landing Gear 6-Wire Switch

(1) Cabin pressure dump; completes electrical circuit to dump valve when the aircraft is on the ground.

E. Right Main Landing Gear 8-Wire Switch

(1) Right engine air/ground start circuit selection.

(2) Nose wheel steering; circuit is disabled in flight. Both main landing gear must be on the ground to obtain power to the nose wheel steering servo. On aircraft that have complied with SB32-037 steering is disabled when gear starts to retract. Both main landing gear down lock switches must be engaged to obtain power to the nose wheel steering servo.

LANDING GEAR SAFETY SWITCHES – MAINTENANCE PRACTICES

1. Adjustment/Test – Landing Gear Safety Switches

A. Adjustment/Test – Switch Adjustment

(1) Adjust each safety switch to actuate when the strut is compressed 3/4 inch from fully extended.

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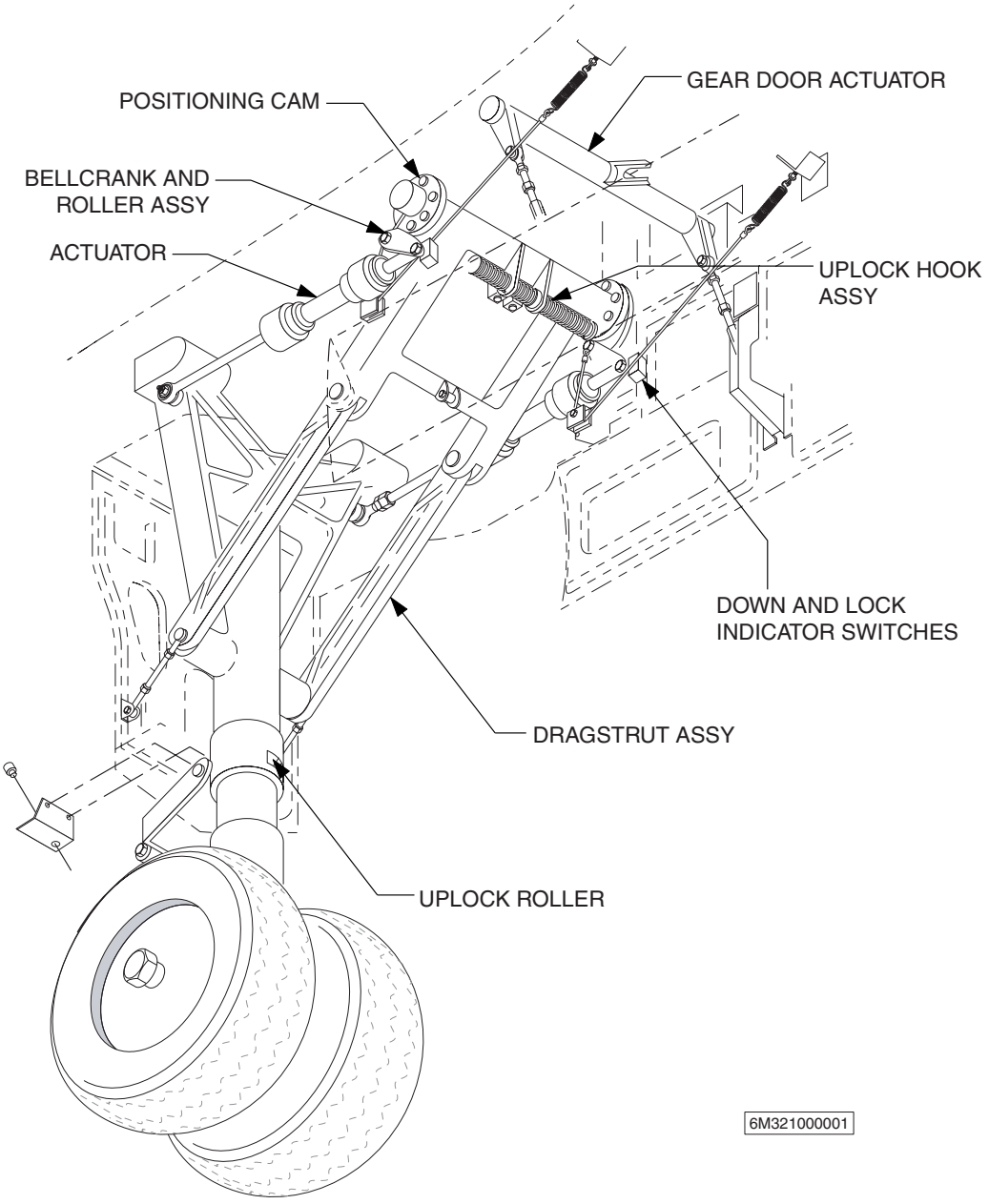
MAIN LANDING GEAR – DESCRIPTION AND OPERATION

1. General

A. The main landing gear is attached to the wing nacelle structure and retracts forward into the wheel well. Landing gear extension and retraction is electrically controlled and hydraulically actuated.

B. Each of the two main landing gear consists of an air-oil telescoping shock strut, a folding drag brace assembly attached to the aircraft structure and shock strut, two identical double acting hydraulic actuators, and dual wheels and brakes.



Main Landing Gear Assembly
Figure 1

C. A spring-loaded uplock hook and mounting rod is attached to the upper portion of the drag brace and functions as the mechanical uplock when the gear is in the fully retracted position.

D. An uplock roller located on each gear engages the spring-loaded uplock hook to mechanically lock the gear in the retracted position. Two positioning cams are mounted on each drag brace assembly to insure locking of the drag strut in the overcenter position when the gear is extended. A spring assisted bellcrank and roller assembly is attached to the cylinder end of each gear actuator to assist the positioning cams in their operation.

E. The landing gear doors consists of a left and right door, hinge-mounted to each side of the wheel well forward of each gear assembly. Landing gear doors are mechanically operated. An actuator mechanism, installed on the upper portion of the drag brace, has a roller that engages the aft fork on the gear door actuator mechanism and closes the doors after the gear has extended.

F. Upon retraction of the gear, a roller at a lower point on the drag brace contacts the forward fork of the door actuator mechanism closing the door. Each main gear incorporates:

(1) Two down position switches installed at each side attached to the wheel well structure adjacent to the bellcrank hardstop.

(2) One up position switch mounted adjacent to mechanical uplock hook.

(3) One up limit switch mounted adjacent to mechanical uplock hook alongside the up position switch.

(4) Two landing gear safety switches installed on each main landing gear scissors.

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MAINTENANCE MANUAL

MAIN GEAR STRUT – DESCRIPTION AND OPERATION

1. General

A. Each main landing gear strut is composed of two telescoping cylinders with enclosed ends. The two cylinders, when assembled, form an upper and lower chamber. The chambers are separated by a floating piston. The upper cylinder is serviced with hydraulic fluid (MIL-PRF-5606). The upper chamber has an orifice dividing it into two smaller chambers.

B. Hydraulic fluid must pass through this orifice during compression and extension of the strut. Metering of the fluid through the orifice, plus compression of nitrogen in the lower chamber, provides the absorption and dissipation of the energy transmitted to the strut and controls the rate of vertical motion.

C. Each strut contains a number of seals to prevent the loss of nitrogen and hydraulic fluid. A packing gland is installed at the open end of the outer cylinder to seal the sliding joint between the telescoping cylinders. A scraper (in a grove in the upper jacket) keeps the sliding surface of the lower cylinder free of dirt, mud, ice, snow, and other contaminants.

2. Operation

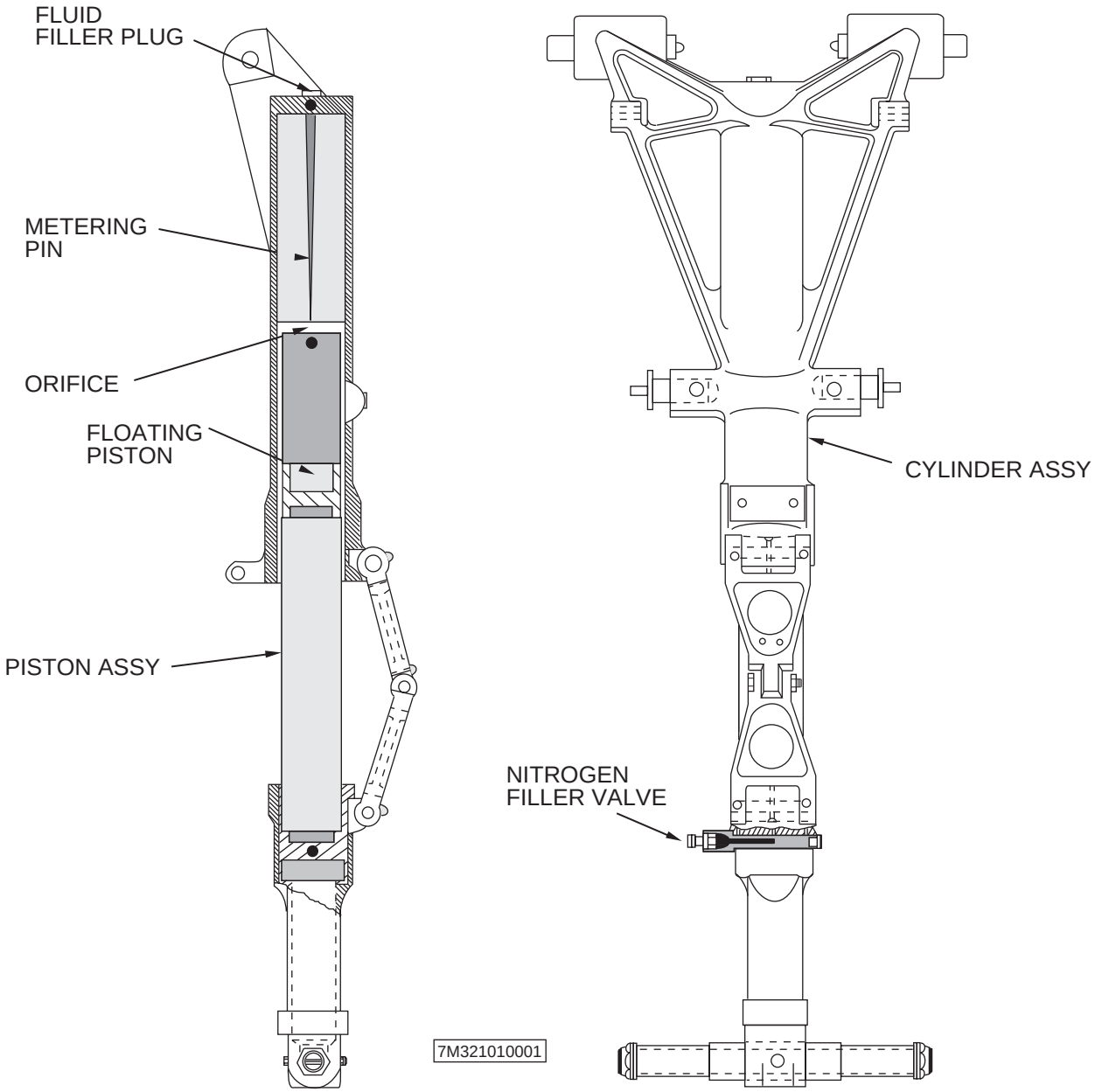
A. As loads are applied to the landing gear the piston forces hydraulic fluid in the cylinder to transfer, through the orifice, to the piston assembly. Since the diameter of the orifice varies due to it's relationship to a tapered plunger, the fluid flow will also vary. As the piston moves towards the top of the cylinder, the orifice becomes smaller, causing a greater restriction of flow and slowing the piston travel.

B. As cylinder hydraulic fluid is displaced, the floating piston moves toward the lower end of the piston assembly. To prevent the floating piston from traveling to the bottom of the piston, the lower portion of the piston is charged with nitrogen.

C. When loads are removed from the gear, the weight of the piston and the attached wheels, causes the hydraulic fluid to be forced back into the cylinder.

D. Small holes are drilled near the top of the piston to permit hydraulic fluid to coat the inner surface of cylinder, providing lubrication.

MAIN GEAR STRUT – DESCRIPTION AND OPERATION



Main Landing Gear Strut
Figure 1

MAIN GEAR STRUT – MAINTENANCE PRACTICES

1. Removal/Installation – Main Gear Strut

A. Removal – Main Gear Strut (Figure 201)

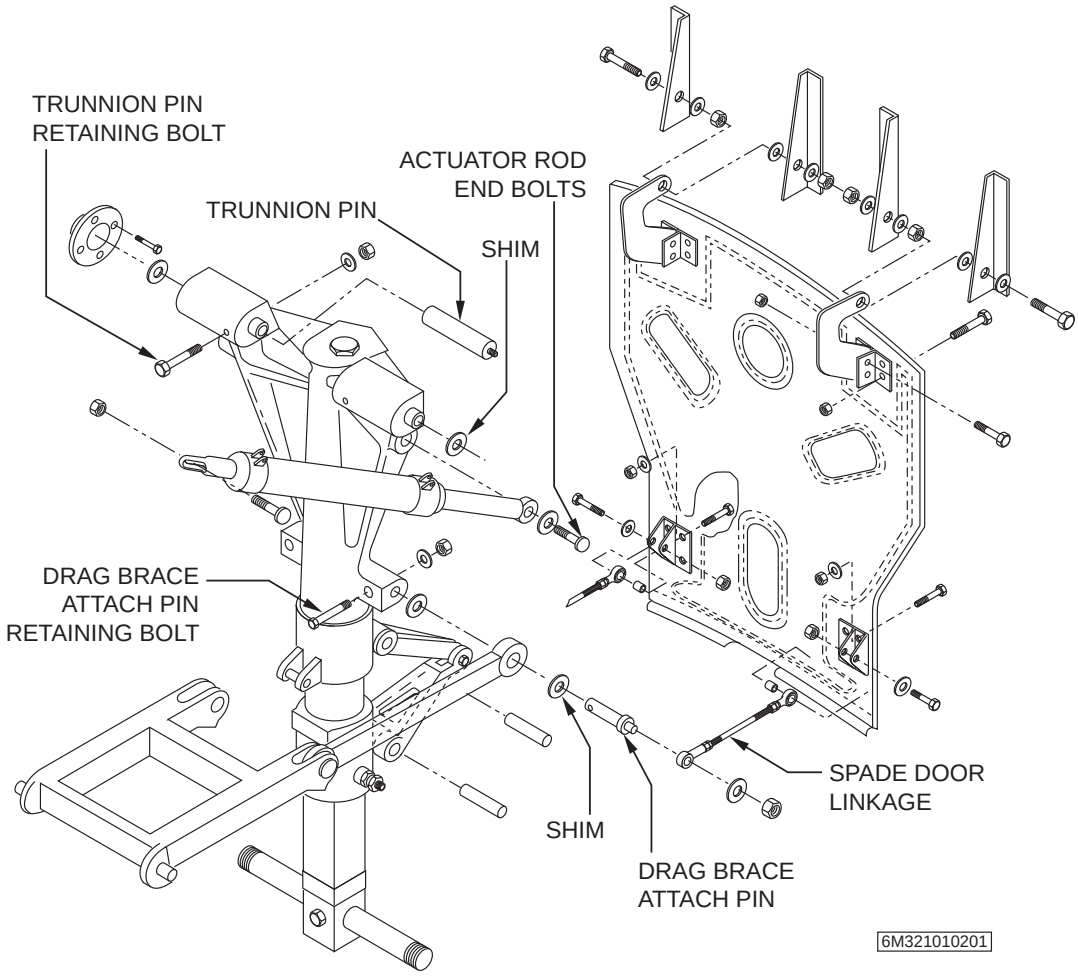
(1) Jack aircraft. (Refer to 07-10-00)

(2) Bleed hydraulic pressure from gear-down side of actuators by disconnecting hydraulic lines to actuator. Cap disconnected hydraulic lines and actuator port to prevent contamination.

(3) Disconnect brake line at top of wheel well and at elbow below scissors assembly. Cap ends to prevent contamination.

NOTE: Brake line may be removed intact with gear strut assembly.

(4) Disconnect scissors switches from mounting bracket at top of scissors.



Main Gear Strut and Door - Removal/Installation
Figure 201

(5) Disconnect clamps securing switch assembly wiring to gear strut assembly and stow away from gear assembly while removing gear.

(6) Remove both actuator rod end nuts, washer and bolts from gear assembly attach points and free both actuators from gear strut assembly.

(7) Remove nuts and washers securing main landing gear spade door linkages to drag brace attach pin and stow linkages away from gear assembly while facilitating gear removal.

(8) Remove drag brace attach pin retaining bolts, washers, and nuts to remove drag brace attach pins.

(9) Remove drag brace attach pins and washers.

(10) Free drag brace assembly from gear assembly.

(11) Remove strut assembly trunnion pin retaining bolt, washers, and nuts.

NOTE: Determine if shims are installed between trunnion assembly and trunnion bushing. Note amount and location of shims before removal to facilitate assembly.

(12) Remove strut assembly trunnion pins by sliding them inboard and carefully lower shock strut assembly.

B. Installation – Main Gear Strut (Figure 201)

NOTE: Before installing main gear strut, inspect trunnion pin and bearing for wear. If diameter of trunnion pin is 1.1210 inch (28.4734 mm) or less, replace pin. If internal diameter of bearing is 1.1255 inch (28.5877 mm) or greater, replace bearing.

(1) Position shock strut assembly into mounting location.

(2) Install shims (AN960-1818L) between the gear trunnion and mount bushings at each side as required. Obtain a combined total minimum clearance of 0.010 inches (0.254 mm) and maximum clearance of 0.032 inches (0.813 mm) before final installation of the trunnion pins.

(3) Position trunnion pins by sliding them outboard into trunnion bushings at each side. Align pins for installation of retaining bolts.

(4) Install trunnion pins retaining bolts, washers, and nuts. Torque 15 to 30 inch-pounds (1.69 to 3.39 N·m).

(5) Position drag brace assembly to gear strut assembly.

(6) Install shims (four each AN960-1616L), one each or more if required, to each side of drag brace assembly before final installation of drag brace attach pins. Obtain a combined total minimum clearance of 0.010 inches (0.254 mm) and maximum clearance of 0.032 inches (0.813mm). Install drag strut attach pins and align pins to install retaining bolts.

(7) Install drag brace attach pin retaining bolts, washers, and nuts. Torque 15 to 30 inch pounds (1.69 to 3.39 N·m).

(8) Connect main landing gear spade door linkages to drag brace attach pins and secure by installing washers, nuts, and cotter pins.

(9) Connect actuator rod ends to gear assembly and install bolts, washers, and nuts. Secure with cotter pins.

(10) Install scissors switches and electrical wiring mounting brackets.

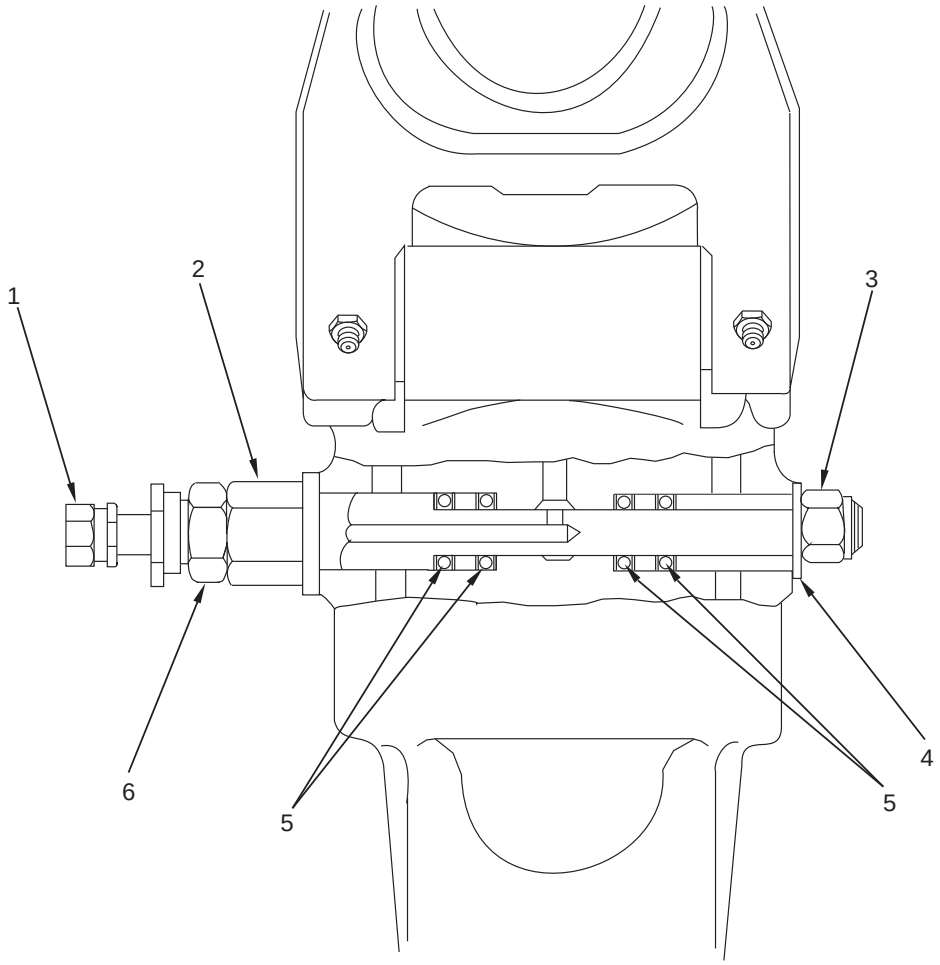
(11) Install brake line connections at previously disconnected points.

(12) Install hydraulic line connections to actuators at previously disconnected points.

(13) Bleed brake system as directed in BRAKE BLEEDING PROCEDURES, Section 32-43-00.

(14) Lubricate main landing gear IAW Chapter 12-20-10.

(15) Perform Adjustment/Test – Retraction and Extension, Section 32-30-00.



Nitrogen Service Valve - Removal/Installation
Figure 202

C. Removal – Nitrogen Service Valve (Figure 202)

(1) Jack aircraft. (Refer to 07-10-00)

(2) Retract gear halfway or approximately 45 degrees. Bleed hydraulic pressure supply of the hydraulic test stand to zero and pull landing gear control circuit breaker on the left essential bus. Place a drip pan, if available, beneath the gear being resealed.

(3) Discharge all nitrogen (air pressure) by opening nitrogen service valve (item 1).

(4) Remove valve body (item 6) from bolt (item 2).

(5) Remove lock nut (item 3) and bushing (item 4).

(6) Remove bolt (item 2) from strut.

(7) Replace seals (item 5) prior to assembly.

D. Installation – Nitrogen Service Valve (Figure 202)

(1) Install bolt (item 2) with new seals into strut.

(2) Slide bushing (item 4) with new seals onto bolt (item 3) and into strut.

(3) Secure bolt (item 2) with lock nut (item 3). Torque lock nut 220 to 300 inch-pounds (24.86 to 33.90 N·m).

(4) Install valve body (item 6) into bolt (item 2). Torque valve body 100 to 125 inch-pounds (11.30 to 14.12 N·m).

(5) Service strut IAW Chapter 12-20-10.

(6) Down jack aircraft and remove jacks. (Refer to 07-10-00)

E. Removal – Main Gear Axle (Figure 203)

(1) Tools and equipment required to remove axle are as follows:

(a) One (1) 0.75 inch (19.05 mm) threaded steel rod 24.5 inches (62.23 cm) long.

(b) Two (2) 0.75 inch (19.05 mm) nuts with threads matching steel rod.

(c) Two (2) 0.25 inch (6.35 mm) thick steel plates (or washers) with 0.75 inch (19.05 mm) holes.

(d) One (1) steel sleeve, 11.5 inches (29.21 cm) long X 1.80 inches (4.57 cm) minimum inside diameter (may be fabricated from length of steel tubing).

(e) Dry ice or CO_2.

(f) Protective washer, 3.0 inches (7.62 cm) O.D X 1.80 inches (4.57 cm) I.D. Make from 0.25 inch (6.35 mm) phenolic or Teflon sheet.

(2) Jack aircraft. (Refer to 07-10-00)

(3) Remove wheels IAW Section 32-40-00.

(4) Remove brakes IAW Section 32-43-10.

(5) Remove nut, washer, and bolt securing axle to strut.

(6) Install protective washer over axle. Slide protective washer against strut yoke.

(7) Chill axle (from inside) with dry ice or CO2.

(8) With axle chilled, position sleeve over axle; insert bolt, washer, and nut on one end through axle as illustrated in Figure 203.

(9) Install plate and second nut. Tighten second nut until axle is clear of strut yoke.

(10) Remove axle.

F. Installation – Main Gear Axle (Figure 203)

WARNING: WEAR PROTECTIVE GLOVES WHEN HANDLING FROZEN PARTS, OR HANDLE WITH SUITABLE TOOLS.

(1) Chill axle with dry ice or CO_2.

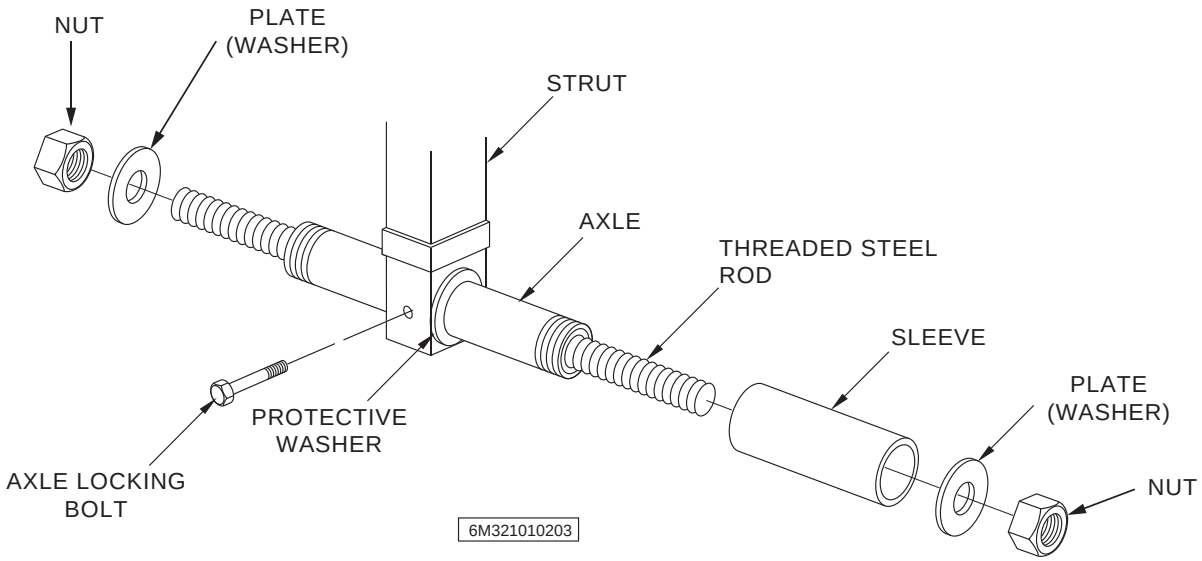
(2) Install axle into strut with axle locking bolt hole aligned with hole in strut yoke.

(3) Secure axle to yoke with axle locking bolt, washer, and nut.

(4) Install brakes. (Refer to 32-43-10)

(5) Install wheels. (Refer to 32-40-00)

(6) Down jack aircraft and remove jacks. (Refer to 07-10-00)



Landing Gear Axle - Removal/Installation
Figure 203

G. Disassembly – Main Landing Gear Strut (Figure 204)

(1) Jack aircraft. (Refer to 07-10-00)

(2) Remove main gear strut IAW instructions in this section.

Note: Accomplish disassembly/assembly of the Main Landing Gear Strut with the strut removed from the airplane if practical. When absolutely necessary, disassembly/assembly may be accomplished with strut installed.

(3) If disassembling strut with gear installed in aircraft:

(a) Connect external electrical power to aircraft. (Refer to 24-40-00, page 201)

(b) Connect external hydraulic power cart to aircraft. (Refer to 29-00-00, page 201)

(c) Retract nose gear halfway (approximately 45°). Bleed external hydraulic power cart supply pressure to zero (0) and pull landing gear control circuit breaker.

(4) Place a drip pan beneath gear being disassembled.

(5) Discharge nitrogen (air pressure) by opening nitrogen air valve.

(6) Remove upper filler plug. Install adapter with hose to filler plug hole and route free end of hose to a catch can or bucket.

(7) Close nitrogen valve and slowly charge strut cylinder with nitrogen (air), forcing hydraulic fluid out of cylinder.

(8) Discharge residual hydraulic fluid by compressing strut.

(9) Discharge nitrogen (air pressure) by opening air valve.

(10) Remove cotter pin, nut, washer and pivot bolt securing torque links (scissors) and separate scissors.

(11) Remove retaining ring and washer from cylinder.

(12) Pull strut piston out of cylinder.

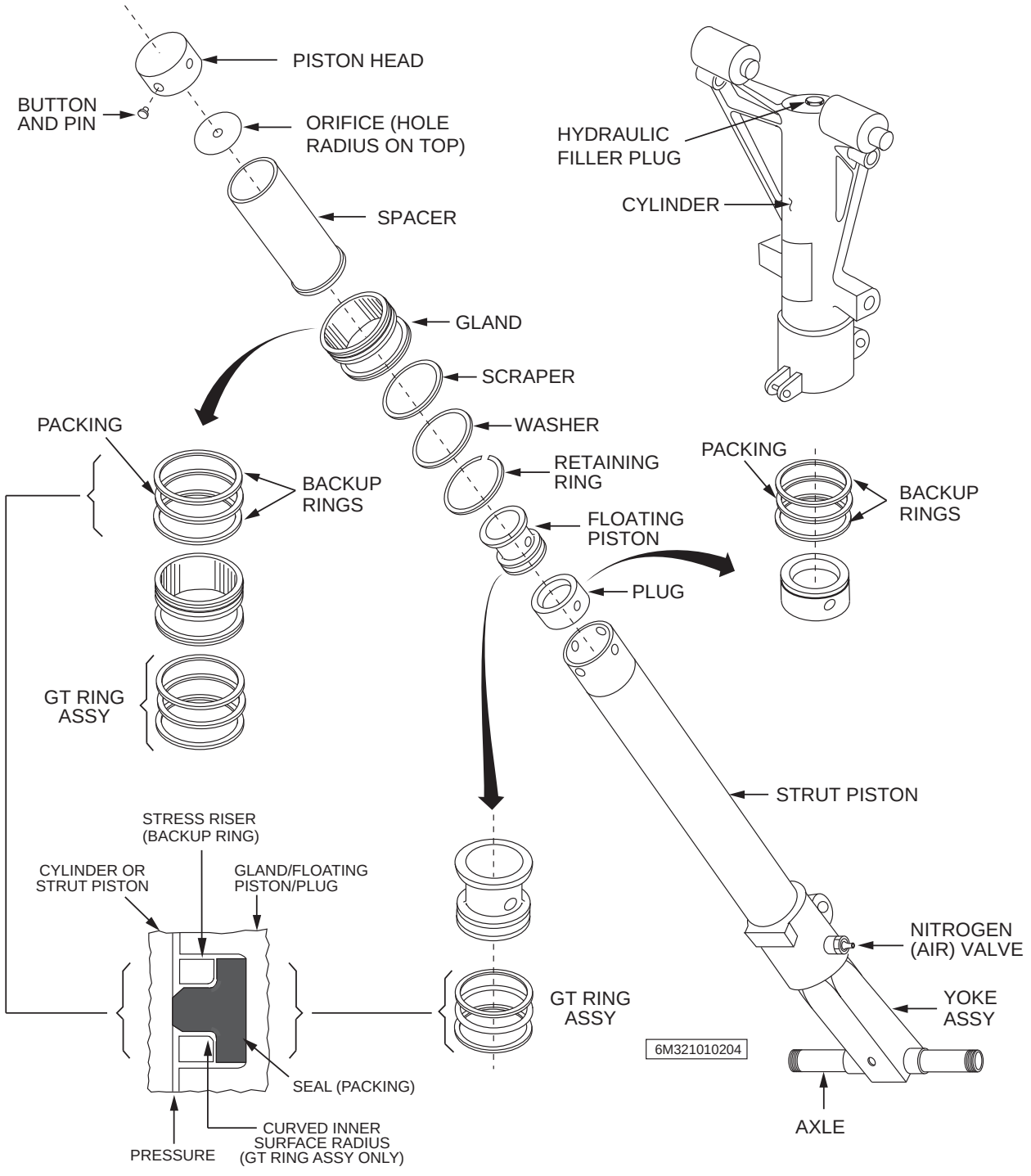
(13) Remove buttons and pins securing piston head to top of strut piston. Making sure to capture orifice from underneath piston head, remove piston head and orifice.

(14) Remove spacer from strut piston.

(15) Remove gland and scraper from strut piston.

WARNING: IT MAY BE NECESSARY TO APPLY AIR PRESSURE TO AIR VALVE TO FORCE FLOATING PISTON FROM STRUT PISTON. DO NOT USE EXCESSIVE PRESSURE WHILE ATTEMPTING TO REMOVE FLOATING PISTON. IT CAN ACT AS A PROJECTILE, POSSIBLY RESULTING IN PERSONNEL INJURY.

(16) Remove floating piston from strut piston chamber.



Main Landing Gear Strut (Resealing)
Figure 204

H. Re-packing and Re-sealing (Figure 204)

CAUTION: DO NOT ALLOW RUBBER SEALS TO CONTACT DEGREASING SOLVENT. SOLVENT MAY CAUSE PREMATURE DETERIORATION OF SEALS.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK DOES CONSTITUTE A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(1) Rinse all parts in degreasing solvent (MPK or equivalent).

(2) Check piston head orifice and all disassembled parts for contamination, damage, or obstruction. Replace if necessary.

(3) Before assembly, soak all seals in hydraulic fluid (MIL-PRF-5606).

CAUTION: O-RING ASSEMBLIES ARE GT RING ASSEMBLIES THAT HAVE A CURVED INNER SURFACE. INSTALL STRESS RISER SO CURVED PART OF STRESS RISER FITS INTO CURVED PART OF SEAL.

(4) Replace packing, backup rings and GT ring assembly on piston gland.

(5) Replace GT ring assembly on floating piston.

(6) Replace scraper.

(7) Replace air valve packings.

(8) Replace hydraulic filler plug packing.

(9) Replace four buttons in piston head.

(10) Inspect piston cylinder and cylinder bore for scoring. Superficial scratches and nicks can be removed with #600 emery cloth.

(11) Lubricate all disassembled parts, piston chamber, piston surfaces, and inner diameter of cylinder assembly with hydraulic fluid (MIL-PRF-5606) before assembly.

I. Assembly (Figure 204)

(1) Install floating piston into piston chamber.

(2) Install retaining ring and washer over piston.

(3) Install scraper over piston beveled side down.

(4) Install gland over piston.

(5) Install spacer over piston.

(6) Install orifice on piston with hole radius on top.

(7) Install piston head and orifice onto strut piston and secure with buttons and pins.

(8) Slide strut piston assembly into cylinder.

(9) Secure strut piston assembly to cylinder with washer and retaining ring in grooves in bottom of cylinder bore.

(10) Align upper and lower scissors and install pivot bolt, washer and nut. Bottom out nut and back off to nearest cotter pin slot. Maximum torque to scissors pivot attachment is 5 inch-pounds (0.56 N·m). Install cotter pin.

(11) Install nitrogen service valve IAW instructions in this section.

(12) Replenish strut with hydraulic fluid. (Refer to 12-10-30).

(13) Leak check strut.

NOTE: Strut may be horizontal or vertical for this test.

(a) Wipe down entire strut assembly with a clean cloth.

(b) Service strut to 350 psi (24.61 kg/cm^2) with nitrogen.

(c) After one (1) hour, verify nitrogen pressure is still 350 psi (24.61 kg/cm^2).

(d) Verify no hydraulic fluid is leaking from between cylinder and strut piston or from filler plug.

(e) Slowly open nitrogen service valve to bleed off pressure and verify strut shows no signs of hydraulic fluid leakage from valve.

(f) Close nitrogen service valve.

(14) If main gear strut was removed to perform this task, reinstall IAW instructions in this section.

(15) If main gear strut was NOT removed to perform this task, perform 32-00-00, Functional Check – Normal and Emergency Systems Extension and Retraction. Adjust normal and/or emergency system rigging if necessary IAW 32-30-00, Adjustment/Test.

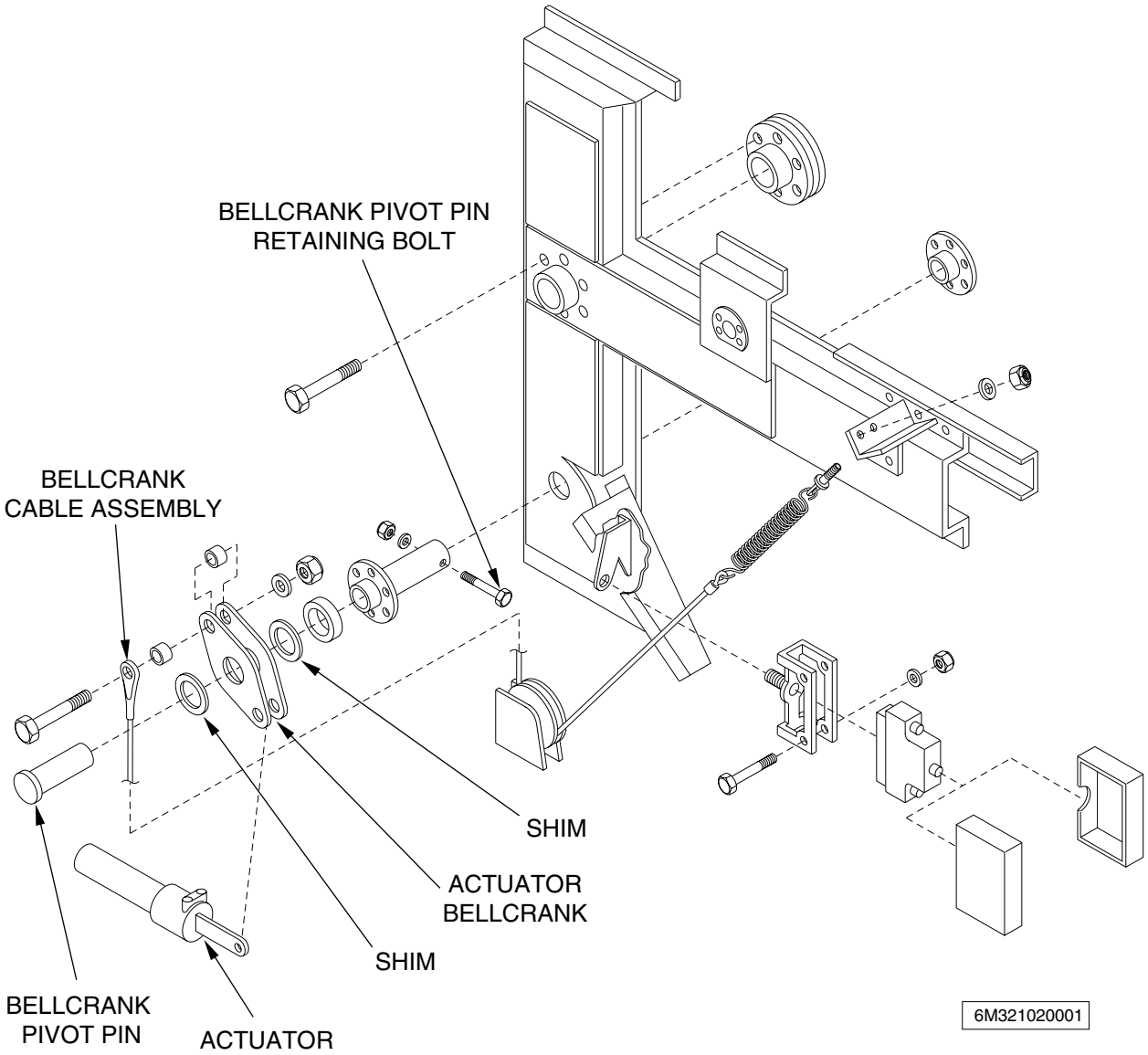
2. Approved Repairs – Main Gear Strut

A. Approved repairs for Main Gear Strut can be found in SA226/227 Structural Repair Manual, Chapter 54-90-50.

MAIN GEAR BELLCRANK AND ACTUATOR – DESCRIPTION AND OPERATION

1. General

A. Each actuator cylinder is attached to an actuator bellcrank and roller assembly through which the initial actuator movement is transmitted to unlock the drag brace during retraction and to release the mechanical uplock hook assembly during gear extension. Movement of the actuator in the selected direction continues until the bellcrank and roller assembly hits a hardstop. Each actuator piston is attached to its respective gear strut. After initial unlocking action by the actuator, its piston moves to retract or extend the gear, as selected.



Bellcrank Pivot
Figure 1

MAIN GEAR BELLCRANK AND ACTUATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Main Gear Bellcrank and Actuator

A. Removal – Main Gear Bellcrank

(1) Jack aircraft.

(2) Bleed hydraulic pressure from gear-down side of actuator by disconnecting hydraulic lines to actuators. Cap disconnected lines and actuator port to prevent contamination.

(3) Remove access panel on nacelle to gain access to retaining bolt securing pivot pin.

(4) Connect spring removal tool, SAT1391, and remove bellcrank spring. See Illustrated Tool and Equipment manual for tool.

(5) Remove retaining bolts, washers, nut, and spacer securing bellcrank cable assembly to bellcrank. Disconnect cable assembly.

(6) Remove bellcrank pivot pin retaining bolt.

(7) Remove bellcrank pivot pin.

NOTE: Care should be taken to note location of, and retain shims inboard and outboard of bellcrank.

(8) Remove actuator rod end nuts, washers, and retaining bolts from gear assembly. Remove actuators.

B. Installation – Main Gear Bellcrank

NOTE: Several actuators used on SA226 series aircraft have been provided by different manufacturers. Actuators by different manufacturers or with different part numbers should not be mixed in any installation (see Illustrated Parts Catalog).

(1) Position actuator and bellcrank into mounting location and install bellcrank pivot pin.

(2) Shim bellcrank assemblies so that both the cam and striker fall on the roller of the bellcrank.

(3) Install bellcrank pivot pin retaining bolt. Torque 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(4) Install bellcrank cable assembly and secure with retaining bolt, washer, spacer, nut, and cotter pin.

(5) Install bellcrank spring using tool SAT1391.

(6) Install actuator rod end to gear assembly and install bolts, washers, nuts, and cotter pin.

(7) Install hydraulic line connections to actuators.

(8) Functional check landing gear IAW LANDING GEAR RIGGING PROCEDURES, IAW Section 32-30-00.

(9) Install access panel on nacelle.

C. Removal – Main Gear Actuator

(1) Jack aircraft.

(2) Bleed hydraulic pressure from down side of actuator by disconnecting hydraulic lines from actuator. Cap disconnected lines and actuator port to prevent contamination.

(3) Remove spring between bellcrank and drag brace mount assembly.

(4) Remove bellcrank pivot pin and retaining bolt.

(5) Remove bellcrank pivot pin.

(6) Remove actuator rod end nuts, washers, and retaining bolts from gear assembly. Remove actuators.

D. Installation – Main Gear Actuator

(1) Position actuator and bellcrank into mounting location and install bellcrank pivot pin.

(2) Install bellcrank pivot pin retaining bolt, washers, and nuts. Torque 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(3) Install spring between bellcrank and drag brace mount assembly.

(4) Install actuator rod ends to gear assembly and install bolts, washers, and nuts. Torque bolts 480 to 690 inch-pounds (54.23 to 77.96 N·m).

(5) Secure with cotter pins.

(6) Install hydraulic lines to actuators.

(7) Perform actuator adjustment and landing gear functional check.

2. Inspection Check – Main Gear Bellcrank and Actuator

A. Inspection – Bellcrank and Bushings

(1) Inspect bellcrank and bushings for cracks and wear.

(a) Replace bushing if worn beyond maximum limit of 0.7535 inch (19.139 mm) diameter.

(2) Inspect shoulder bolt for evidence of corrosion, cracks, and wear.

(b) Replace shoulder bolt if worn beyond maximum limit of 0.7475 inch (18.987 mm) diameter.

B. Inspection and Repair – Bellcrank Pivot Pin and Mounting

NOTE: Perform this inspection whenever main landing gear assemblies are removed from the aircraft.

(1) Jack aircraft.

(2) Disconnect bellcrank springs on both main landing gear.

(3) Remove oval access panels on inboard and outboard of each engine nacelle at N.S. 130.

(4) For each pivot pin, disassemble and inspect as follows:

(a) Remove bellcrank pivot pin retaining bolt, washer, and nut. Remove retaining bolt.

(b) Remove shoulder bolt, and support gear actuator clear of access to bolt hole.

CAUTION: HOLE TOLERANCE OF FITTING MUST NOT EXCEED 0.7519 INCH (19.0983 mm), AND MUST NOT BE ENLARGED OR ELONGATED BY REAMING.

(c) Measure diameter of hole through the web and doubler.

(5) If holes through both web and doubler are between 0.385 and 0.397 inch (9.7790 and 10.0838 mm) diameter, assemble as follows:

(a) Install shoulder bolt, washer, and nut.

(b) Install AN960-616 washers under bolt shank as required to maintain a gap (at bolt head) between 0.005 and 0.020 inch (0.127 to 0.508 mm).

(c) Torque nut 160 to 190 inch-pounds (18.08 to 21.47 N·m).

(d) Install pivot pin retaining bolt, washer, and nut.

(e) Torque nut 20 to 25 inch-pounds (2.26 to 2,82 N·m).

MAIN GEAR DRAG BRACE – DESCRIPTION AND OPERATION

1. General (Figure 1)

A. The drag brace is attached to the aircraft structure and the landing gear shock strut. When the main gear is fully extended, the drag brace joints (elbows) are designed to be over-center to lock the gear down. The drag brace is held in this locked position by spring-assisted actuator bellcranks positioned against cams on the upper drag brace trunnions.

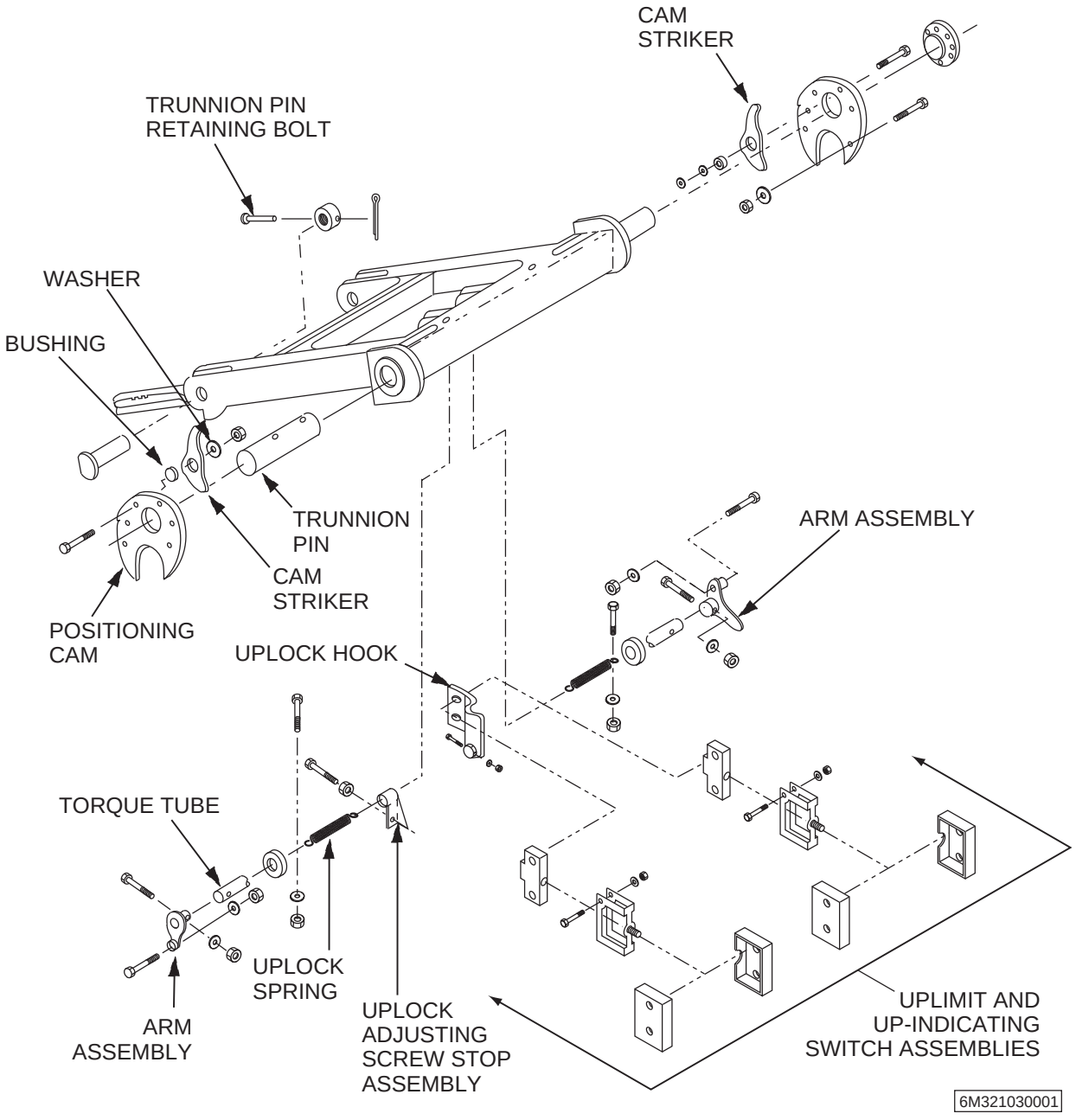
B. There are two positioning cams, one mounted at each drag brace assembly upper trunnion mount. A striker lever is installed on each positioning cam and is linked to the uplock assembly. The actuator bellcrank roller engages the striker during initial normal gear extension to rotate the uplock assembly and release the strut uplock roller. When the gear is extended the actuator bellcranks rotate with the positioning cam. The bellcrank rollers ride over a lobe and ultimately, at full extension, end up in a depression in the cam to lock the drag brace in the over-center position.

C. The uplock assembly consists of a spring loaded uplock hook mounted on a torque tube. The assembly is attached to the upper drag brace on each gear and functions as a mechanical lock in the retracted position. An uplock arm is attached to the outboard end of the uplock assembly torque tube and is engaged by the striker lever during initial normal gear extension to rotate the uplock hook and release the uplock roller on the main gear strut. An emergency release arm is attached to the inboard end of the torque tube and is engaged by the actuating arm of the emergency cable release system during initial emergency gear extension. When actuated, the release arm rotates the uplock hook to release the uplock roller on the main gear strut.

D. The upper drag brace and lower links are a matched fitted set. They can be identified by a stamped match set number where they connect at the center pivot hinge area.

E. Grease fittings are installed at each upper and lower trunnion and at each joint for periodic servicing.

MAIN GEAR DRAG BRACE – DESCRIPTION AND OPERATION



Main Gear Drag Brace
Figure 1

MAIN GEAR DRAG BRACE – MAINTENANCE PRACTICES

1. Removal/Installation – Main Gear Drag Brace

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Removal – Drag Brace (Figure 201)

NOTE: The lower and upper drag braces are a matched fitted set and are identified by a stamped match set number where they connect at center pivot hinge area.

(1) Remove inboard and outboard main gear bellcranks and actuators. (Ref 32-10-20)

NOTE: Care should be taken to note location of and retain any shims (10) (AN960-1616L washers) installed on drag brace trunnion pins during the next step.

(2) At both lower drag brace links (5) to main gear strut trunnions, remove sealant brush coat (if applied) and remove retaining bolt (3), washer and nut. Remove lower trunnion pins (4).

(3) Remove drag brace lower links (5) by removing drag brace center hinge pins (6) and associated hardware (cotter pin, washer, retaining pin and ring).

(4) Remove up-limiting and up-indicating switch assemblies from uplock hook and stow out of the way to facilitate further maintenance.

(5) Remove both drag brace trunnion pin retaining bolts (7), washers, and nuts.

NOTE: Care should be taken to note location of and retain any shims (9) (AN960-1816L washers) installed on drag brace trunnion pins during the next step.

(6) While holding upper drag brace (1), grab grease pins on trunnion pins (8) and push pins inboard.

(7) Rotate drag brace forward to remove door actuator roller from door actuator hook.

(8) Carefully slide upper drag brace down and out of wheel well.

(9) If drag brace, uplock assembly and/or positioning cam are being replaced, see instructions in this section.

B. Installation – Drag Brace (Figure 201)

(1) Position upper drag brace assembly (1) into mounting location.

(2) Install AN960-1816L washers as shims (9) between drag brace trunnion and trunnion mount collar at each side of drag brace as required. Obtain a combined total minimum clearance of 0.010 to 0.038 inch (0.254 to 0.965 mm) before final installation of trunnion mount pins.

(3) Grab grease pins on trunnion pins (8) and push trunnion pins outboard into trunnion mount collar at each side. Align holes in pins and upper drag brace (1) to install retaining bolts (7).

(4) Install trunnion pin retaining bolts (7), washers and nuts. Torque 15 to 30 inch pounds (1.69 to 3.39 N·m).

NOTE: The lower and upper drag braces are a matched fitted set and are identified by a stamped match set number where they connect at center pivot hinge area.

(5) Insert drag brace lower links (5) into upper drag brace (1) and secure with drag brace center hinge pins (6) and associated hardware (cotter pin, washer, retaining pin and ring).

(6) Attach drag brace assembly to gear strut assembly with pins (4) and AN960-1616L washers as shims (10), one (or more as required) to each side of drag brace lower links (5). Shim as required to obtain a combined total minimum clearance of 0.010 to 0.038 inch (0.254 to 0.965 mm).

(7) Align holes in pins (4) and strut (2) and install retaining bolts (3) wet with sealant in accordance with SA226/227 Structural Repair Manual 51-30-03, Section 2.F.(3)(g). Install washers and nuts and torque bolt 15-30 inch pounds.

(8) Brush coat the head and nut end of retaining bolts (3) in accordance with SA226/227 Structural Repair Manual 51-30-03, Section 2.F.(3)(g).

(9) Install up-limiting and up-indicating switch assemblies to the drag brace uplock hook and install all wiring clamps and connections.

(10) Install inboard and outboard main gear bellcranks and actuators and downlock springs. (Ref 32-10-20)

(11) Perform Adjustment/Test - Landing Gear Extension and Retraction (Refer to 32-30-00, Page 201).

C. Removal – Uplock Assembly (Figure 202)

(1) Remove drag brace assembly following instructions in Step 1.A. in this section.

(2) Remove retaining bolts (1) securing arm assemblies (items 2 and 3) mounted at each end of uplock assembly torque tube.

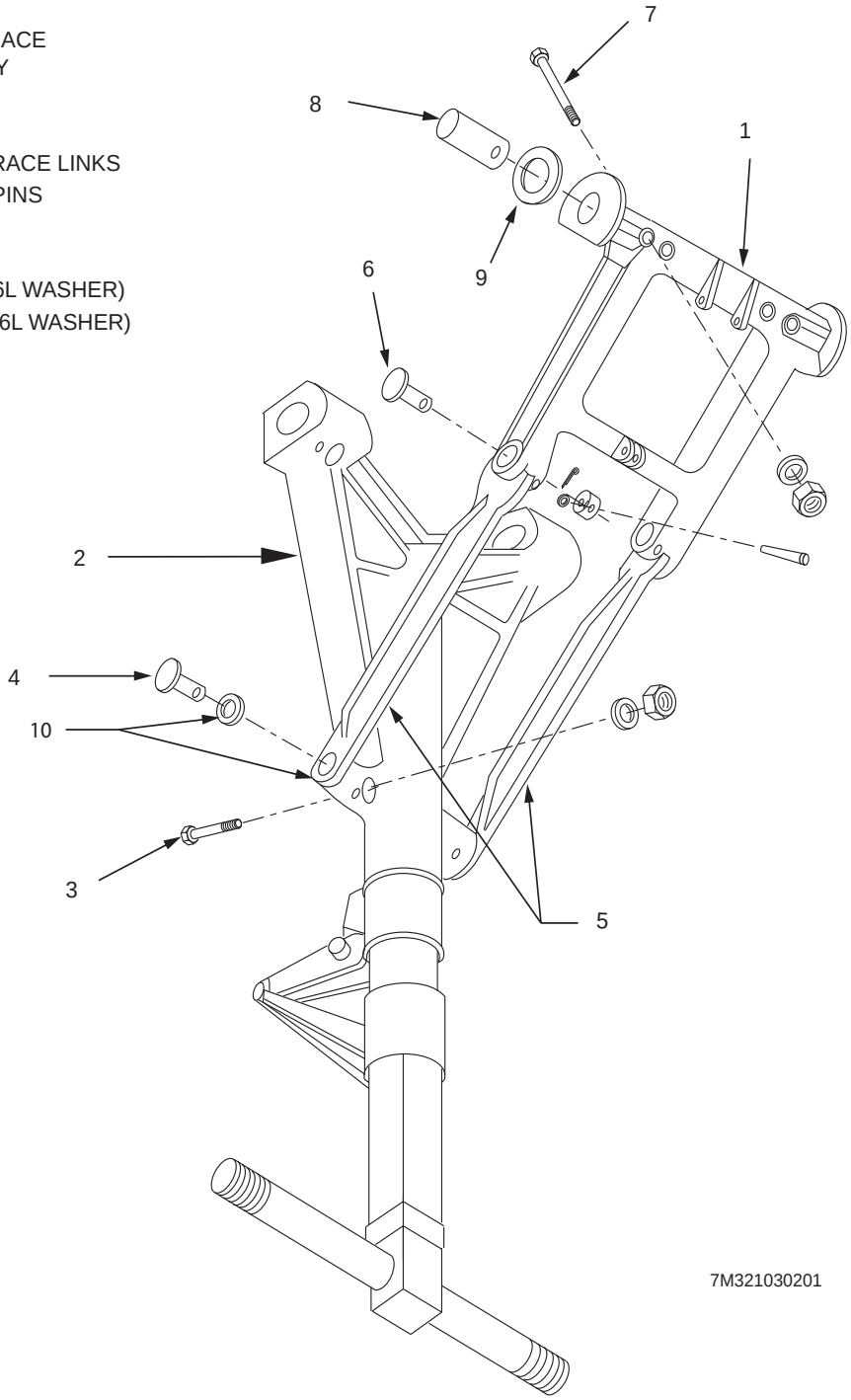
(3) Remove retaining bolt (4) securing uplock adjust screw stop assembly (5) and uplock spring (6).

(4) Remove retaining bolt (10) securing the other uplock spring (6).

(5) Remove retaining bolts (7) securing hook assembly (8).

(6) With all retaining bolts removed, torque tube (9) and all attached parts may be removed from drag brace by sliding tube from drag brace assembly.

1. UPPER DRAG BRACE
2. MLG STRUT ASSY
3. RETAINING BOLT
4. TRUNNION PIN
5. LOWER DRAG BRACE LINKS
6. CENTER HINGE PINS
7. RETAINING BOLT
8. TRUNNION PIN
9. SHM (AN960-1816L WASHER)
10. SHIM (AN960-1616L WASHER)



Drag Brace
Figure 201

D. Installation – Uplock Assembly (Figure 202)

NOTE: The uplock springs must be twisted around the torque tube two times upon final installation of uplock assembly to the upper drag brace assembly.

NOTE: Tighten all springs in direction required to apply tension of arm assemblies on cam strikers.

(1) Slide torque tube (9) through mount bosses on drag brace, uplock hook assembly (8), uplock adjust screw stop assembly (5) and springs (6) (see Illustrated Parts Catalog 32-10-30 for part numbers).

(2) Install uplock arm (2) on outboard end and emergency release arm (3) on inboard end of torque tube (9).

(3) If uplock assembly is not new and is being reinstalled:

(a) Install the attaching hardware for items installed onto torque tube with hardware retained during removal procedure (see Illustrated Parts Catalog for part numbers for each item) turning nuts until snug without final torque applied.

(b) Verify the hook assembly (8) and arm assemblies (2 and 3) do not have rotational play due to elongated bolt holes in these assemblies or in the torque tube. If play is evident, corrective action must be taken prior to further flight.

(c) Tighten all attaching hardware on uplock assembly to final torque.

(d) Tighten all hardware to final torque.

(4) If uplock assembly is new:

(a) Loose install attaching hardware for items installed onto torque tube (see Illustrated Parts Catalog for part numbers for each item).

(5) Install main gear drag brace following instructions in Step 1.B. in this section. If uplock assembly is new, down lock springs should not be installed at this time and both actuators should be left disconnected at the MLG strut.

(6) For aircraft with new uplock assembly, the uplock assembly should have pilot holes drilled in the uplock hook (8), stop assembly (5), arms (2 and 3) and torque tube (9). The following steps establish the position of each of these items so bolt holes for each can be properly located:

(a) With clearance between the uplock roller and uplock hook (8) at 0.000 to 0.040 inch, drill two holes through two pilot holes provided on new uplock hook. Drill through hook and torque tube with a number 13 drill bit.

(b) Install uplock hook retaining bolts (7), washers, and nuts. Torque 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(c) With uplock hook stop bolt at mid travel in nutplate on stop assembly (5), position stop assembly so there is zero clearance between back of uplock hook (8) and uplock roller with no load on hook. Verify the uplock hook stop bolt is in contact with drag brace when zero clearance is obtained between hook and uplock roller.

(d) Drill a 0.190 to 0.194 inch (4.826 to 4.927 mm) diameter hole through stop assembly (5) and torque tube (9) for retaining bolt (4). Reserve final installation of retaining bolt for installation of uplock spring (6) (also secured to bolt).

(e) Drill a 0.190 to 0.194 inch (4.826 to 4.927 mm) diameter hole through pilot hole provided in torque tube (9) on the opposite side of the stop assembly (5) for the uplock spring retaining bolt (10).

(f) With bearings on arm assemblies (items 2 and 3) engaged securely against cam strikers, drill through predrilled holes in the arm assemblies and through torque tube (9) with a number 40 drill bit. Complete drilling with a size D drill bit and install retaining bolts (1), washers and nuts in both arm assemblies.

(g) Secure stop assembly (5) and left downlock spring (6) as follows:

1 Wrap a loop of 0.032 inch safety wire around torque tube (9) twice in same direction spring needs to be turned to apply tension of arm assemblies on cam strikers.

2 Ensure boss end of downlock spring is wrapped over mount boss on drag brace.

3 Attach safety wire loop end to opposite end of spring and pull safety wire to twist spring around torque tube twice.

4 Insert bolt (4) through spring (6) end and stop assembly (5).

5 Secure bolt (4) with washer and nut and tighten to final torque.

(h) Secure right downlock spring (6) as follows:

1 Wrap a loop of 0.032 inch safety wire around torque tube (9) twice in same direction spring needs to be turned to apply tension of arm assemblies on cam strikers.

2 Ensure boss end of downlock spring is wrapped over mount boss on drag brace.

3 Attach safety wire loop end to opposite end of spring and pull safety wire to twist spring around torque tube twice.

4 Insert retaining bolt (10) through opposite end of spring and torque tube (9)

5 Secure bolt (10) with washer and nut and tighten to final torque.

(7) Perform Adjustment/Test - Extension and Retraction. (Refer to 32-30-00, page 201)

E. Removal – Positioning Cam (Figures 202)

(1) Remove drag brace assembly per instructions in this section.

(2) Note orientation of positioning cam to drag brace and identify as inboard or outboard position.

(3) Remove retaining hardware securing positioning cam (12) to drag brace assembly.

(4) If positioning cam is being replaced, remove retaining bolt securing striker (11) to cam and remove striker.

F. Installation – Positioning Cam (Figures 202)

(1) If positioning cam is being reinstalled:

(a) Place cam on drag brace at same orientation and inboard or outboard position as removed.

(b) Install cam retaining bolts, washers and nuts.

(2) If positioning cam is being replaced with new cam:

(a) Position new cam (12) between drag brace and nacelle trunnion mount and install drag brace assembly per instructions in this section.

(b) With drag brace and actuator bellcrank installed, hold the drag brace knee in the over-centered position (push up on center of drag brace).

(c) Rotate cam to obtain 0.002 to 0.015 inch (0.051 to 0.381 mm) clearance (gap) between the bellcrank roller and the positioning cam with gear down.

CAUTION: EXERCISE EXTREME CARE WHEN ENLARGING PILOT HOLES TO PREVENT ELONGATING DRAG BRACE HOLES.

NOTE: During the next step, use a drill bushing of the correct size to properly locate center of pilot holes and prevent undesired reaming or elongation of drag brace holes.

(d) Locate and drill two #40 (0.0980 inch [2.489 mm]) pilot holes through cam.

(e) Open pilot holes to 0.1865 to 0.1875 inch (4.7371 to 4.7625 mm) and install two retaining bolts, washers and nuts to secure cam to drag brace.

(f) Remove drag brace and cam as a single unit.

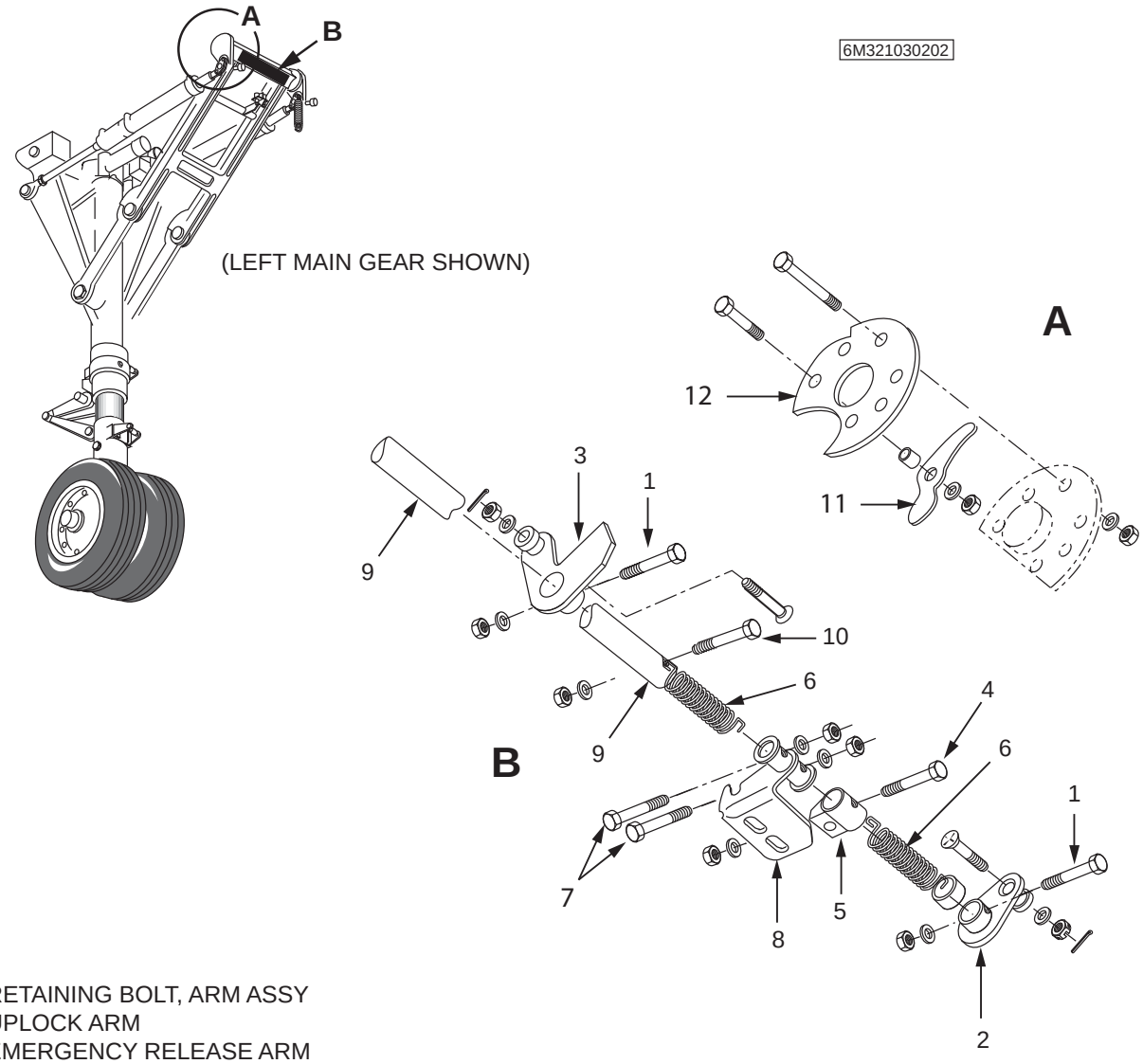
(g) Locate and drill the remaining bolt holes in cam.

(h) Install cam retaining bolts, washers and nuts.

(i) Install striker (11) to cam with bolt, bushing, washer and nut.

(3) Install drag brace per instructions in this section.

(4) Perform Adjustment/Test - Landing Gear Extension and Retraction for the main landing gear. (Refer to 32-30-00, page 201).



1. RETAINING BOLT, ARM ASSY
2. UPLOCK ARM
3. EMERGENCY RELEASE ARM
4. RETAINING BOLT, STOP ASSY
5. ADJUST SCREW STOP ASSEMBLY
6. UPLOCK SPRINGS
7. RETAINING BOLT/TAPER PIN, UPLOCK HOOK
8. UPLOCK HOOK ASSY
9. TORQUE TUBE
10. RETAINING BOLT, SPRING
11. STRIKER
12. POSITIONING CAM

Uplock Assembly
Figure 202

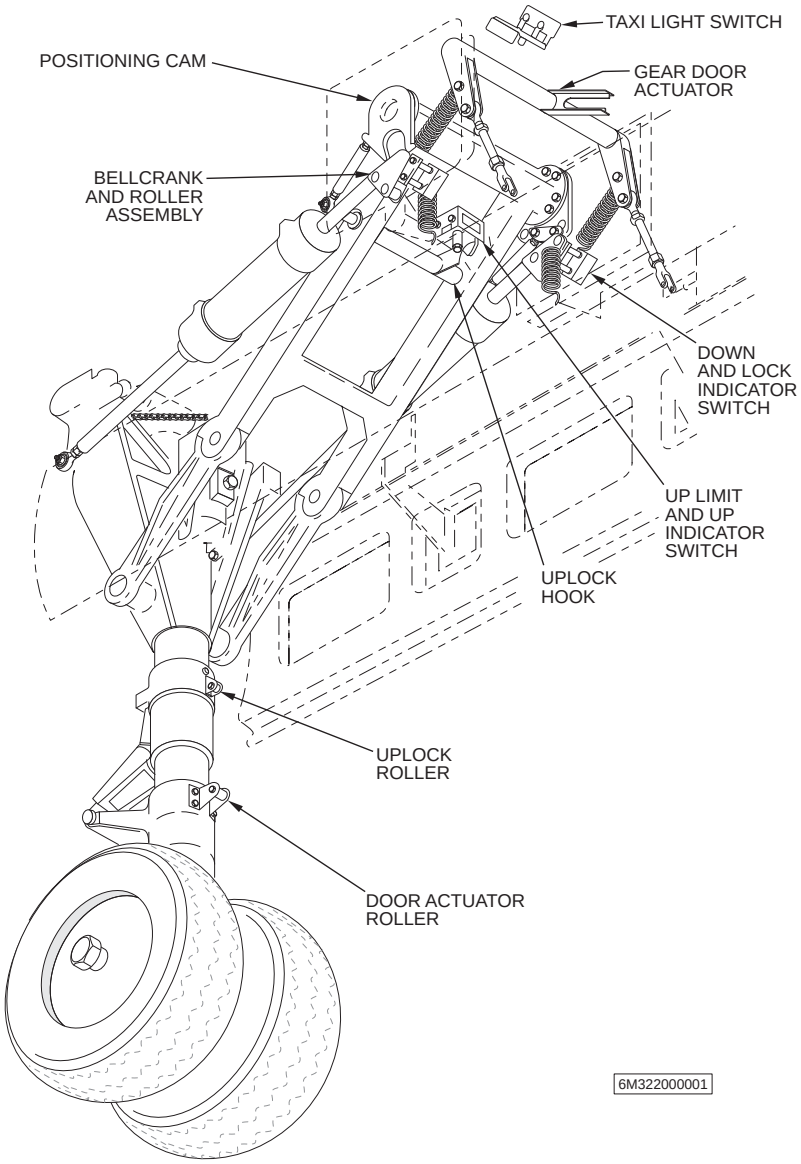
NOSE LANDING GEAR – DESCRIPTION AND OPERATION

1. General

A. The nose landing gear is mounted immediately forward of the forward pressure bulkhead and is retracted forward into the wheel well. Landing gear extension and retraction is electrically controlled and hydraulically actuated.

B. The nose landing gear consists of an air-oil telescoping shock strut, a folding drag brace assembly attached to the aircraft structure and shock strut, two identical double acting hydraulic actuators, and dual wheels.

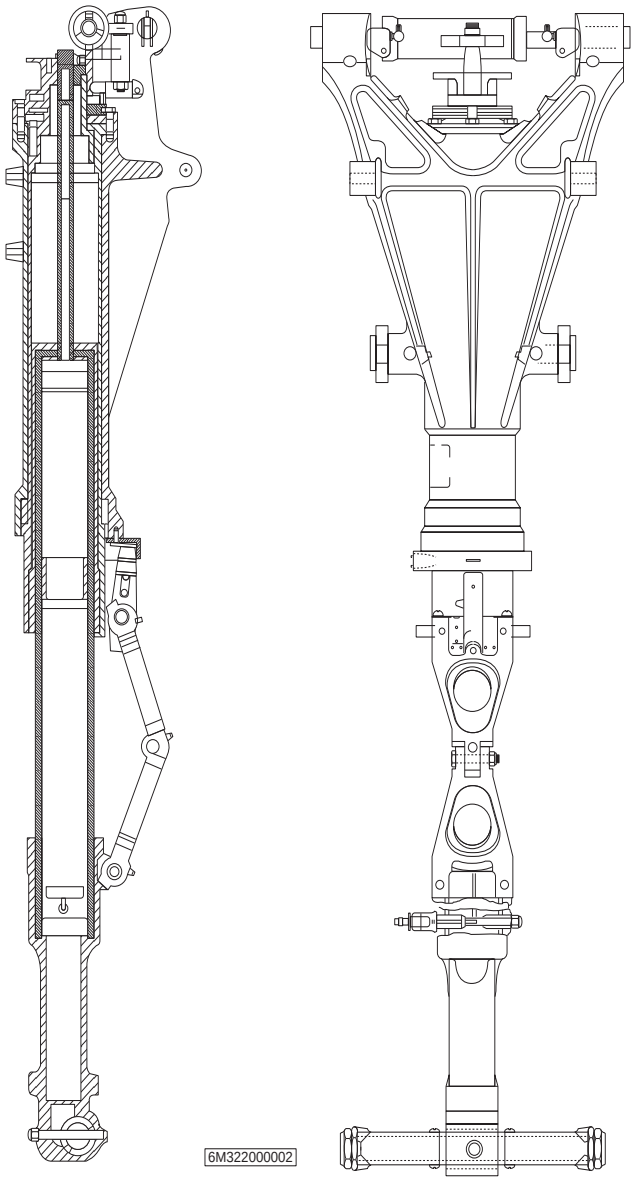
C. The gear incorporates a taxi light and either an electric nose steering actuator installed on earlier model aircraft or a hydraulic steering actuator installed on later models.



Nose Landing Gear
Figure 1

D. A shimmy damper unit is installed on earlier model aircraft using the electric nose steering actuator. The nose steering actuator and shimmy damper are mounted atop the nose strut. The piston rod of the shimmy damper is attached to the stationary upper section of the strut. The cylinder of the shimmy damper is attached to the feedback pulley of the electric nose steering actuator.

E. A nose wheel centering device is installed on the nose landing gear. The device consists of a fixed cam attached to the stationary upper section of the strut and a follower arm and roller attached to the scissors between the two sections of the strut. A spring-loaded uplock hook and mounting rod is attached to the upper portion of the drag brace and functions as the mechanical uplock when the gear is fully retracted.



Nose Gear
Figure 2

F. An uplock roller is located on the gear which engages the spring-loaded uplock hook to mechanically lock the gear in the retracted position. Two positioning cams are mounted on each side of the drag brace to insure locking the drag brace overcenter when the gear is extended.

G. A spring assisted bellcrank and roller assembly is attached to the cylinder end of each gear actuator to assist the positioning cams in their operation. The left and right nose landing gear doors are spring-loaded open when the gear is extended, and mechanically closed by a gear door actuator during retraction.

H. The nose landing gear incorporates two down position switches installed at each side attached to the wheel well structure next to the bellcrank hardstop, one up position switch mounted adjacent to the mechanical uplock hook, one up limit switch mounted next to the mechanical uplock hook along side of the up position switch, and a taxi light on-off position switch located on the right keelson in the nose wheel well.

I. Nose Landing Gear Strut

(1) The nose landing gear strut is identical to the main landing gear strut, except for the addition of a metering pin at the orifice which creates a variable orifice. The size of the orifice and the restriction of fluid varies with the amount of compression and extension of the strut. The metering pin is not included on some early SA226-AT and SA226-TC airplanes. A shimmy damper, taxi light, nose wheel steering actuator, and a nose wheel centering device are installed on the nose gear strut.

J. Nose Wheel Centering Device

(1) The nose wheel centering device consists of a fixed cam attached to, the stationary upper section of the strut and a follower arm, with roller attached to the scissors between the two sections of the strut. As the weight of the airplane is removed from the nose gear, the weight of the gear plus the force of the nitrogen pressure causes the strut to extend. These extension forces are also transmitted through the scissors to the follower arm and roller assembly which tracks to the center of the cam and moves the steerable portion of the nose gear to the centered position.

K. Door Actuator Roller

(1) The nose landing gear doors are spring loaded overcenter to the open position when gear is extended. The door actuator hook is mechanically linked to the nose gear doors. A roller mounted on the nose gear yoke assembly engages the hook when the gear is retracted to close the doors during retraction of the gear.

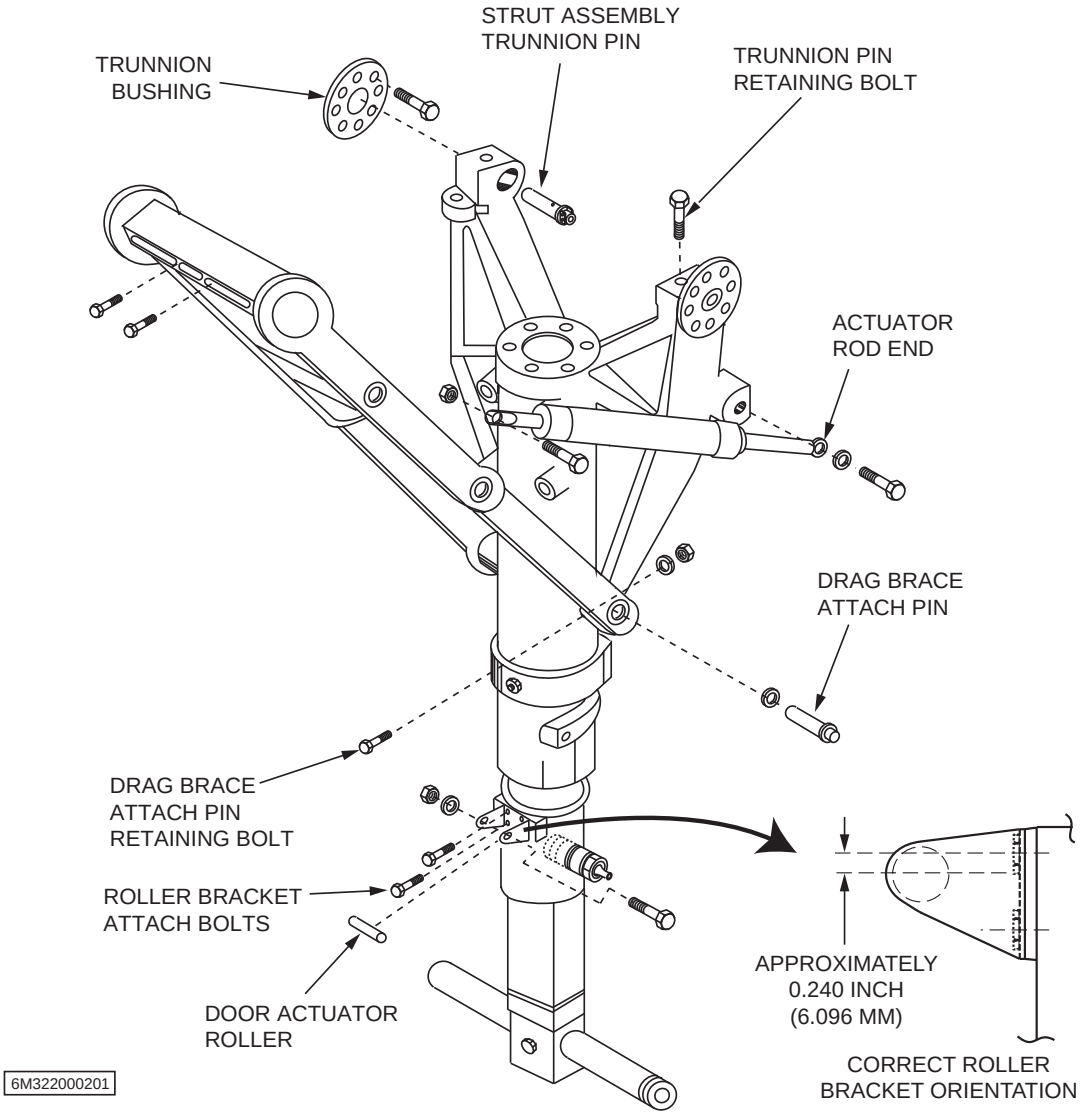
1. Removal/Installation – Nose Landing Gear

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Removal – Nose Gear Strut (Figure 201)

(1) Jack aircraft (Refer to 07-10-00).

(2) On Pre SB 226-32-016 aircraft, disconnect and remove nose gear steering cable.



Nose Gear
Figure 201

(3) On Post SB 226-32-016 aircraft:

(a) Bleed off hydraulic reservoir head pressure and bleed off hydraulic system pressure IAW procedures in Section 12-10-20.

(b) Disconnect and cap hydraulic lines and fittings to nose gear steering actuator.

(4) Disconnect nose gear steering and taxi light electrical connectors.

(5) Remove both actuator rod end nuts, washers, and bolts from gear assembly attach points. Free both actuators from gear strut assembly.

CAUTION: IF SERVICE BULLETIN 226-32-057 HAS BEEN INCORPORATED, STOW AND SUPPORT DRAG BRACE ASSEMBLY TO THE NORMAL AFT POSITION TOWARDS THE STRUT ASSY TO PREVENT DAMAGE TO THE NLG GEAR UP-LIMIT AND UP-INDICATING SWITCH ACTUATOR.

(6) Remove drag brace attach pin retaining bolts, washers, and nuts to remove drag brace attach pins. Remove attach pins and washers and free drag brace assembly from gear assembly.

(7) Remove strut assembly trunnion pin retaining bolts, washers, and nuts.

NOTE: Note location and amount of shims installed between trunnion assembly and trunnion bushing before removal to facilitate assembly.

(8) Remove trunnion pins by sliding them inboard and carefully lower shock strut assembly.

B. Installation – Nose gear Strut (Figure 201)

(1) Position shock strut assembly into mounting location.

(2) Install shims (AN960-1616L) between gear trunnion and trunnion bushings at each side, as required. Obtain a combined total minimum clearance of 0.010 inch (0.254 mm) and maximum clearance of 0.038 inch (0.965 mm) before final installation of trunnion mount pins.

(3) Install trunnion mount pins by sliding them outboard into trunnion mount bushings at each side. Align pins for installation of retaining bolts.

(4) Install trunnion pin retaining bolts, washers, and nuts. Torque 15 to 30 inch-pounds (1.69 to 3.39 N·m).

(5) Install drag brace assembly to gear strut assembly and install shims (AN960-1416L) each side, as required to obtain a combined total minimum clearance or 0.010 inch (0.254 mm) and maximum clearance of 0.038 inch (0.965 mm). Install drag brace attach pins and align bolt holes in pins to install retaining bolts.

(6) Install drag strut attach pin retaining bolts, washers, and nuts. Torque 15 to 30 inch-pounds (1.69 to 3.39 N·m).

(7) Install actuator rod ends to gear assembly and install bolts, washers, and nuts. Secure with cotter pins.

(8) On Pre SB 226-32-016 aircraft, install and connect nose gear steering cable.

(9) On Post SB 226-32-016 aircraft, install hydraulic lines to nose gear steering actuator.

(10) Install nose gear steering and taxi light electrical connectors.

(11) Lubricate nose landing gear (Refer to 12-10-30).

(12) If necessary, service nose gear with hydraulic fluid and nitrogen (Refer to 12-10-30).

(13) Perform nose landing gear rigging IAW Section 32-30-00.

C. Removal – Nitrogen Service Valve (Figure 202)

(1) Jack aircraft (Refer to 07-10-00).

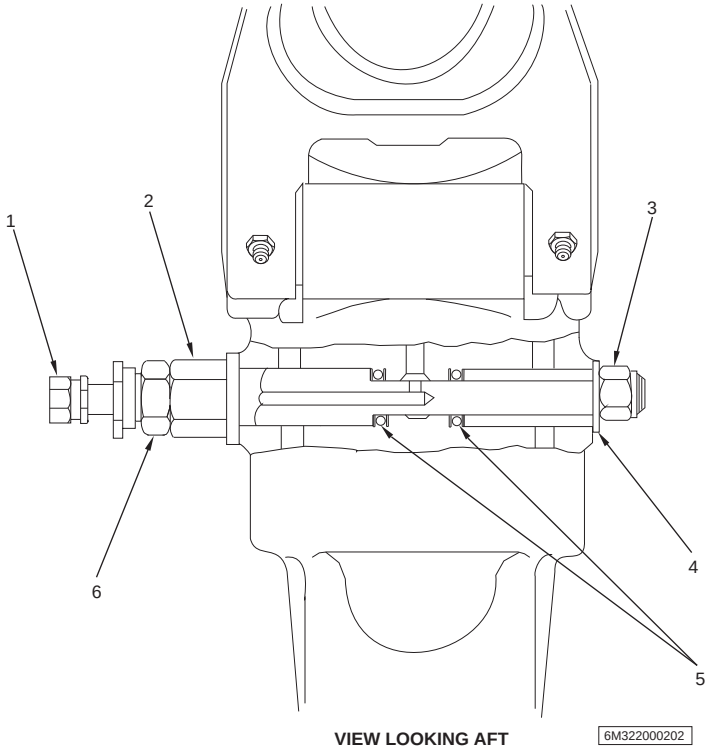
(2) Discharge all nitrogen (air pressure) by removing cap and opening nitrogen service valve (item 1).

(3) Remove valve body (item 6) from bolt (item 2).

(4) Remove lock nut (item 3) and bushing (item 4).

(5) Remove bolt (item 2) from strut.

(6) Replace seals (item 5) prior to assembly.



Nitrogen Service Valve
Figure 202

D. Installation – Nitrogen Service Valve (Figure 202)

(1) Install bolt (item 2) with new seals into strut.

(2) Slide bushing (item 4) with new seals onto bolt (item 2) and into strut.

(3) Secure bolt (item 2) with lock nut (item 3). Torque lock nut 220 to 300 inch-pounds (24.86 to 33.90 N·m).

(4) Install valve body (item 6) into bolt (item 2). Torque valve body 100 to 125 inch-pounds (11.30 to 14.12 N·m).

(5) Down jack aircraft IAW Section 07-10-00.

(6) Service strut with nitrogen IAW Section 12-10-30 and install cap on service valve (1).

E. Wear Limit/Removal – Nose Gear Axle (Figure 203)

(1) Wear – Axle

(a) Manufacturing tolerance for the outside diameter of the nose landing gear axle is 1.4990 to 1.4995 inches (38.0746 to 38.0863 mm). If axle diameter is less than 1.4990 inches (38.0746 mm), replace axle.

(2) Tools and equipment required to remove axle are as follows:

(a) One (1) 0.75 inch (19.05 mm) threaded steel rod 24.5 inches (62.23 cm) long.

(b) Two (2) 0.75 inch (19.05 mm) nuts with threads matching steel rod.

(c) Two (2) 0.25 inch (6.35 mm) thick steel plates (washers) with 0.75 inch (19.05 mm) holes.

(d) One (1) steel sleeve, 11.5 inches (29.21 cm) long X 1.80 inches (4.57 cm) minimum inside diameter (may be fabricated from length of steel tubing).

(e) Dry ice or CO_2.

(f) Protective washer, 3.0 inches (7.62 cm) O.D X 1.80 inches (4.57 cm) I.D. Make from 0.25 inch (6.35 mm) phenolic or Teflon sheet.

(3) Jack aircraft (Refer to 07-10-00).

(4) Remove nose gear wheels (Refer to 32-42).

(5) Remove nut, washer, and axle locking bolt securing axle to strut.

(6) Install threaded steel rod through axle with one steel plate (or washer) and one nut fully threaded onto one end.

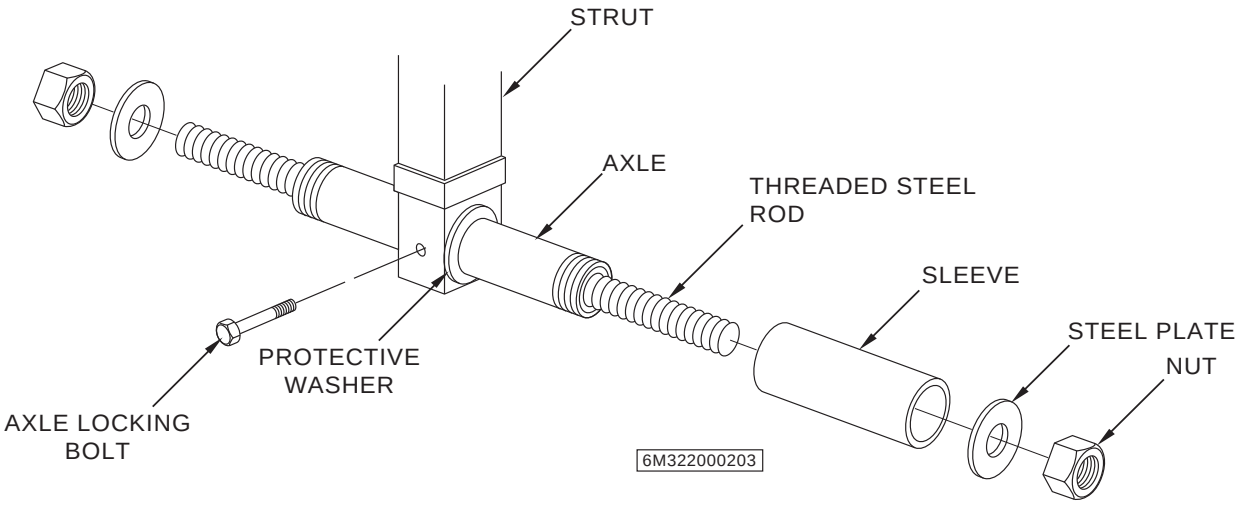
(7) Install protective washer over opposite end of axle against strut yoke.

WARNING: WEAR PROTECTIVE GLOVES WHEN HANDLING FROZEN PARTS OR HANDLE WITH SUITABLE TOOLS.

(8) Chill axle (from inside) with dry ice or CO_2.

(9) With axle chilled, position sleeve onto axle against the protective washer, then install one steel plate (or washer) and one nut on opposite end of steel rod.

(10) Tighten nut on sleeve end of steel rod until axle is clear of strut yoke. Remove axle.



Nose Gear Axle Removal and Installation
Figure 203

F. Installation – Nose Gear Axle (Figure 203)

WARNING: WEAR PROTECTIVE GLOVES WHEN HANDLING FROZEN PARTS OR HANDLE WITH SUITABLE TOOLS.

(1) Chill axle with dry ice or CO_2. Install axle into strut with axle locking bolt hole aligned with hole in strut yoke.

(2) Secure axle to yoke with axle locking bolt, washer, and nut.

(3) Install nose gear wheels (Refer to 32-42).

(4) Down jack aircraft and remove jacks (Refer to 07-10-00).

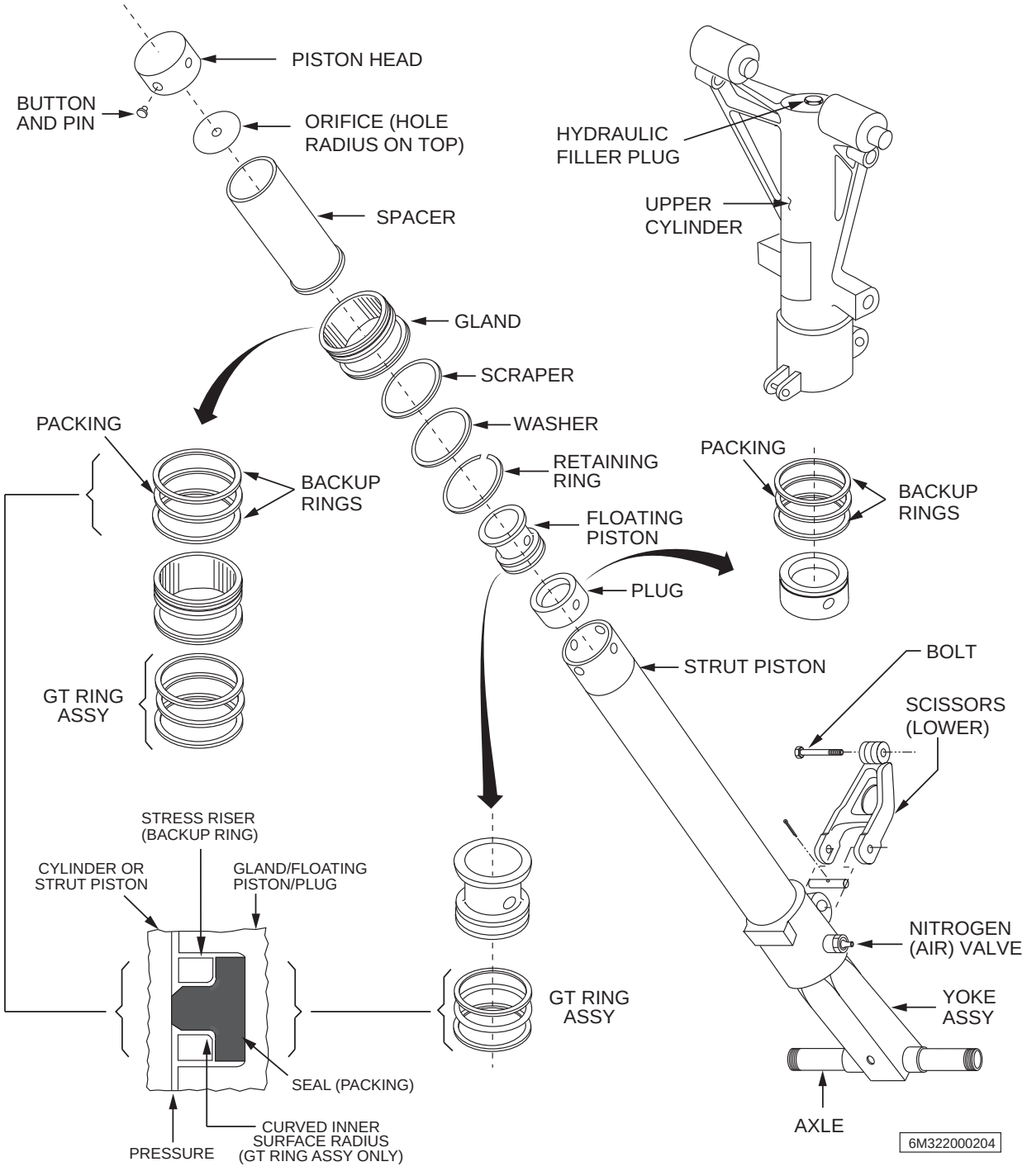
2. Disassembly/Assembly – Nose Landing Gear

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Disassembly – Nose Landing Gear Strut (Lower) (Figure 204)

(1) Jack aircraft (Refer to 07-10-00).

(2) Remove nose gear strut IAW instructions in this section.



Nose Landing Gear Lower Strut Assembly
Figure 204

NOTE: Accomplish disassembly/assembly of the Nose Landing Gear Strut with the strut removed from the airplane if practical. When absolutely necessary, disassembly/assembly may be accomplished with strut installed.

(3) If disassembling strut with gear installed in aircraft:

(a) Connect external electrical power to aircraft (Refer to 24-40-00).

(b) Connect external hydraulic power cart to aircraft (Refer to 29-00-00).

(c) Retract nose gear halfway (approximately 45°). Bleed external hydraulic power cart supply pressure to zero (0) and pull landing gear control circuit breaker.

(4) Place a drip pan beneath gear being disassembled.

(5) Discharge nitrogen (air pressure) by opening nitrogen air valve.

(6) Remove hydraulic filler plug. Install adapter with hose to filler plug hole and route free end of hose to a catch can or bucket.

(7) Close nitrogen valve and slowly charge strut with nitrogen, forcing hydraulic fluid out of cylinder.

(8) Discharge residual hydraulic fluid by compressing strut.

(9) Discharge nitrogen (air pressure) by opening air valve.

(10) Remove cotter pin, nut, washer and pivot bolt securing torque links (scissors) and separate scissors.

(11) Remove retaining ring and washer from cylinder.

(12) Pull strut piston out of cylinder.

(13) Remove buttons and pins securing piston head to top of strut piston. Making sure to capture orifice from underneath piston head, remove piston head and orifice.

(14) Remove spacer from strut piston.

(15) Remove gland and scraper from strut piston.

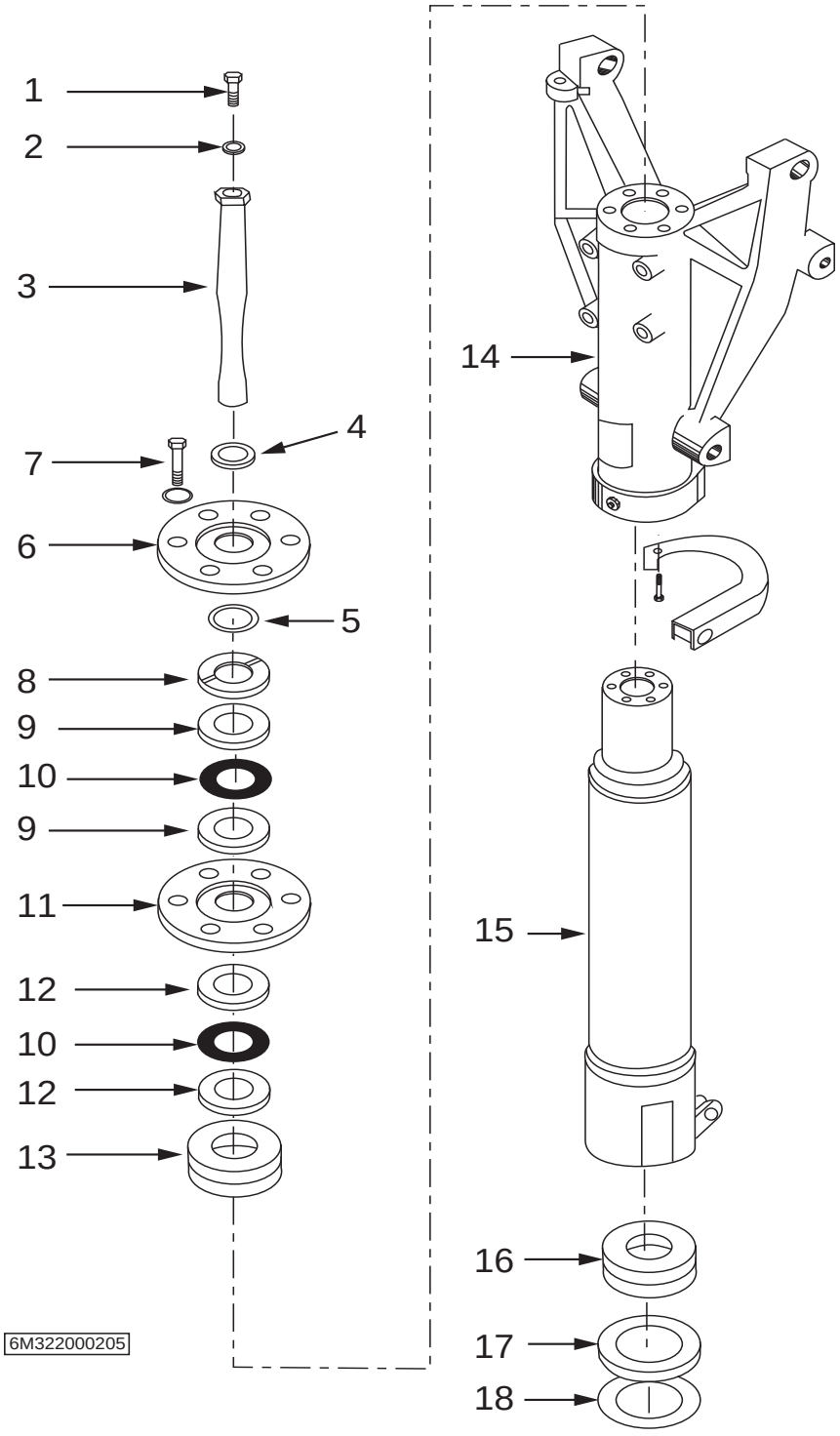
WARNING: IT MAY BE NECESSARY TO APPLY AIR PRESSURE TO AIR VALVE TO FORCE FLOATING PISTON FROM STRUT PISTON. DO NOT USE EXCESSIVE PRESSURE WHILE ATTEMPTING TO REMOVE FLOATING PISTON. IT CAN ACT AS A PROJECTILE, POSSIBLY RESULTING IN PERSONNEL INJURY.

(16) Remove floating piston from inside of piston chamber.

B. Disassembly – Nose Landing Gear Upper Strut (Figure 205)

(1) Perform **2.** A. Disassembly – Nose Landing Gear Lower Strut.

(2) Remove metering pin (item 3) by unscrewing it from assembly and discard packing (item 4).



Nose Landing Gear Upper Strut Assembly
Figure 205

(3) Remove retaining bolts (item 7) and washers, then remove upper flange (item 6). Discard packing (item 5).

(4) Remove split ring (item 8) securing inner cylinder assembly (item 15) to outer cylinder assembly (item 14).

(5) Remove thrust bearing (item 10) and bearing races (item 9) from top of lower flange (item 11).

(6) Remove lower flange.

(7) Remove thrust bearing (item 10) and bearing races (item 12) from top of outer cylinder assembly.

(8) Remove inner cylinder assembly (item 15) from outer cylinder assembly (item 14).

(9) Remove packing gland (item 17) and packing (item 18) from bottom of outer cylinder assembly and discard.

(10) Remove upper bearing (item 13) and lower bearing (item 16) from outer cylinder assembly.

C. Re-sealing and Assembly – Nose Landing Gear Upper Strut Assembly (Figure 205)

(1) Replace O-ring (item 5) on upper flange (item 6).

(2) Replace packing gland (item 17) and packing (item 18) at bottom of outer cylinder assembly.

(3) Replace upper and lower needle bearings (items 13 and 16) and thrust race assemblies (items 9 and 12) as necessary.

(4) Install upper and lower bearings (items 13 and 16) into outer cylinder assembly (item 14).

(5) Install inner cylinder assembly (item 15) into outer cylinder assembly.

(6) Install thrust race assembly (thick races, item 12) to top of outer cylinder.

(7) Install lower flange (item 11) to top of outer cylinder assembly.

(8) Install thrust race assembly (thin races, item 9) to top of lower flange.

(9) Install split ring (item 8) securing inner cylinder to outer cylinder.

(10) Install upper flange (item 6) and secure to outer cylinder assembly (item 14) with bolts (item 7) and washers. Torque retaining bolts 220 to 300 inch-pounds (24.86 to 33.90 N·m).

(11) Replace O-ring (item 4) on metering pin (item 3) and install metering pin into upper strut assembly. Torque 400 to 450 inch-pounds (45.19 to 50.84 N·m).

(12) Replace O-ring (item 2) on filler plug (item 1) and install filler plug.

(13) Perform step **2.** D. or proceed with step **2.** E. as necessary.

D. Re-packing and Re-sealing - Nose Landing Gear Lower Strut (Figure 204)

CAUTION: DO NOT ALLOW RUBBER SEALS TO CONTACT DEGREASING SOLVENT. SOLVENT MAY CAUSE PREMATURE DETERIORATION OF SEALS.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK DOES CONSTITUTE A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(1) Rinse all parts in degreasing solvent (MPK or equivalent).

(2) Check floating piston head orifice and all disassembled parts for contamination, damage, or obstruction. Replace if necessary.

(3) Before assembly, soak all seals in hydraulic fluid (MIL-PRF-5606).

CAUTION: O-RING ASSEMBLIES ARE GT RING ASSEMBLIES THAT HAVE A CURVED INNER SURFACE. INSTALL STRESS RISER SO CURVED PART OF STRESS RISER FITS INTO CURVED PART OF SEAL.

(4) Replace packing, backup rings and GT ring assembly on gland.

(5) Replace GT ring assembly on floating piston.

(6) Replace scraper inside gland beveled side down.

(7) Replace nitrogen (air) valve packings.

(8) Replace four buttons in piston head.

(9) Inspect strut piston cylinder and upper cylinder bore for scoring. Superficial scratches and nicks can be removed with #600 emery cloth.

(10) Lubricate all disassembled parts, piston chamber, piston surfaces and inner diameter of cylinder assembly with hydraulic fluid (MIL-PRF-5606) before assembly.

E. Assembly - Nose Landing Gear Lower Strut (Figure 204)

(1) Install floating piston into strut piston.

(2) Install retaining ring and washer over strut piston.

(3) Install gland over piston with scraper at bottom.

(4) Install orifice on strut piston with hole radius on top.

(5) Install piston head and orifice onto strut piston and secure with buttons and pins.

(6) Slide strut piston assembly into upper cylinder.

(7) Secure strut piston assembly to cylinder with washer and retaining ring in grooves in bottom of cylinder bore.

(8) Align upper and lower scissors and install pivot bolt, washer and nut. Bottom out nut and back off to nearest cotter pin slot. Maximum torque to scissors pivot attachment is 5 inch-pounds (0.56 N·m). Install cotter pin.

(9) Install nitrogen (air) service valve (Refer to 32-00-00).

(10) Service nose gear strut with hydraulic fluid and nitrogen (Refer to 12-10-30).

(11) Leak check nose gear assembly:

NOTE: Strut may be horizontal or vertical for this test.

(a) Wipe down entire strut assembly with a clean cloth.

(b) Service strut to 350 psi (24.61 kg/cm^2) with nitrogen.

(c) After one (1) hour, verify nitrogen pressure is still 350 psi (24.61 kg/cm^2).

(d) Verify no hydraulic fluid is leaking from between cylinder and strut piston or from filler plug.

(e) Slowly open nitrogen service valve to bleed off pressure and verify strut shows no signs of hydraulic fluid leakage from valve.

(f) Service nose gear strut with nitrogen (Refer to 12-10-30).

(g) Close nitrogen service valve.

(12) Check nose gear strut assembly trunnion bushing and mount pin for wear (Figure 201).

(a) If outside diameter of trunnion pin is less than 0.995 inch (25.2730 mm), replace pin.

(b) If inside diameter of trunnion bearing exceeds 1.0010 inch (25.6540 mm), replace bearing.

(13) Install nose gear strut assembly IAW instructions in this section.

(14) Disconnect external hydraulic power cart from aircraft (Refer to 29-00-00).

(15) Disconnect external electrical power from aircraft (Refer to 24-40-00).

F. Removal – Door Actuator Roller/Brackets (Figure 201)

(1) Remove cotter pin, nut and bolt securing roller to brackets and remove roller.

(2) Remove the two bolts securing each door actuator roller bracket to yoke. Retain shims if installed and retain for installation.

(3) Remove brackets from yoke.

G. Installation – Door Actuator Roller/Brackets (Figure 201)

NOTE: Refer to Figure 201 for proper bracket orientation.

(1) Position door actuator roller brackets in place on yoke with any removed shims.

(2) Install two bolts to secure each bracket to yoke.

(3) Install nut and bolt securing roller to brackets. Install cotter pin.

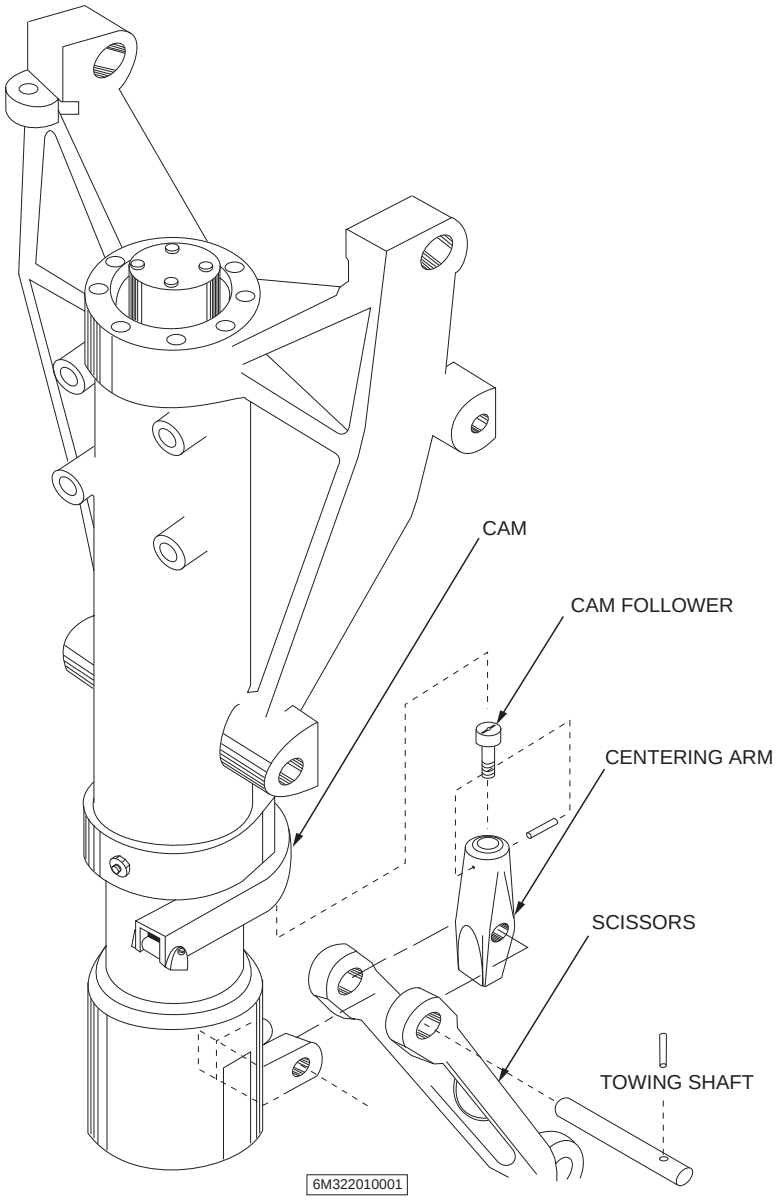
(4) Verify nose landing gear door rigging IAW Section 32-30-00.

NOSE GEAR CENTERING ASSEMBLY – DESCRIPTION AND OPERATION

1. General

A. The nose wheel centering device consists of a fixed cam attached to the stationary upper section of the strut and follower arm, with a roller, attached to the scissors between two (2) sections of the strut.

B. When the weight of the airplane is removed from the nose gear, the weight of the gear plus force of nitrogen pressure causes the strut to extend. These extension forces are also transmitted through the scissors to the follower arm and roller assembly tracking to the center of the cam, and moves the steerable portion of the nose gear to the centered position.



Nose Gear Centering Assembly
Figure 1

1. Removal/Installation – Nose Gear Centering Assembly

A. Removal – Nose Wheel Centering Assembly

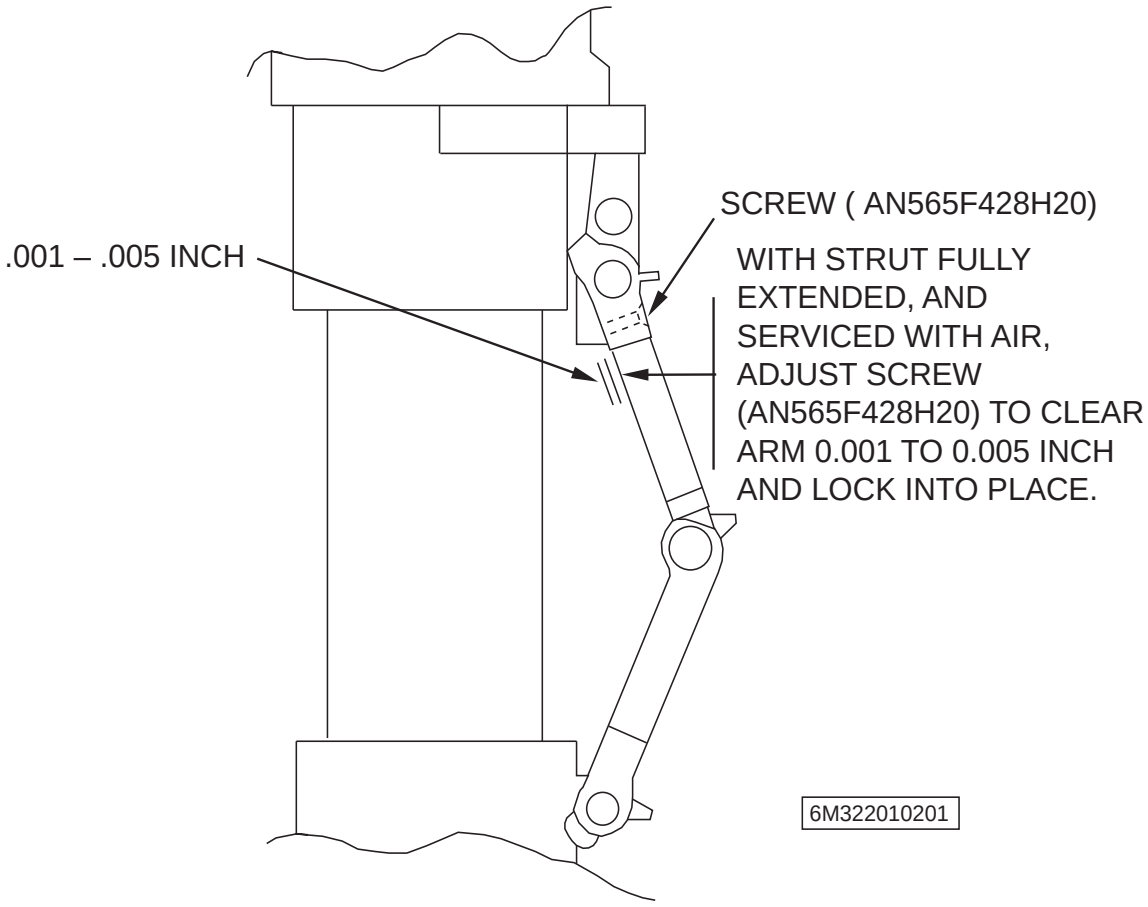
(1) Disconnect scissors assembly.

(2) Remove lockwire from spring pin securing towing shaft through upper scissors assembly. Remove spring pin.

(3) Remove towing shaft securing centering arm and upper scissors assembly. Remove both assemblies.

(4) To remove cam follower from centering arm assembly, remove cam follower spring pin and remove follower.

(5) Remove bolts and lock washers securing cam to strut cylinder. Remove cam.



Nose Gear Scissor Assembly
Figure 201

B. Installation – Nose Wheel Centering Assembly

(1) Install cam to strut assembly and secure with three cam mounting bolts.

(2) Torque cam mounting bolts 50 to 70 inch-pounds (5.65 to 7.91 N·m).

(3) Install cam follower onto centering arm assembly. Secure by installing spring pin.

(4) Install centering arm and upper scissors assembly into position. Secure by installing towing shaft through both assemblies.

(5) Install spring pin securing towing shaft. Lockwire spring pin.

(6) Connect scissors assembly.

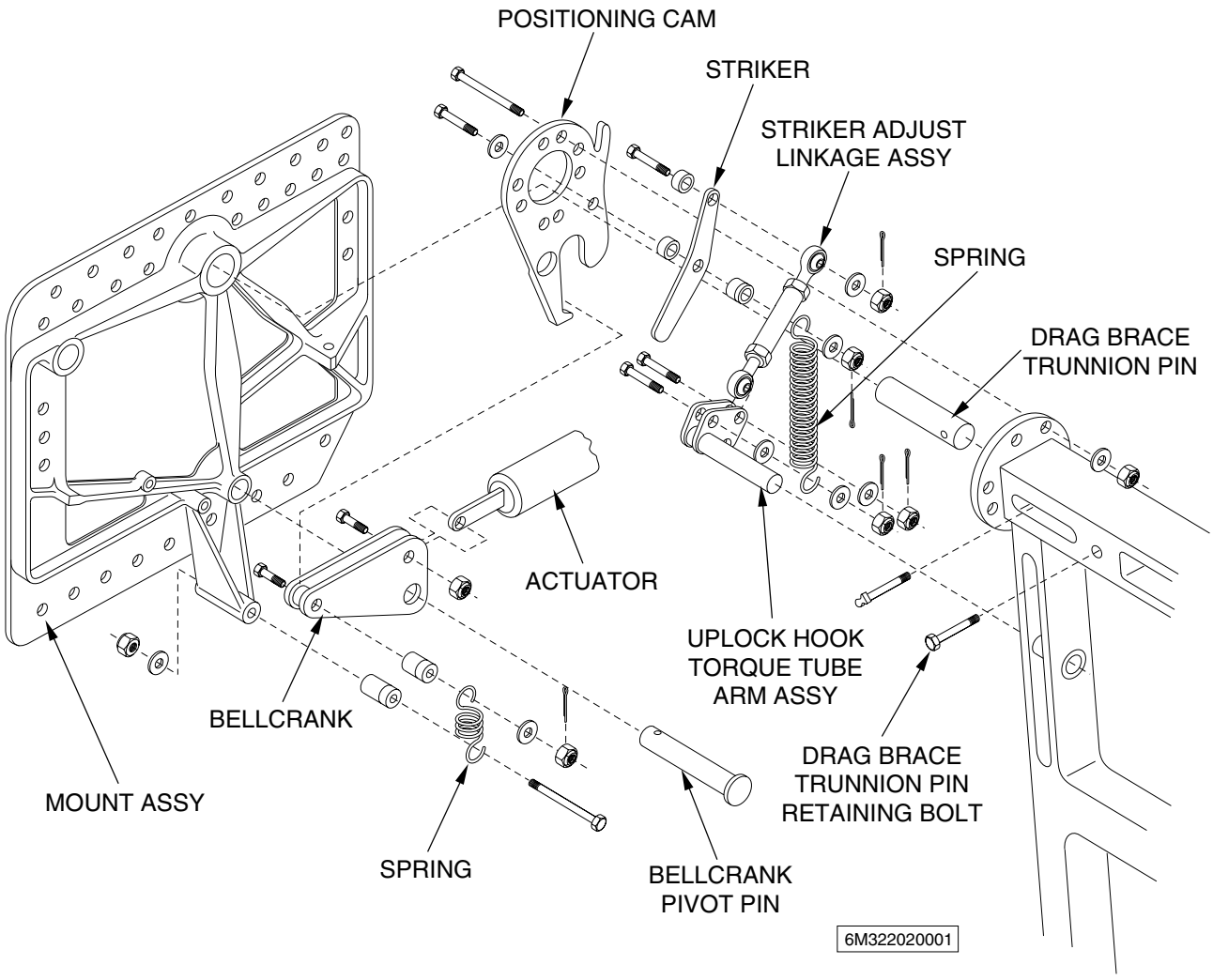
NOTE: Do not overtighten bolt. Bottom nut and back off to nearest cotter pin slot. Max allowable torque required to pivot scissors about attachment is five inch-pounds.

NOSE GEAR BELLCRANK AND ACTUATOR – DESCRIPTION AND OPERATION

1. General

A. Each actuator cylinder is attached to an actuator bellcrank and roller assembly through which the initial actuator movement is transmitted to unlock the drag brace during retraction and to release the mechanical uplock hook assembly during gear extension.

B. Movement of the actuator in the selected direction continues until the bellcrank and roller assembly contacts a hardstop. Each actuator piston is attached to its respective gear strut. After initial uplocking action by the actuator, its piston moves to retract or extend the gear, as selected.



Nose Gear Actuator
Figure 1

NOSE GEAR BELLCRANK AND ACTUATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Gear Bellcrank and Actuator

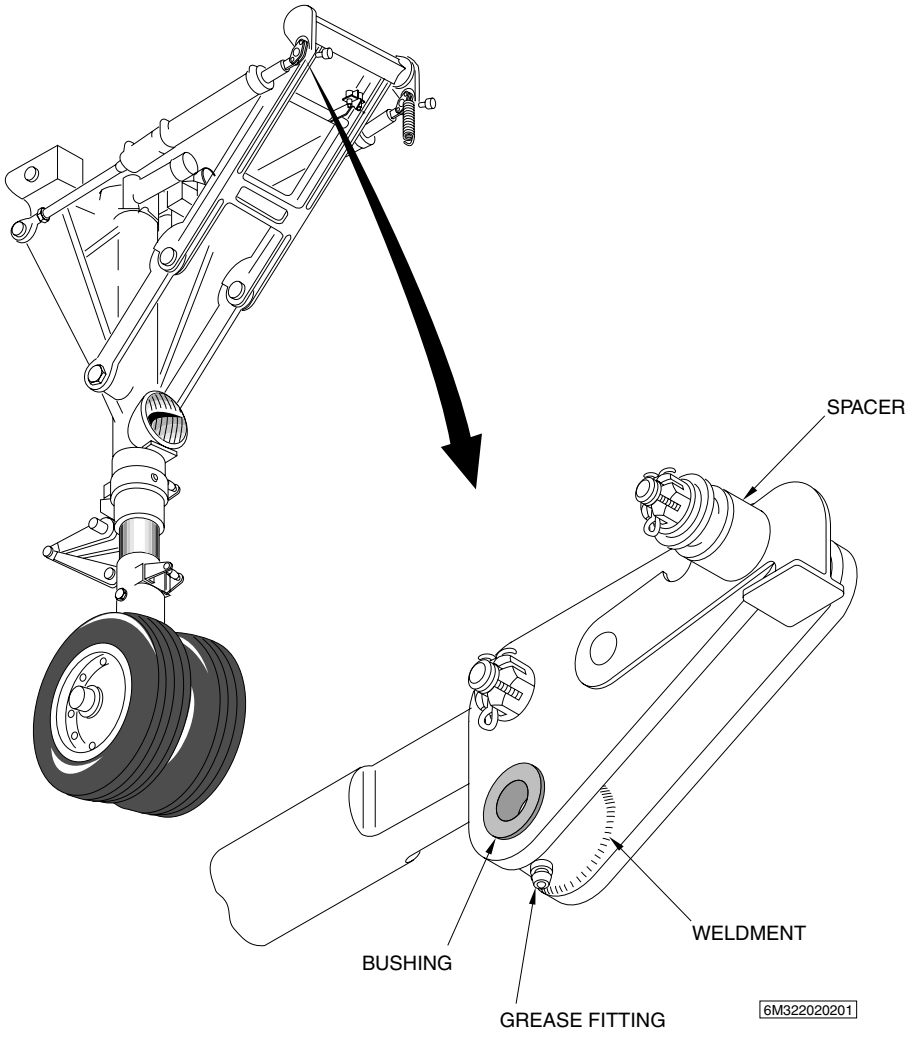
WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Removal – Nose Gear Bellcrank

(1) Jack aircraft.

(2) Bleed hydraulic pressure from down side of actuators by disconnecting hydraulic lines to the actuators. Cap disconnected hydraulic lines and actuator ports to prevent contamination.

(3) Remove forward baggage compartment floorboard access panels to gain access to bellcrank pivot pin retaining bolts.



Nose Gear Bellcrank
Figure 201

(4) Remove spring between bellcrank and drag brace mount assembly.

NOTE: If a slow down indication is obtained when nose gear is lowered, actuator bellcrank springs may need to be replaced with heat treated corrosion resistant steel springs.

(5) Remove bellcrank pivot pin retaining bolts.

(6) Remove bellcrank pivot pin.

(7) Remove bellcrank.

(8) If bushing is worn beyond limits in Para. 2, remove bushing by placing the bellcrank horizontally against two level surfaces on either side of bushing and carefully tap out one bushing at a time. Repeat the procedure for the other bushing.

(9) Inspect the bellcrank for cracks and wear.

B. Installation – Nose Gear Bellcrank

(1) If bushing requires replacement (see Inspection/Check- Nose Gear Actuator and Bellcrank):

(a) Carefully align and ream bearing hole in the bellcrank.

(b) Insert bushings and tap into place.

(c) After bushing installation, ream 0.6265 to 0.6255 inch (15.91 to 15.89 mm).

(d) Install bellcrank bolt, washer, spacer, nut and cotter pin. Torque 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(2) If bushing does not require replacement (see Inspection/Check- Nose Gear Actuator and Bellcrank):

(a) Position bellcrank into mounting location.

(b) Install bellcrank bolt, washer, spacer, nut, and cotter pin. Torque 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(3) Shim bellcrank INBD so that release arm and cam fall on bellcrank roller.

CAUTION: DURING INSTALLATION, DO NOT STRETCH SPRING BEYOND 7.35 INCHES (186.69 mm) SINCE PERMANENT DEFORMATION COULD RESULT.

(4) Install spring, bellcrank and drag brace mount assembly.

(5) Install actuator rod ends to gear assembly. Install bolts, washers and nuts. Torque bolts 480 to 690 inch-pounds (54.23 to 77.96 N·m). Secure nuts with cotter pins.

(6) Install hydraulic lines to actuators.

(7) Perform actuator adjustment and landing gear functional check.

(8) Install floorboard access panels in nose baggage compartment.

NOSE GEAR BELLCRANK AND ACTUATOR – MAINTENANCE PRACTICES

C. Removal – Nose Gear Actuator

(1) Jack aircraft.

(2) Bleed hydraulic pressure from gear-down side of actuators by disconnecting hydraulic lines to the actuator. Cap disconnected hydraulic lines and actuator port to prevent contamination.

(3) Remove front baggage compartment floorboard IAW Chapter 53 to gain access to bellcrank pivot pin retaining bolts.

(4) Remove spring between bellcrank and drag brace mount assembly.

(5) Remove bellcrank pivot pin retaining bolt.

(6) Remove bellcrank pivot pin.

(7) Remove actuator rod end nuts, washers, and retaining bolts from gear assembly. Remove actuators.

D. Installation – Nose Gear Actuator

NOTE: Several actuators used on SA226 series aircraft have been provided by different manufacturers. Accordingly, actuators by different part numbers should not be mixed in any installation (see Illustrated Parts Catalog).

(1) Position actuator and bellcrank into mounting location and install bellcrank pivot pin.

(2) Install bellcrank pivot pin retaining bolt, washers and nuts. Torque 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(3) Install spring between bellcrank and drag brace mount assembly.

(4) Install actuator rod ends to gear assembly and install bolts, washers, and nuts. Torque bolts 480 to 690 inch-pounds (54.23 to 77.96 N·m). Secure nuts with cotter pins.

(5) Install hydraulic lines to actuators.

(6) Perform actuator adjustment and landing gear functional check IAW LANDING GEAR RIGGING PROCEDURES, Section 32-30-00.

(7) Install floorboard access panels in nose baggage compartment IAW Chapter 53.

2. Inspection/Check – Nose Gear Bellcrank and Actuator

A. Inspection – Bellcrank and Bushings

(1) Inspect bellcrank and bushings for cracks and wear.

(a) Replace bushing if worn more than 0.6285 inch (15.964 mm) diameter.

(2) Inspect bellcrank pivot pin for evidence of corrosion, cracks, and wear.

(a) Replace pin if worn more than 0.6226 inch (15.814 mm) diameter.

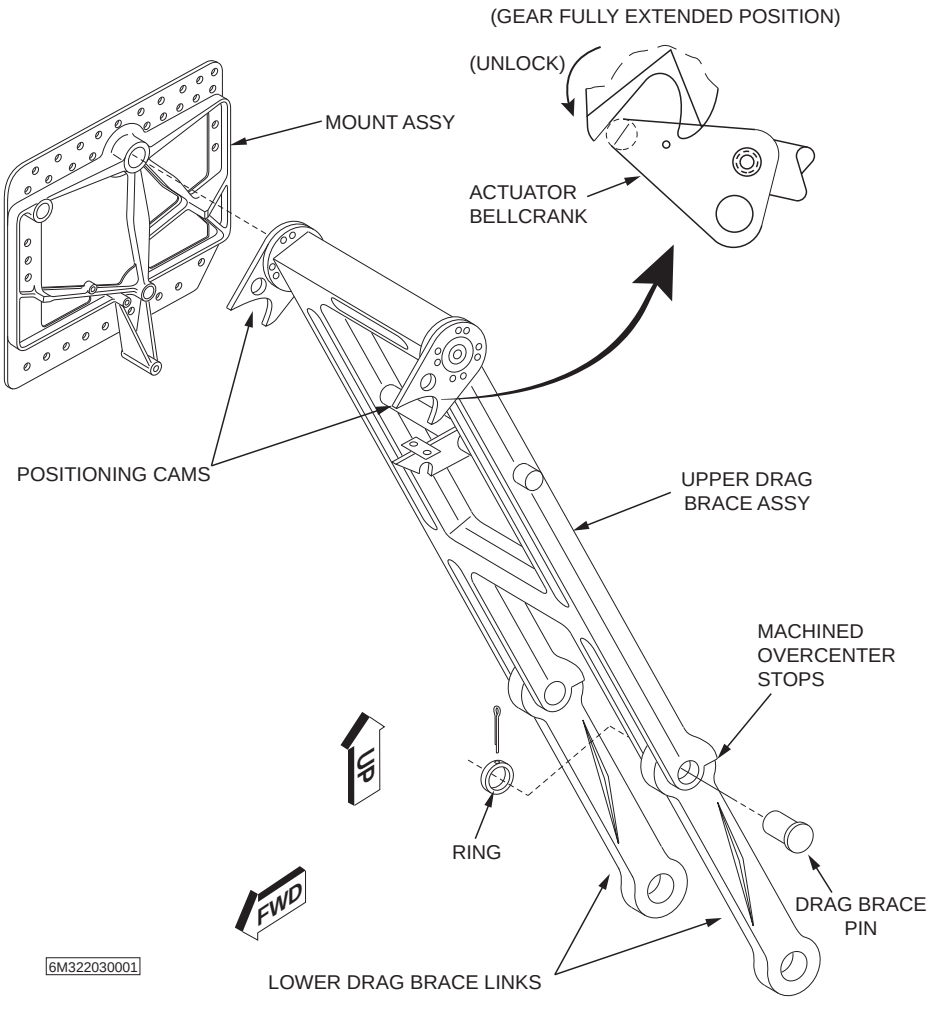
NOSE GEAR DRAG BRACE – DESCRIPTION AND OPERATION

1. General

A. The drag brace is attached to the aircraft structure and the landing gear shock strut. Positioning the drag brace assembly in an overcenter condition locks the gear in the extended position. The drag brace is held in the overcenter condition by the spring assisted bellcrank and roller assembly and hydraulic pressure trapped in the down side of the landing gear actuator.

B. The uplock hook assembly consists of a spring loaded uplock hook and a mounting rod. The assembly is attached to the upper drag brace on each gear and functions as a mechanical lock in the retracted position. A release lever is attached to the mounting rod and is engaged by the actuating arm of the manual cable release system when the gear is retracted. When actuated, the release lever rotates the uplock hook and releases the gear (see LANDING GEAR RETRACTION AND EXTENSION, 32-30-00, this chapter).

C. There are two positioning cams mounted on each of the drag strut assemblies. When the gear is extended the bellcrank rollers engage a notch in the cam to lock the drag strut in the overcenter position.



Nose Gear Drag Brace
Figure 1

NOSE GEAR DRAG BRACE – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Gear Drag Brace

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Removal – Drag Brace

(1) Jack aircraft (Refer to 07-10-00).

(2) Remove nose gear actuators and bellcranks (Refer to 32-20-20).

(3) At both lower drag brace links to main gear strut trunnions, remove sealant brush coat (if applied) and remove pin retaining bolts, washers and nuts. Remove lower trunnion pins freeing drag brace assembly from gear assembly.

(4) Remove upper drag brace trunnion pin retaining bolts. While holding upper drag brace, grab grease pins on trunnion pins and push pins inboard.

(5) Carefully slide upper drag brace down and out of wheel well.

B. Installation – Drag Brace

NOTE: Upper and lower drag braces must be assembled as matched sets. See matched set numbers stamped near drag brace pivot pin holes, on each lower leg of upper brace and on upper end of each lower brace.

(1) Position drag brace assembly into mounting location.

(2) Install shims (AN960-1616L) between drag brace trunnion and trunnion bushings at each side as required. Obtain a combined total minimum clearance of 0.010 inch (0.254 mm) and maximum clearance of 0.038 inch (0.965 mm) before final installation of trunnion mount pins.

(3) Apply a coat of grease (Refer to 12-20-10) to trunnion bushings.

(4) Install trunnion pins by sliding them outboard into trunnion mount bushings at each side.

(5) Align pins to install retaining bolts.

(6) Install trunnion pin retaining bolts, washers, and nuts. Torque 15 to 30 inch-pounds (1.69 to 3.39 N·m).

(7) Position both lower drag brace links to gear strut.

(8) Install shims (AN960-1416L) symmetrically (inboard and outboard of each lower drag link) as required to obtain a combined total lateral clearance of 0.010 inch (0.254 mm) to 0.038 inch (0.965 mm) between strut and each lower drag brace link.

(9) Install drag brace attach pins and align holes in pins to install retaining bolts.

(10) Install drag brace attach pin retaining bolts with sealant in accordance with SA226/227 Structural Repair Manual 51-30-03, Section 2.F.(3)(g). Install washers and nuts and torque bolt 15 to 30 inch-pounds (1.69 to 3.39 N·m).

(11) Brush coat the head and nut end of retaining bolts in accordance with SA226/227 Structural Repair Manual 51-30-03, Section 2.F.(3)(g).

(12) Install nose gear actuators and bellcranks (Refer to 32-20-20).

(13) Down jack aircraft (Refer to 07-10-00).

C. Removal – Uplock Assembly (Figure 201)

(1) Remove drag brace assembly IAW step **1.** A.

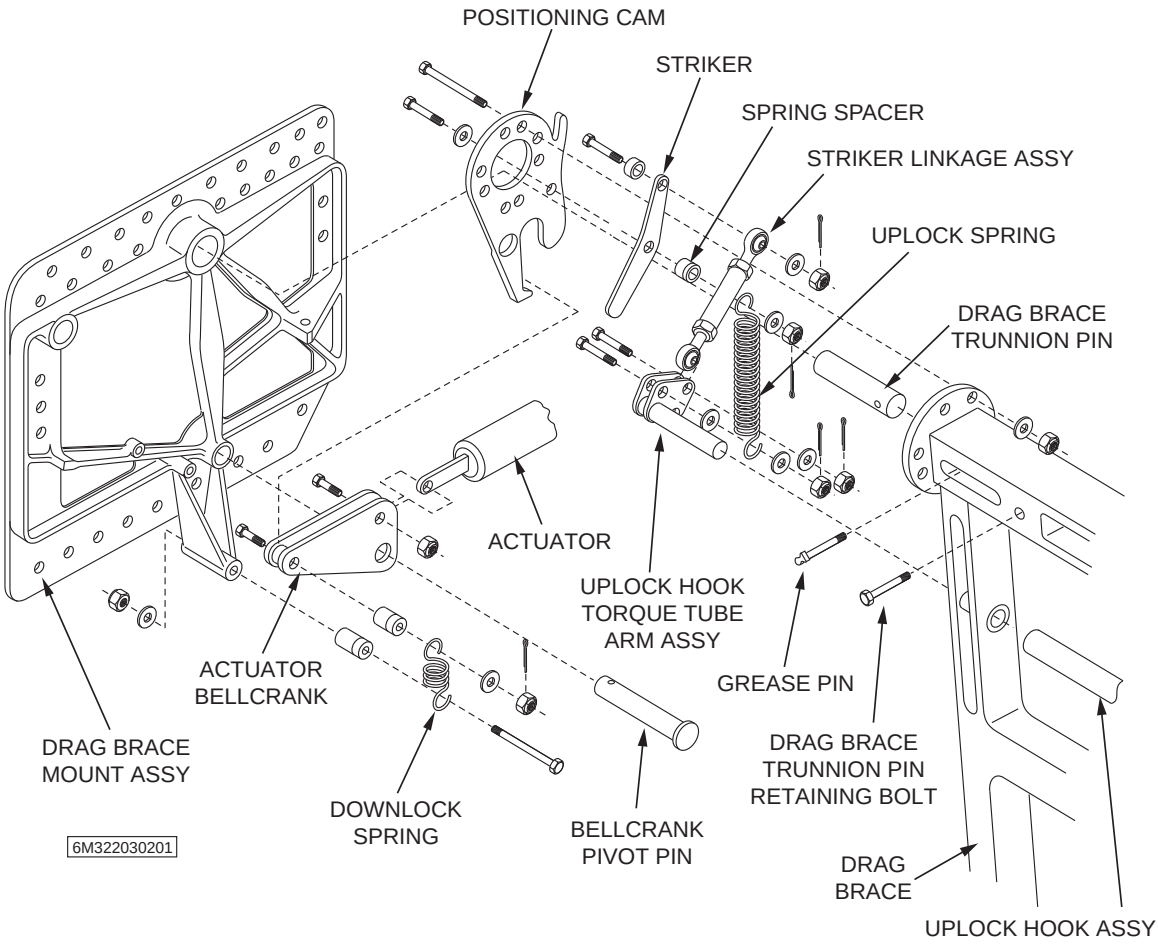
(2) Remove retaining bolts in uplock hook torque tube securing uplock hook arm assemblies.

(3) Remove retaining bolt securing spring to arm assembly at striker. Remove spring.

(4) Remove retaining bolts securing striker linkage assembly and arm assembly to the striker. Remove linkage assembly.

(5) Remove arm assembly by sliding it from drag brace and uplock hook torque tube.

(6) Remove uplock hook assembly.



Nose Gear Uplock Assembly Removal/Installation
Figure 201

D. Installation – Uplock Assembly (Figure 201)

NOTE: Verify bushings in drag brace for uplock hook shaft are not worn before installation of uplock hook assembly. If replacement of bushings is required, confirm Service Bulletin status to determine bushing part number, and if bushings need to be drilled for lubrication or not. Refer to Service Bulletins 226-32-052 and 226-32-060.

(1) Position uplock assembly into the mounting location on the drag brace.

(2) Install shims (AN960C1016L or AN960PD1016L) as required to obtain a maximum clearance of 0.016 inch (0.406 mm) between uplock assembly and drag brace (see Figure 202).

(3) Slide left and right uplock hook torque tube arm assemblies through each side of drag brace and into uplock hook assembly.

NOTE: When installing arm assembly into torque tube, it may be necessary to ream torque tube before arm assembly will fit into torque tube. Ream torque tube by hand with 0.625 inch (15.875 mm) reamer.

(4) For Aircraft Pre SB 226-32-053, pre-adjust left and right striker link assemblies to 4.00 inch (101.6 mm).

(5) For Aircraft Post SB 226-32-053, pre-adjust left and right striker link assemblies to 4.25 inch (107.95 mm).

(6) Connect striker linkage assemblies onto left and right uplock hook torque tube arm assemblies and onto striker arms. Install uplock springs and verify strikers operate freely.

(7) Install drag brace assembly IAW Drag Brace Installation Steps in **1**.B.(1) through (9) in this section.

NOTE: New uplock assemblies must be drilled.

CAUTION: BEFORE DRILLING THROUGH UPLOCK HOOK TORQUE TUBE AND UPLOCK TORQUE TUBE ARM ASSEMBLIES, ENSURE SUFFICIENT CLEARANCE BETWEEN LINKAGE ASSEMBLY RETAINING NUT/COTTER PIN AND DRAG BRACE ASSEMBLY (SEE FIGURE 202).

(8) If uplock assembly is new, establish holes for bolts securing left and right uplock hook torque tube arm assemblies to uplock hook assembly as follows:

(a) Manually rotate landing gear to retracted position until uplock hook is fully engaged with uplock roller with no load on the hook. Gap between roller and back of hook must be 0.0 inch. The uplock hook adjust bolt must be in contact with the drag brace when the zero clearance is obtained.

(b) Adjust striker link rods as necessary to obtain zero clearance requirement in previous step with striker levers in contact with their respective actuator bellcrank rollers (Refer to 32-30-00, Figure 203).

(c) If proper adjustment cannot be made, verify rigging of nose gear emergency release system is not interfering with striker rod adjustment (Refer to 1. H. Adjustment - Emergency Release System in section 32-30-00).

(d) With uplock hook fully engaging uplock roller (touching but with no load on hook), and with striker levers contacting their respective actuator bellcrank rollers, drill holes through pilot holes at each end of new uplock hook torque tube and through uplock hook torque tube arm assemblies.

(e) Drill and ream bolt holes to 0.189 inch (4.80 mm) maximum diameter. Install retaining bolts, washers, and nuts. Torque nuts 20 to 25 inch pounds (2.26 to 2.82 N·m).

(f) Manually disengage uplock roller from hook and extend nose gear until drag brace is overcenter.

(9) If uplock assembly is not new:

(a) Install bolts through uplock hook torque tube and both uplock hook torque tube arm assemblies. Install washers and nuts onto bolts, turning nuts until they just contact uplock hook assembly torque tube.

(b) Verify there is no rotational play between hook assembly and either torque tube arm. If play is evident, isolate and correct discrepancy before proceeding.

NOTE: It is recommended to comply with Service Bulletin 226-32-053 which installs a new uplock hook assembly consisting of torque tube arms with solid rods instead of tubes. This modification reduces the possibility of rotational play in hook assembly.

(c) Tighten nuts on bolts at both ends of hook assembly and ensure all attaching hardware is tight. Verify sufficient clearance between striker link hardware and drag brace as shown in Figure 202.

(10) Perform remaining Drag Brace Installation Steps in **1**.B.(10) through (13).

E. Removal – Positioning Cam (Figure 201)

(1) Remove drag brace assembly IAW Step **1.** A.

(2) Remove retaining bolts, washers and nuts securing cam to drag brace assembly.

(3) Remove positioning cam.

(4) If necessary, remove hardware securing striker to cam and remove striker.

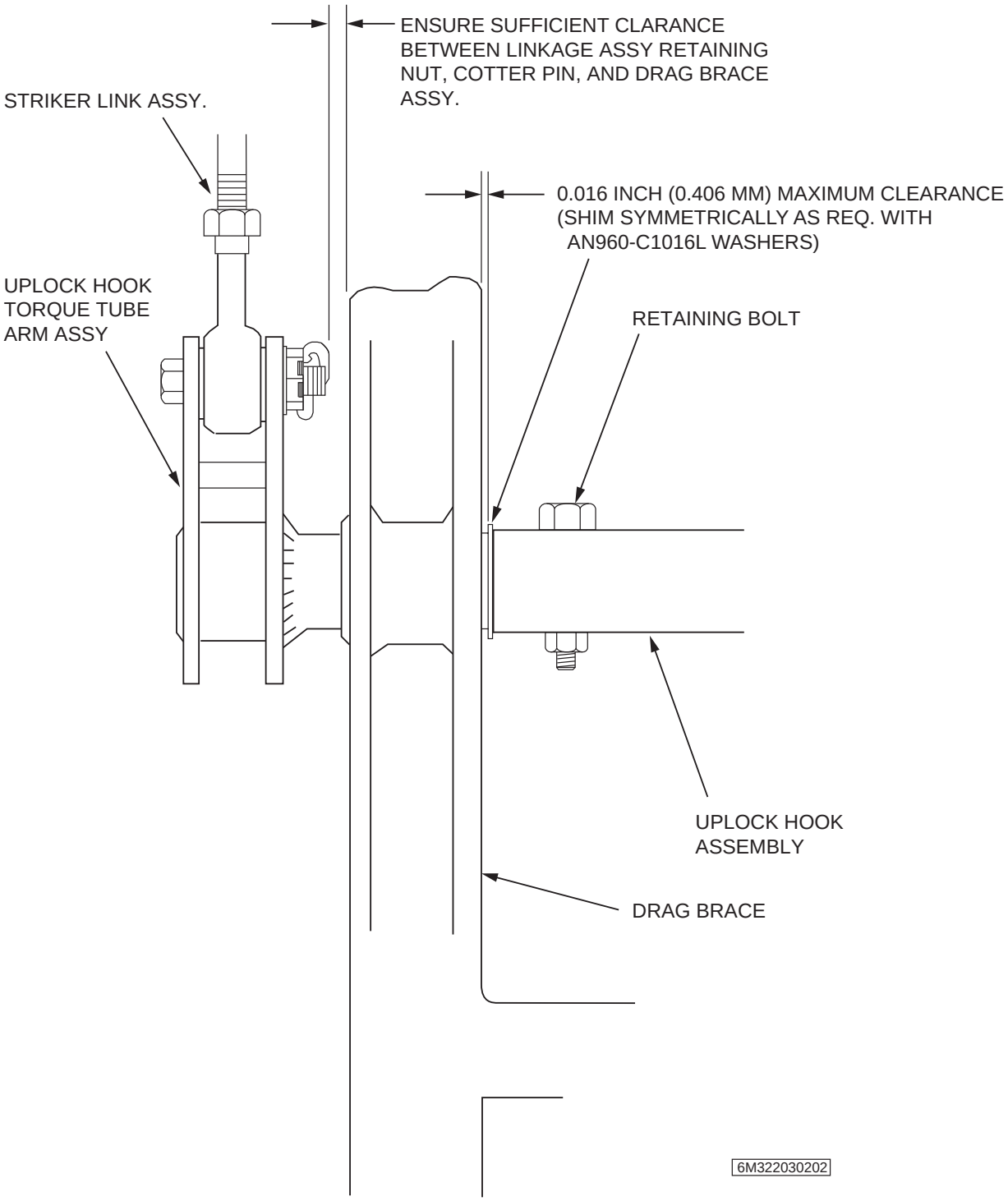
F. Installation – Positioning Cam (Figure 201)

(1) If positioning cam is being replaced with new cam:

(a) Position new cam between drag brace and nacelle trunnion mount and install drag brace assembly per instructions in this section.

(b) Hold the drag brace knee in the over-centered position (push up on center of drag brace).

(c) Rotate cam to specified clearance (gap) between the bellcrank roller and the positioning cam with gear down (see Section 32-30-00, Figure 205).



Drag Brace Arm Assembly
Figure 202

CAUTION: EXERCISE EXTREME CARE WHEN ENLARGING PILOT HOLES TO PREVENT ELONGATING DRAG BRACE HOLES.

NOTE: During the next step, use a drill bushing of the correct size to properly locate center of pilot holes and prevent undesired reaming or elongation of drag brace holes.

(d) Locate and drill two #40 (0.098 inch [2.489 mm]) pilot holes through cam.

(e) Open pilot holes to 0.1895 to 0.1915 inch (4.8133 to 4.8641 mm) and install two retaining bolts, washers and nuts to secure cam to drag brace.

(f) Remove drag brace and cam as a single unit.

(g) Locate and drill the remaining bolt holes in cam.

(h) Mark the position of cam to drag brace, then remove the two retaining bolts, washers and nuts installed earlier.

(i) Deburr all holes and prime with zinc chromate primer.

(2) If positioning cam is being reinstalled:

(a) Place cam on drag brace at same orientation and inboard or outboard position as removed.

(3) Secure cam to drag brace with retaining bolts, washers and nuts. Torque retaining nuts 20 to 25 inch pounds (2.26 to 2.82 N·m).

NOTE: Because of close tolerance of holes, bolts may require gentle tapping into place. Be careful not to damage threads of bolts.

(4) Install striker with spring spacer, onto cam with retaining bolt, washer, nut and cotter pin.

(5) Install drag brace assembly IAW Step **1.** B.

LANDING GEAR RETRACTION AND EXTENSION – DESCRIPTION AND OPERATION

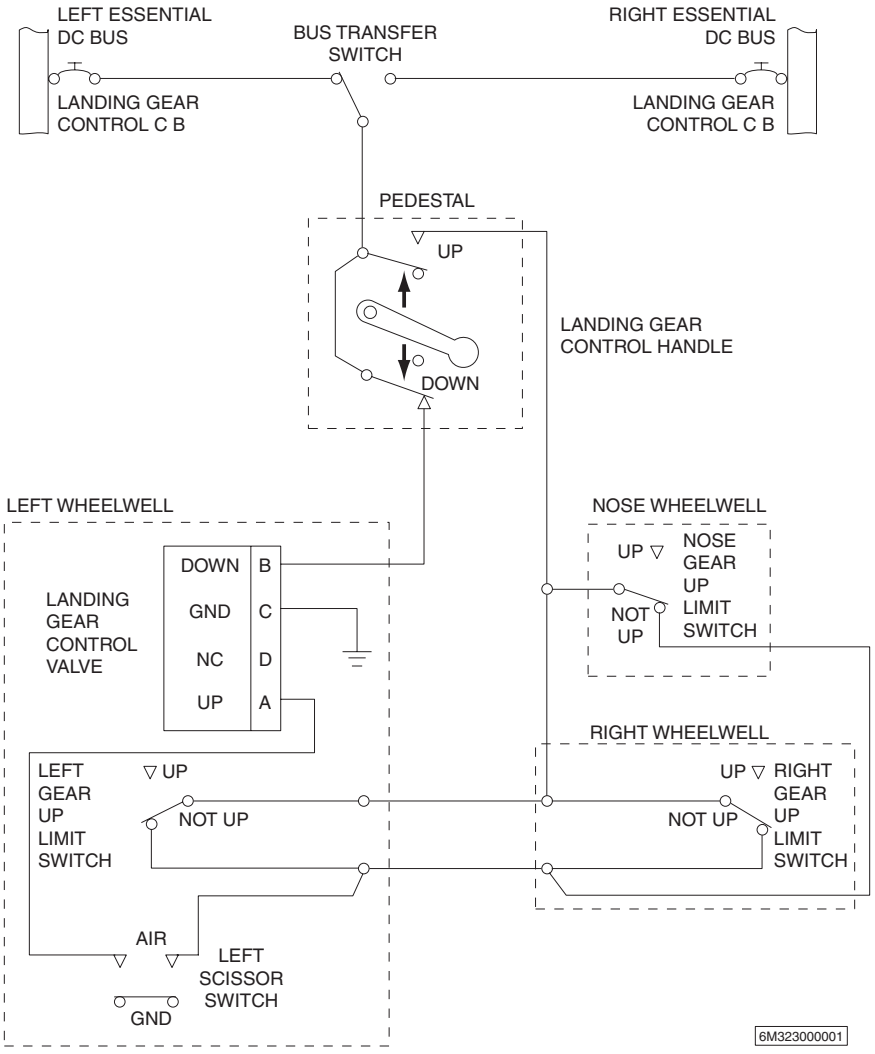
1. General

A. Landing gear retraction and extension is electrically controlled and hydraulically actuated.

(1) The landing gear control handle, located on the cockpit pedestal, is used to direct 28.5 VDC electrical power from either the left or right essential bus, as selected to the landing gear selector valve mounted underneath the hydraulic power pack in the left wheel well.

(2) The selector valve, when actuated, directs hydraulic pressure to the landing gear actuators for either retraction or extension of the gear, as selected.

(3) Two actuators are installed on each gear. Both actuators are used for gear retraction. Hydraulic pressure is directed to the left actuator of the nose landing gear and outboard actuators of each main landing gear during normal gear extension and by use of the hand pump, hydraulic pressure (if available) is applied to the right actuator of the nose landing gear and inboard actuators of each main landing gear during emergency extension of the gear.



Normal Landing Gear Control
Figure 1

B. As each gear is fully retracted, it engages a mechanical uplock hook. When the last of the three gear fully retracts, electrical power is disconnected from the selector valve. The selector valve moves to the closed (OFF) position, and a hydraulic lock exists in the actuator lines.

C. As each gear moves to the fully extended (down) position, its dual drag strut unfolds and the drag strut joints (or elbows) move to an overcenter position. This overcenter position of the extended drag strut is locked by the mechanical interference between a bellcrank and roller assembly attached to each actuator cylinder and a cam attached to each side of the dual drag strut.

(1) Normal hydraulic pressure, when available, is applied to the down side of the main landing gear outboard actuators, and to the nose landing gear left actuator until the selector valve closes due to shutdown of both engines (hence causing loss of normal hydraulic pressure), or the electrical power is shutoff. When the selector valve closes a hydraulic lock is formed.

(2) When the emergency hand pump is used, hydraulic pressure is applied to the down side of the right actuator on the nose landing gear and the inboard actuator of each main landing gear. Emergency hand pump pressure is not routed through the selector valve.

D. The emergency gear extension system includes provisions for a manual release of the mechanical uplocks, manual repositioning of valves to bypass the gear selector valve, and the hydraulic hand pump which applies hydraulic pressure to the down side of the right actuator of the nose landing gear and inboard actuator of each main landing gear.

(1) The manual emergency gear release handle is located on the cockpit floor adjacent to the left side of the copilot's seat.

(2) On SA226 series aircraft T201 through T227, AT001 through AT008, and TC201 through TC204, cables and pulleys are installed to release the mechanical uplock on all gear and to reposition the two gear bypass valves beneath the forward side of the hydraulic reservoir.

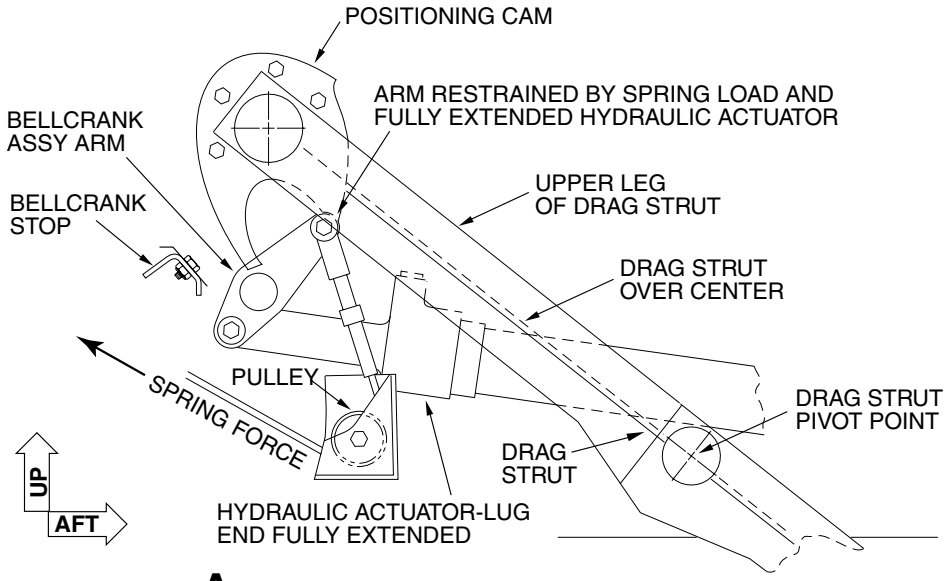
(3) On SA226 series aircraft, T228 and up, AT009 and up, and TC205 and up, a shielded cable system is installed replacing the cables and pulleys, from the emergency release handle to the nose gear uplock.

E. Operation

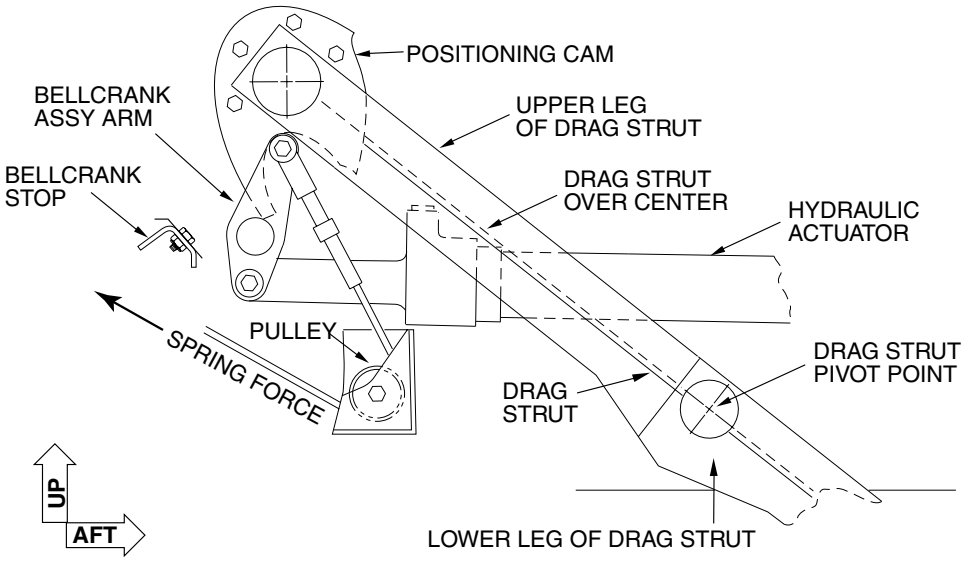
(1) Before emergency extension procedures are implemented, reduce the aircraft speed to maximum gear extension speed (176 knots calibrated air speed – KCAS) or less. Place the landing gear handle in the DOWN position in the event normal gear control should return. Move the emergency release lever counterclockwise approximately 90 degrees. This action moves the cables to release the mechanical uplock on each gear and reposition two gear bypass valves at the hydraulic reservoir.

(2) Repositioning these two valves allows the fluid trapped in the up line of the actuators used for normal retraction, to bypass the gear selector valve and return to the reservoir. The gears then free fall of their own weight and the force of the airstream carries the gear to the down and locked position.

(3) A hydraulic hand pump is located on the cockpit floor adjacent to the pilot's seat. Before the hand pump can be operated, remove the safety pin in order to rotate an emergency selector valve lever. Rotating the selector valve lever 90 degrees counterclockwise permits supply of hydraulic fluid from the reservoir to the down side of the right actuator of the nose landing gear and to the inboard actuators of the main landing gear. The hand pump can then be actuated to provide hydraulic pressure to the down side of the actuators to further ensure positive locking of the gear.



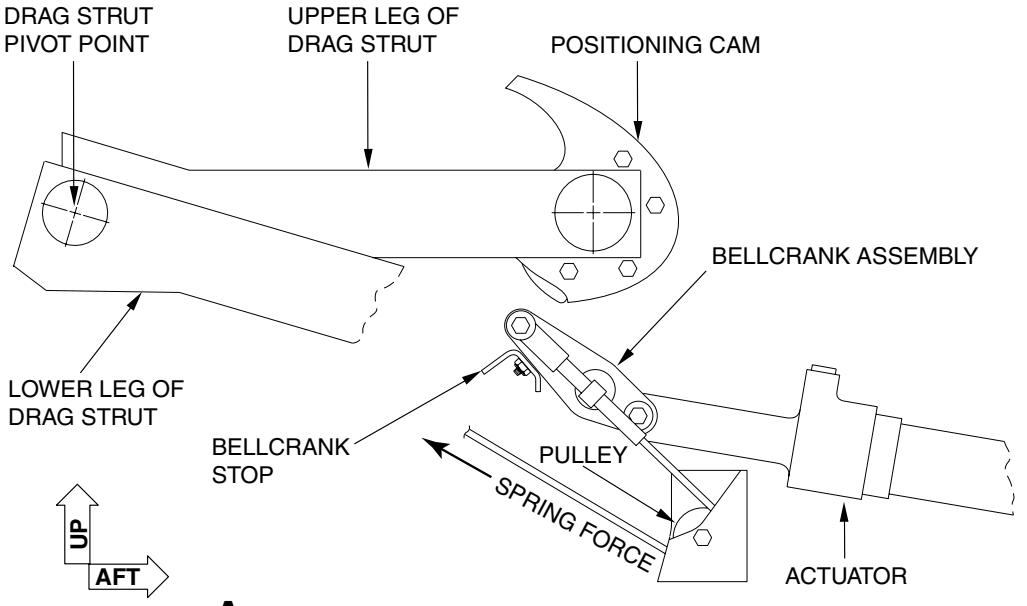
A LANDING GEAR DOWN & LOCKED



B LANDING GEAR - INITIAL UNLOCKING ACTION 6M323000002

Main Gear Operation
Figure 2

(4) A shuttle valve is installed between an engine driven pump pressure line and the emergency hand pump pressure line. The shuttle valve is actuated by hydraulic pressure to direct either normal hydraulic pressure or hand pump pressure to the hydraulic pressure gauge located on the copilot's instrument panel. Thus, if normal hydraulic pressure is lost the hydraulic pressure gauge will indicate hand pump pressure.



Main Gear Operation
Figure 3

2. Control – Retraction and Extension

A. Normal control of the landing gear is accomplished by an electrical system that uses a control handle, located on the cockpit pedestal, to control a solenoid valve.

B. The control handle actuates an up and a down microswitch. The microswitches receive power from the LAND GEAR CONT circuit breakers on the left and right essential buses. The power can be connected to either bus by a transfer switch on the rear portion of the left console.

C. When the control handle is placed in the down position, power goes directly to the down solenoid of the control valve. When the control handle is placed in the UP position power must first pass through the normally closed (not up) contacts of at least one of the uplimit switches.

D. The uplimit switches open when the gear is up and locked. On aircraft not complying with SB226-32-057, the uplimit switches are mounted adjacent to the up position switches on the uplock hook of each landing gear. On aircraft complying with SB226-32-057, the uplimit switches are mounted on the left and right drag brace mount fittings. From the uplimit switches, power must then pass through a scissor switch on the left gear to the up solenoid of the control valve.

E. The scissor switch operates by strut extension. The switch is open on the ground disabling the up circuit of gear selector valve solenoid when the aircraft is on the ground and closes when the wheels are off the ground to complete the circuit.

F. A service bulletin (32-025) was issued to replace present landing gear switches (main and nose) with environmentally sealed switches, as well as all switch brackets. The service bulletin also replaced the taxi light switch and bracket with a sealed switch; added a terminal strip for nose gear switches; and replaced a landing gear indicator with replaceable bulb type. Refer to service bulletin effectivity SA226 series T201 through 275, 277 through 291, T(B) 276, 292, through 335, AT001 through 074, TC201 through 334.

G. To improve nose landing gear uplock operation, service bulletin 32-053 is recommended for all aircraft prior to SA226 serial number 334E, which have already accomplished service bulletin 32-025. Service bulletin 32-053 replaces downlock switches, downlock switch brackets, and wire harness. The service bulletin also relocates new switches, brackets, and wiring harness to reduce flexing of the wiring and improve the mechanical advantage of the switch actuation. Effectivity is SA226 T(B) 336-417, and TC334E-419, see service bulletin.

H. Service Bulletin 32-057 was issued for all SA226 effectivities to replace the main gear uplock switches with more reliable switches and to relocate the main and nose gear uplock switches from the drag brace to the drag brace mounts.

3. Main Gear Operation – Retraction and Extension

A. Down and Locked

(1) Each main landing gear is locked down by its drag strut extending to an overcenter position. The drag brace cannot fold nor move from its overcenter position, because the actuator bellcrank has been positioned so as to prohibit movement of the drag strut positioning cam (reference Figure 2A). Hydraulic pressure, from either the normal or emergency system, will fully extend the landing gear actuator thereby locking the actuator bellcrank in place. In the absence of hydraulic pressure, a spring attached to the keelson web and the actuator bellcrank provides the force necessary to ensure the actuator bellcrank stays properly positioned.

B. Retraction

(1) Hydraulic pressure is applied to the retraction side of each landing gear actuator. Initial retraction of each actuator causes its respective bellcrank to rotate forward. As the bellcrank rotates it moves away from the positioning cam to unlock the drag brace. Further rotation of the bellcrank causes it to contact the forward side of the positioning cam. This action causes the drag brace to fold and gear retraction begins. As the drag brace folds, the positioning cam is rotated which allows the bellcrank to rotate forward and contact a stop installed on the keelson. Continued retraction of the actuators causes the drag brace to continue folding and the landing gear strut to rotate forward into the wheel well. The gear continues traveling into the well until the uplock hook is engaged.

C. Up and Locked

(1) As the drag link continues folding and the strut enters the wheel well, a roller installed on the forward portion of the strut will contact the uplock hook assembly installed on the upper part of the drag brace. The uplock hook assembly consists of the hook, positioning springs, position indicating switch, a hydraulic power pack control switch and two actuating arms. The lower surface of the hook is a tapered ramp. As the uplock roller contacts this ramp, the hook is moved away against the spring tension until the roller enters the opening in the hook. At this time, spring force will move the hook aft and trap the uplock roller (see Figure 3C). Each of the two bosses that provide the mounting for the uplock roller now contacts its respective electrical switch assembly. One (position) switch will provide an up indication in the cockpit while the other (limit) switch supplies a signal to control the hydraulic power pack. When the last gear is up and locked, the hydraulic power pack pressure will be shut off. Each gear is secured in its wheel well by the hydraulic lock in the actuators and by the mechanical lock of the uplock hook.

D. Normal Extension

(1) As hydraulic pressure is applied to the down side of the left nose landing gear actuator and the outboard actuator of each main landing gear, extension of the actuator causes the respective bellcrank to rotate aft. During this rotation, the bellcrank will contact the striker installed on the positioning cam. The striker then pushes the actuating arm on the drag brace forward. The movement of the actuating arm, which is on a common shaft with the uplock hook, causes the hook to move forward and release the uplock roller. The strut now extends aft towards full vertical, and the drag brace unfolds. The bellcrank continues rotation until the strut is fully down and locked, and the drag brace is locked overcenter by the bellcrank roller.

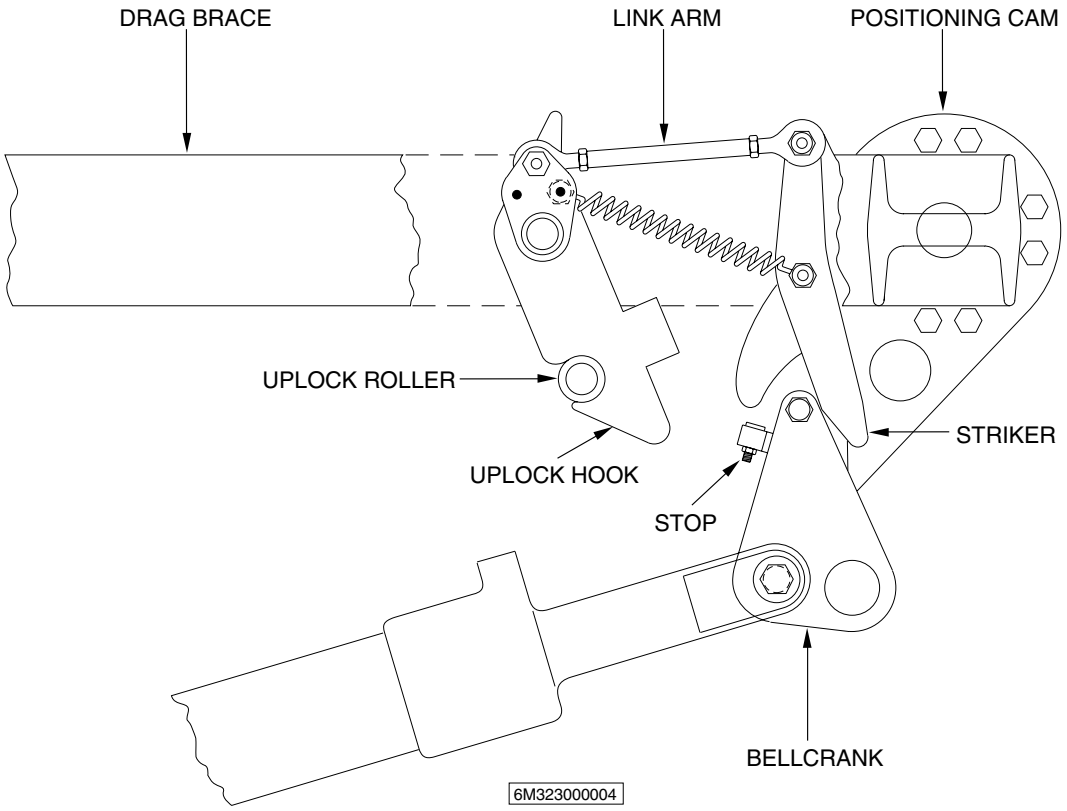
4. Nose Gear Operation – Retraction and Extension

A. Down and Locked

(1) The nose landing gear is locked down by its drag strut extending to an overcenter position. The drag brace cannot fold nor move from its overcenter position because the actuator bellcrank has been positioned to prohibit movement of the drag brace positioning cam (see Figure 4). Hydraulic pressure, from either the normal or emergency system, will fully extend the landing gear actuator thereby locking the actuator bellcrank in place. In the absence of hydraulic pressure, a spring attached to the keelson web and the actuator bellcrank provides the force necessary to ensure the actuator bellcrank stays properly positioned.

B. Retraction

(1) Hydraulic pressure is applied to the retraction side of each landing gear actuator. Initial retraction of each actuator causes its respective bellcrank to rotate aft. As the bellcrank rotates it moves away from the positioning cam to unlock the drag brace. Further rotation of the bellcrank causes it to contact the aft side of the positioning cam. This action causes the drag brace to fold and gear retraction begins. As the drag brace folds, the positioning cam is rotated which allows the bellcrank to rotate forward and contact a stop installed on the keelson. Continued retraction of the actuators causes the drag brace to continue folding and the landing gear strut to rotate forward into the wheel well. The gear continues traveling into the well until the uplock hook is engaged.



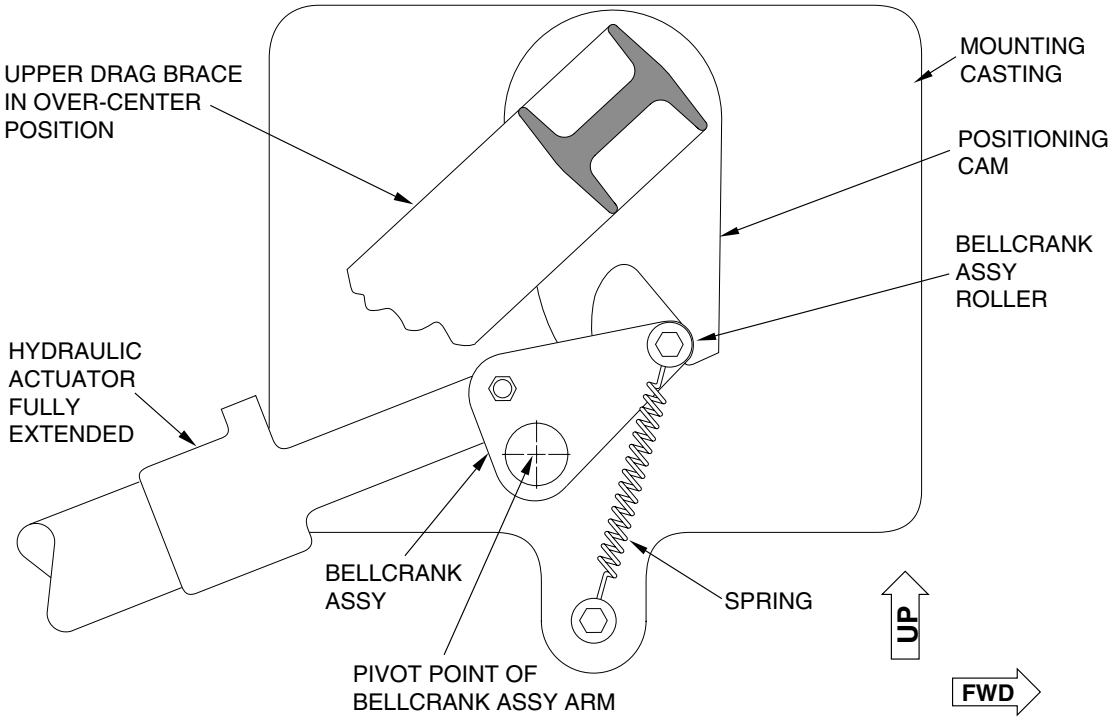
Nose Gear Operation
Figure 4

C. Up and Locked

(1) The initial rotation of the bellcranks release a striker arm installed on each positioning cam. The striker arms are mechanically connected to the uplock hook assembly. Springs installed on either side of the drag brace will rotate the uplock hook aft when the bellcrank rollers release the striker arms. As the nose gear strut enters the wheel well, a roller on the left of the strut will begin contacting the uplock hook. The hook is pushed forward by the roller until the roller aligns with the opening in the hook. Spring action then moves the hook aft to capture the roller. On aircraft not complying with SB32-057, two electrical switches installed on the uplock hook contact the boss on either side of the uplock roller. Closure of one (position) switch provides an up and locked indication in the cockpit while closure of the second (limit) switch controls electrical power to the hydraulic pack. On aircraft complying with SB32-057, the switches are mounted on the left and right drag brace mounts and are actuated by the positioning cam (main gear) or an actuator installed on the positioning cam (nose gear). When all three gears are up and locked, electrical power to the hydraulic pack is removed. Fluid trapped in the actuators and lines provides a hydraulic lock while the uplock hook provides a mechanical lock.

D. Normal Extension

(1) Hydraulic pressure causes the left nose landing gear and the outboard main landing gear actuators to extend. The first action as the actuators extend is a forward rotation of each actuators bellcrank. As the bellcrank moves forward it contacts its respective uplock striker. Movement of the striker causes the uplock assembly to rotate forward and release the gear. The strut falls aft and the drag strut unfolds. The bellcrank continues rotation until the gear is down and locked, and the drag strut is locked overcenter by the bellcrank roller.



NOTE: ARM RESTRAINED BY SPRING LOAD AND FULLY EXTENDED HYDRAULIC ACTUATOR

Nose Gear Operation
Figure 5

LANDING GEAR RETRACTION AND EXTENSION – FAULT ISOLATION

1. Troubleshooting – Landing Gear Retraction and Extension

PROBLEM:

LANDING GEAR FAILS TO RETRACT

PREREQUISITES:

Aircraft on jacks, emergency gear selector in NORM Position, Landing Gear Control Handle UP, Hydraulic Power and Electrical Power Applied.

START:

Verify correct hydraulic fluid level as shown on sight gauge on left outboard side of left nacelle.
- NO ► Service hydraulic system with fluid.
- YES ▼

Check for 28 VDC at landing gear control handle microswitches. Is there 28 VDC?
- NO ► Repair or replace landing gear control circuit breaker.
- YES ▼

With LAND GEAR CONT circuit breaker disengaged and BATT switch OFF, verify that control handle microswitches are operating. Are microswitches OK?
- NO ► Replace landing gear control handle assembly.
- YES ▼

With aircraft on jacks, check for 28 VDC on terminal 3 left wheel well terminal strip. Is there 28 VDC?
- NO ► Adjust or replace left main gear scissors switch.
- YES ▼

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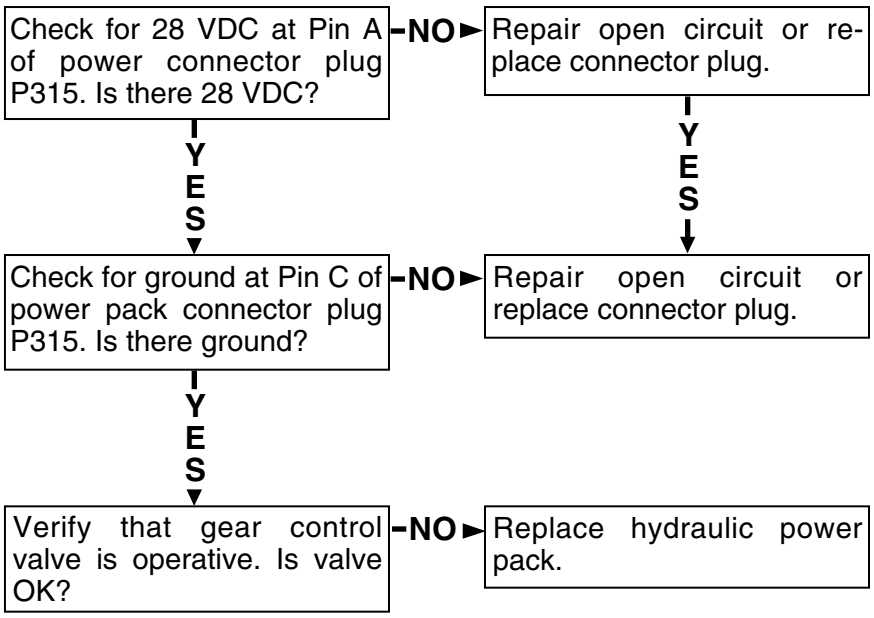

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LANDING GEAR RETRACTION AND EXTENSION – FAULT ISOLATION

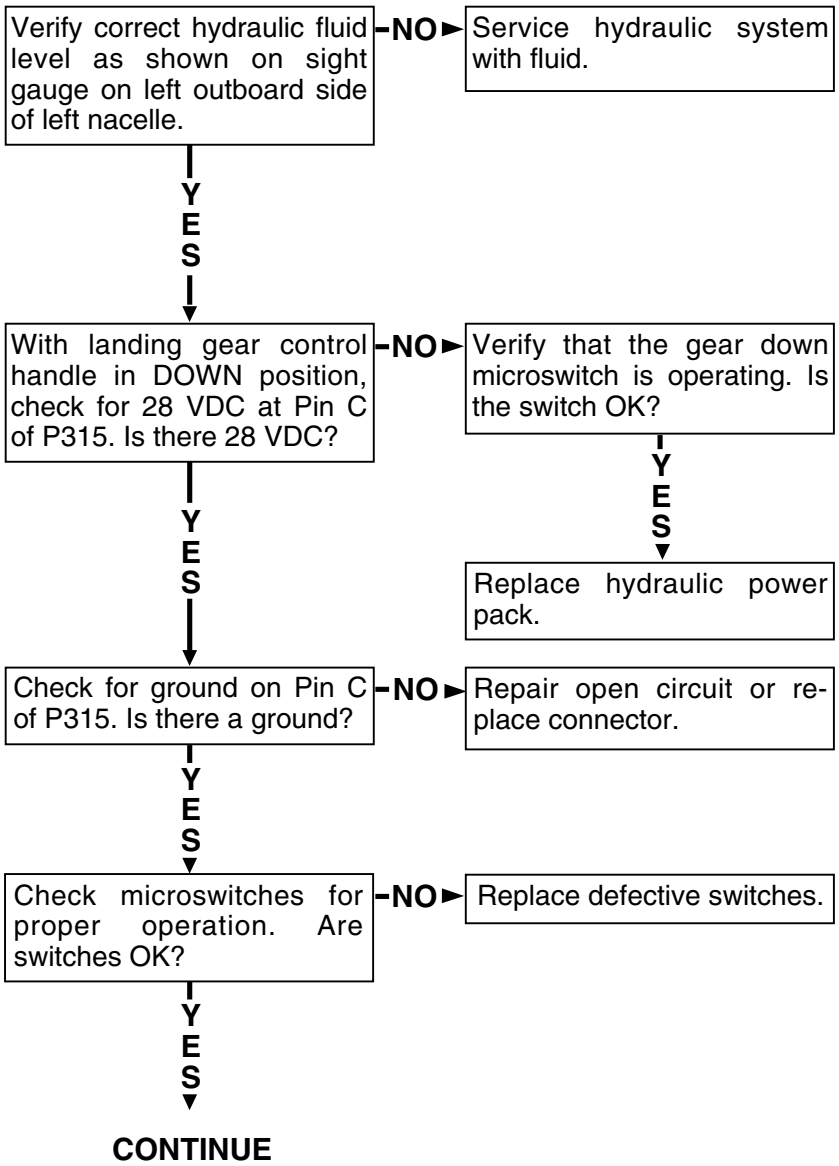
PROBLEM:

LANDING GEAR FAILS TO EXTEND UNDER NORMAL OPERATION

PREREQUISITES:

Aircraft on jacks, emergency gear selector in NORM Position, Landing Gear Control Handle DOWN, Hydraulic and Electrical Power Applied.

START:



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Check landing gear control circuit breaker (CB57 or CB58). Are circuit breakers OK?	–NO►	Replace circuit breakers or repair open circuit.

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END

LANDING GEAR RETRACTION AND EXTENSION – FAULT ISOLATION

PROBLEM:

LANDING GEAR SLOW TO EXTEND

PREREQUISITES:

Emergency gear selector in NORM Position.
Relubricate pivot points.

START:

Check hydraulic fluid level. Is fluid level OK?
- NO ► Service hydraulic system.
- YES ▼

Check engine driven pumps for proper operation and prime. Are pumps OK?
- NO ► Replace or prime engine driven pumps.
- YES ▼

Check for leaks in suction lines. Are there leaks?
- NO ► Repair all leaks and bleed system.
- YES ▼

Check hydraulic actuators for internal leaks. Are actuators OK?
- NO ► Replace or repair faulty actuators.
- YES ▼

Check actuator bellcrank springs for wear and tear. Are they OK?
- NO ► Replace Springs.
- YES ▼

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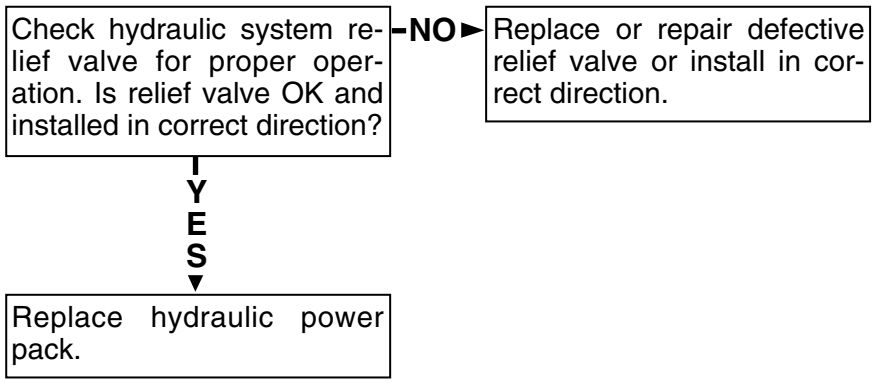

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END

LANDING GEAR RETRACTION AND EXTENSION – FAULT ISOLATION

PROBLEM:

NOSE LANDING GEAR SHIMMIES

PREREQUISITES:

Nose steering actuator disarmed.

START:

Check nose tires for proper balance. Are tires balanced?
- NO ► Balance tires.
- YES ▼

Check nose tires for proper inflation. Are tires properly inflated?
- NO ► Service tires.
- YES ▼

Check nose gear wheel bearings for excessive wear. Are bearings OK?
- NO ► Replace wheel bearings.
- YES ▼

Check nose gear wheel bearing nut for proper torque. is nut tight?
- NO ► Retorque wheel bearing nut.
- YES ▼

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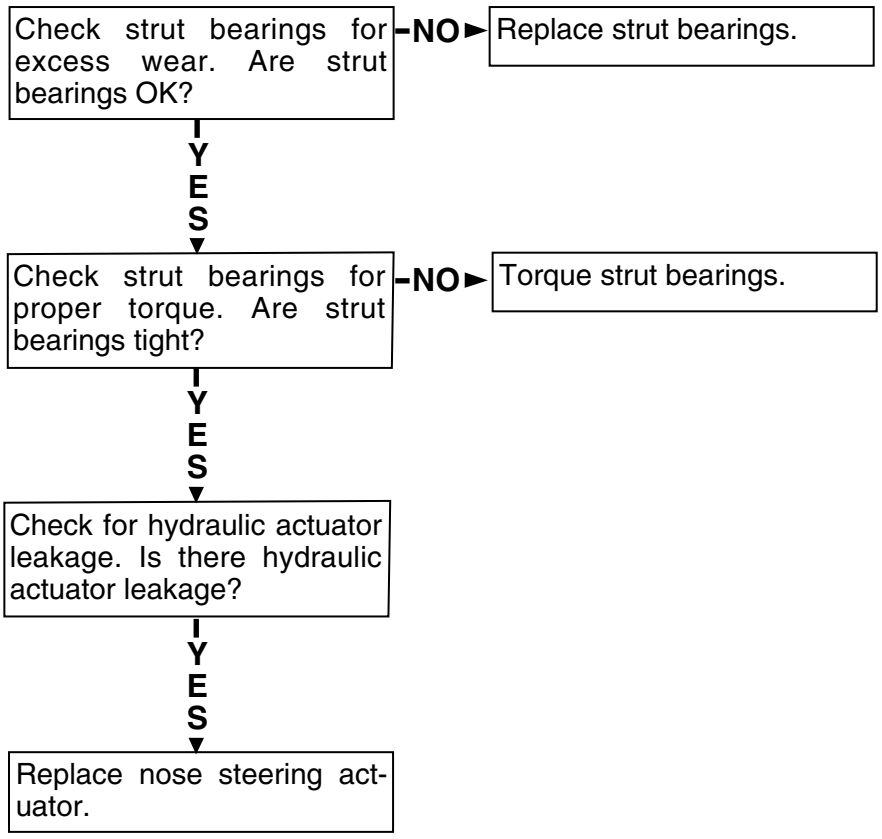

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END

1. Adjustment/Test – Landing Gear Retraction and Extension

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. Rigging – Preparation

(1) Jack aircraft.

(2) Remove access panel left bottom engine nacelle.

(3) Connect hydraulic test stand.

(4) Connect APU.

(5) Pressurize hydraulic system to approximately 2,000 psi (140.6 kg/cm^2).

CAUTION: AFTER PERFORMING A FREE FALL CHECK OR COMPLETION OF ANY MAINTENANCE THAT WOULD ALLOW AIR INTO THE LANDING GEAR HYDRAULIC SYSTEM AND PRIOR TO REMOVING THE JACKS FROM THE AIRCRAFT, CYCLE THE GEAR AT LEAST FOUR TIMES WITH THE NORMAL SYSTEM AND INSPECT ALL THREE LANDING GEAR TO ENSURE THAT THEY ARE DOWN AND LOCKED.

(6) Perform functional test (landing gear).

NOTE: The landing gear should be cycled several times and observed for smoothness of operation, chafing of the flexible lines, and flush fitting of all doors.

CAUTION: TAPE OR WIRE THE LANDING GEAR DOORS TO THE FULL OPEN POSITION WHEN THE ACTUATOR LINKAGE IS DISCONNECTED.

(7) Disconnect the landing gear door actuator linkage.

NOTE: On MLG doors, release quick release catch to open doors.

B. Rigging – Main Gear

(1) Landing gear down and locked, and hydraulic pressure applied to each actuator down side (see Note below). Insert rigging pins through bellcranks into upper drag brace mount rigging holes (if equipped with rigging pin holes) or measure bellcrank roller position (see Figure 201).

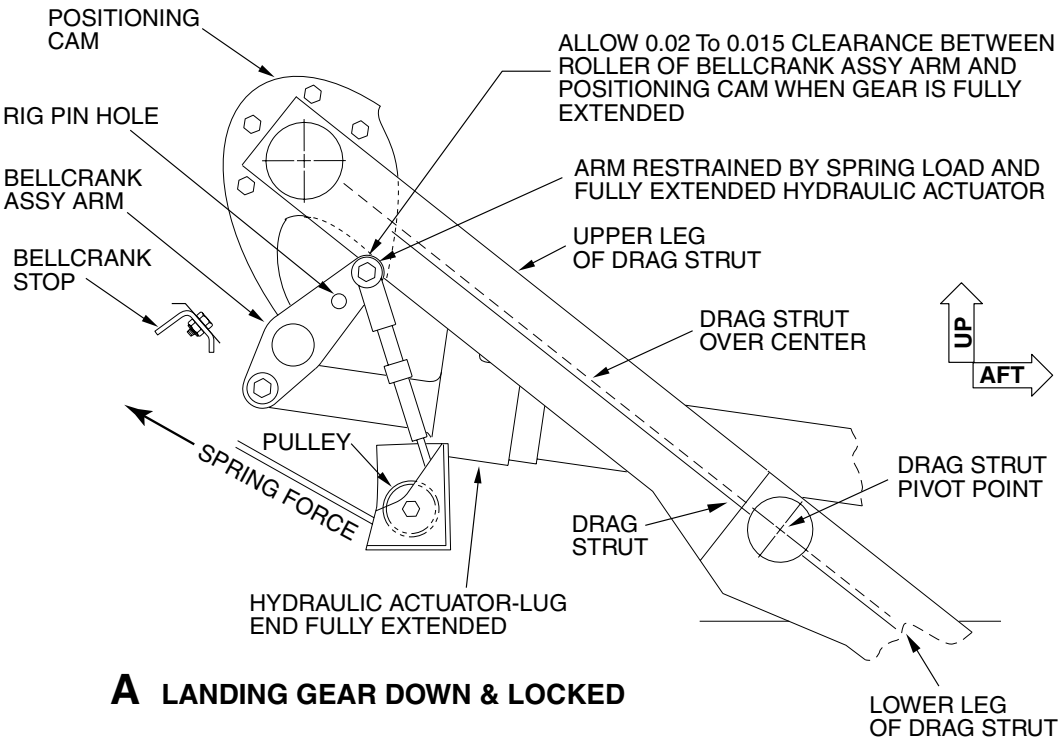
NOTE: The outboard actuators extend using the normal hydraulic system pressure. The emergency hand pump extends the inboard actuators. All actuators retract using the normal hydraulic system pressure.

(2) To adjust the landing gear actuators.

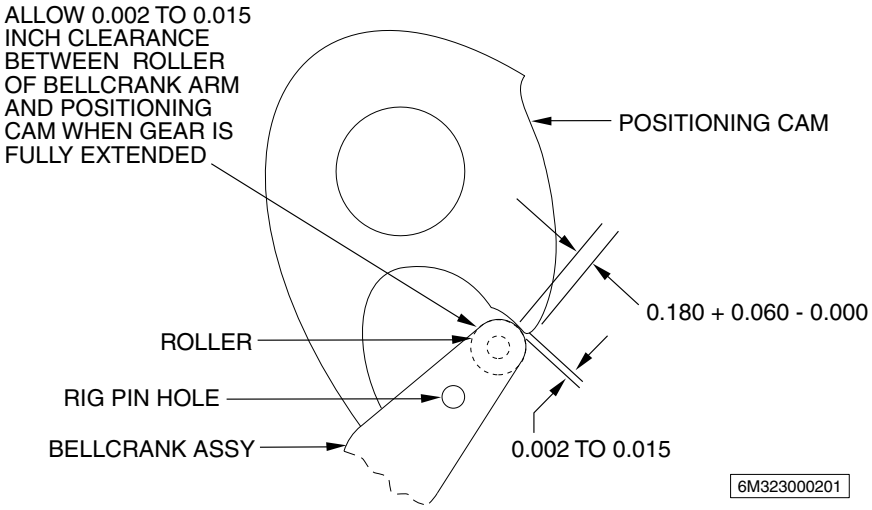
(a) Remove hydraulic pressure from actuators.

(b) Leave rod ends connected to landing gear.

(c) Loosen the rod end jam nut.



A LANDING GEAR DOWN & LOCKED



Main Gear Rigging
Figure 201

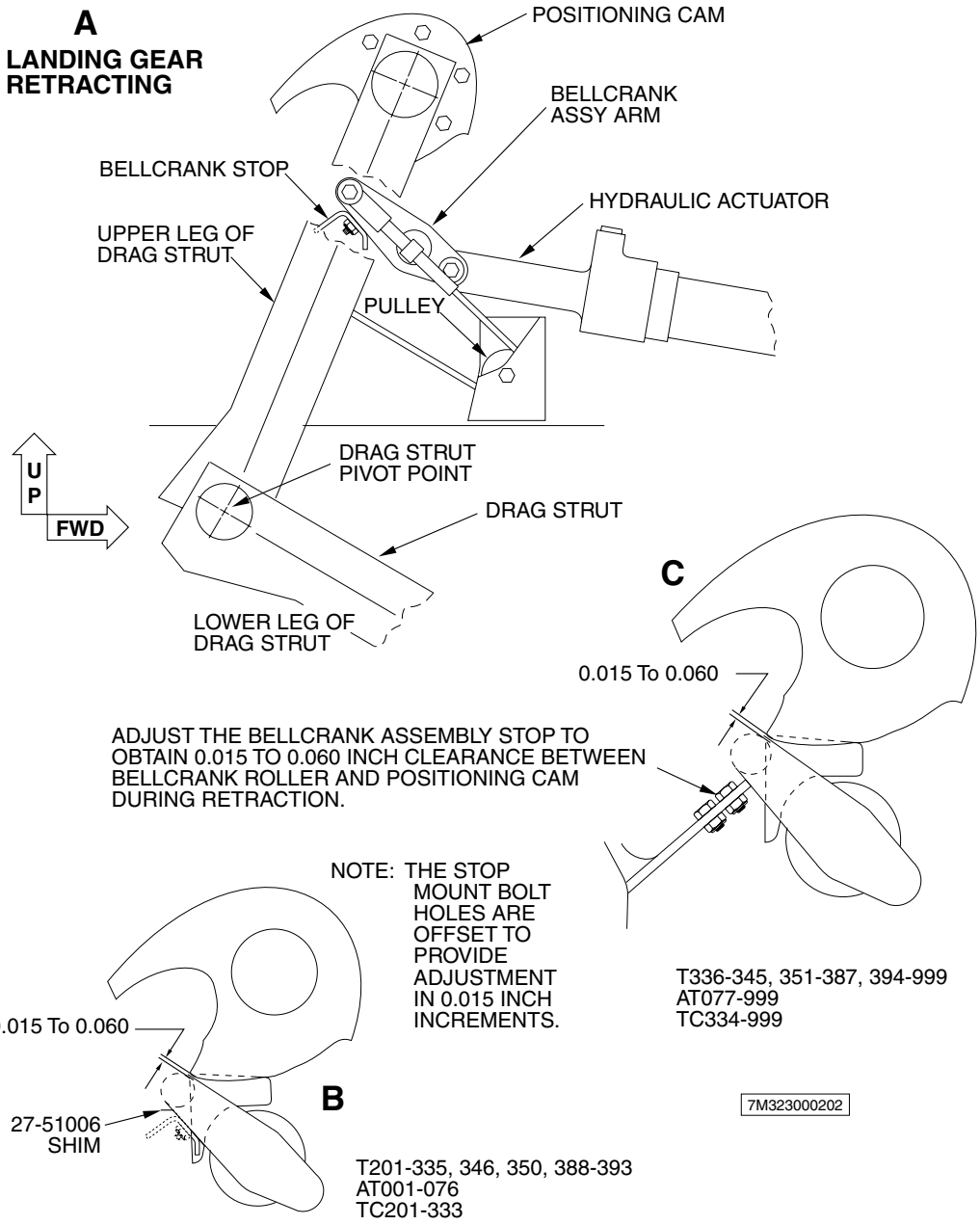
(d) Turn the actuator piston until rigging pin inserts freely, or the desired position of bellcrank roller is obtained (see Figure 201B).

(e) Apply hydraulic pressure to the actuators down side, and recheck rigging.

(f) Lube-torque actuator rod end jam nuts 210 to 240 inch-pounds (23.73 to 27.12 N·m) and safety.

NOTE: One fourth turn of the actuator piston equals 0.036 inch bellcrank movement.

(3) Hold the drag brace knee in the overcentered position (push up on center of drag brace). Clearance between the cam face and bellcrank roller should be 0.002 to 0.015 inches (see Figure 201B).

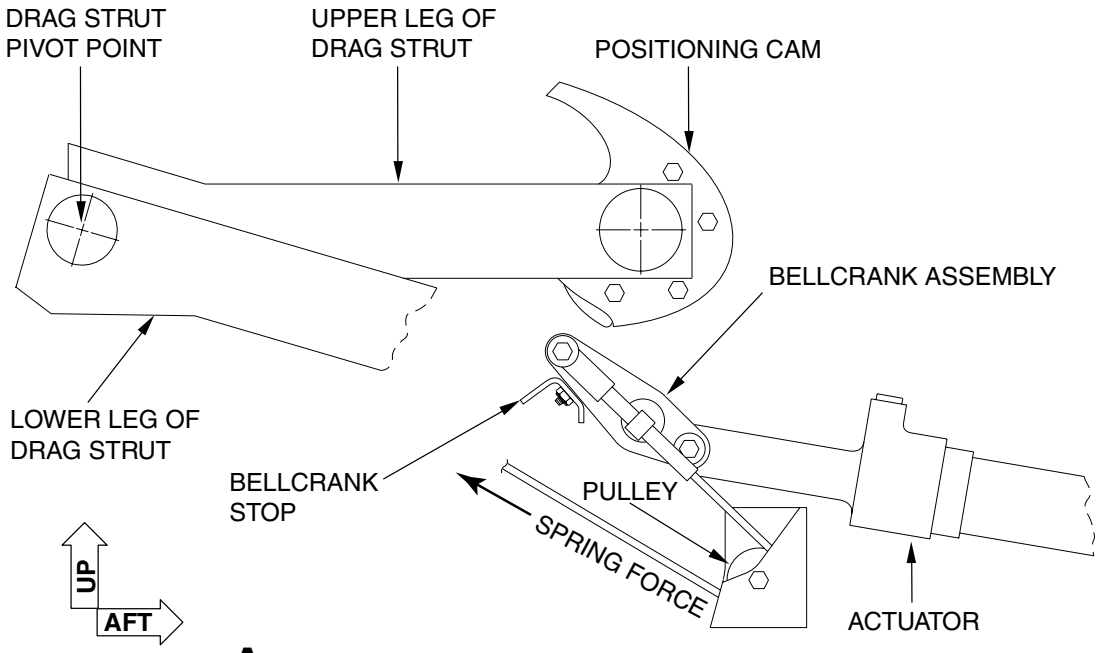


Main Gear Rigging
Figure 202

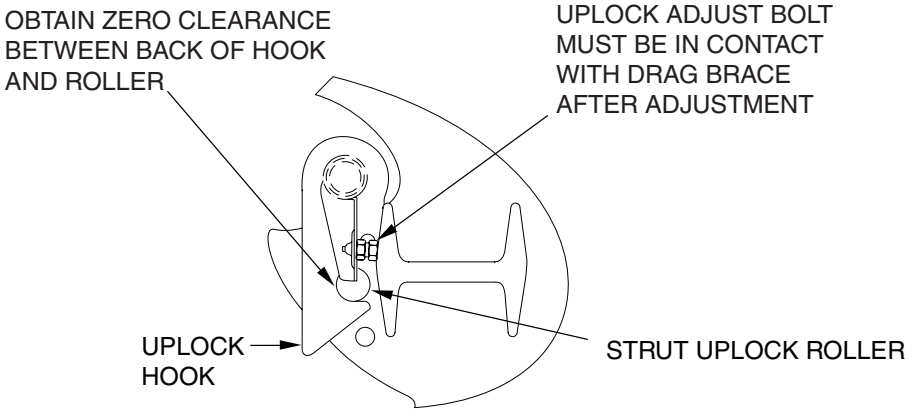
(4) If clearance is not within tolerance.

(a) Check drag brace trunnion pins and bushings for excessive wear. Replace as required.

(b) Check bellcrank pivot bolt and bushing for wear. Replace as required.



A LANDING GEAR - RETRACTING



B

Main Gear Rigging
Figure 203

(c) Check wheel well structure for evidence of distortion, or damage due to hard landing or excessive loads.

NOTE: If minimum clearance is below 0.002 inch, remove minimum material required to achieve 0.002 to 0.005 inch clearance and polish cam face to a smooth, scratch-free surface. (A bright appearance, accomplished with 600-A grit wet or dry corborundum abrasive paper should result in the desired finish). If maximum clearance exceeds 0.015 inch, replace cam.

(5) Retract the gear slowly, stop the gear at point where the positioning cam tip is nearest to the bellcrank roller. Clearance should be 0.015 to 0.060 inches (see Figure 202).

(6) Shim the bellcrank stop (see Figure 202B) or rotate the bellcrank stop (see Figure 202C) to adjust clearance.

NOTE: This adjustment also affects the up position of the gear. The bellcrank stop must be adjusted within the specified clearance and allow the gear to travel to the up position so that the uplock hook fully engages the uplock roller.

(7) With the gear up and locked, support the weight of the gear and adjust the uplock hook adjust bolt to obtain zero clearance between the back of the uplock hook and the uplock roller with no load on the hook (see Figure 203B). The uplock hook adjust bolt must be in contact with the drag brace when the zero clearance is obtained. If proper adjustment cannot be made, check rigging of the emergency release system in this chapter.

(8) Extend the gear and adjust each downlock switch to actuate (click) approximately 7/32 inch before the downlock is fully positioned (see Section 32-60-30).

(9) Retract the gear and remove hydraulic pressure to allow the roller to rest in the hook.

(10) Adjust each switch until the plunger makes contact with the positioning cam, then adjust each switch, with four full turns of the adjustment nut, toward the cam (four turns equal 0.13 inch). Tighten nuts.

NOTE: This is initial adjustment only. Switches may have to be adjusted again during operational check.

(11) Cycle gear several times using normal hydraulic system pressure. Ensure gear operates smoothly. Check for proper cockpit indication.

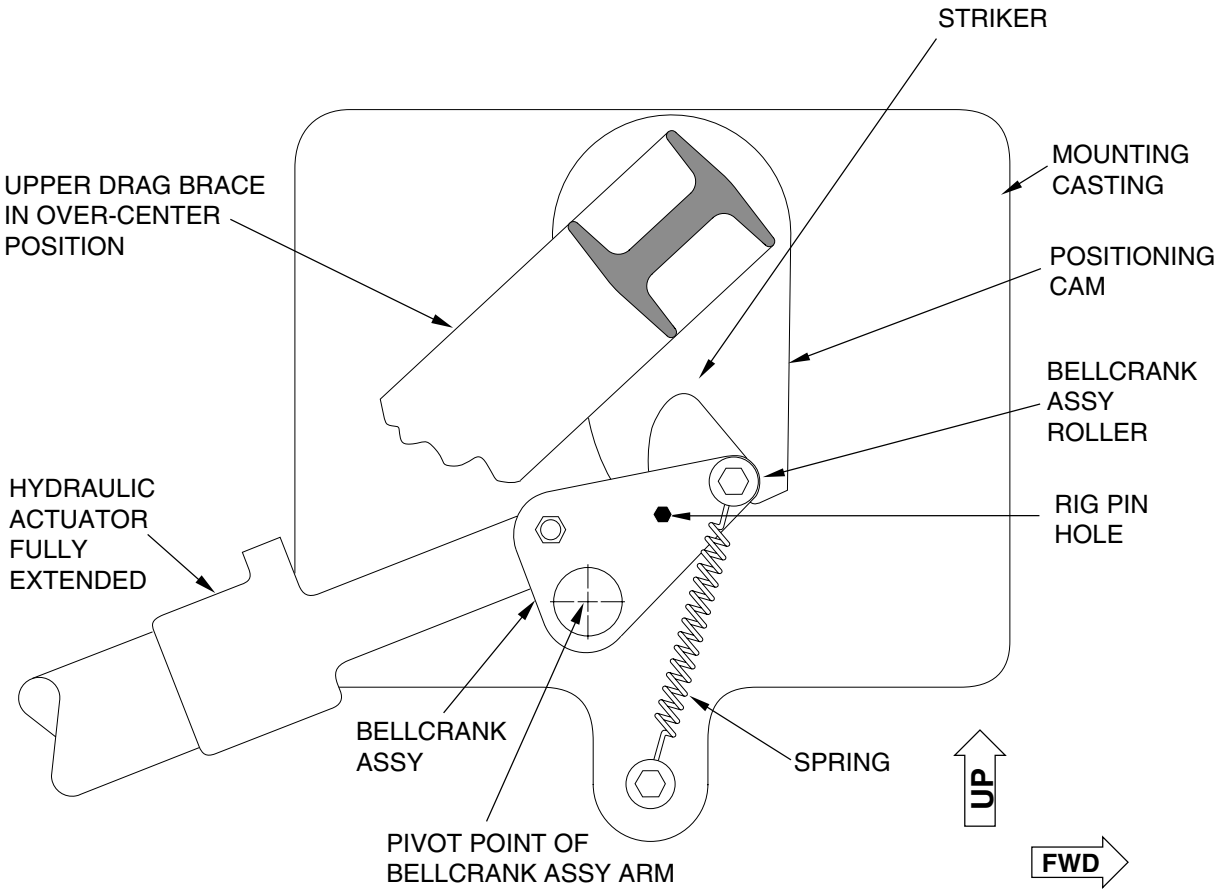
(12) Reconnect the landing gear door actuator linkage.

(13) Cycle the gear and check door for proper fit, flushness, and operation (see Door Rigging, in this Chapter).

(14) Adjust safety switch (outboard switch on the left main landing gear scissors). Switch should operate (click) when strut is compressed 3/4 inch from full extend.

(15) Check landing gear emergency release rigging (see Emergency Release System), IAW step **1.**, G.

C. Nose Landing Gear Rigging



Nose Gear Rigging
Figure 204

(1) Landing gear down and locked, hydraulic pressure applied to each actuator down side (see Note below). Insert rigging pins through the bellcranks into the keelson web rigging holes (if equipped with rig pin holes) or measure bellcrank roller position (see Figure 205).

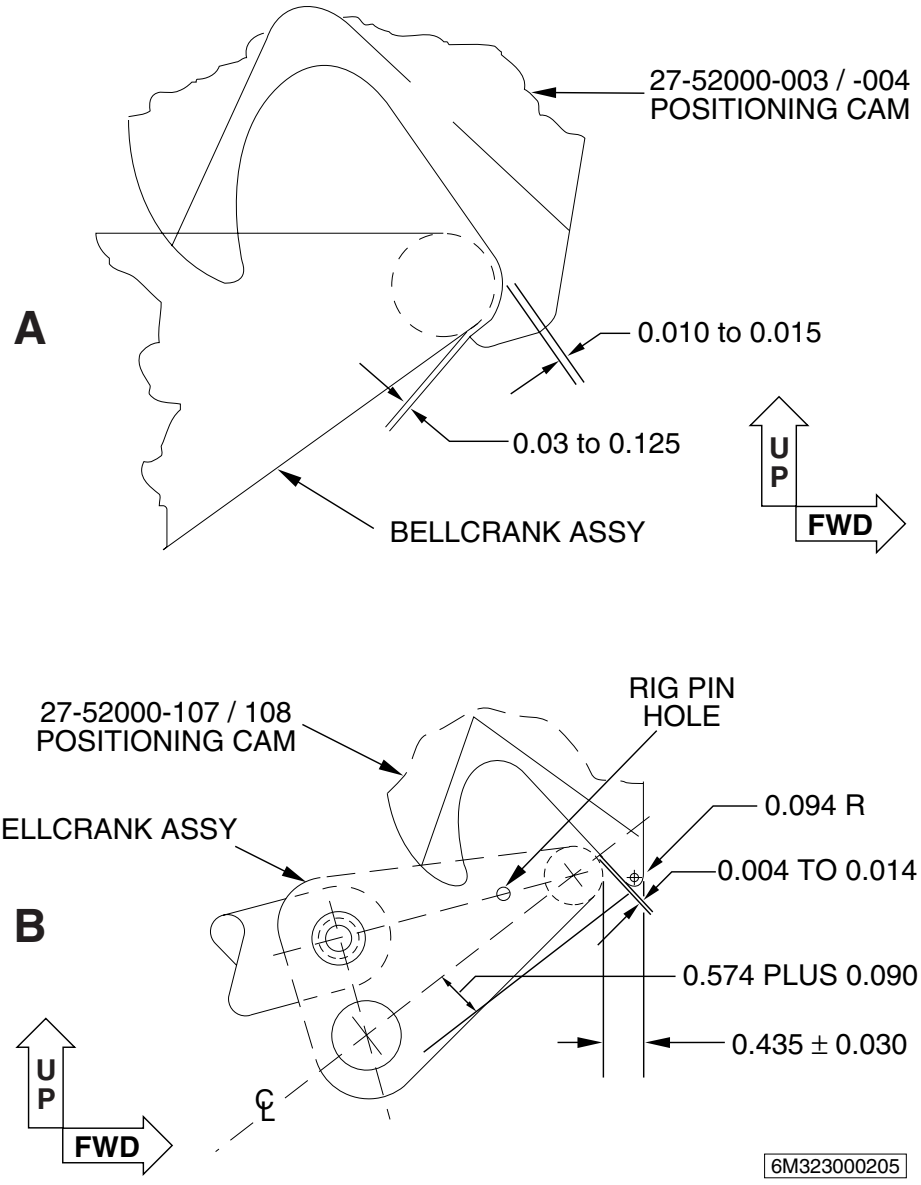
NOTE: The left actuator extends using the normal hydraulic system pressure. The emergency hand pump extends the right actuator. Both actuators retract using the normal hydraulic system pressure.

(2) To adjust landing gear actuators.

(a) Remove hydraulic pressure from actuators.

(b) Leave rod end connected to the gear.

(c) Loosen rod end jam nut.



Nose Gear Rigging
Figure 205

(d) Turn actuator piston until rigging pin inserts freely or the specified position of the bellcrank roller is obtained.

1 Aircraft S/N's T201-310, AT001-073, and TC201-999 with positioning cam P/N 27-52000-003/004 use position specified in Figure 205A.

2 Aircraft S/N's T311-999, AT074-999, and TC300-999 with positioning cam P/N 27-52000-107/108 use position specified in Figure 205B.

(e) Apply hydraulic pressure to actuators down side, recheck rigging.

(f) Lube-torque actuator rod end jam nut 210 to 240 inch-pounds (23.73 to 27.12 N·m) and safety.

NOTE: One-fourth turn of the actuator piston equals .040 inch bellcrank movement.

(3) Hold drag brace knee in the overcentered position (pull down on center of the drag brace). Clearance between the positioning cam face and bellcrank roller should be as specified in Figure 205, detail A or B.

(4) If clearance is not within tolerance.

(a) Check drag brace trunnion pins and bushings for excessive wear. Replace as required.

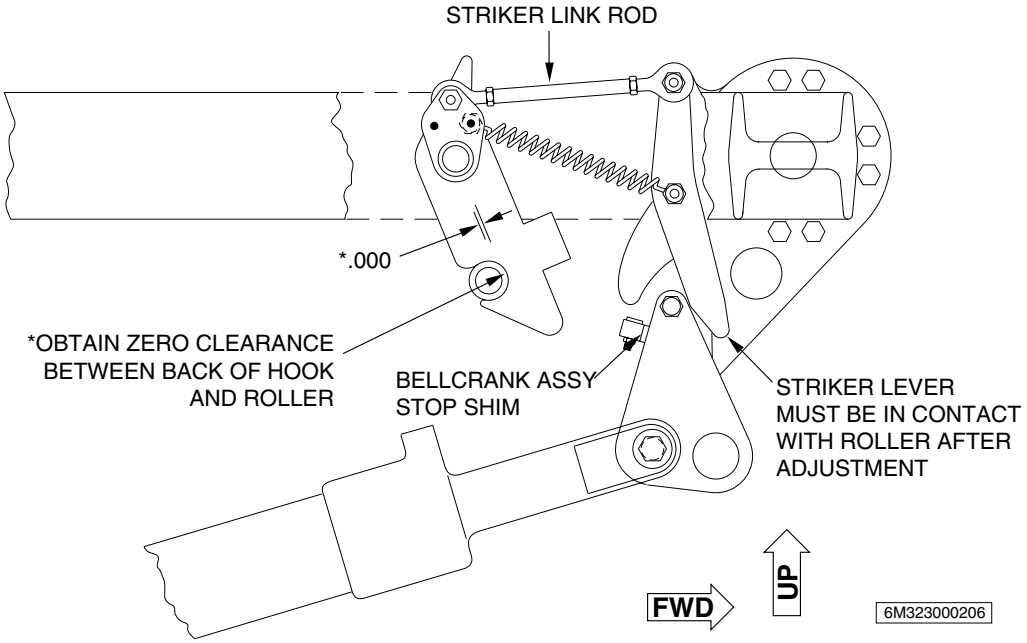
(b) Check bellcrank pivot bolts and bushings for excessive wear. Replace as required.

(c) Check wheel well structure for evidence of distortion, or damage due to hard landing or excessive loads.

NOTE: If clearance is below minimum, remove minimum material required to achieve minimum clearance and polish cam face to a smooth, scratch-free surface. A bright appearance, accomplished with 600-A grit wet or dry corborundum abrasive paper should result in the desired finish. If clearance exceeds maximum specified, replace cam.

(5) Retract the landing gear. Check that the gear travels to the up position so that the uplock hook fully engages uplock roller.

(6) If necessary, shim the bellcrank stops as required to adjust the gear to uplock position. Do not install more shims than required to allow the uplock hook to fully engage the uplock roller.



Nose Gear Rigging
Figure 206

(7) With the gear up and locked, support the weight of the gear and adjust striker link rods to obtain zero clearance between back of uplock hook and uplock roller (see Figure 206). The striker levers must be in contact with the bellcrank rollers after this adjustment. If proper adjustment cannot be made, check rigging of the emergency release system, in this chapter.

(8) Extend the gear and adjust each down lock switch to actuate (click) approximately 7/32 inch before the down lock is fully positioned (see Section 32-60-30).

(9) Retract the gear and remove the hydraulic pressure to allow the roller to rest in the hook.

(10) Adjust each switch until the plunger makes contact with the positioning cam, then adjust each switch with four full turns of the adjustment nut, toward the cam (4 turns equal 0.13 inch). Tighten nuts.

NOTE: This is initial adjustment only. Switches may have to be adjusted again during operational check.

(11) Cycle the gear several times using normal hydraulic system pressure. Ensure gear operates smoothly. Check for proper cockpit indication.

(12) Reconnect the landing gear door actuator linkage.

(13) Cycle the gear and check door for proper fit, flushness and operation (see Door Rigging, in this Chapter.

(14) Check landing gear emergency release rigging (see Emergency Release System), IAW step **1.**, G.

D. Main Gear Door Rigging – Landing Gear Door Rigging

NOTE: After door installation or maintenance on doors or door operating components, accomplish the following rigging procedures prior to other operations of the landing gear.

(1) Support the aircraft on jacks.

(2) Disconnect the landing gear door linkage at the push-pull rod to swivel fitting connection.

NOTE: Tape or wire the landing gear doors in the full open position when the door actuating linkage is disconnected, to prevent possible damage due to contact during gear movement.

CAUTION: DURING THE FOLLOWING PROCEDURE, OPERATE THE LANDING GEAR BY MEANS WHICH WILL PROVIDE SLOW TRAVEL WHICH CAN BE STOPPED AT ANY TIME. OBSERVE ALL MOVEMENT OF THE LANDING GEAR AND STOP MOVEMENT IF NECESSARY TO AVOID DAMAGE DUE TO CONTACT WITH THE DOOR COMPONENTS.

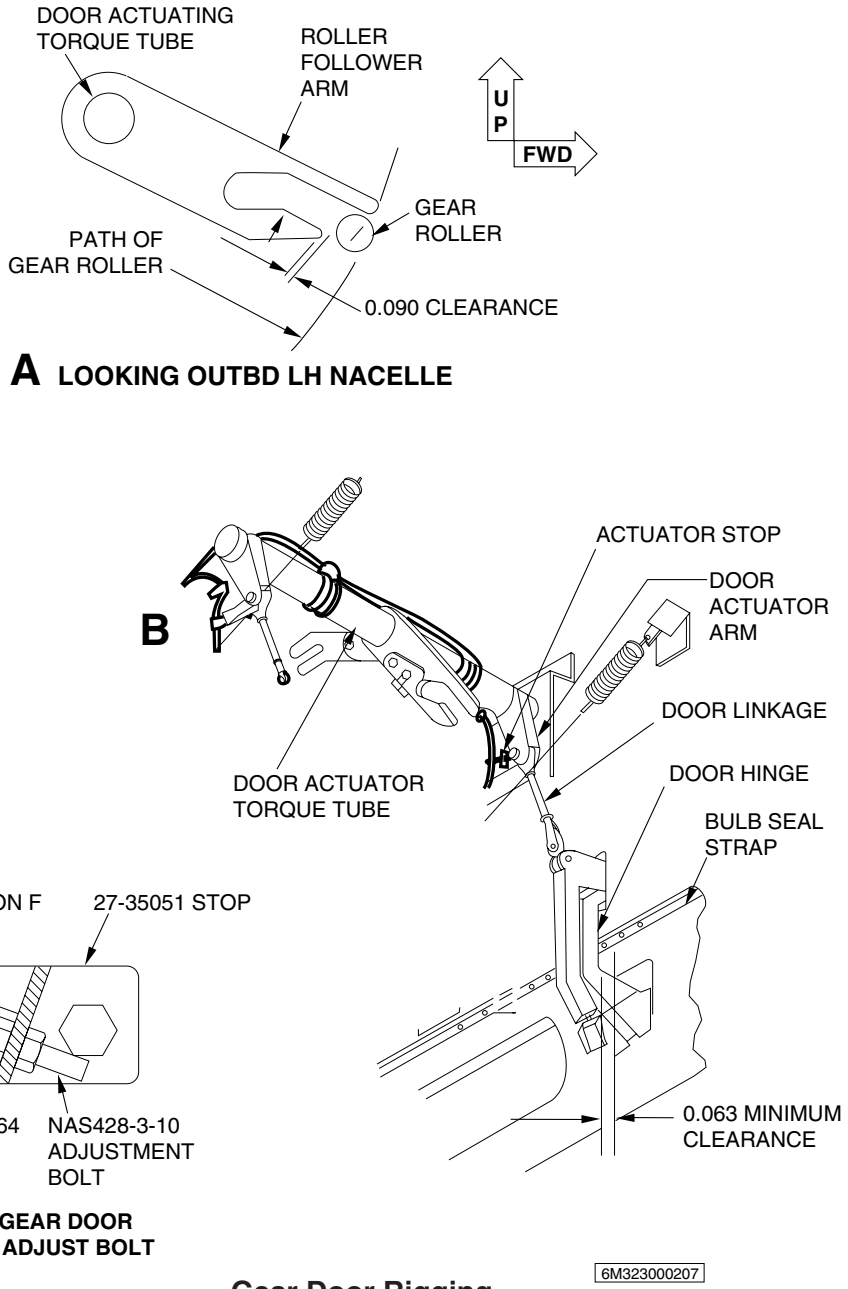
(3) Partially retract landing gear. Verify minimum clearance of 0.090 inch is maintained between lower lip of roller follower arm of door actuator and gear roller (see Detail A, Figure 207) when roller passes lower lip at nearest point.

NOTE: If 0.090 inch clearance is not correct, ensure that component wear or other rigging problems are not the cause.

(4) If adjustment is required, remove the door actuator stop bolt retaining jacket (see Figure 207). Adjust both the inboard and outboard stop bolts against the door actuator arms to obtain the 0.090 inch minimum clearance.

CAUTION: WHEN RETRACTING GEAR, DO NOT ALLOW GEAR DOORS TO MAKE CONTACT NOR GO OVER CENTER IN THE CLOSED POSITION.

(5) Fully retract the landing gear. With weight of gear resting on the uplock hook, close each door individually and adjust the door actuating linkage to obtain a flush fit.



Gear Door Rigging
Figure 207

(6) Extend landing gear. With gear door actuator arms against stop bolts, connect door actuator linkage and verify minimum clearance of 0.063 inch between door hinge and bulb seal retaining strap along keelson (see Figure 207).

NOTE: If adjustment is required, adjust the stop bolt against the actuator arms. Recheck the 0.090 clearance, step **1.**, D., 3.

(7) Retract the landing gear.

(a) Inspect each door in closed position to ensure doors fit flush and mate with, but do not contact, each other at door parting line.

(b) Ensure that forward edge of each door is 0.0625 inch aft of nacelle removable panel.

(c) Check doors for proper fit of the rubber seals.

(8) Extend landing gear. Ensure that door actuator travels far enough aft to be over center and doors cannot be closed without lifting door actuator from overcenter position.

(9) Cycle landing gear several times to ensure smoothness of operation and proper cockpit indication.

(10) Install all hardware and safety. Install door stop bolt retaining jacket and apply Devcon F. Ensure bolt head is free of Devcon F within striking surface of door actuator arms.

E. Nose Gear Door Rigging – Landing Gear Door Rigging

NOTE: After door installation or maintenance on doors or door operating components, accomplish the following rigging procedures prior to other operations of the landing gear.

(1) Support the aircraft on jacks.

(2) Disconnect the landing gear door linkage.

NOTE: Tape or wire the landing gear door in the full open position when the door actuating linkage is disconnected, to prevent possible damage due to contact during gear movement.

CAUTION: DURING THE FOLLOWING PROCEDURES, OPERATE THE LANDING GEAR BY MEANS WHICH WILL PROVIDE SLOW TRAVEL WHICH CAN BE STOPPED AT ANY TIME. OBSERVE ALL MOVEMENT OF THE LANDING GEAR AND STOP MOVEMENT IF NECESSARY TO AVOID DAMAGE DUE TO CONTACT WITH THE DOOR COMPONENTS.

(3) Partially retract the landing gear. Verify minimum clearance of 0.090 inch is maintained between lower lip of roller follower arm of door actuator and gear roller (see Detail A, Figure 207). when roller passes lower lip at nearest point.

NOTE: The nose gear door actuator stops are not adjustable. If clearance is not obtained, check strut servicing, and door actuator for wear or damage.

CAUTION: WHEN RETRACTING GEAR, DO NOT ALLOW GEAR DOORS TO MAKE CONTACT NOR GO OVER CENTER IN THE CLOSED POSITION.

(4) Fully retract landing gear. With weight of gear resting on uplock hook, close each door individually and adjust door actuating linkage to obtain flush fit.

(5) Extend landing gear. With gear door actuator arms against stop bolts, connect door actuator linkage and verify minimum clearance of 0.063 inch between door hinge and bulb seal retaining strap along keelson (see Figure 207).

(6) Retract the landing gear.

(a) Inspect each door in closed position to ensure doors fit flush and mate with, but do not contact, each other at door parting line.

(b) Ensure that forward edge of each door is 0.0625 inch aft of nose skin.

(c) Check doors for proper fit of rubber seals.

(7) Extend landing gear. Ensure that door actuator travels far enough aft to be over center and doors cannot be closed without lifting door actuator from overcenter position.

(8) Cycle landing gear several times to ensure smoothness of operation and proper cockpit indication.

(9) Ensure all hardware is replaced and safetied.

F. Operational Check – Emergency Mode

(1) Place the emergency gear extension handle in the normal position.

(2) Ensure that all gears are down and locked.

(3) Place emergency shutoff valve in the emergency position.

(4) Starting with zero system pressure, use the hand pump to raise the pressure to 1,000 psi as indicated on the panel gauge.

(5) Check the bleed down rate. If the pressure diminished to 700 psi or less in less than five minutes time, the system is unsatisfactory.

(6) With the emergency gear extension handle in the EMERGENCY position, emergency gear shutoff valve in EMERGENCY position, and 700 psi on the indicator, apply apply 2,000 psi to the system with the auxiliary hydraulic unit. Position the pedestal gear actuating handle to the gear up position. Gauge pressure will increase to 2,000 psi and the landing gear may or may not unlock, but it should not retract in either case.

(7) Return all emergency handles to the normal position and retract the gear.

(8) With gears up and locked and hydraulic pressure off and reduced to zero, place the emergency gear shutoff valve in the emergency position. Rotate emergency extension handle counterclockwise (gears should release).

(9) Build up pressure with the hand pump. Gears should extend to the down and locked position. Check for proper cockpit indication.

(10 Return all emergency valves to the normal position and cycle gear in a normal manner several times to bleed any trapped air from system.

G. Emergency Release System – Functional Test

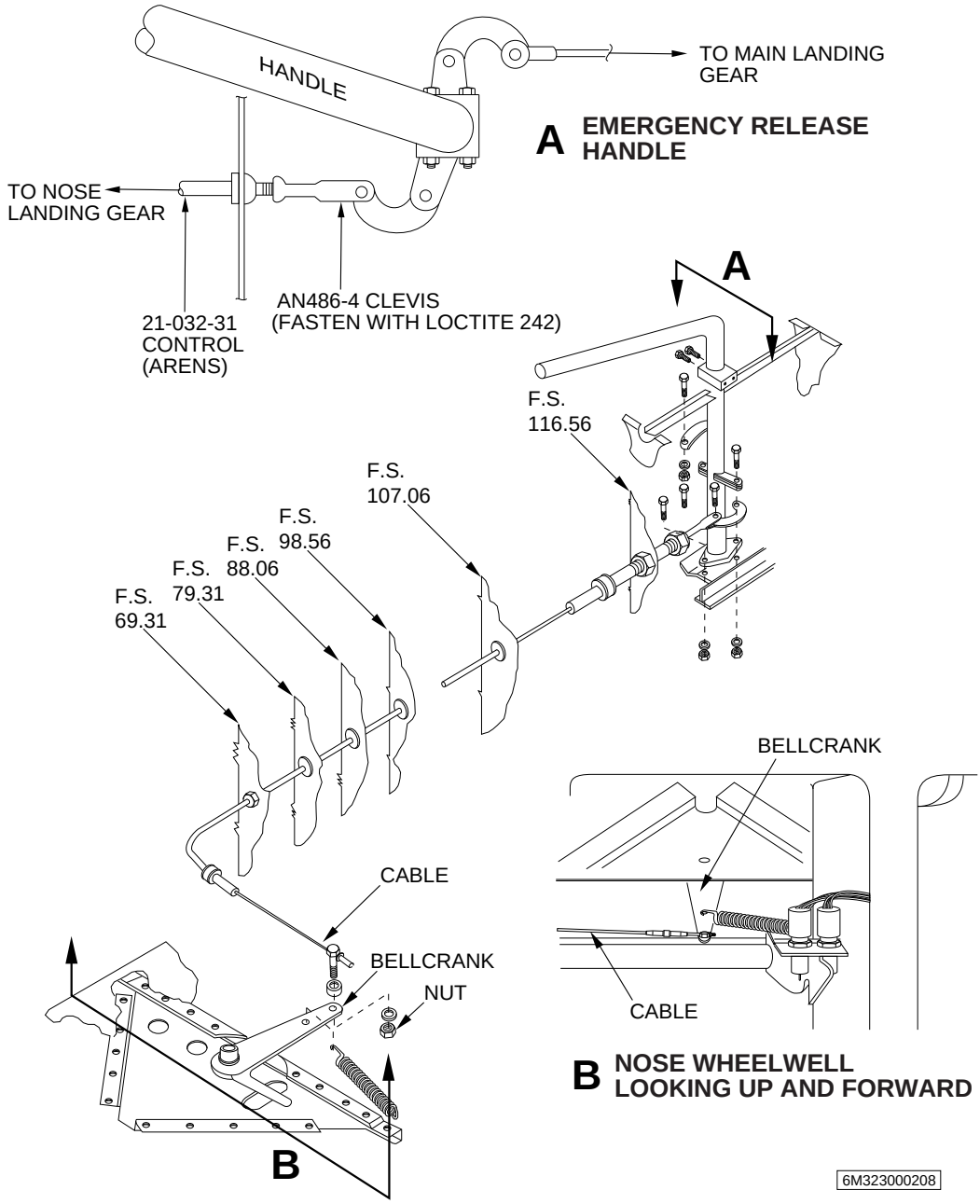
(1) Check landing gear free fall on jacks as follows:

(a) Place gear selector up with hydraulic pressure on. Retract gear and leave gear selector in up position with hydraulic pressure on.

(b) Rotate emergency gear extension handle slowly rearward (counterclockwise looking down).

(c) Observe that the gear uplocks release before 45 degrees of manual release handle travel.

(d) Observe that the hydraulic power pack shuts off prior to or simultaneous with the gear lock release (by-pass valve actuates).



Emergency Extension System
Figure 208

(e) Rig the main gear locks to release both main gear simultaneously to allow symmetrical extension (see Adjustment-Emergency Release, in this section.

(f) Observe that the emergency gear handle mechanism goes overcenter and remains in the maximum travel position.

(g) Place gear selector to down position, turn off hydraulic pressure and return all emergency handles to normal position.

(h) Apply hydraulic pressure and retract the gear with the gear selector handle.

(i) Remove the hydraulic pressure and place the gear selector handle to the down position.

(j) Perform or check for compliance with items **1.**, G., (b) through (f).

(k) Functional check landing gear free fall.

NOTE: If, during free fall, any of the three gears do not fall into the down and locked condition, simulate an airload by applying not more than 25 pounds of force at the lower end of the strut (between the wheels) pulling rearward. If more than 25 pounds of force is required to get the gear down and locked, problems exist which must be identified and eliminated.

H. Adjustment – Emergency Release System

(1) Remove the right cabin floorboards (between station 154.03 and 171.406 on series SA226 – T and at station 254.521 on series SA226 – AT and TC to adjust the cable tension between the actuating arm and release levers to each main gear.

(2) Retract the landing gear.

CAUTION: DO NOT TENSION CABLES SO AS TO INTERFERE WITH THE UPLOCK HOOK RIGGING (SEE LANDING GEAR RIGGING).

(3) Adjust the cable tension to obtain a positive contact between the actuating arm and the release lever on each gear.

(4) Remove the access panel inside the left main gear well on the inboard side for the bypass valve turnbuckle.

(5) With the bypass valve in normal position, adjust the cable tension to provide proper operation.

(6) Safety turnbuckles after adjustment has been made.

NOTE: On SA226 series aircraft, T – 228 through T – 999, AT – 009 through AT – 999, and TC – 205 through TC – 999, a shielded cable system is installed replacing the cables and pulley arrangement from the emergency release handle to the nose gear uplock lever. Adjustment can be made either the clevis, attached to the control to the emergency release handle (see Figure 208), or at the bellcrank assembly within the nose landing gear wheel well.

(7) Fasten the AN486-4 clevis (see Figure 208) with loctite #242. Once loctite has been applied, adjustment at this point may be difficult, therefore, adjustment can be made at the bellcrank in the nose wheel well.

(a) Loosen the nut to the bolt securing the control assembly cable to the bellcrank assembly.

(b) Remove slack from the cable by pulling the cable taut. Tighten the nut to the bolt securing the control assembly cable to the bellcrank assembly.

(8) Perform functional test IAW step **1.**, G.

(9) Install access panels and floorboards.

WHEELS AND TIRES – DESCRIPTION AND OPERATION

1. General

A. SA226 series aircraft use dual wheel assemblies on each landing gear. Wheel assemblies are split type used for tube and tubeless tires. Split wheel assemblies require careful work and strict attention to safety details.

1. Removal/Installation – Wheels and Tires

A. Removal – Main Landing Gear Wheel

(1) Jack aircraft.

(2) Deflate tire.

WARNING: EXERCISE CARE WHEN REMOVING VALVE CORE FROM VALVE STEM (VALVE CORE SHOULD BE REMOVED TO ENSURE COMPLETE DEFLATION).

(3) Remove cotter key, keyed spacer, spacer, and axle nut.

(4) Remove outer seal and cone assembly.

(5) Remove wheel and tire assembly. Remove inner seal and bearing cone assembly.

B. Wheel Balance

(1) It is important that aircraft wheels be well balanced. If a wheel is heavier on one side than the other the heavy side will remain down so that it always strikes the runway first in landing. This results in severe wear on just one section of the tire. Unbalanced wheels can also produce vibration when an aircraft is moving on the ground.

(2) Balance marks are placed on many inner tubes to indicate the heavy portion of the tube. These marks are often colored strips about 1/2 inch wide by two inches long. When a tube is installed its balance marks should be located next to the light side balance mark of the tire (often a colored dot). If the tube has no balance mark place its valve next to the tire balance mark. When mounting tubeless tires the balance mark on the tire is normally placed in line with the wheel valve unless otherwise specified by the manufacturer.

(3) With some split wheels, the light side of the wheel halves are marked with an L stamped on the flange. In assembling these wheels, position the L's 180° apart.

(4) Should additional dynamic or static balancing be required after tire mounting, accessory balancing weights should be installed around the circumference of the flange.

C. Installation – Main Landing Gear Wheel

(1) Mount wheel, tire, and bearing assembly onto axle. Ensure that brake disc lugs engage torque key slots of inner wheel half of wheel assembly.

(2) Install spacer, keyed spacer, and axle nut. Torque axle nut to 32 foot pounds while rotating wheel to ensure proper seating of bearings.

(3) Back axle nut off to zero torque. While rotating wheel retighten to 16 foot-pounds in one continuous rotation of axle nut.

(4) Check for alignment of locking holes. If holes are not aligned, continue tightening axle nut to first locking hole and secure with cotter key.

2. Inspection/Check – Wheels and Tires

A. Inspection – Wheels and Tires

(1) Inspect wheel half flanges for cracks, corrosion, and deformations.

(2) Check tire for proper inflation and condition.

(3) Check brake assemblies and hoses for evidence of over-heating or leakage and security. Inspect torque keys for looseness or wear (main wheels).

(4) Check wheel retaining nut for condition and safety.

(5) Inspect anti-skid hubcap and wiring for condition and security (if equipped).

B. Each time wheel is removed:

(1) Clean and repack wheel bearings.

TIRES – DESCRIPTION AND OPERATION

1. General

A. Due to various customer operational requirements, several main and nose wheel tire combinations have been installed on the SA226 series aircraft. These combinations consist of different tire sizes and ply ratings. For standard installation or customer option installations and aircraft effectivities, refer to Table 201.

B. Effect of Temperature

(1) The tire develops an internal temperature rise as it rolls along the ground under load. Hot spots can occur to the tire because rubber is a poor conductor of heat. These hot spot temperatures will occur dependent on the duration of the roll and the speed or RPM of the tire.

(2) High temperature rise can shorten the life of the tire because materials now in use in fabricating tires are affected by temperatures. The temperature rise in the tire can also weaken the strength of the tread rubber and its adhesion to the tire carcass. Tread wear increases directly proportionate with rise in temperature.

C. Ambient Temperature

(1) The temperature in the immediate vicinity of the tire which can be affected by such things as proximity to engines or brake heat.

D. Heat Soak

(1) The most serious problem facing the aircraft tire is the effect of temperature rise due to the air friction of high-speed. The duration of exposure to high temperature is an important factor. Since rubber is a poor heat conductor, if a tire starts a flight at a low temperature, it will take some time for the external heat to penetrate (heat soak) to critical parts of the tire structure. Critical parts are at the tire shoulders, immediately above the rim flanges, and at the approximate center of the tire bulk. Obviously, the shorter the duration of the heat soak, the higher ambient temperature the tire will withstand.

E. Base Temperature

(1) The temperature where flight operations are most often conducted. In situations where aircraft may be flown in extreme varying temperature changes, consideration should be given towards initial ground inflation of tires. Except where the aircraft is in transit for long periods, base temperature is to be used to determine ambient temperature for normal tire inflation procedures.

F. Changes in Pressure Versus Temperature

(1) Use Figure 201 to determine pressure adjustments for tires where flight operations expose the tires to varying ambient air temperatures. In certain conditions the range of ambient air temperature can become so adverse that it can seriously affect tire performance unless tire pressure adjustment is made.

(2) Figure 201 begins by using an assumed base temperature of 70°F. This chart may, of course, be used for any other base temperature dependent upon the base temperature at the users base of flight operation.

(3) The chart should not be used for transient temperature changes but only in situations whereas complete heat soak occurs in which the contained air of the tire has reached the ambient outside temperature. Example: In the wheel well of an aircraft in flight, the ambient temperature of a nose wheel tire has decreased to -50°F. The tire was previously inflated to 70 psi on the ground in an ambient temperature of 70°F.

(4) On the chart, mark the 70 psi point (1) on the vertical 70°F line (2). Through this point, draw a line to the left (3) paralleling the pressure change line (4) already depicted on the chart. This new line intersects the -50°F vertical line (5) at 51 psi (6). This is what the tire inflation pressure will become after completed exposure to -50°F. To compensate for this pressure loss and to determine what the original ground inflation should have been, reverse the preceding procedure. On the -50°F vertical line (5), mark the 70 psi point (7). Draw a line to the right (8) parallel to the pressure change line (9) until it intersects (10) the ground temperature 70°F vertical line.

FACTORY STANDARD INSTALLATION

HIGH PRESSURE TIRES						
TIRE SIZE	PLY RATING	GEAR	TIRE PRESSURE (PSI)	EFFECTIVITY T	AT	TC
18 X 5.5	8	Main	105	201-397	001-074	201-419
18 X 5.5	10	Main	140	CUSTOMER OPTION		
16 X 4.4	6	Nose	80	201-397	–––	–––
16 X 4.4	6	Nose	70	–––	001-074	201-397
LOW PRESSURE TIRES						
19.5 X 6.75-8	10	Main	62	201-417	–––	–––
			70	–––	001-074	210-397
			75	–––	–––	398-419
18 X 4.4	6	Nose	70	201-417	–––	–––
			46	–––	001-074	201-397
			50	–––	–––	398-419
18 X 4.4 (Chine)	10	Nose	50	–––	001-074	201-397
			54	–––	–––	398-419

NOTE: Main tires other than 18 X 5.5 and nose tires other than 16 X 4.4 are not compatible with above standard installations. Modification of aircraft structure is required prior to installation. Reference Fairchild Aircraft Corporation Service Bulletin 32-007.

Aircraft originally equipped with low pressure tires may use high pressure tires except for Metro IIA aircraft which must use low pressure tires.

Assumed base temperature of 70°F

CAUTION: DO NOT MIX PLY RATINGS BETWEEN TIRES ON THE NOSE GEAR OR BETWEEN TIRES ON THE MAIN GEAR.

ALUMINUM AND MAGNESIUM MAIN GEAR WHEELS MAY BE INTERMIXED ON AN AIRCRAFT. HOWEVER, TUBE AND TUBLESS TIRES MAY NOT BE INTERMIXED.

Tire Specification
Table 201

TIRES – MAINTENANCE PRACTICES

1. Servicing – Tires

A. Tools and accessories

NOTE: Mounting, dismounting, and maintaining aircraft wheels and tires is a specialized job that is best done with special equipment. Some of the more widely used special tools and accessories are:

(1) Tire tread gauges.

(2) Remote inflator – for remote inflation of newly mounted high pressure tires.

(3) Valve tools – core keys, repair tool, deflator cap, broken valve remover, and tube valve retrieval tube.

(4) Valve extensions.

(5) Pressure gauges.

(6) Rubber mallets.

(7) Tire levers.

(8) Bead spreaders – for inspecting inside of tires.

(9) Tire and wheel balancing machine (available from many manufacturers).

(10) Tubeless tire bead press (generally custom made to individual needs).

B. Tire Change Precautions

(1) Check tires for proper inflation and condition.

(2) Inspect for cracks and tread wear when inflating tires.

NOTE: Proper inflation will help avoid damage during landing shock or contact with ground ruts, prevent cuts from sharp stones, and minimizes tread wear.

(3) Always deflate a tire before removing it from an aircraft. This eliminates the danger of an accident during handling or dismounting an inflated tire.

(4) Use caution when unscrewing valve core at full tire pressure. Use a deflator cap or release most of the pressure before removing the valve core.

WARNING: PLACE TIRE IN A SAFETY CAGE BEFORE INFLATING. THIS WILL PREVENT SERIOUS INJURY IN CASE THE TIRE OR WHEEL SEPARATES DUE TO INCORRECT MOUNTING OR HIDDEN DEFECTS.

(5) When first inflating a freshly mounted tire, use a clip–on chuck and an extension base.

(6) Use dry air or nitrogen to service tires.

NOTE: Nitrogen precludes condensation buildup associated with air inflation. It will not sustain combustion, and will reduce degradation of inner liner material due to oxidation. Fairchild Aircraft recommends servicing tires with nitrogen.

(7) Measure newly mounted tires with 27-95026 template before installation on landing gear.

NOTE: It is imperative that the tire size be controlled to help prevent interference between tire and surrounding aircraft structure when gear is retracted. Tires should be measured after they are mounted on the wheels and prior to installation on the landing gear to verify sufficient clearance.

(a) Mount tire and inflate to recommended tire pressure.

(b) Let inflated tire stand for a minimum of 12 hours (optimum of 24 hours).

(c) Verify maximum tire sizes are as follows:

1 Maximum Outside Tread Diameter .....19.20 inches (48.77 cm)

2 Maximum Shoulder Width .....6.35 inches (16.13 cm)

3 Maximum Diameter at Outside Edge of Tread.....17.20 inches (43.69 cm)

4 Maximum Tire Width.....5.60 inches (14.22 cm)

C. Deflation

(1) Always deflate the tire before removing it from an aircraft. This eliminates the danger of an accident during handling or demounting an inflated tire.

WARNING: EXERCISE CARE WHEN REMOVING VALVE CORE FROM VALVE STEM. VALVE CORES CAN BE EJECTED LIKE A BULLET.

(2) Valve core should be removed to ensure complete deflation. Use a deflater cap or release most of the pressure before removing valve core.

D. Inflation

WARNING: AN INFLATED TIRE IS A POTENTIALLY EXPLOSIVE DEVICE. PLACING THE TIRE IN A SAFETY CAGE BEFORE INFLATING WILL PREVENT SERIOUS INJURY IN CASE THE TIRE OR WHEEL SEPARATES DUE TO INCORRECT MOUNTING OR HIDDEN DEFECTS.

(1) When first inflating a freshly mounted tire, a clip-on chuck and an extension are recommended.

(2) Use dry air or nitrogen to service tires.

NOTE: Nitrogen precludes condensation build-up associated with air inflation. It will not sustain combustion and will reduce degradation of inner liner material due to oxidation. M7 Aerospace recommends servicing tires with nitrogen.

E. Tire Inflation Pressure

(1) Each combination of tires used on SA226 series aircraft requires different tire pressures for optimum efficiency and safety of operation. Table 201 shows the tire pressure required of each type of tire and the combination. This chart represents an assumed base temperature of 70°F.

(2) Inflation pressure should be checked only when tires are cool or at normal ambient temperature. Allow one (1) hour cooling before checking inflation pressure. (Two (2) hours in hot weather).

NOTE: For other base temperatures, refer to Figure 201.

(3) The maximum allowable air loss for tubeless tires permitted is 5% within 24 hours. This does not include possible air losses in the wheel assembly or the initial 24-hour stretch conditioning during which 3% to 5% drop in air pressure may result (temperature constant). In service, the pressure drop should be a maximum of 5% for any 24-hour period, averaging between 2% to 3%. If more than 5% pressure drop is indicated for any 24-hour period, check the following:

(a) Accuracy of air pressure gauge.

(b) Valve cores.

(c) Paint tire with a soapy solution. Continuous seepage of small bubbles at a very slow rate is not a basis for rejection. All tubeless tires normally will slowly seep air. Leakage at a fast rate with large bubbles indicates damaged or defective tire.

NOTE: Immersion is recommended only as a last resort. Ensure tire and wheel assembly is thoroughly dried immediately after immersion.

(4) Should the tire show leakage other than that considered normal, check for wheel damage, defect, or porosity, and O-ring or gasket leakage by immersion in water. If no wheel leakage is evident, air loss may be due to:

(a) Low ambient temperature affecting wheel shrinkage and O-ring or gasket seal.

(b) Movement between wheel halves during rolling.

(c) Tire leakage during rolling.

F. Fuel and Solvent Hazards

(1) Tires should not come in contact with oil, gasoline, jet fuel, hydraulic fluids, or any type of rubber solvent. All are natural enemies of rubber and cause it to disintegrate rapidly. Do not stand or lay tires on floors that are covered with oil or grease. When working on engines or landing gear, cover tires so oil or solvent will not drip onto them.

G. Moisture-Ozone

(1) Wet or moist conditions have a rotting effect and may be even more damaging when the moisture contains foreign elements that are further detrimental to rubber and cord fabric. Avoid strong air currents as they increase the supply of oxygen and often carry ozone, both of which cause rapid aging of rubber. Tires should be kept away from electric motors, battery chargers, electric welding equipment, electric generators, and similar equipment which create ozone.

H. Storage

(1) The ideal location for tire or tube storage is a cool, dry, and reasonable dark location, free from air currents and dirt. While low temperatures (not below 32°F) are not objectionable, high room temperatures (80°F) are detrimental and should be avoided.

(2) Whenever possible, store tires in regular tire racks which hold them up vertically. The surface of the tire rack against which the weight of the tire rests should be flat and, if possible, three to four inches wide to avoid permanent distortion of the tire. If tires are piled on top of one another, do not stack too high as this will cause distortion and can cause trouble when the tires are put into service.

2. Inspection/Check – Tires

A. Systematic inspection of mounted tires is recommended for both safety and tire economy. The frequency of the inspection should be determined by the operator.

B. Marking Damage

(1) When inspecting a mounted tire, it is advisable to mark bulges, loose tread areas, cuts, etc. with chalk or crayon before removing and dismounting.

C. Tread Wear

(1) Inspect treads visually or with an approved depth gauge.

(2) Replace tires when tread has worn to the base of any groove at any spot. The tire can be considered safe as long as casing plies are not exposed.

D. Uneven Wear

(1) Check for uneven tread wear.

(a) If wear is not excessive, tire may be dismounted and turned around.

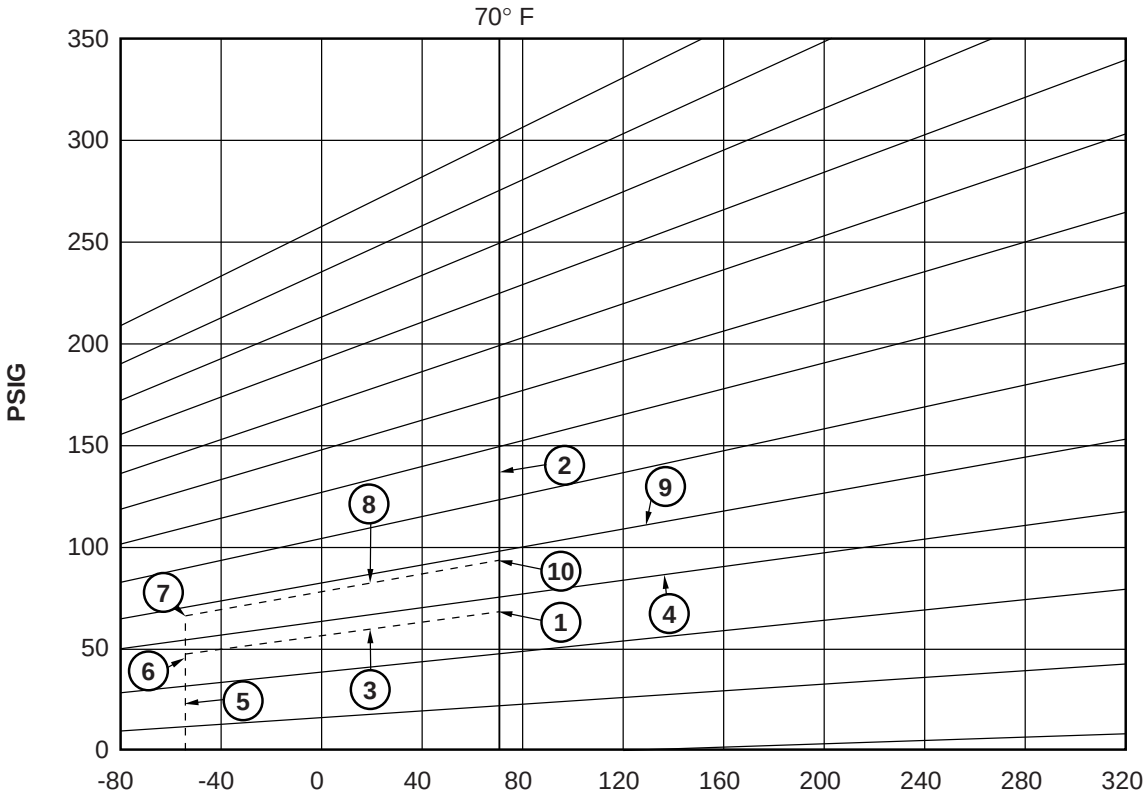
(b) Correct gear misalignment.

E. Cuts and Cracks

(1) Inspect treads and sidewalls for cuts and cracks. Remove tire if cuts or cracks expose casing ply, or if they extend across more that 50% of any rib.

F. Bulges

(1) Bulges in any part of the tire, or an area of loose tread or sidewall rubber, indicate a damaged tire body with ply or tread separation. Replace tires with bulges.



Tire Pressure Chart
Figure 201

G. Flat Spots

(1) Flat spots or skid burns are acceptable unless casing plies are exposed. If severe unbalance results, remove tire.

H. Beads

(1) Inspect bead areas next to wheel flanges for excessive heat damage.

I. Tire Clearance

(1) Check to make sure that nothing is caught between gear and tire.

(2) Look for marks on tires, gear, and in wheel wells that might indicate rubbing or inadequate clearance when retracted.

(3) Tire sizes must be controlled to help prevent interference of mounted tires to the surrounding structure.

J. Wheels

(1) Check wheels for damage. Replace cracked or damaged wheels.

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MAIN LANDING GEAR WHEEL (ALUMINUM) – DESCRIPTION AND OPERATION

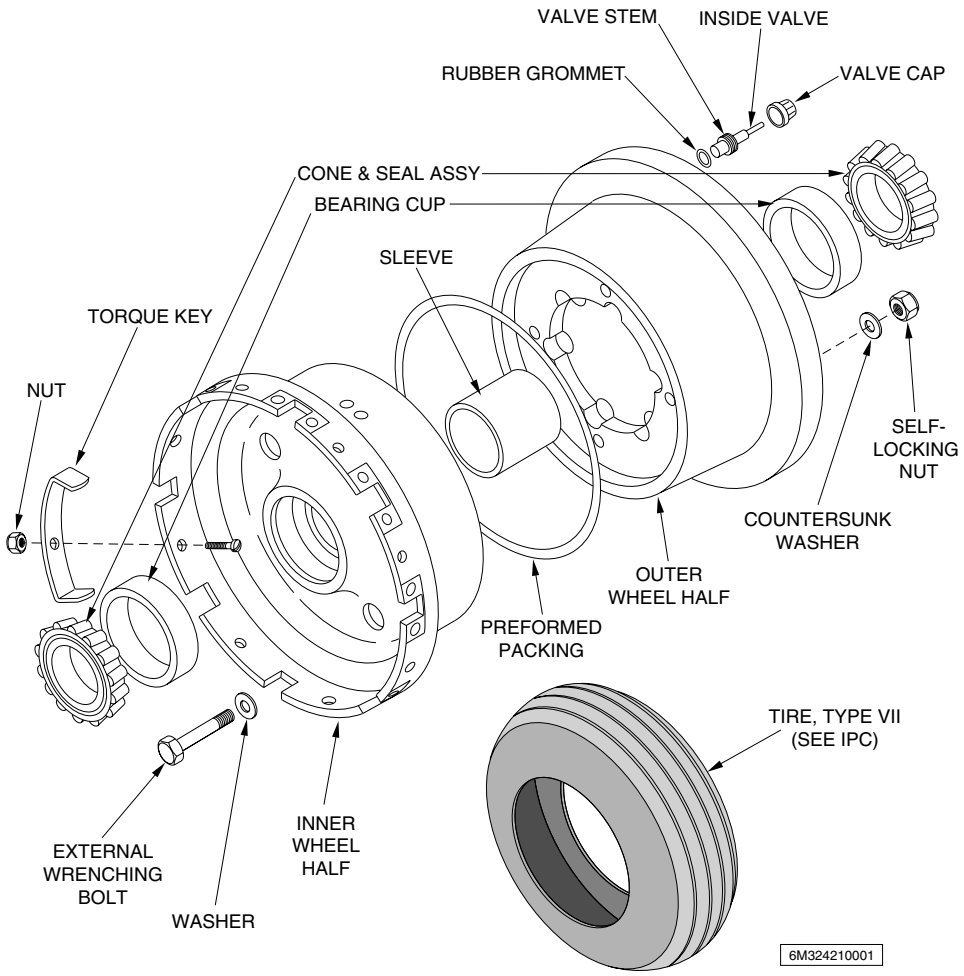
1. General

A. Aluminum 3-1357 Wheel Assembly

(1) The aluminum wheel assembly consists of a forged aluminum inner and outer wheel half bolted together with eight external wrenching bolts, 16 countersunk washers, and eight self-locking nuts. Washers are used beneath the nuts and bolts to prevent galling and stress concentration. A preformed packing in the groove of the inner wheel half seals the halves against air leakage. A sleeve remains between the wheel halves to keep contaminants out of the bearings. Bearing cups and bearing cones with seals are installed in the hub area.

(2) On the inner wheel half, eight torque keys are installed in slots in the flange area and each is secured with a nut and screw. The torque keys retain the lugs of the brake disc which rotates with the wheel.

(3) For tubeless tires, a tubeless tire valve assembly on the outer surface of the outer wheel half is used to inflate and deflate the tubeless tire.



3-1357 Wheel Assembly
Figure 1

MAIN LANDING GEAR WHEEL (ALUMINUM) – MAINTENANCE PRACTICES

1. Servicing – Main Landing Gear Wheel (Aluminum)

A. Disassembly – Aluminum 3-1357 Wheel Assembly

WARNING: TIRE MUST BE DEFLATED BEFORE REMOVING TIE BOLTS.

CAUTION: DO NOT USE TIRE IRONS. THEY MAY DAMAGE THE WHEEL FLANGES OR TIRE BEADS AND PREVENT PROPER AIR RETENTION.

(1) Break tire beads away from wheel flanges with a bead breaker or pneumatic tire dismounter.

(2) Remove nuts, washers, and bolts from wheel.

(3) Separate the wheel halves and remove tire.

(4) If the preformed packing is to be reused, place it on a clean sheet of paper for cleaning.

NOTE: Torque keys should only be removed if excessively worn, cracked, loose, or for Zyglo inspection of the wheel half.

B. Cleaning – Aluminum 3-1357 Wheel Assembly

WARNING: USE SOLVENTS IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND SKIN CONTACT WITH SOLVENT. DO NOT USE COMPRESSED AIR OVER 30 PSI (2.11 KG/CM^2). PROTECT EYES.

CAUTION: DO NOT USE CLEANING SOLVENTS ON RUBBER PARTS.

(1) Clean all metal parts in dry cleaning solution (chlorinated cleaner, acetone, methyl ethyl ketone, or equivalent) and dry with compressed air.

CAUTION: DO NOT SPIN DRY BEARINGS OR HANDLE BEARING WITH BARE HANDS.

(2) Wipe bearing seals with clean dry cloth.

(3) Wash bearing cones in uncontaminated cleaning solution. Rotate bearing cage by hand while submerged in solution.

(4) Dry bearings with a clean cloth.

(5) Repack bearing with grease immediately after cleaning to prevent corrosion and seal in a clean container.

C. Inspection – Aluminum 3-1357 Wheel Assembly

(1) Refer to the Vendor's Component Maintenance Manual for Main Landing Gear Wheel Inspection procedures.

D. Lubrication – Aluminum 3-1357 Wheel Assembly

(1) Bearings – pack grease (MIL-G-3545) into bearing cones and smear grease on ends of rollers. Do not over lubricate. Spread a thin coat of grease on surface of bearing cups.

(2) Bolts, nuts, and washers – lubricate threads of bolts, nuts, and face of washers with thread compound (MIL-T-5544).

(3) Preformed packing – supply a thin layer of bearing grease (MIL-G-3545) to packing before installing in wheel.

E. Assembly – Aluminum 3-1357 Wheel Assembly

(1) Install preformed packing and sleeve between wheel halves before rejoining wheel halves.

(2) Install wheel halves into tire.

(3) Install bolts, washers, and retaining nuts into wheel securing wheel halves together.

NOTE: Retaining nuts are installed on valve core side of wheel half.

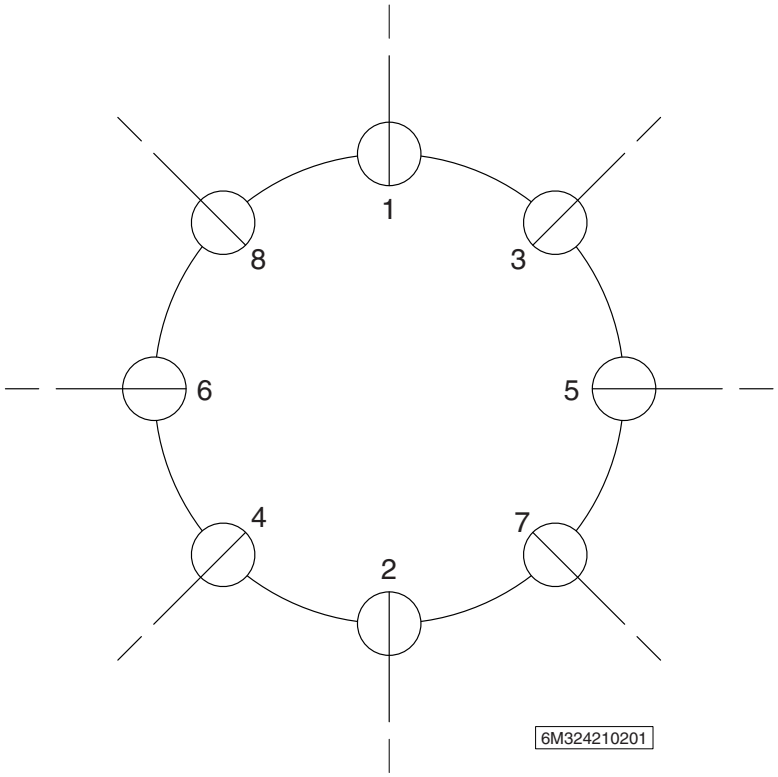
(4) Torque bolts in two stages as follows:

(a) Tighten tie bolts in cross sequence (see Figure 201) 200 to 208 inch-pounds (22.60 to 23.39 N·m).

(b) Tighten tie bolts in cross sequence (see Figure 201) 250 to 260 inch-pounds (28.25 to 29.38 N·m) using a preset hand torque wrench.

NOTE: All values given are Lubtork values.

After tire growth has stabilized, the wheel should hold normal inflation pressure for a minimum of 24 hours with not more than four percent loss in pressure.



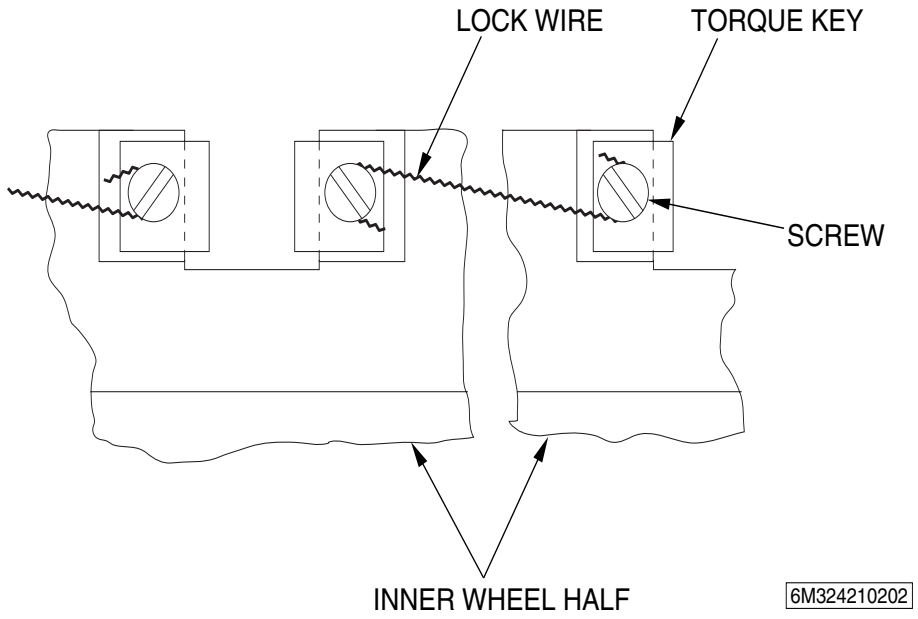
Torque Sequence
Figure 201

(5) Clean and lubricate bearings.

(6) Install torque keys (if removed).

NOTE: When installing torque keys, oversize screws and nuts may be utilized at the discretion of the user, IAW B.F. Goodrich SB No. 377. For part numbers refer to SA226 series Illustrated Parts Catalog.

(7) Verify torque before inflating tire.



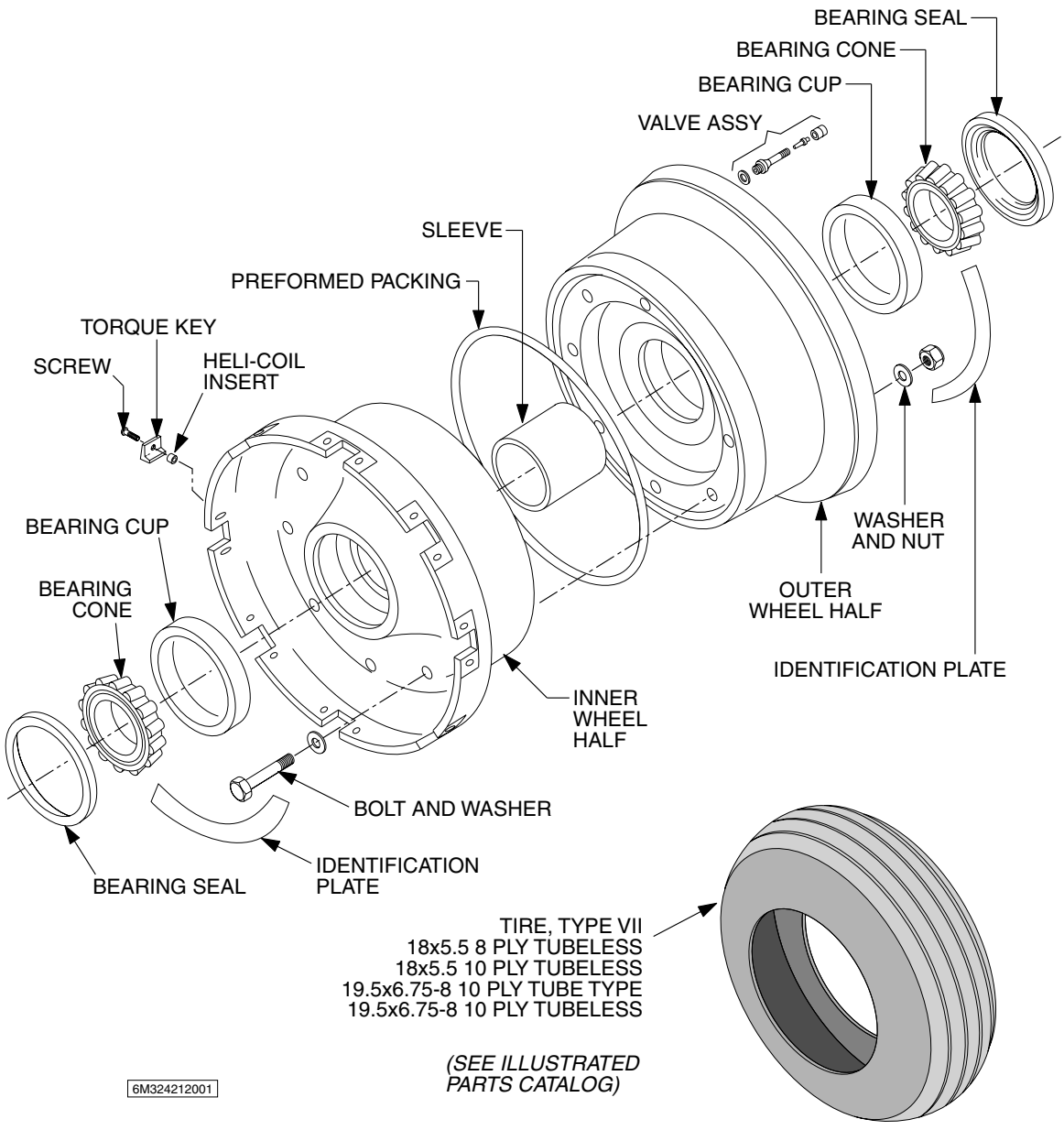
Safety-Wiring Torque Key Mounting Screws
Figure 202

MAIN LANDING GEAR WHEEL (MAGNESIUM) – DESCRIPTION AND OPERATION

1. General

A. Magnesium 3-1283 Wheel Assembly

(1) The magnesium wheel assembly consists of a cast magnesium inner and outer wheel half assembled with eight bolts, 16 countersunk washers, and eight self-locking nuts. The washers are used beneath the nuts and bolt heads to prevent galling and stress concentration.



3-1283 Wheel Assembly
Figure 1

(2) A preformed packing in the inner wheel half seals the wheel halves against air leakage. A sleeve between the wheel halves keeps contaminants out of bearings. A bearing cup, two bearing cones, and bearing seal are installed in each hub. A self-adhesive identification plate is fastened to each wheel half outer surface.

(3) On the inner wheel half, 16 torque keys (two to eight slots in the flange area) are secured with a screw. Each screw is fastened to a heli-coil insert in the slotted flange surface. Torque keys hold the lugs of the brake disc. The brake disc rotates with the wheel when the wheel and brake are mounted on the landing gear axle. For tubeless tires, a tubeless tire valve assembly is installed on the outer surface of the outer wheel half and is used to inflate and deflate the tubeless tire.

MAIN LANDING GEAR WHEEL (MAGNESIUM) – MAINTENANCE PRACTICES

1. Servicing – Main Landing Gear Wheel (Magnesium)

A. Disassembly – Main Landing Gear Wheel (Magnesium)

WARNING: TIRE MUST BE DEFLATED BEFORE REMOVING TIE BOLTS.

(1) Break tire beads away from wheel flanges with bead breaker or pneumatic tire dismounter.

CAUTION: DO NOT USE TIRE IRONS. THEY MAY DAMAGE WHEEL FLANGES OR TIRE BEADS AND PREVENT PROPER AIR RETENTION.

(2) Remove nuts, washers, and bolts from wheel.

(3) Separate wheel halves and remove tire.

(4) If preformed packing is to be resused, place it on a clean sheet of paper for cleaning.

NOTE: Torque keys should only be removed if excessively worn, cracked, loose, or for zyglo inspection of wheel half.

B. Cleaning – Main Landing Gear Wheel (Magnesium)

WARNING: USE SOLVENTS IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND SKIN CONTACT WITH SOLVENT. DO NOT USE COMPRESSED AIR OVER 30 PSIG. PROTECT EYES.

CAUTION: DO NOT USE CLEANING SOLVENTS ON RUBBER PARTS.

(1) Clean all metal parts in dry cleaning solution (chlorinated cleaner, acetone, methyl ethyl ketone, or equivalent) and dry with compressed air.

CAUTION: DO NOT SPIN DRY BEARINGS OR HANDLE BEARING WITH BARE HANDS.

(2) Wipe bearing seals clean with clean dry cloth.

(3) Wash bearing cones in uncontaminated cleaning solution. Rotate bearing cage by hand while submerged in solution.

(4) Dry bearings with a clean cloth.

(5) Repack bearing with grease immediately after cleaning to prevent corrosion and seal in a clean container.

C. Inspection – Main Landing Gear Wheel (Magnesium)

(1) Refer to the Vendor's Component Maintenance Manual for Main Landing Gear Wheel Inspection procedures.

D. Lubrication – Main Landing Gear Wheel (Magnesium)

(1) Bearings – pack grease (MIL-G-3545) into bearing cones and smear grease on ends of rollers. Do not over lubricate. Spread a thin coat of grease on surface of bearing cups.

(2) Bolts, nuts, and washers – lubricate threads of bolts, nuts, and face of washers with thread compound (MIL-T-5544).

(3) Preformed packing – supply a thin layer of bearing grease (MIL-G-3545) to packing before installing in wheel.

E. Assembly – Main Landing Gear Wheel (Magnesium)

(1) Install preformed packing and sleeve between wheel halves before rejoining wheel halves.

(2) Install wheel halves into tire.

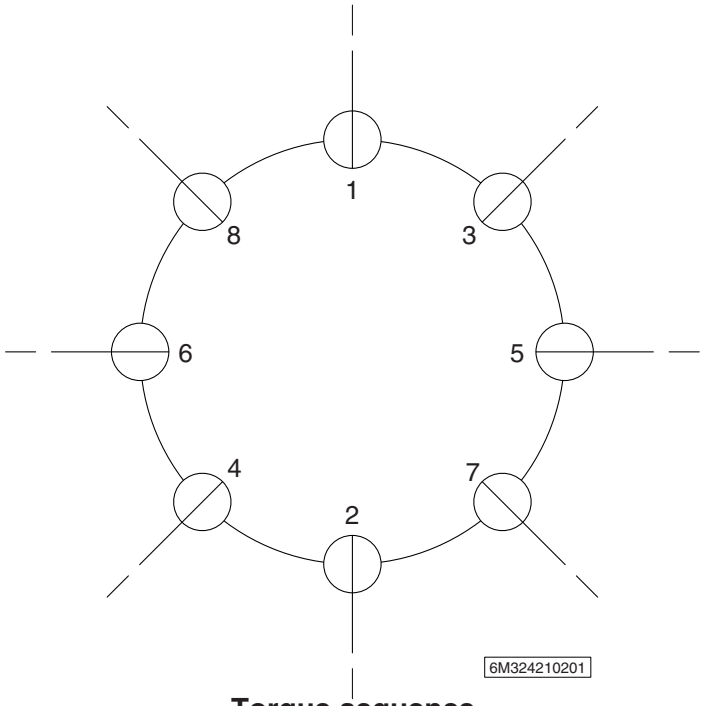
(3) Assemble wheel halves securing with bolts, washers, and retaining nuts.

(4) Torque bolts in two stages as follows:

NOTE: Retaining nuts are installed on valve core side of wheel half. Torque values are Lubtork values.

(a) Tighten tie bolts in cross sequence (see Figure 201) to 80% of the final torque.

(b) Tighten tie bolts in cross sequence (see Figure 201) to full specified torque of 140 to 160 inch-pounds (15.82 to 18.08 N·m) using a preset hand torque wrench.



Torque sequence
Figure 201

(6) Clean and lubricate bearings.

(7) Verify torque before inflating tire.

(8) Install torque keys (if required) and safety wire screws in pairs.

NOTE: After tire growth has stabilized, the wheel should hold normal inflation pressure for a minimum of 24 hours with not more than four percent loss in pressure.

MAINTENANCE MANUAL

NOSE LANDING GEAR WHEEL – DESCRIPTION AND OPERATION

1. General

A. SA226 series aircraft utilize dual nose wheels. As standard installation, a Jay-Em Corporation magnesium wheel, P/N 214A99M, was provided for use with a pressure tube-type 16 X 4.4 tire. In order to adequately meet individual operational requirements, chine tires and larger wheel assemblies for use with low pressure tires are offered as optional equipment. One optional wheel assembly is the Goodyear magnesium wheel, P/N 9544207-1, which is used with a low pressure tubeless 18 X 4.4 tire. A second optional wheel assembly is the B.F. Goodrich aluminum wheel, P/N 3-1418, which is also used with a low pressure tubeless 18 X 4.4 tire.

B. The 3-1418 wheel may be used to replace the 9544207-1 wheel but should not be intermixed on installation. See Figure 1 for the 9544207-1 wheel assembly, Figure 2 for the 214A99M wheel, and Figure 3 for the 3-1418 wheel.

C. Standard tires are 6 ply rated and the optional chine tire is 10 ply rated. The chine tire, utilized on AT and TC model aircraft, is a tire design to deflect water and slush to the side away from engine intakes. For further information and breakdown of wheels and tires, refer to the Illustrated parts Catalog (section 32-40-00). For tire servicing instructions, refer to Maintenance Procedures, Tires, in this chapter.

MAINTENANCE MANUAL

NOSE LANDING GEAR WHEEL – DESCRIPTION AND OPERATION

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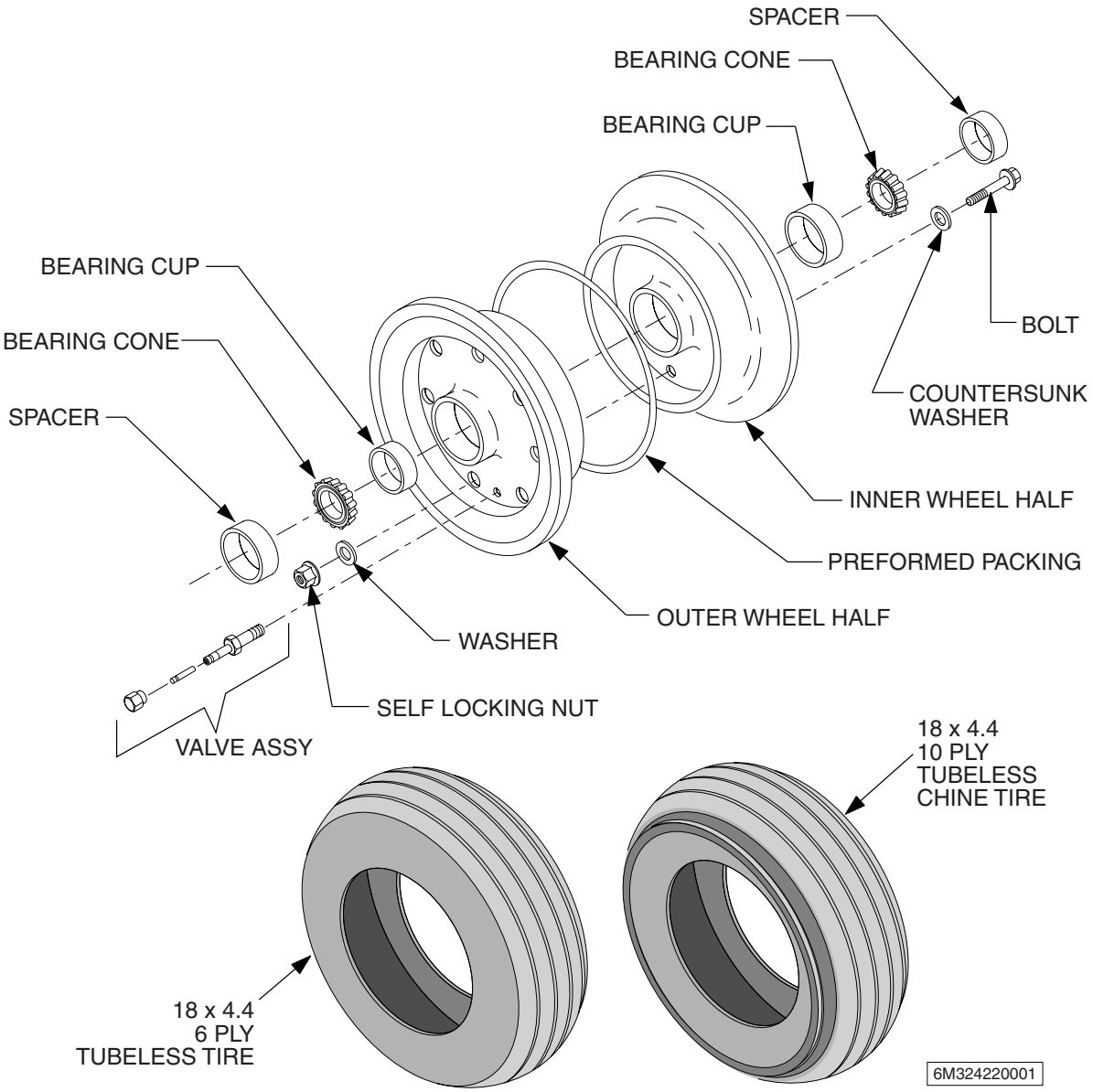
NOSE LANDING GEAR WHEEL (GOODYEAR) – DESCRIPTION AND OPERATION

2. Nose Landing Gear Wheel (Goodyear)

A. The Goodyear nose landing gear wheel assembly consists of a cast magnesium inner and outer wheel half bolted together with eight external wrenching bolts, 8 countersunk washers installed on the inner wheel half eight flat washers installed on the outer wheel half, and eight self-locking nuts. The washers are used beneath the nuts and bolt heads to prevent galling and stress concentration.



Nose Landing Gear Wheel (Goodyear) Figure 1

NOSE LANDING GEAR WHEEL (GOODYEAR) – DESCRIPTION AND OPERATION

B. A preformed packing, in the groove of the inner wheel half seals the halves against air leakage. Bearing cups and bearing cones with seals with attached seals are installed in the hub area. Because the Goodyear wheel is not as wide as the Jay-Em wheel, wheel spacers are provided. During installation, one wheel spacer is installed onto the axle prior to installing the wheel. The outer spacer is installed prior to installing the axle washer and nut.

C. A tubeless tire valve assembly is installed on the outer surface of the outer wheel half and is used to inflate and deflate the tubeless tire.

NOSE LANDING GEAR WHEEL (GOODYEAR) – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Landing Gear Wheel (Goodyear)

A. Removal – Nose Landing Gear Wheel (Goodyear)

(1) Jack the aircraft.

(2) Deflate tire.

WARNING: EXERCISE CARE WHEN REMOVING VALVE CORE FROM VALVE STEM. THE VALVE CORE SHOULD BE REMOVED TO ENSURE COMPLETE DEFLATION.

(3) Remove cotter key, axle nut, and keyed washer.

(4) Remove the outer spacer.

(5) Remove wheel from axle.

B. Installation – Goodyear Nose Landing Gear Wheel (Goodyear)

(1) Check for burrs or rough threads on the axle and axle nut.

(2) Apply thread compound (MIL–T–5544) to threads and all bearing surfaces of washers and nuts prior to installation.

(3) Install inboard wheel spacer onto axle.

(4) Install wheel assembly.

(5) Install outboard wheel spacer onto axle.

(6) Install keyed washer and axle nut. Lube torque the axle nut to 40 inch-pounds (4.5 N·m) while rotating the wheel to ensure proper seating of bearing cones.

(7) Back the axle nut off to zero torque (without rotating wheel). Torque nut to 20 inch-pounds (2.26 N·m) While rotating wheel.

(8) Check for alignment of locking holes.. If holes are not aligned, continue tightening axle nut to the first locking hole and secure with cotter key.

NOTE: Axle nut lube torque is 20 inch-pounds (2.26 N·m) minimum.

2. Inspection/Check – Nose Landing Gear Wheel (Goodyear)

A. Inspection – Nose Landing Gear Wheel (Goodyear)

(1) Visually inspect each wheel half for distortion, cracks, active corrosion, and other similar damage. Use a fluorescent or dye penetrant to detect cracks. Distortion is usually at the register of the wheel halves and the flange outside diameter. Cracks usually are found in the flange and tie bolt area. Corrosion will be evident at areas of dissimilar metals such as bearing seats, lock ring grooves, tie bolt areas, and tube valve area. Inspect the painted surfaces of each wheel half. The wheel halves must be covered with a protective coating of acrylic lacquer to prevent corrosion.

(2) Inspect bearing cups for cracks, excessive scoring, discoloration which would indicate overheating, distortion, or other similar damage. Check each bearing cup for a sure and positive fit in the wheel half.

(3) Inspect bearing cones and rollers for cracks, mars, broken or badly worn rollers, discoloration which would indicate overheating, and other similar damage. Check each roller for free and easy movement.

(4) Flat spots: wheels outside the following limits should be replaced.

(a) Radial run-outs = 0.010 max.

(b) Lateral run-outs = 0.012 max.

(c) Out of round = 0.030 max.

B. Lubrication – Nose Landing Gear Wheel (Goodyear)

(1) Pack bearing cones and roller with grease (MIL-T-81322). Work grease thoroughly between and around the rollers.

(2) Soak felt grease retainer in lubricating oil (MIL-L-7870 or Silicone Fluid, MIL-S-21568) before assembly.

(3) Lubricate the tie bolt threads and bearing surfaces of the nut, bolt head, and washers with anti-seize compound (MIL-T-5544) at installation and with relubrication at each subsequent torque application.

(4) Apply a thin layer of bearing grease (MIL-G-81322) to the packing before installaing the wheel.

C. Disassembly – Nose Landing Gear Wheel (Goodyear)

WARNING: TIRES MUST BE DEFLATED BEFORE LOOSENING TIE BOLTS.

NOTE: To remove bearing cones from the Goodyear wheel, only the wheel spacer preceeds the bearing cone, and the cone is usually removed from the wheel upon wheel removal from the axle.

CAUTION: DO NOT USE TIRE IRONS. THEY MAY DAMAGE THE WHEEL FLANGES OR TIRE BEADS AND PREVENT PROPER AIR RETENTION.

(1) Break tire beads away from wheel flanges with a bead breaker or pneumatic tire dismounter .

(2) Remove nuts, washers, and bolts from wheel.

(3) Separate the wheel halves and remove from the tire.

(4) The Goodyear nose landing gear wheel has a preformed packing which seals the two wheel halves. If it is to be reinstalled, place it on a clean sheet of paper for cleaning.

NOTE: Bearing cups are shrunk fit into the wheel halves and should not be removed unless replacement is necessary.

CAUTION: IT IS RECOMMENDED THAT GLOVES BE WORN WHEN HANDLING WHEEL HALVES AFTER THEY HAVE BEEN EXPOSED TO HIGH TEMPERATURE.

(5) If a bearing cup is to be replaced, submerge the landing gear wheel halves in boiling water. When the wheel halves have reached water temperature, remove from the water and drive the bearing cup out of the casting with a brass rod.

D. Assembly – Nose Landing Gear Wheel (Goodyear)

(1) Install preformed packing between wheel halves before rejoining wheel halves.

(2) Install bolts, washers, and retaining nuts into wheel securing wheel halves together.

(3) Clean and lubricate bearings as detailed in the Lubrication procedures of this chapter.

CAUTION: IT IS RECOMMENDED THAT GLOVES BE WORN WHEN HANDLING WHEEL HALVES AFTER THEY HAVE BEEN EXPOSED TO HIGH TEMPERATURE.

(4) To replace bearing cups, submerge the landing gear wheel halves in boiling water and, if possible, chill the bearing cups with dry ice. When the landing gear wheel halves have reached water temperature, remove from the water and push bearing cups into place using a plug which is slightly smaller in diameter than the diameter of the bearing cups.

(5) Lube torque nuts on tie bolts 120 inch-pounds Using a preset hand torque wrench in criss-cross sequence across the diameter. Recheck the torque once around before inflating tire.

E. Inflation – Goodyear Nose Landing Gear Wheel (Goodyear)

NOTE: Refer to the Tire Maintenance Practices procedures detailed in this chapter for correct inflation procedures of the tire.

(1) After the inflation has been accomplished and tire growth has stabilized, the wheel shall be required to hold the normal inflation pressure for 24 hours with not more than four percent loss in pressure.

3. Cleaning/Painting – Nose Landing Gear Wheel (Goodyear)

A. Paint Removal – Nose Landing Gear Wheel (Goodyear)

NOTE: Commercial paint strippers other than those listed may be used if not harmful to magnesium.

(1) To strip paint:

(a) Blast with abrasive of ground walnut hulls (MIL-G-5634), Type II or III may be used. This method will not remove the DOW surface treatment and does not require masking or removal of the bearing cup.

(b) Chemical strip using formula N0. 1006 (purchased from Allied Ketite Chemical Co., 2400 East Devon Avenue, Des Plaines, Illinois. 60018).

(c) Methylene Chloride (avoid hand or skin contact).

(d) Acetone (flammable, use in well ventilated area).

CAUTION: REMOVE BEARING CUPS IF CHEMICAL PAINT STRIPPERS ARE USED.

(2) After stripping wheel halves, inspect using zyglo and make necessary repairs (blending of nicks or corrosion and remachining around tie bolt halves).

B. Cleaning – Nose Landing Gear Wheel (Goodyear)

CAUTION: DO NOT SPIN DRY THE BEARINGS OR HANDLE BEARINGS WITH BARE HANDS.

(1) Clean all parts with either dry cleaning solvent (P-D-680, Type I), or vapor degrease with trichlorethylene (MIL-T-27602) and dry with compressed air.

(2) Degrease bearing cups and cones with solvent (P-D-680, Type I or VV-K-211) and rinse in isopropyl alcohol.

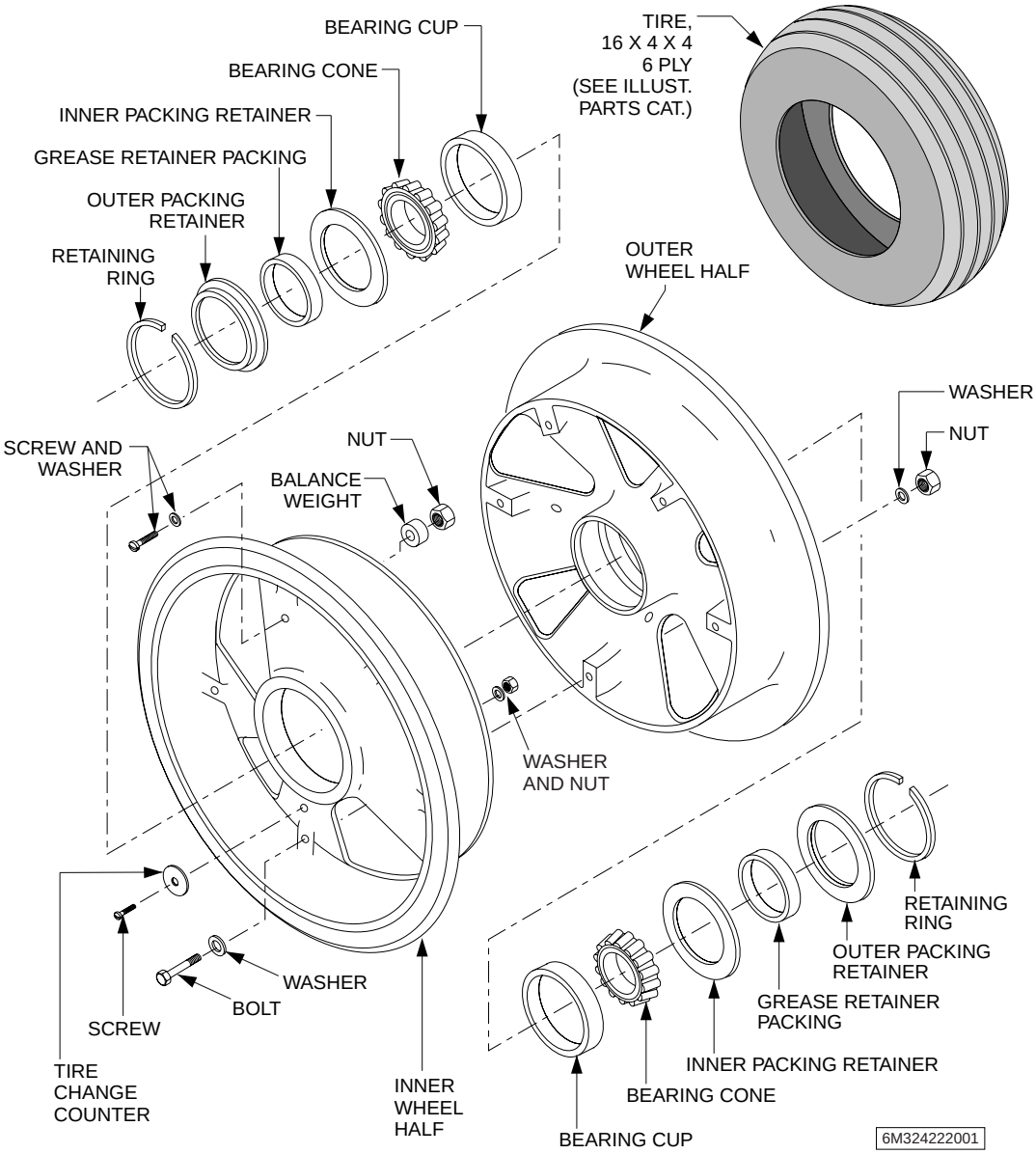
(3) Repack bearing cone with grease immediately after cleaning to prevent corrosion and place in a clean, closed container. (See Lubrication procedures step **2.**, B.

NOSE LANDING GEAR WHEEL (JAY-EM) – DESCRIPTION AND OPERATION

1. General

A. Nose Landing Gear Wheel (Jay-Em)

(1) The standard nose landing gear wheel assembly consists of a cast magnesium inner and outer wheel half bolted together with five bolts, ten washers, and five self-locking nuts. The washers are used beneath the nuts and bolt heads to prevent galling and stress concentration.



Nose Landing Gear Wheel (Jay-Em)
Figure 1

NOSE LANDING GEAR WHEEL (JAY-EM) – DESCRIPTION AND OPERATION

(2) Two bearing cups, two bearing cones, two inner packing retainers, two grease retainer packings, two outer packing retainers, and two retaining rings are installed, one set in each hub area.

NOSE LANDING GEAR WHEEL (JAY-EM) – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Landing Gear Wheel (Jay-Em)

A. Removal – Nose Landing Gear Wheel (Jay-Em)

(1) Jack the aircraft.

(2) Deflate tire.

WARNING: EXERCISE CARE WHEN REMOVING VALVE CORE FROM VALVE STEM. THE VALVE CORE SHOULD BE REMOVED TO ENSURE COMPLETE DEFLATION.

(3) Remove cotter key, axle nut, and keyed washer.

(4) Remove wheel from axle.

B. Installation – Nose Landing Gear Wheel (Jay-Em)

(1) Check for burrs or rough threads on the axle and axle nut.

(2) Apply thread compound (MIL–T–5544) to threads and all bearing surfaces of washers and nuts prior to installation.

(3) Install wheel assembly.

(4) Install keyed washer and axle nut. Lube torque the axle nut 150 to 200 inch-pounds (16.95 N·m) while rotating the wheel to ensure proper seating of the bearings.

(5) Back the axle nut off to zero torque value and final torque the nut to 40 inch-pounds (4.52 N·m).

(6) Check for alignment of locking holes.. If holes are not aligned, continue tightening axle nut to the first locking hole and secure with cotter key. key.

NOTE: Axle nut lube torque is 40 inch-pounds (4.52 N·m) minimum.

2. Inspection/Check – Nose Landing Gear Wheel (Jay-Em)

A. Inspection – Nose Landing Gear Wheel (Jay-Em)

(1) Visually inspect each wheel half for distortion, cracks, active corrosion, and other similar damage. Use a fluorescent or dye penetrant to detect cracks. Distortion is usually at the register of the wheel halves and the flange outside diameter. Cracks usually are found in the flange and tie bolt area. Corrosion will be evident at areas of dissimilar metals such as bearing seats, lock ring grooves, tie bolt areas, and tube valve area. Inspect the painted surfaces of each wheel half. The wheel halves must be covered with a protective coating of acrylic lacquer to prevent corrosion.

(2) Inspect bearing cups for cracks, excessive scoring, discoloration which would indicate overheating, distortion, or other similar damage. Check each bearing cup for a sure and positive fit in the wheel half.

(3) Inspect bearing cones and rollers for cracks, mars, broken or badly worn rollers, discoloration which would indicate overheating, and other similar damage. Check each roller for free and easy movement.

NOSE LANDING GEAR WHEEL (JAY-EM) – MAINTENANCE PRACTICES

B. Surface Treating – Nose Landing Gear Wheel (Jay-Em)

(1) Surface treat reworked wheel halves by brushing locally repaired areas or dipping entire wheel in tank by:

(a) Neutralizing corrosion with Chromic Acid Pickle (Dow 1). Pickling solution consists of 1 pound chromic acid, 1.3 pounds nitric acid, and one gallon of water.

(b) Applying solution (Dow 1) by brushing, dipping, spraying, or with a pad. The solution will complete its action in about two minutes. Wash off with cold water.

(c) Follow the Dow 1 treatment with Dow 19 surface treatment using the same procedure. Use 1-1/3 ounce of chromic acid, one ounce of calcium sulfate to one gallon of water to make Dow 19 Chromic Acid Treatment.

NOTE: Dow 19 solution may be kept safely in glass, stainless steel, aluminum containers, or vinyl, polyethruane rubber lined tanks.

(2) After surface treatment, paint wheel halves with two coats of zinc chromate primer (MIL-P-8585) and one coat of aluminum lacquer (MIL-L-7178). Reinstall bearing cup.

C. Lubrication – Nose Landing Gear Wheel (Jay-Em)

(1) Pack bearing cones and roller with grease (MIL-GO-81322). Work grease thoroughly between and around the rollers.

(2) Soak felt grease retainer in lubricating oil (MIL-L-7870 or Silicone Fluid, MIL-S-21568) before assembly.

(3) Lubricate the tie bolt threads and bearing surfaces of the nut, bolt head, and washers with anti-seize compound (MIL-T-5544) at installation with relubrication and at each subsequent torque application.

(4) Apply a thin layer of bearing grease (MIL-G-81322) to the packing before installing the wheel.

C. Disassembly – Nose Landing Gear Wheel (Jay-Em)

WARNING: TIRES MUST BE DEFLATED BEFORE LOOSENING TIE BOLTS.

(1) To remove bearing cones from the Jay-em wheel, remove the retaining ring, outer packing retainer (dust cover), grease retainer packing (felt ring), inner packing retainer, and bearing cone.

(2) Break tire beads away from wheel flanges with a bead breaker or pneumatic tire dismounter.

CAUTION: DO NOT USE TIRE IRONS. THEY MAY DAMAGE THE WHEEL FLANGES OR TIRE BEADS AND PREVENT PROPER AIR RETENTION.

(3) Remove nuts, washers, and bolts from wheel

(4) Separate the wheel halves and remove from the tire.

NOTE: Bearing cups are shrunk fit into the wheel halves and should not be removed unless replacement is necessary.

CAUTION: IT IS RECOMMENDED THAT GLOVES BE WORN WHEN HANDLING WHEEL HALVES AFTER THEY HAVE BEEN EXPOSED TO HIGH TEMPERATURE.

(5) If a bearing cup is to be replaced, submerge the landing gear wheel halves in boiling water. When the wheel halves have reached water temperature, remove from the water and drive the bearing cup out of the casting with a brass rod.

D. Assembly – Nose Landing Gear Wheel (Jay-Em)

(1) Install wheel halves into tire.

(2) Install bolts, washers, and retaining nuts into wheel securing wheel halves together.

(3) Clean and lubricate bearings as detailed in the Lubrication procedures of this chapter.

CAUTION: IT IS RECOMMENDED THAT GLOVES BE WORN WHEN HANDLING WHEEL HALVES AFTER THEY HAVE BEEN EXPOSED TO HIGH TEMPERATURE.

(4) To replace bearing cups, submerge the landing gear wheel halves in boiling water and, if possible, chill the bearing cups with dry ice. When the landing gear wheel halves have reached water temperature, remove from the water and push bearing cups into place using a plug which is slightly smaller in diameter than the diameter of the bearing cups.

(5) Torque nuts on tie bolts to 75 inch-pounds Using a preset hand torque wrench in criss-cross sequence across the diameter. Recheck the torque once around before inflating tire.

E. Inflation – Nose Landing Gear Wheel (Jay-Em)

NOTE: Refer to the Tire Maintenance Practices procedures detailed in this chapter for correct inflation procedures of the tire.

(1) After the inflation has been accomplished and tire growth has stabilized, the wheel shall be required to hold the normal inflation pressure for 24 hours with not more than four percent loss in pressure.

3. Cleaning/Painting – Nose Landing Gear Wheel (Jay-Em)

A. Paint Removal – Nose Landing Gear Wheel (Jay-Em)

NOTE: Commercial paint strippers other than those listed may be used if not harmful to magnesium.

(1) To strip paint:

(a) Blast with abrasive of ground walnut hulls (MIL-G-5634), Type II or III. This method will not remove the DOW surface treatment, and does not require masking or removal of the bearing cup.

(b) Chemical strip using formula N0. 1006 (purchased from Allied Ketite Chemical Co., 2400 East Devon Avenue, Des Plaines, Illinois. 60018).

(c) Methylene Chloride (avoid hand or skin contact).

(d) Acetone (flammable, use in well ventilated area).

CAUTION: REMOVE BEARING CUPS IF CHEMICAL PAINT STRIPPERS ARE USED.

(2) After stripping wheel halves, inspect using zyglo and make necessary repairs (blending of nicks or corrosion and remachining around tie bolt halves) within limits specified in the Repair and Replacement procedures of this chapter.

B. Cleaning – Nose Landing Gear Wheel (Jay-Em)

CAUTION: DO NOT SPIN DRY THE BEARINGS OR HANDLE BEARINGS WITH BARE HANDS.

(1) Clean all parts with either dry cleaning solvent (P-D-680, Type I), or vapor degrease with trichlorethylene (MIL-T-27602) and dry with compressed air.

(2) Degrease bearing cups and cones with solvent (P-D-680, Type I or VV-K-211) and rinse in isopropyl alcohol.

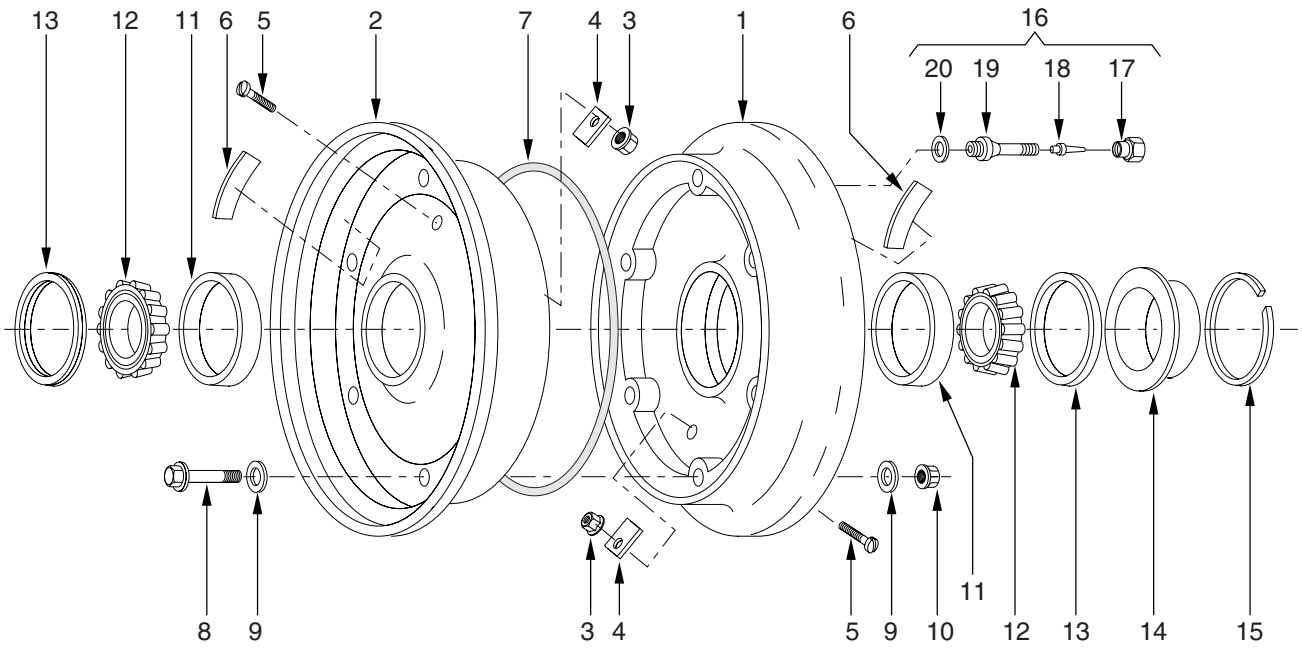
(3) Repack bearing cone with grease immediately after cleaning to prevent corrosion and place in a clean, closed container. (See Lubrication procedures in this chapter).

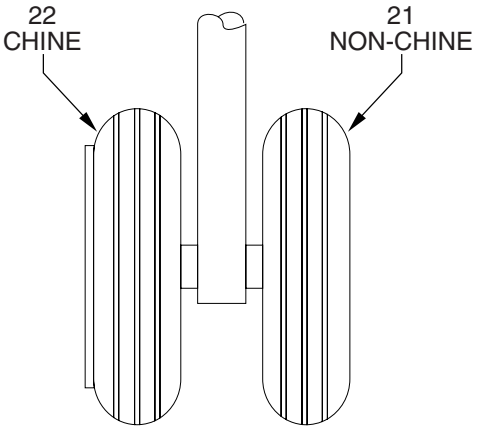
NOSE LANDING GEAR WHEEL (B.F. GOODRICH) – DESCRIPTION AND OPERATION

1. General

A. Nose Landing Gear Wheel (B.F. Goodrich)

(1) The 3-1418 nose landing gear wheel is a split type wheel, designed for a 18 X 4.4, type VII, tubeless tire. A tubeless tire inflation valve installed in the outer wheel half is used to inflate the tire.





1. WHEEL HALF, OUTER
2. WHEEL HALF, INNER
3. NUT, SELF LOCKING
4. WEIGHT, BALANCE
5. SCREW, MACHINE
6. INSTRUCTION PLATE
7. PACKING, PREFORMED
8. BOLT
9. WASHER, COUNTERSUNK
10. NUT
11. CUP, BEARING
12. CONE, BEARING
13. SEAL ASSY
14. HUB CAP
15. RING, RETAINING
16. VALVE ASSY
17. VALVE CAP
18. INSIDE VALVE
19. VALVE STEM
20. GROMMET, RUBBER
21. TIRE, TUBELESS 6 PLY
22. TIRE, TUBELESS 10 PLY SINGLE CHINE

6M324224001

Nose Landing Gear Wheel
B.F. Goodrich, 3-1418
Figure 1

NOSE LANDING GEAR WHEEL (B.F. GOODRICH) – DESCRIPTION AND OPERATION

(2) The cast aluminum alloy wheel halves are fastened together by six equally spaced tie bolts, secured with washers and nuts. A packing, in the groove of the inboard wheel half seals the wheel against air leakage.

(3) Tapered roller bearing cones and cups installed in the wheel half hub areas support the wheel on the axle. The bearings are protected against moisture and dirt contamination by grease seals that are self retained by axle spacers.

(4) A hub cap is secured to the outer wheel half hub by a retaining ring. The inner and outer wheel halves are balanced to within one ounce inches with balanced weights fastened to the wheel web by screws and nuts.

NOSE LANDING GEAR WHEEL (B.F. GOODRICH) – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Landing Gear Wheel (B.F. Goodrich)

A. Removal – Nose Landing Gear Wheel (B.F. Goodrich)

(1) Jack the aircraft.

(2) Deflate tire.

WARNING: EXERCISE CARE WHEN REMOVING VALVE CORE FROM VALVE STEM. THE VALVE CORE SHOULD BE REMOVED TO ENSURE COMPLETE DEFLATION.

(3) Remove retaining ring and hub cap.

(4) Remove cotter pin, axle nut, and washer.

(5) Remove wheel from tire.

B. Installation – Nose Landing Gear Wheel (B.F. Goodrich)

(1) Install grease seal on axle.

(2) Install inboard bearing cone on axle.

(3) Install wheel and tire assembly on axle.

(4) Install outboard bearing cone.

(5) Install washer and axle nut.

(6) While rotating wheel, torque axle nut 200 inch-pounds (22.60 N·m) to properly seat bearing cones.

(7) Loosen axle nut to zero torque. While rotating wheel, retorque axle nut 110 inch-pounds (12.43 N·m).

(8) Check for alignment of locking holes in axle nut and axle. If holes are not aligned, continue tightening axle nut until locking holes are aligned.

(9) Secure axle nut with cotter pin.

(10) Position hub cap over axle and secure to wheel with retaining ring. Be sure retaining ring completely engages groove in bore wheel.

2. Inspection/Check – Nose Landing Gear Wheel (B.F. Goodrich)

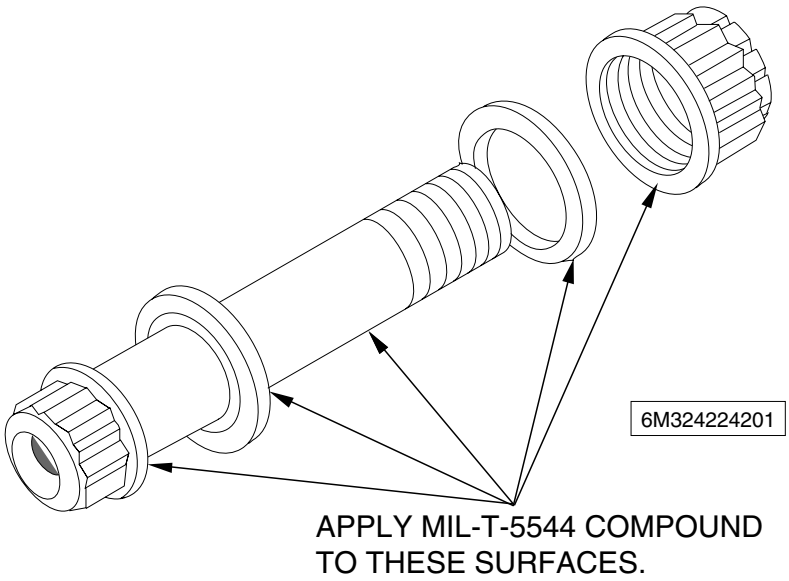
A. Refer to Component Maintenance Manual (CMM) for Nose Landing Gear Wheel Inspection procedures.

B. Lubrication – Nose Landing Gear Wheel (B.F. Goodrich)

(1) Bearings – pack bearing cones and cups with grease (Aero Shell No. 5 or equivalent. Be sure there is ample grease in and around rollers and cage.

(2) Lubricate nuts, and washers and thread of bolts with anti-seize compound IAW specification MIL-T-5544. Relubricate at each subsequent torque application, refer to Figure 201.

(3) Apply a thin layer of grease (Aero Shell No. 5 or equivalent) to wheel half packing before installation.



Lubtork Application
Figure 201

C. Disassembly – Nose Landing Gear Wheel (B.F. Goodrich)

WARNING: TIRE MUST BE COMPLETELY DEFLATED BEFORE LOOSENING ANY TIE BOLTS AND NUTS.

(1) Use a bead breaker or pneumatic tire dismounter to break tire beads away from wheel flanges.

CAUTION: DO NOT USE TIRE IRONS. DAMAGED WHEEL FLANGES OR TIRE BEADS MAY PREVENT PROPER AIR RETENTION.

(2) Remove nuts, washers, and bolts from wheel assembly.

NOTE: If an air wrench is used, hold bolts and turn nuts.

(3) Separate wheel halves and remove tire and packing.

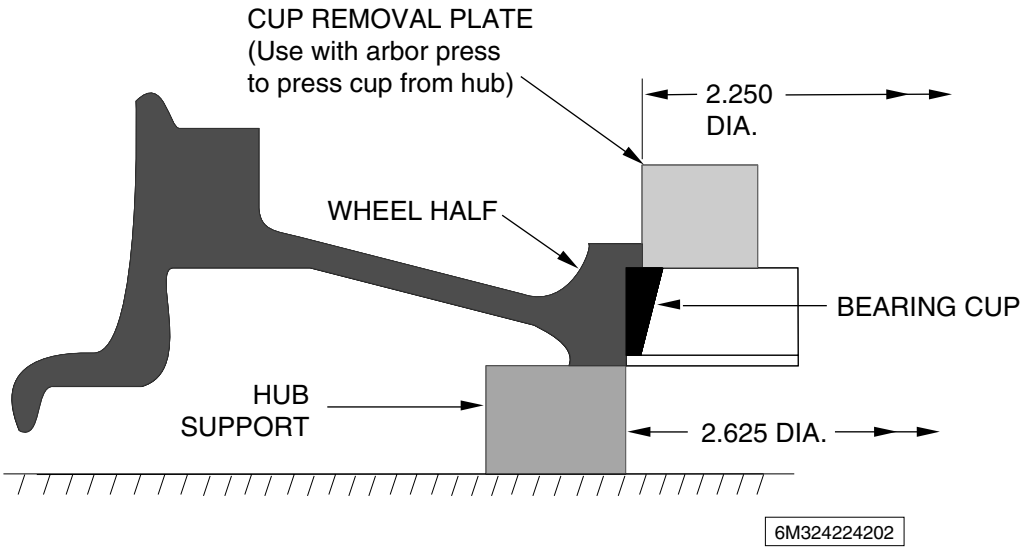
(4) Remove valve stem and grommet from outer wheel half assembly.

(5) If necessary, remove balance weights and attaching hardware from inner and outer wheel half assemblies.

NOTE: Replacement of balance weights in exact number and location from which removed, will avoid necessity of rebalancing wheel halves.

(6) Bearing cups are press fit into wheel halves and are to be removed only for replacement or wheel surface treatment. If cups are to be replaced, heat the wheel half to 175°F (79°C) maximum for 20 minutes and remove IAW Figure 202.

NOTE: Support the wheel hub while removing cups.



Bearing Cup Removal
Figure 202

D. Assembly – Nose Landing Gear Wheel (B.F. Goodrich)

(1) If bearing cups have been removed, install in wheel half bores, as shown in Figure 203.

(2) If necessary, install balance weights in same location from which removed to maintain original wheel half balance.

NOTE: If balancing of wheel halves is required, use standard balance procedure and balance within one ounce-inches.

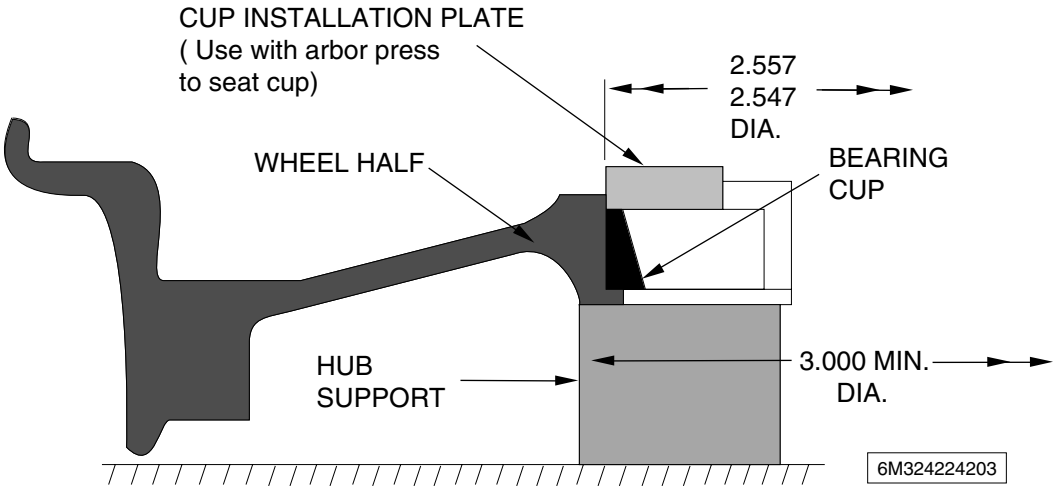
(3) Assemble grommet on valve stem. Thread valve stem into hole in inner wheel half. Torque valve stem 35 to 65 inch-pounds (3.95 to 7.34 N·m) dry torque.

(4) Install prelubricated (IAW step **3.,** B., (3), under Lubrication) packing into O-ring groove of inner wheel half. Make sure packing is properly seated before continuing assembly.

(5) Place tire in inner wheel half with red mark on tire bead adjacent to the tubeless tire valve stem previously installed in the wheel half.

(6) Align outer wheel half assembly with inner wheel half assembly so light sides (impression stamped L on outer flanges) are approximately 180° apart and bolt holes match. Assemble outer and inner wheel half assemblies.

NOSE LANDING GEAR WHEEL (B.F. GOODRICH) – MAINTENANCE PRACTICES



Bearing Cup Installation
Figure 203

(7) Apply a generous coat of thread compound, specification MIL-T-5544, to threads of bolts, faces of washers, and bearing face of nuts (see Figure 202). Countersunk face washers should be toward head of bolt. Torque to 100 inch-pounds (11.30 N·m), two nuts diametrically opposite. Stagger remaining nuts to 100 inch-pounds (11.30 N·m) torque.

NOTE: Do not use impact wrench when applying final torque. Use present hand torque wrench only.

(8) Using a pre-set hand torque wrench, final torque nuts 120 to 140 inch-pounds (13.52 to 15.82 N·m) in criss-cross sequence. Recheck bolts for proper torque before inflating tire.

WARNING: DO NOT INFLATE TIRE UNTIL ALL BOLTS ARE INSTALLED AND PROPERLY TIGHTENED. OVER-INFLATION OF TIRE MAY RESULT IN PERSONAL INJURY AND/OR PROPERTY DAMAGE.

(9) Verify inner valve is installed and tightly seated in valve stem.

(10) Make sure bearing cups and cones have been properly lubricated, see **3.**, B., Lubrication.

NOTE: If bearing cups have been removed, re-install in wheel half bores as shown in Figure 203.

(11) Wrap bearing cones and seals in waxed or greased moisture-proof paper. Insert each wrapped bearing cone and seal in applicable bearing bore of wheel. Seal opening with a paper dust cover secured to bearing bore with masking tape until ready for installation on aircraft.

NOSE LANDING GEAR WHEEL (B.F. GOODRICH) – MAINTENANCE PRACTICES

E. Testing – Nose Landing Gear Wheel (B.F. Goodrich)

WARNING: IT IS RECOMMENDED THAT ALL MOUNTED TIRES BE INFLATED INSIDE A METAL CAGE TO AVOID POSSIBLE INJURY TO PERSONNEL. OVER-INFLATION OF TIRE MAY RESULT IN PERSONAL INJURY AND/OR PROPERTY DAMAGE.

(1) Test tire and wheel assembly for air retention by inflating tire to specified operating pressure IAW tire specifications, as shown in Section 32-41-00, Table 201.

(2) Allow assembly to stand and stabilize for not less than 12 hours.

(3) Check for possible air leakage around tire beads and valve by applying a soap-water solution to these areas. Look for air bubbles. It is normal for air bubbles to foam at the sidewall vent holes.

(4) Check tire pressure and re-inflate to proper pressure to compensate for decrease in pressure caused by stretching and stabilizing of tire body.

(5) Allow tire and wheel assembly to stand an additional 24 hours. Check tire pressure. If pressure drop does not exceed five percent of full inflation pressure, proceed with installation of assembly on aircraft.

(6) If pressure drop exceeds five percent of full inflation pressure, check inner valve for proper seating, inflation valve grommet for leaking, or preformed packing for leaking, Conduct soap test.

3. Cleaning/Painting – Nose Landing Gear Wheels (B.F. Goodrich)

A. Cleaning – Nose Landing Gear Wheel (B.F. Goodrich)

WARNING: USE SOLVENTS IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND SKIN CONTACT WITH SOLVENT, PROTECT EYES.

CAUTION: DO NOT USE CLEANING SOLVENTS TO CLEAN RUBBER PARTS.

(1) Clean all metal parts in dry cleaning solvent (P-D-680, Type II or equivalent. Dry with low pressure Compressed air.

(2) Wipe grease seals and packing with a clean, dry cloth.

CAUTION: DO NOT SPIN DRY BEARING OR HANDLE WITH BARE HANDS.

(3) Wash bearing cones in uncontaminated cleaning solvent. Manually rotate bearing cage while submerged in cleaning solution. Dry bearing cones and repack with grease immediately after cleaning to prevent corrosion and seal in a clean container.

(4) Paint may be stripped by either an abrasive method or chemical stripper not harmful to aluminum. Avoid removing surface treatment under paint. Both methods of paint stripping are described as follows:

(a) Abrasive blasting IAW MIL-G-5634, Type II or III is recommended. This method will not remove surface treatment and does not require masking or removal of bearing cup.

NOSE LANDING GEAR WHEEL (B.F. GOODRICH) – MAINTENANCE PRACTICES

WARNING: USE PAINT STRIPPERS IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND SKIN CONTACT WITH STRIPPER.

(b) Paint may be removed from wheel halves using A-127-B solvent and lacquer thinner (Allied-Kelite Products Division) or equivalent. A-127-B stripper may be used full strength.

NOTE: Remove or mask bearing cup prior to paint stripping wheel half.

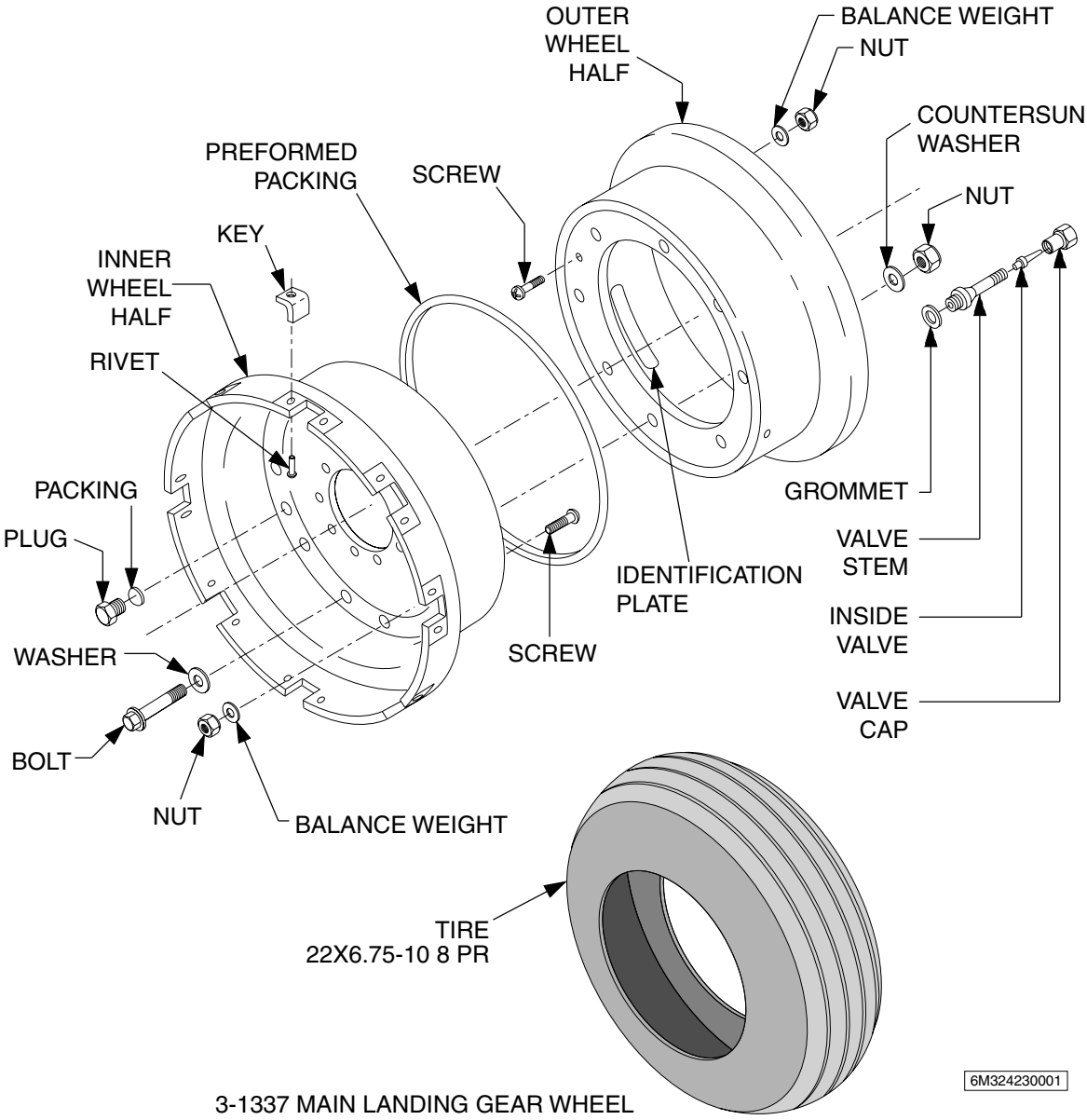
(c) Thoroughly rinse stripping compound from parts with clean, cold water. Dry parts with low pressure compressed air.

COROTATING MAIN LANDING GEAR WHEEL – DESCRIPTION AND OPERATION

1. General

A. 3-1337 Main Landing Gear Wheel (Corotating)

(1) The 3-1337 wheel is designed for use with a 22 X 6.75-10 eight ply tubeless tire and is mounted on a corotating wheel axle. This configuration is in use on SA226 series aircraft T210, 213, 253, and 254 only and is no longer a customer option item.

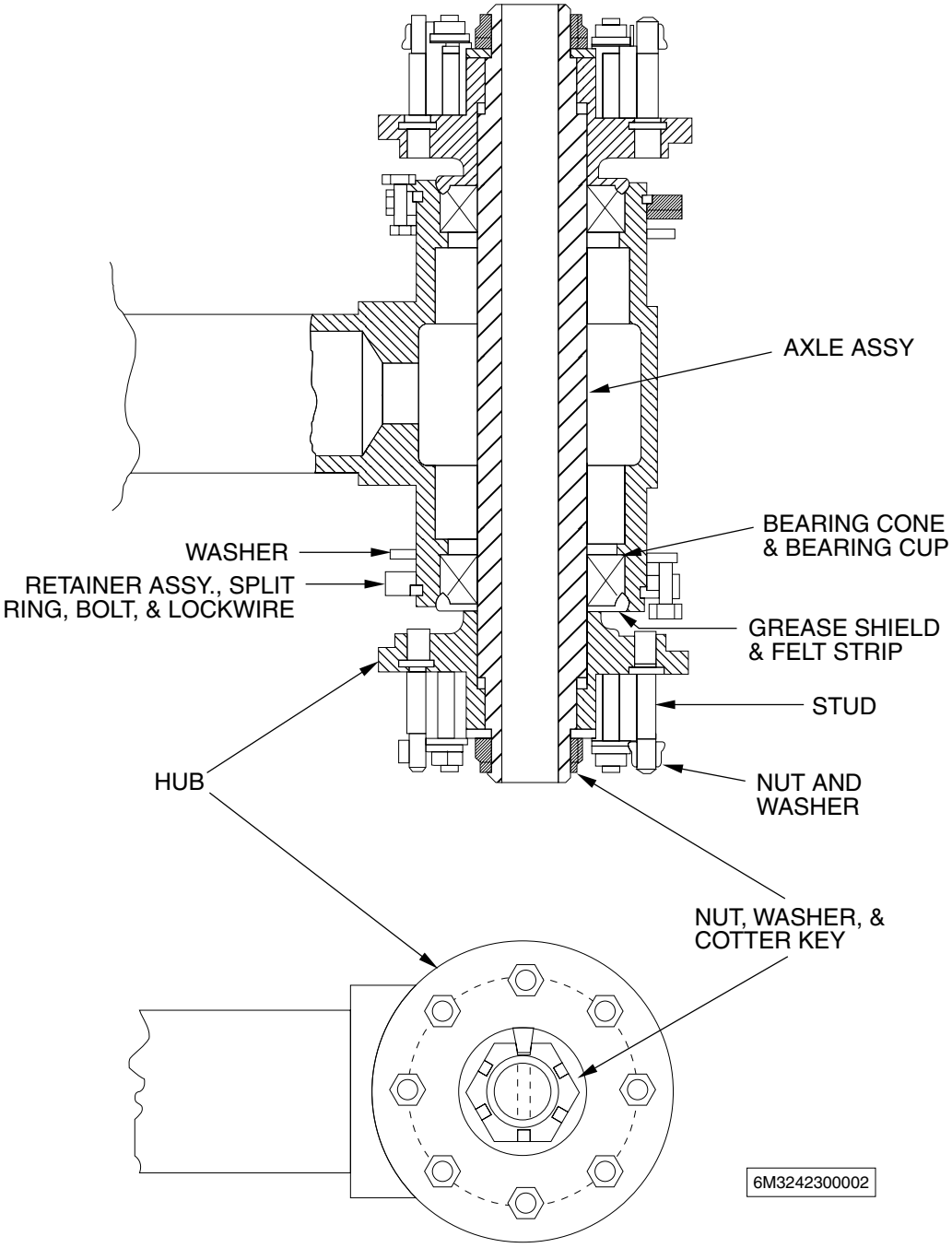


3-1337 MAIN LANDING GEAR WHEEL

3-1337 Main Landing Gear Wheel (Corotating)
Figure 1

COROTATING MAIN LANDING GEAR WHEEL – DESCRIPTION AND OPERATION

(2) The wheel consists of a an inner and outer wheel half bolted together with nuts and bolts. Washers are used beneath the nuts and bolt heads to prevent galling and stress concentration.



Corotating Wheel
Figure 2

(3) A packing in the groove inner wheel half seals the wheel halves against air leakage. An identification plate is fastened by adhesive to the outer surface of the outer wheel half.

(4) The following components are mounted on the inner wheel half. Torque keys are installed in slots in the flange area with rivets. The torque keys retain the lugs of the brake disc which rotates with the wheel, when the wheel and brake are mounted on the landing gear axle. Balance weights are installed on the inner and outer wheel half. The balance weights are held in place by screws and nuts.

(5) Aircraft manufactured with low pressure tires size 22 X 6.75 incorporate main gear struts with two corotating wheels on a common axle for each strut. The ends of the axle are made with a hexagonal shape, which fits closely into matching hexagonal holes in the center of the wheel hubs, causing both wheels and the axle to turn as one unit.

2. Axle Bearings

A. The axle is held in position in its housing by the two bearing inner races; the inner races are attached to the axle by means of a tight friction fit. Removal of the axle and inner races is accomplished by means of a puller, Fairchild P/N L00323-1. Installation may be accomplished with an inserter, Fairchild P/N L00323-7.

COROTATING MAIN LANDING GEAR WHEEL – MAINTENANCE PRACTICES

1. Removal/Installation – Corotating Main Landing Gear Wheel

A. Removal – Wheel to Hub

(1) Remove wheel to hub retaining nuts and washers.

(2) Separate wheel assembly from the hub.

B. Installation – Wheel to Hub

(1) Place wheel and hub together so that the hub studs protrude through the wheel attach holes from inner (brake disc) side of wheel assembly.

(2) Install stud washers and nuts; snug the nuts in a criss-cross pattern.

(3) After all nuts are snug, lube torque the stud nuts 120 to 130 inch-pounds (13.56 to 14.69 N·m), using the same criss-cross pattern.

C. Removal – Corotating Wheel and Hub

(1) Jack the aircraft.

(2) Deflate the tires.

(3) If tires, wheels, and hubs are to be disassembled, make orientation marks (chalk, china marker, tape, etc.) to provide for later reassembly of parts in the same relative position.

(4) Remove cotter pins from the axle ends and loosen the axle nuts.

NOTE: Loosen the nut on the outboard end of the axle before loosening the axle nut on the inboard end.

(5) Remove axle nuts and wheel assemblies.

D. Installation – Corotating Wheel and Hub

(1) Check for burrs or rough threads on axle and axle nuts.

(2) Apply MIL-T-5544 thread compound to all threads and bearing surfaces of washers and nuts prior to installation.

(3) Mount hub, wheel, and tire assemblies on the axle. Take care to ensure engagement of brake disc lugs in the torque key lined slots of the inner wheel half flange.

(4) Install and tighten the axle nut and washer on the inboard wheel to a snug fit; align the locking hole; and install the cotter pin. Hold the nut on the inboard wheel and install the second washer and nut on the outboard end of the axle. Lube torque the outboard axle (wheel retaining) nut 40 to 50 foot-pounds (54.24 to 67.80 N·m).

(5) Loosen the outboard axle nut to zero torque. Then final lube torque 20 to 30 foot-pounds (27.12 to 40.68 N·m). If locking holes are not aligned, tighten the axle nut to align with the first hole.

(6) Install cotter pin at the outboard axle nut and final check for free rotation of wheels and axle.

E. Disassembly

WARNING: EXERCISE CARE WHEN REMOVING VALVE CORE FROM VALVE STEM. THE VALVE CORE SHOULD BE REMOVED TO ENSURE COMPLETE DEFLATION.

CAUTION: DO NOT USE TIRE IRONS. THEY MAY DAMAGE THE WHEEL FLANGES OR TIRE BEADS AND PREVENT PROPER AIR RETENTION.

(1) Break tire beads away from wheel flanges by using a bead breaker.

(2) Remove nuts, washers, and bolts from wheel. Remove flange nuts.

(3) Separate wheel halves and remove the tire. Remove hub flange.

(4) If packing is to be reinstalled, place it on a clean sheet of paper. Wipe clean with a dry cloth and relubricate using wheel bearing grease before installation.

NOTE: Torque keys should only be removed if excessively worn, cracked, loose, or for zyglo inspection of the wheel half.

F. Removal – Axle Bearings

(1) After removing both wheels from the strut, install the puller P/N L00323-1 over either end of the axle, with the three puller bolt heads pointing away from the strut.

(2) Install a wheel retaining washer and nut to transmit force from the puller to the axle.

(3) Apply pulling pressure between the axle and its housing by turning the three puller bolts alternately in small increments.

(4) When the axle pulls free. the grease shield and bearing inner race nearest the puller remain with the axle. The second grease shield and inner race are easily removed from the axle housing.

NOTE: If the inner race remaining with the axle is to be removed, measure carefully and make a notation describing the position of the race on the axle. This information will be useful when reinstalling a race on the axle prior to reassembly to the strut.

(5) The race remaining on the axle may be removed by putting the axle into its housing and using the puller at the other end or by means of a standard bearing press.

(6) After removing the axle and inner races, the bearing outer races may be removed from the axle housing with the aid of a slide hammer.

G. Installation – Axle Bearings

(1) Press both bearing outer races into the axle housing with the large diameter of the conical surface toward the open ends of the housing.

(2) Install one bearing inner race on the axle with the apex of the cone toward the center of the axle's length. Position the race using the dimension previously measured during the removal.

(3) Inspect the axle and bearings and ensure that all surfaces are clean and free of contamination before applying grease.

(4) Before installing the axle and bearings into the axle housing, pack the bearings and inner races with MIL-G-81322 grease (Aeroshell Grease 22, Mobil 28, Royco 22S, or other grease meeting the specification). Apply a light coat of the grease to the bare axle and outer race surfaces.

(5) Install the axle with the inner race into the housing. Make sure the proper inner and outer cones are matched.

(6) Install the remaining inner cone using the inserter tool (Fairchild P/N L00323-7). The inserter inner hub screws onto the axle end threads. The outer hub bears against the bearing inner race and force is applied by turning the three bolts alternately in small increments. When the axle and bearing assembly is properly installed in the housing, the axle should tun freely with no end or side play. The axle ends must extend evenly from either side of the housing to assure proper installation of the wheel hubs.

(7) Inspect the grease shields for dents, cracks, or other obvious defects. Replace with new parts if found to be defective. If still serviceable, clean the felt strip around the edge or replace with a new item Before installing the grease shield, the felt strip must be thoroughly dried and packed with MIL-G-25760A grease (Aeroshell #16).

(8) Install the grease shields, one at a each end of the axle housing. Put the deeper cupped face toward the bearings.

2. Inspection/Check – Corotating Main Landing Gear Wheel

A. Inspection (while equipment is on aircraft)

(1) Inspect wheel half flanges for cracks and corrosion.

(2) Check for loose bolts and nuts and retighten or replace as required. Check flange nuts also.

NOTE: No repair or replacement is recommended while equipment is on the aircraft except the tightening or replacing of loose bolts, nuts, and flange nuts.

(3) For tire change only, deflate tire, remove the eight nuts and washers that hold the hub to the wheel. Remove tire and wheel from axle and mount new tire and wheel, reinstall the eight nuts and washers. Inflate the tire. (See Tire Maintenance practices Section 32-41-00 for proper inflation).

B. Inspection (while disassembled)

(1) Magnaflux bolts for cracks and breaks.

(2) Inspect wheel halves for cracks and breaks with dycheck or fluroescent penetrant, noting in particular the bead seat, tube well, and web junction areas.

(3) Visually inspect all metal parts for pitting, corrosion, cracks, breaks, eneven wear, and other surface defects.

(4) Inspect packing sealing surfaces for surface smoothness.

(5) Inspect packings for pits, cuts, and other defects. Replace as necessary.

(6) Inspect torque keys for wear at the area where the brake disc lugs contact the torque keys. Replace torque keys if cracked, worn, or broken.

3. Approved Repairs – Corotating Main Landing Gear Wheel

A. Repair and Replacement

(1) Repair scratches, nicks, and other surface blemishes on wheel halves by sanding with emery cloth, removing as little metal as possible. Polish the repaired surface.

(2) Paint repaired areas with two coats of zinc chromate primer MIL-P-8585 and one coat aluminum lacquer MIL-L-7178.

NOTE: Apply only one coat of zinc chromate primer and no finish coat on packing groove and mating surfaces of the wheel halves.

(3) Replace all parts damaged beyond repair.

(4) Replace packings at each overhaul.

B. Lubrication

(1) Lubricate threads of bolts, nuts, and bearing surface of washers with thread compound, MIL-T-5544.

(2) Lubricate packings with a light coat of bearing grease prior to installation.

C. Assembly

(1) If balance weights have been removed, reinstall in their original location.

(2) Tighten nuts 120 to 130 inch-pounds (13.56 to 14.69 N·m). Lube torque using a preset hand wrench in criss-cross sequence across the diameter. Recheck the torque once around before inflating tire.

(3) Recheck torque once around before inflating tire (see 32-41-00 for tire inflation procedures).

NOTE: Ensure all bolts, nuts, and washers have been properly lubricated before assembling.

D. Testing

(1) After tire growth has been stabilized, ensure that the wheel holds normal pressure for 24 hours with not more than four percent (4%) loss in pressure.

MAINTENANCE MANUAL

COROTATING MAIN LANDING GEAR WHEEL – MAINTENANCE PRACTICES

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MAINTENANCE MANUAL

BRAKE SYSTEM – DESCRIPTION AND OPERATION

1. General

A. Except for the anti-skid/power brake system (see 32-43-20), the standard brake system installed on SA226 series aircraft is operated manually by depressing the toe of the brake pedals mounted on pilot's and copilot's rudder pedals. The force exerted on the brake pedals is transmitted to the respective wheel brakes via hydraulic brake lines.

B. Fluid for the system is supplied from a small brake system hydraulic reservoir mounted in the nose baggage compartment.

(1) The reservoir is serviced via a reservoir cap located on the exterior of the left side of the nose baggage compartment with MIL-H-83282 hydraulic fluid.

(2) The hydraulic system for the brakes is completely independent of the main (2,000 psi) hydraulic system used for landing gear and wing flap actuation.

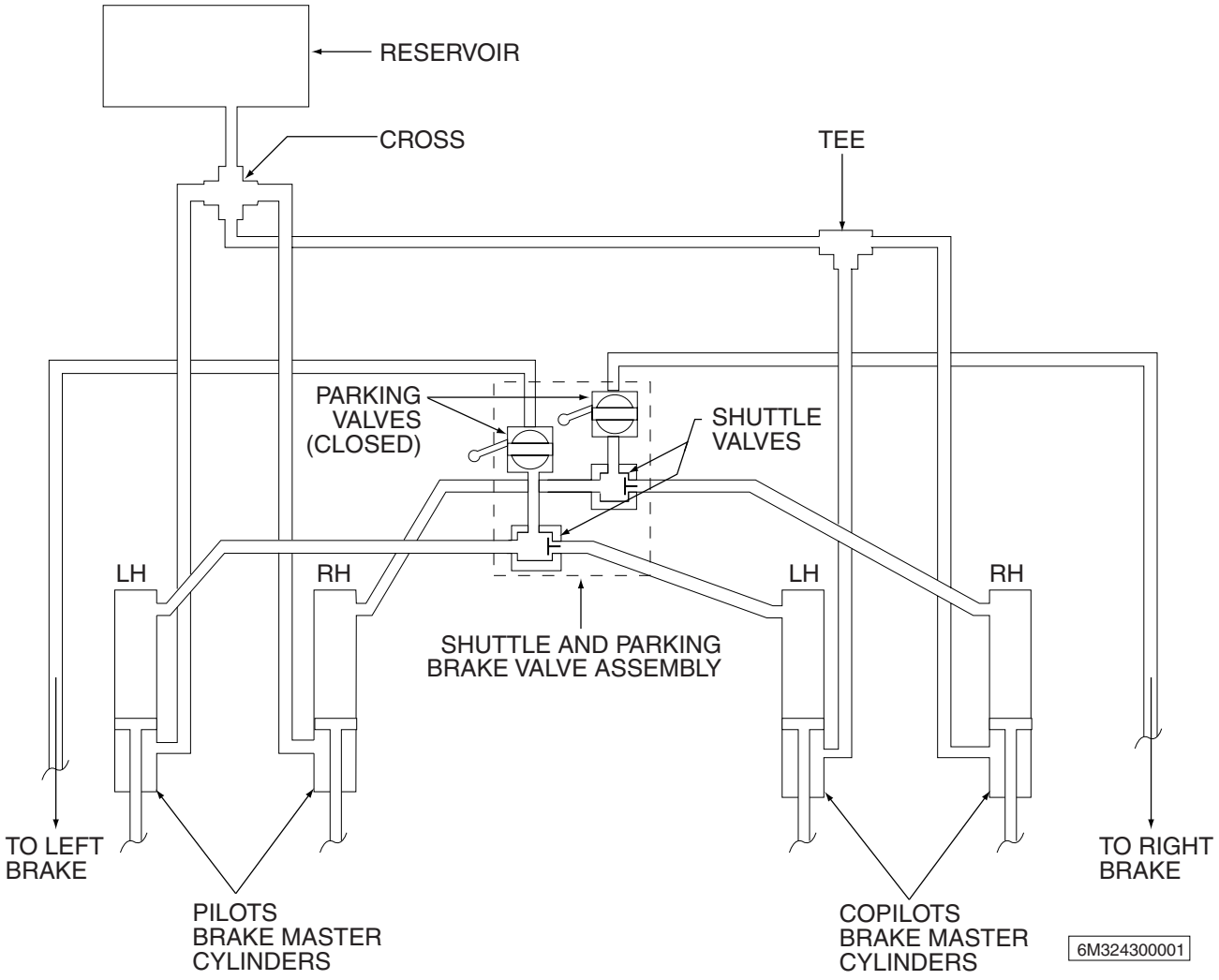
C. The brake reservoir supplies fluid to four brake master cylinders and to the hydraulic lines from the master cylinders, through brake shuttle valves, to the wheel brakes. Reservoir capacity is approximately one pint and total brake system capacity is approximately three pints of fluid.

D. On SA226 T201 through T265, T167, AT001 through AT-45, and TC201 through TC222, unless retrofitted, one hydraulic line provides fluid from the reservoir to the left brake master cylinder (see Figure 1).

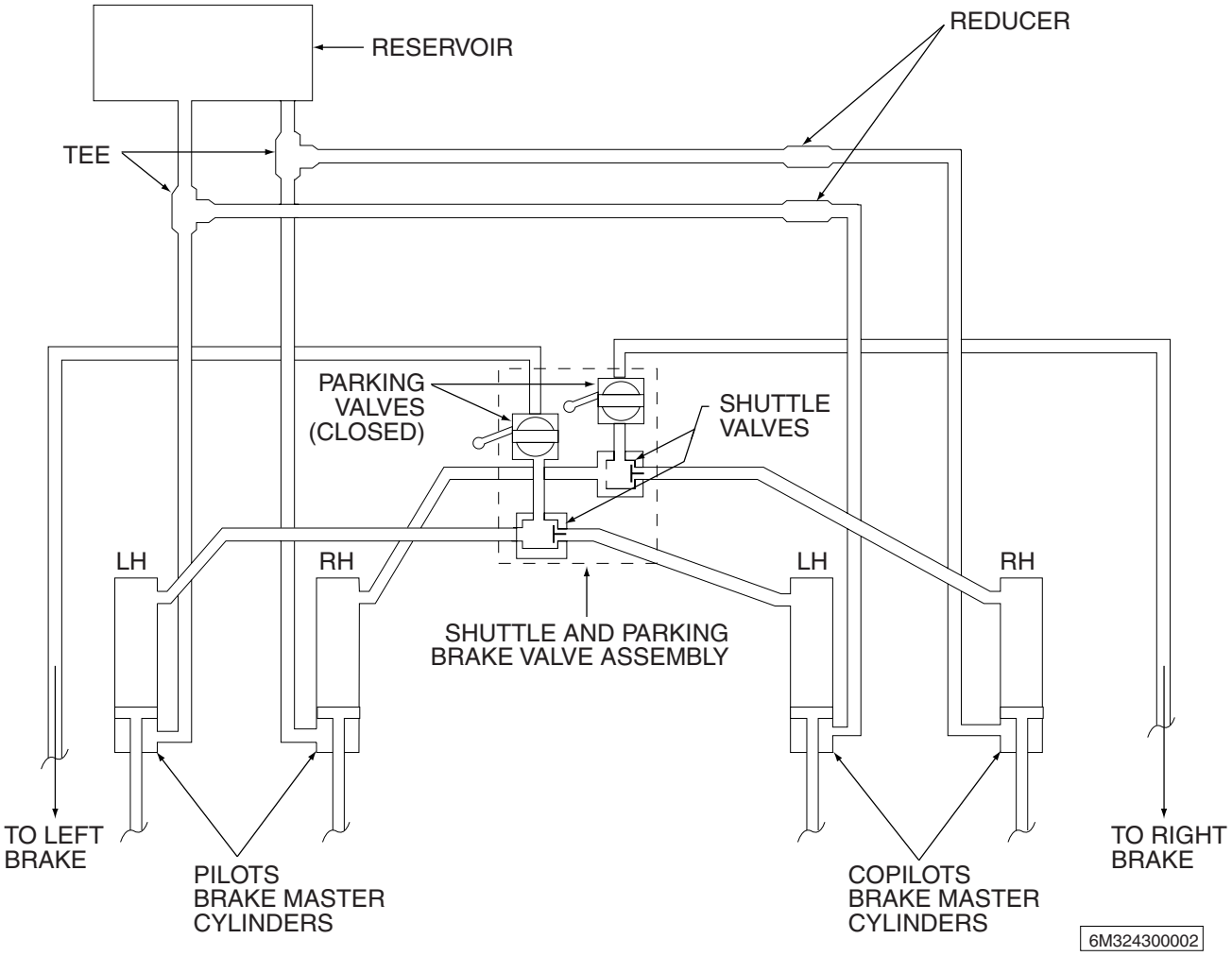
E. On SA226 T266, T268 and up, AT046 and up, TC222E and up, and retrofitted airplanes, one hydraulic line provides fluid from the reservoir to the left brake master cylinders and a second hydraulic line is installed to provide fluid from the reservoir to the right brake master cylinders. The purpose of installing two lines instead of one is for more efficient and reliable bleeding of the brake system (see Figure 2).

F. Two types of brake assemblies are in use on SA226 series aircraft. The B.F. Goodrich 2-1203 brake assembly is standard for all SA226 aircraft; whereas, the B.F. Goodrich 2-1268 brake assembly is a customer option installation and is used only on aircraft employing corotating wheels (see Maintenance Practices – Wheels). Separate maintenance procedures applicable to each type of brake assembly are provided for in this chapter.

G. On anti-skid/power brake equipped aircraft, the 2-1203 brake assembly is used but has a different automatic brake adjuster assembly. Thus the brake in this instance has been redesignated as a 2-1203-1 brake assembly.



Brake System Schematic
Figure 1



Brake System Schematic
Figure 2

MAINTENANCE MANUAL

BRAKE SYSTEM – MAINTENANCE PRACTICES

1. Servicing – Brake System

A. Servicing Equipment – Brake System

(1) MIL-H-83282 Hydraulic Fluid.

(2) Brake Pressure Type Bleeder Device (Pressure Pot).

(3) 0-1000 psi Hydraulic Gauge.

(4) 7.0 psi Air Pressure Source.

(5) Feeler Gauge (0.016 to 0.019 inch).

B. Servicing Procedure – Brake System

WARNING: THE BRAKES ARE DESIGNED TO FUNCTION AT SPECIFIED PRESSURES (UP TO 750 PSI) (52.73 KG/CM^2). EXCESSIVE PRESSURE CAN CAUSE STRUCTURAL DEFLECTION AND OVERCOMPENSATION BY THE AUTOMATIC ADJUSTERS, RESULTING IN A TIGHT BRAKE CONDITION. SHOULD THIS CONDITION OCCUR, THE BRAKE ASSEMBLY MUST BE REMOVED FROM THE AIRCRAFT, DISASSEMBLED AND REASSEMBLED TO RESET THE ADJUSTERS.

NEW BRAKES MAY NOT HOLD AIRCRAFT FROM MOVING AT HIGH POWER SETTINGS UNTIL BRAKES ARE WORN IN.

OVERHEATED BRAKES SHOULD BE INSPECTED FOR DAMAGE PRIOR TO NEXT FLIGHT.

CAUTION: DO NOT ALLOW RESERVOIR TO BECOME EMPTY DURING BLEEDING PROCEDURE.

DO NOT TAMPER WITH BRAKE ADJUSTER SETTING

(1) Pressure Bleeding the System:

NOTE: If equipment is available, system may be backfilled from brake bleeder port. Brakes should be bled until pedal becomes firm.

(a) Ensure the brake system reservoir is fully serviced with brake fluid (MIL-H-83282 Hydraulic Fluid).

NOTE: The reservoir filler cap is located at the left side aft of the nose baggage compartment door. Brake reservoir capacity is one pint of fluid. Total brake system capacity is approximately 3 pints of fluid.

(b) Fill reservoir as needed (fluid should be level with the bottom of the filler cap port).

CAUTION: WHEN PRESSURE BLEEDING OF THE BRAKE SYSTEM IS ATTEMPTED WITH THE BRAKE RESERVOIR CAPPED, THE RESULT WILL BE VENTING OF HYDRAULIC BRAKE FLUID INTO THE COCKPIT CABIN AREA. PRIOR TO BLEEDING THE SYSTEM, THE RESERVOIR VENT LINE SHOULD BE DISCONNECTED FROM THE RESERVOIR. (ACCESS MAY BE GAINED TO THE BRAKE RESERVOIR THROUGH THE NOSE BAGGAGE COMPARTMENT.

(c) Connect clear plastic tubing to reservoir vent port.

(d) Vent the opposite end of the tubing into a clean container in order to retrieve vented fluid during pressure bleeding.

NOTE: Clear plastic tubing should be utilized in order to observe unwanted aerated fluid in the system.

(e) Connect pressure port fluid lines to both brake bleed valve ports (one located on each of the two brake assemblies of each main landing gear strut) at the wheel being serviced.

(f) Carefully and slowly apply bleed pressure by operating both brake pedals of the system (pilot and copilot) to ensure proper bleeding of the parallel brake master cylinder arrangement.

(g) With the brake pedal fully depressed and brake bleed valve ports opened (master cylinder piston bottomed), check rigging of rudder/brake pedal and brake master cylinder (see Figure 201).

(h) Repeat procedures 1., B., (1), e, f, and g for opposite main landing gear brakes.

(2) Gravity Bleeding the System:

(a) Ensure the brake system reservoir is fully serviced.

(b) Connect an overflow tube to the bleed valve port of brake assembly being bled in order to retrieve aerated fluid into a clean container.

(c) While applying the brakes with a steady even pressure upon the pedals, open the bleeder valve slowly forcing hydraulic fluid and any existing air out of the system.

(d) Close the bleeder valve before brake pressure is exhausted. Repeat this procedure until one–half pint of hydraulic brake fluid runs free of air bubbles.

(e) Tighten bleeder valve after bleeding is completed.

(f) Check reservoir level and add fluid as necessary as air is expelled from system.

(g) Repeat steps 1., B., (2), c, d, and e for opposite main landing gear brake assembly.

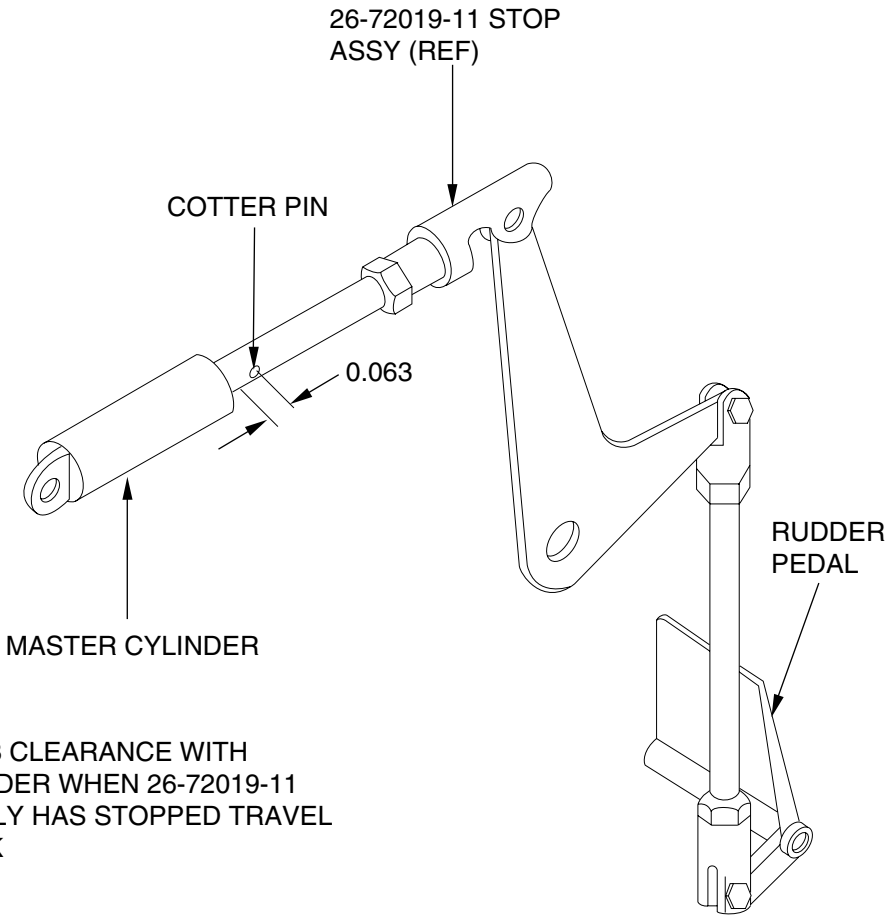
(3) Bench Bleeding the System:

NOTE: The following procedures are recommended to remove aerated fluid from the piston housing when the brake assembly is not on the aircraft.

BRAKE SYSTEM – MAINTENANCE PRACTICES

(a) With the brake assembly removed from the aircraft, use a hydraulic hand pump capable of pressurizing to 400 psi (28.12 kg/cm^2).

(b) Install hydraulic fluid pressure line of the hand pump to the supply port of the brake assembly.



CAUTION: PRESSURE BLEEDING WITH THE RESERVOIR CAPPED WILL RESULT IN VENTING FLUID INTO THE CABIN. PRIOR TO BLEEDING THE SYSTEM, DISCONNECT THE RESERVOIR VENT LINE FROM THE PRESSURE BULKHEAD AND ATTACH VENT TUBING TO COLLECT OVERFLOW.

CHECK RESERVOIR LEVEL AND ADD AS NECESSARY AS AIR IS EXPELLED FROM THE SYSTEM.

6M324300201

Brake Master Cylinder
Figure 201

(c) Install clear plastic tubing to bleeder port of the brake assembly placing opposite end of tubing into clean container for fluid retrieval.

NOTE: Use clear plastic tubing to observe any aerated fluid being bled from the brake assembly.

(d) With the brake assembly placed in an upright position (as installed on aircraft), and bleeder port closed, apply 400 psi (28.12 kg/cm^2) hand pump pressure.

(e) Slowly open bleeder port to relieve pressure and any aerated fluid.

(f) Place the brake assembly onto a flat and level work area with the piston housing side of the brake assembly facing up (this is done to insure all air in the assembly rises to the top of the piston bore wells). Apply 400 psi (28.12 kg/cm^2) pressure with the bleeder port closed.

(g) With the brake assembly held in the above configuration, agitate the assembly in all directions. Return to the above configuration to further assure all trapped air within the piston bore channels has risen to the top of the piston bore wells.

(h) Slowly open the bleeder port. Observe discharged aerated fluid. Apply 400 psi (28.12 kg/cm^2) pump pressure and repeat bleeding until aerated fluid is no longer observed.

(i) Place the brake assembly in an upright position (as when installed on aircraft) and apply hand pump pressure to 400 psi (28.12 kg/cm^2) with bleeder port closed.

(j) Rotate the brake assembly back and forth and open the bleeder port. Repeat until aerated fluid is no longer observed.

(k) With the brake in an upright position and bleeder port closed, disconnect the hand pump supply line and install the supply port fitting to brake assembly.

(l) Install the brake assembly onto the aircraft (see Section 32-43-10).

(m) Connect hydraulic fluid supply line from the aircraft to the brake assembly's supply port while a slight fluid pressure is being maintained upon the brake pedals.

(n) After completion of brake assembly installation, bleed the brakes again as outlined in the Servicing procedures of this chapter.

2. Adjustment/Test – Brake System

A. Functional Test – Brake System

(1) Apply normal brake pedal force and observe brake pedal linkage operation.

(2) Verify there no binding, interference or overcenter operation of the brake pedal system.

MAINTENANCE MANUAL

BRAKE SYSTEM – MAINTENANCE PRACTICES

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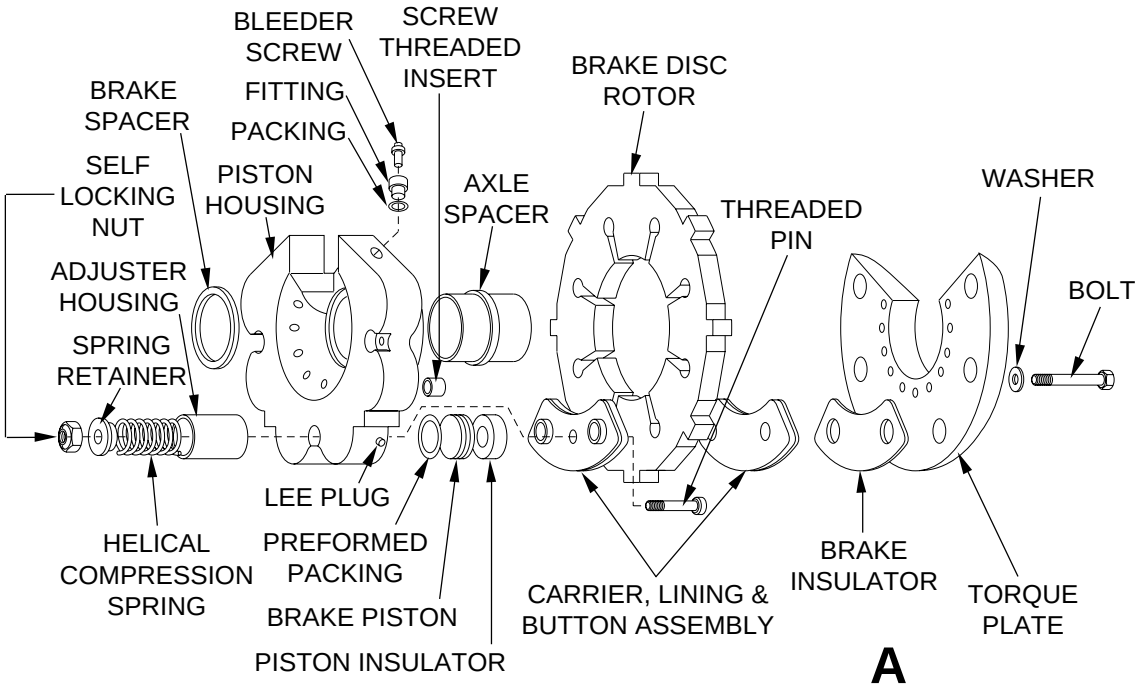
MAINTENANCE MANUAL

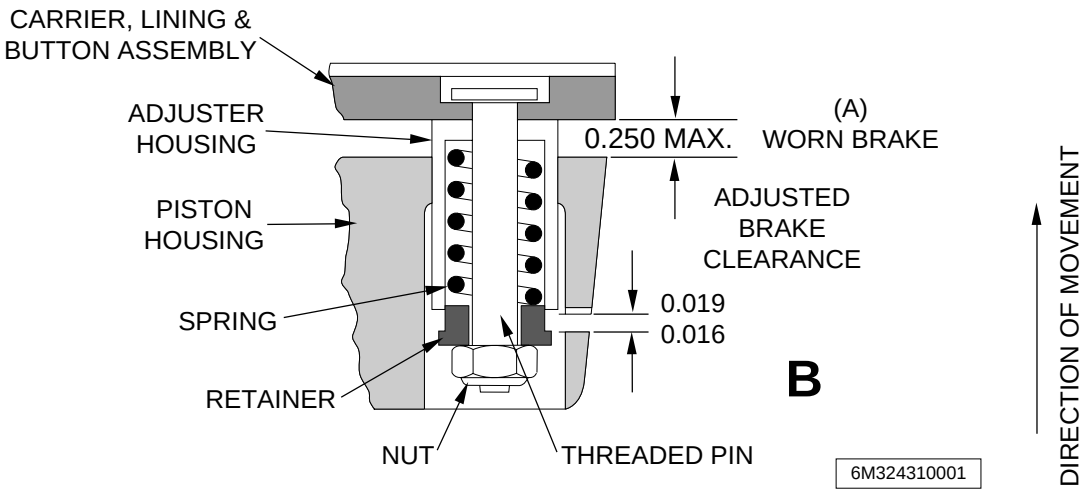
2-1203 MAIN LANDING GEAR BRAKE – DESCRIPTION AND OPERATION

1. General

A. The brake is designed for use with MIL-H-83282 hydraulic fluid and to withstand 750 psi (52.73 kg/cm^2) operating pressure with zero back pressure.

B. The forged aluminum torque plate is secured to the forged aluminum piston housing assembly with nine bolts and washers (see Figure 1A, only one each is shown).





Automatic Brake Adjuster
Figure 1

2-1203 MAIN LANDING GEAR BRAKE – DESCRIPTION AND OPERATION

C. Three stationary segmented carrier, lining, and torque button assemblies (only one assembly is shown) are installed on each side of the brake rotor disc.

(1) The torque button assemblies act as anti-rotation drives and are inserted in mating holes in the torque plate and the piston housing. Heat is shielded from the torque plate and piston housing assembly by the three brake insulators and six piston insulators (one of each is shown).

D. An axle spacer and brake spacer provides proper seating of the brake when mounted on the landing gear axle. A bleeder assembly, consisting of the bleeder screw, fitting, and packing is installed on the piston housing to facilitate bleeding of the brakes.

E. Six brake pistons are installed within bores in the piston housing. A preformed packing, seated in the piston outer diameter groove, seals against hydraulic fluid leakage from the piston housing.

F. Three automatic adjusters seated in the piston housing wells are provided and each adjuster is comprised of an automatic adjuster housing enclosing a spring retainer, helical compression spring, and threaded pin secured by a self-locking nut. The threaded pin is attached to the carrier, lining, and torque button assembly thereby retracting the assemblies after release of brake pressure. Thus, each adjuster provides retraction as well as adjustment.

G. Hydraulic pressure is applied from the hydraulic master cylinder of the aircraft to the brakes through hydraulic tubing. Hydraulic pressure to the brake assembly forces the brake pistons to press the three carrier, lining, and button assemblies against the brake rotor disc and against the three carrier, lining, and button assemblies installed on the torque plate.

2. Description and operation (2-1203-3 Brake Assembly)

A. Figure 1B shows the automatic brake adjuster retracted to full brake clearance with no pressure applied to the brake. When pressure is applied to the pistons, the carrier lining, and button assembly is forced in the direction of the arrow. The threaded pin is pulled in the same direction by the movement of the carrier, lining, and button assembly compressing the helical compression spring until the spring retainer, held by the self-locking nut, seats against the automatic adjuster housing. At this point all the brake clearance is taken up and braking follows in proportion to the brake pressure applied.

B. As the brake lining wears, the extra movement of the carrier and lining assembly will, under the pressure of braking, pull the automatic adjuster housing assembly forward into the brake body.

C. When the brakes are released, the helical compression spring will push the spring retainer back to the same clearance relation as before, while the automatic adjuster housing stays in the advanced position. The carrier, lining, and button assembly is returned to the preset clearance (against the automatic adjuster housing) by the threaded pin. In this manner, the brake clearance is automatically adjusted.

D. As an option, brakes may be operated without the automatic adjusters, if desired. Without the automatic adjusters, the brake will exhibit a light dragging with the brakes off, due to the pressure head in the brake system. This condition is considered normal and is not detrimental to tire or brake lining life.

2-1203 MAIN LANDING GEAR BRAKE – MAINTENANCE PRACTICES

1. Removal/Installation – 2-1203 Main Landing Gear Brake

NOTE: Wheel assembly 3-1357-1 must be used with 2-1203-3 brake assemblies. Aircraft with anti-skid/power brakes, must be first converted to a manual brake system IAW SB226-32-038, if the 2-1203-3 brake is to be installed.

A. Removal – Brake

CAUTION: PLACE IDENTIFICATION TAGS ON LINES AS THEY ARE REMOVED.

(1) Jack aircraft.

(2) Deflate tire (see wheel removal procedures, Section 32-40-00).

(3) Remove the axle nut and keyed washer and dismount the wheel and tire assembly from the axle of the strut.

(4) After removing the main wheel and tire assembly, remove the hydraulic lines and cap open lines and ports.

(5) Remove the brake assembly by sliding it off over the axle.

B. Installation – Brake

NOTE: Brakes are normally shipped from the factory completely assembled and ready for installation.

(1) Prior to installation, check for burrs or rough threads on axle and axle nut.

(2) Apply thread compound (MIL-T-5544) to threads and all bearing surfaces of washers and nuts prior to installation.

(3) Install the axle spacer into the piston housing of brake assembly.

(4) Mount one brake in place onto the axle making certain that the brake spacer bottoms onto the strut.

NOTE: Apply a thin coat of bearing grease (MIL-G-3545 or MIL-G-81322 on axle nut threads.

(5) Mount the wheel, tire, and bearing assembly IAW 32-40-00 **2.**, B. Installation – Main Wheel.

(6) Remove the shipping plugs from the brake inlet ports and install the brake hydraulic fluid line to the inlet port of the piston housing.

(7) Repeat previous procedures to install remaining brake and wheel assemblies.

(8) Bleed the brakes (see brake servicing procedures, Section 32-43-00).

2-1203 MAIN LANDING GEAR BRAKE – MAINTENANCE PRACTICES

2. Inspection/Check – 2-1203 Main Landing Gear Brake

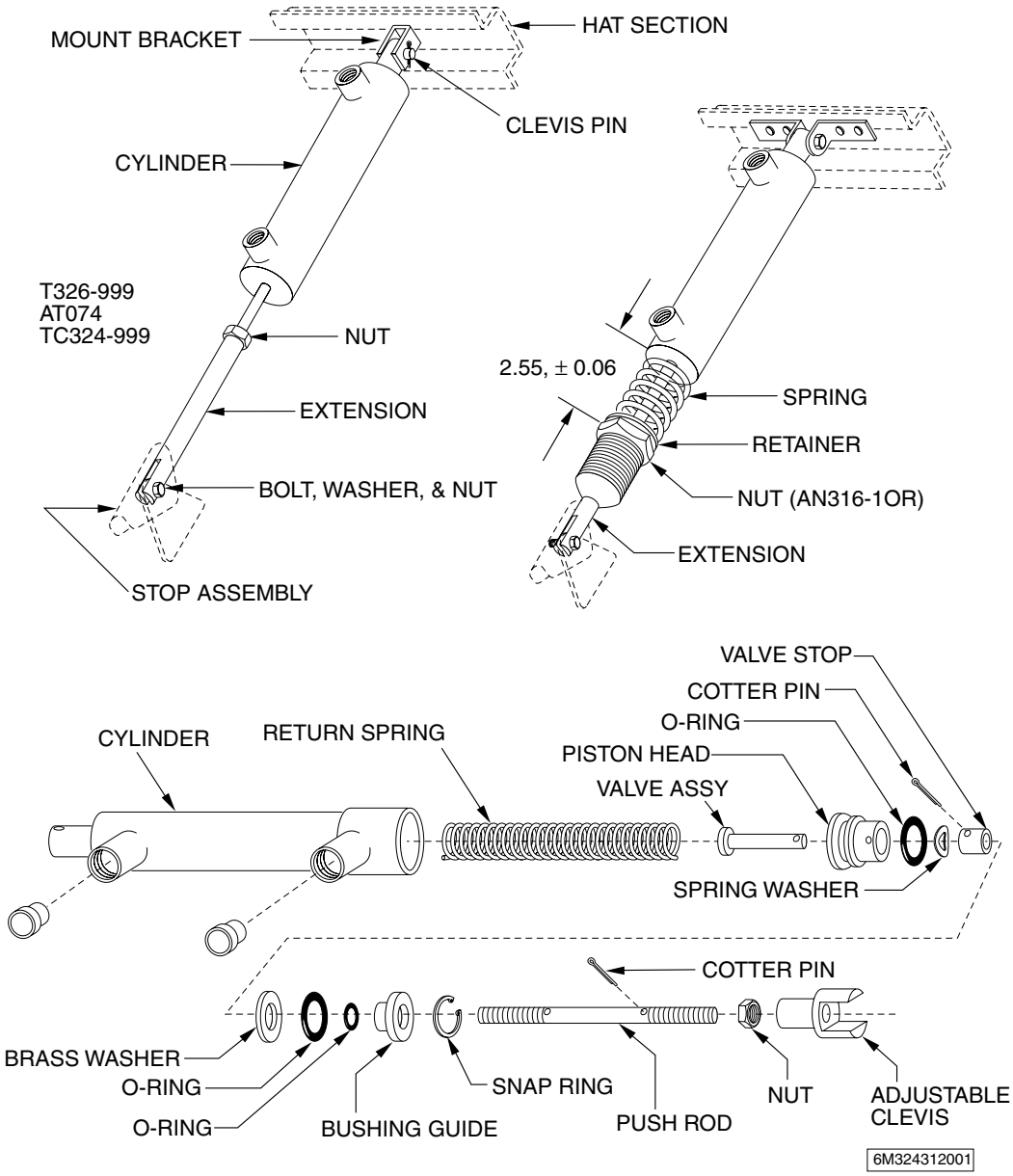
A. Refer to Vendor Inspection Manual for 2-1203 Main Landing Gear Brake Inspection procedures.

MAINTENANCE MANUAL

BRAKE MASTER CYLINDERS – DESCRIPTION AND OPERATION

1. General

A. Four brake master cylinders are mounted on the forward pressure bulkhead. One master cylinder is actuated by each of the four toe brake pedals mounted on the pilot's and copilot's rudder controls. The brake master cylinders are of the compensating type which allow replacement of hydraulic fluid from reservoir in case of system leakage.



Brake Master Cylinder
Figure 1

BRAKE MASTER CYLINDERS – DESCRIPTION AND OPERATION

B. On SA226 series aircraft T271 and up, AT051 and up, and TC222EE and TC225 and up, springs are installed on each brake master cylinder to ensure positive return of brake pedals (see Figure 1). Fairchild Aircraft Service Bulletin 32-011 provides installation procedures for brake return springs on aircraft not so equipped.

MAINTENANCE MANUAL

BRAKE MASTER CYLINDERS – FAULT ISOLATION

1. Troubleshooting – Brake Master Cylinders

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Solid pedal – limited or no braking action	Worn or glazed brake linings	Replace brake linings.
Unable to hold pressure	Leak in system	Repair leak.
	Faulty master cylinder seals	Replace master cylinder.
	Defective master cylinder	Replace master cylinder.
Spongy brake pedals	Air in system	Bleed system.
Parking brakes will not hold	Air in system	Bleed system.
	Defective parking brake valve	Repair or replace valve.
Brakes grab	Brake linings contaminated with oil	Replace linings.
	Brake disc warped	Replace linings.

BRAKE MASTER CYLINDERS – MAINTENANCE PRACTICES

1. Removal/Installation – Brake Master Cylinders

A. Removal – Brake Master Cylinders

(1) Gain access to brake master cylinders forward of pilot/copilot's seats.

CAUTION: PLACE IDENTIFICATION TAGS ON LINES AS THEY ARE REMOVED.

(2) Remove and plug hydraulic brake lines.

(3) Remove upper and lower attaching cotter pins, nuts, washers, and bolts securing cylinder.

(4) Remove cylinder.

B. Installation – Brake Master Cylinders

(1) Position the cylinder and secure with bolts, washers, nuts, and cotter pins.

NOTE: Ensure all bolts installed through the aircraft ends of the rod assembly between the rudder pedals and torque tube bellcrank, and the bolts installed through the stop assembly of the brake master cylinder assembly are free turning. Installation of attaching bolts too tight will restrict the return of the brake master cylinder piston and the rudder pedals to their normal position.

(2) Adjust the AN316-10R nut to ensure length of spring is 2.55, ± 0.06 inches (see Figure 1).

(3) Remove identification tags from brake lines.

(4) Install the brake hydraulic lines.

(5) Bleed the brake system (see brake servicing procedures, Section 32-43-00).

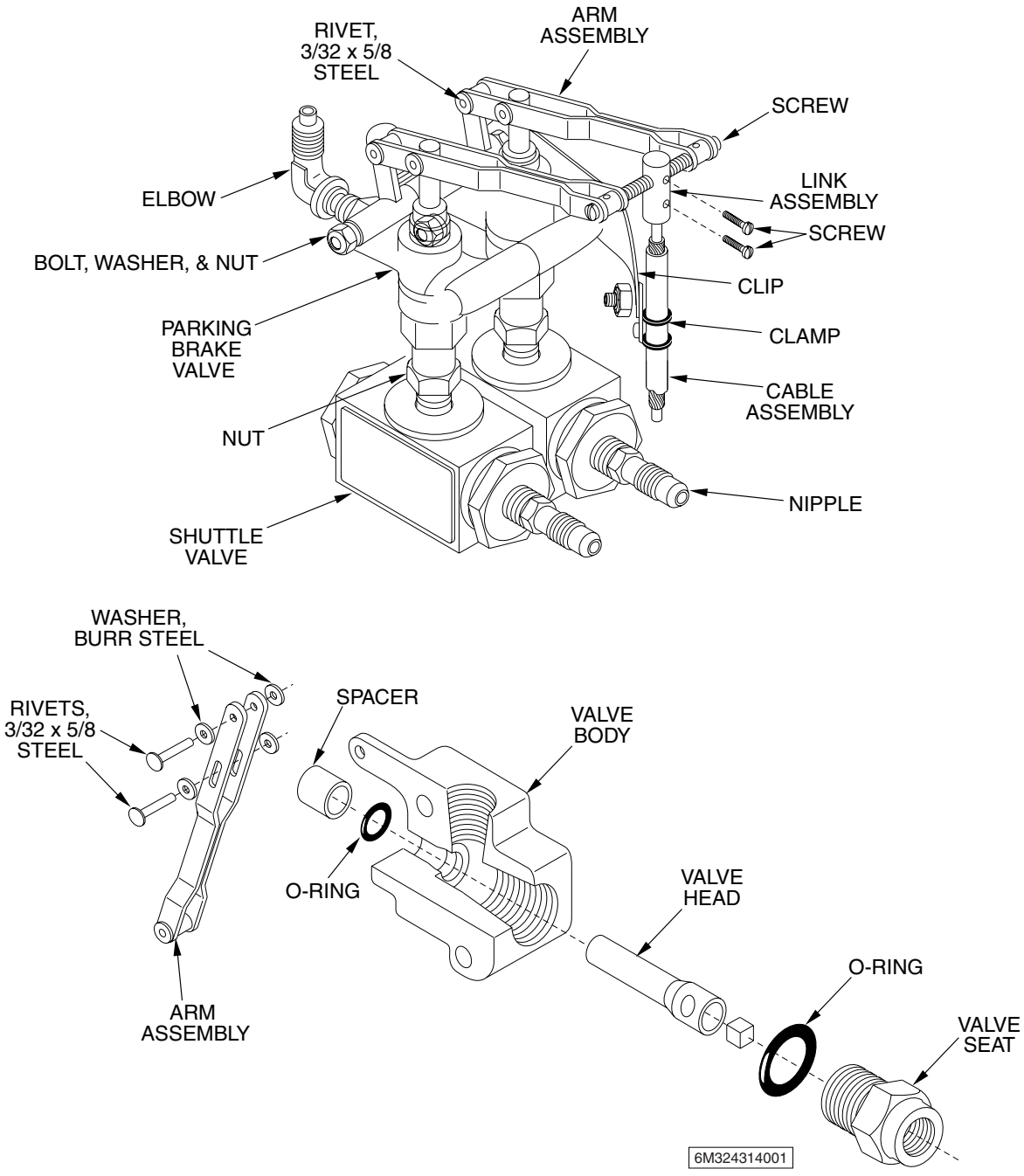
(6) Check for fluid leaks.

(7) Operate the brakes to ensure the spring does not bind.

SHUTTLE AND PARKING BRAKE VALVE ASSEMBLY – DESCRIPTION AND OPERATION

1. General

A. The shuttle valve and parking brake valve assembly is installed forward of the cockpit pedestal on the forward pressure bulkhead. It consists of two shuttle valves and two parking brake valves mounted as an assembly. Each set of valves provides a separate operation.



Shuttle Valve and Parking Brake Valve Assembly Figure 1

2. Shuttle Valves

NOTE: Shuttle valves are not installed on brake system with anti-skid/power brakes.

A. One valve shuttles when pressure is applied to permit either the pilot or copilot (but not both) to apply pressure to the left brakes. The other valve shuttles when pressure is applied to permit either the pilot or copilot (but not both) to apply pressure to the right brakes.

B. In effect, the shuttle valves transfer the braking function to the set of brake master cylinders (pilot's or copilot's) that is actuated first, but prevent simultaneous actuation of a brake by both pilot and copilot.

3. Parking Brake Valves

A. When actuated, these valves trap pressure to the wheel brakes to hold the brakes in the braked (parked) position.

(1) The parking brake control knob is located on the cockpit pedestal and is used to control both parking brake valves simultaneously. When set to the park or on position (parking brake control out), each valve acts as a one way check valve to trap whatever pressure is applied to the respective brake line. Thus, the parking brake control can be set first and then pressure applied via either the pilot's brake pedals or the copilot's brake pedals.

(2) The valves are released by depressing the button in the center of the parking brake control and moving the control fully to the forward position.

NOTE: It is recommended to actuate the brake pedals momentarily after releasing the parking brake control to unseat any component that may have stuck in the brakes on position.

SHUTTLE AND PARKING BRAKE VALVE ASSEMBLY – DESCRIPTION AND OPERATION

1. Removal/Installation – Shuttle and Parking Brake Valve Assembly

A. Removal – Parking Brake Valve

(1) Gain access to the shuttle and parking brake valve assembly mounted forward of the cockpit pedestal on the forward pressure bulkhead.

CAUTION: PLACE IDENTIFICATION TAGS ON LINES AS THEY ARE REMOVED.

(2) Remove and plug the hydraulic brake lines.

(3) Loosen the set screws on the link assembly securing the parking brake control cable.

(4) Remove the clamp securing the parking brake control cable assembly.

(5) Remove bolts, washers, and nuts securing the valve assembly.

(6) Remove the valve assembly.

B. Installation – Parking Brake Valve

(1) Position the valve assembly and install attaching bolts, washers, and nuts.

(2) Install the parking brake control cable into the link assembly and tighten the link assembly set screws.

(3) Install the clamp securing the parking brake control cable assembly to the valve assembly clip.

NOTE: Ensure that the control cable assembly does not bottom out at the link assembly. Otherwise, allow enough clearance to ensure full travel of the valve's arm assembly so that positive opening and closing actuation of the valve is achieved when operating the parking brake control knob.

(4) Install the hydraulic brake lines.

(5) Bleed the brake system (see brake servicing procedures, Section 32-43-00).

(6) Check to ensure that their are no fluid leaks.

ANTI-SKID POWER BRAKE – DESCRIPTION AND OPERATION

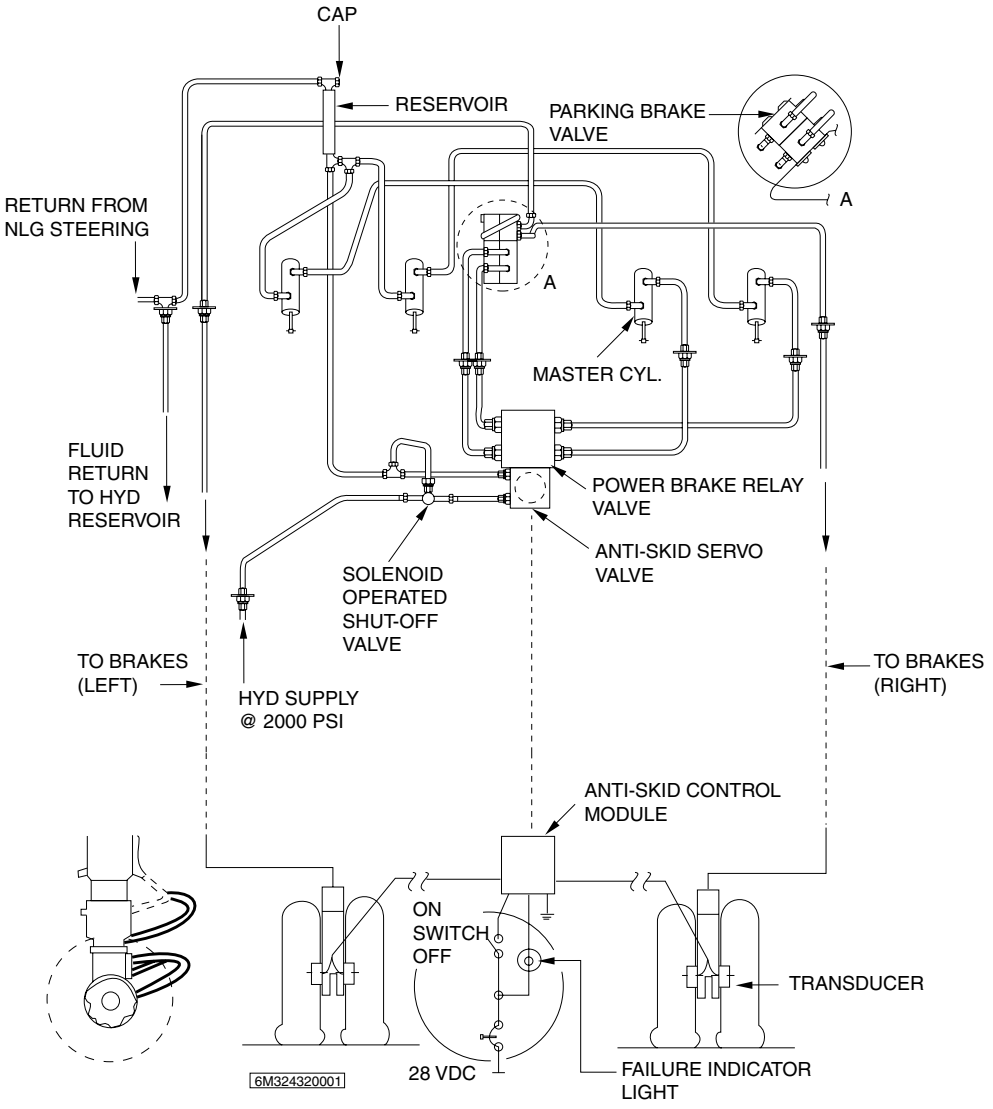
1. General

A. Anti-skid power brakes have been installed on some SA226 series aircraft T276 and is a customer option installation for SA226 series aircraft T291 through 999, AT066 through 999, and TC250 through 999. The anti-skid brake hydraulic pressure is provided by the aircraft's hydraulic power pack and does not use a separate hydraulic system for the brakes such as is installed on aircraft without anti-skid operation.

B. The anti-skid operation consists of:

(1) A reservoir assembly which stores enough hydraulic fluid to provide braking action in the event of normal hydraulic pressure failure.

(2) Larger brake master cylinders having a 1.25 inch (31.75 mm) diameter piston, 1.50 inch (38.10 mm) stroke, and a 1.49 cubic inch (2.51 ccm) displacement.



Anti-Skid Power Brake
Figure 1

(3) A power brake relay valve, anti-skid servo valve, and an electronic anti-skid control module.

(4) An electrically controlled two position hydraulic pressure supply shutoff valve.

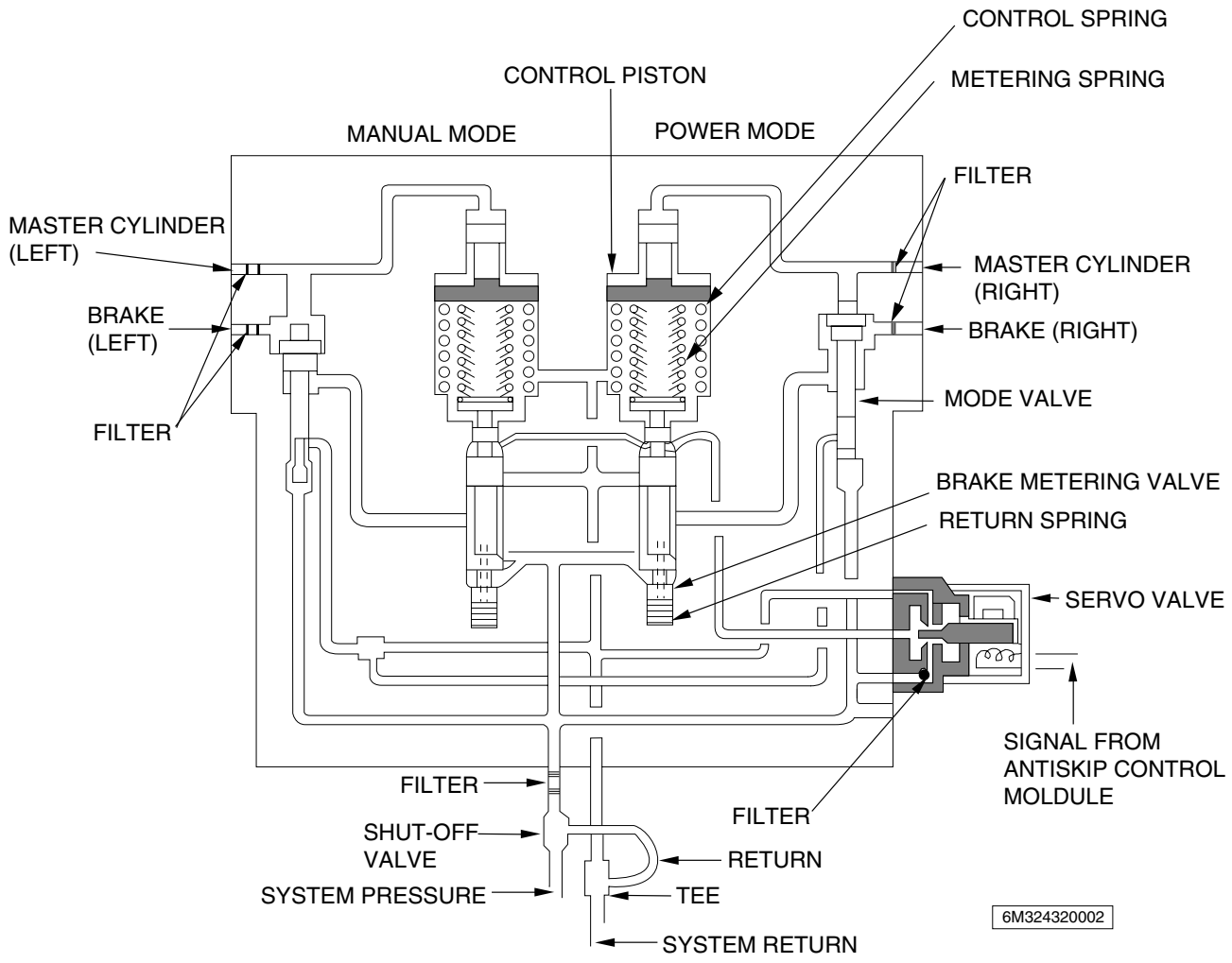
(5) A wheel mounted electrical impulse sending transducer assembly.

(6) A brake adjuster assembly that compensates for the air pressure applied to the top of the hydraulic power pack.

(7) A left console anti-skid ON-OFF switch.

(8) An annunciator panel anti-skid illuminating light and, a left console anti-skid and power brake circuit breaker.

C. The anti-skid electrical system is powered by 28 volt DC power from the left essential bus.

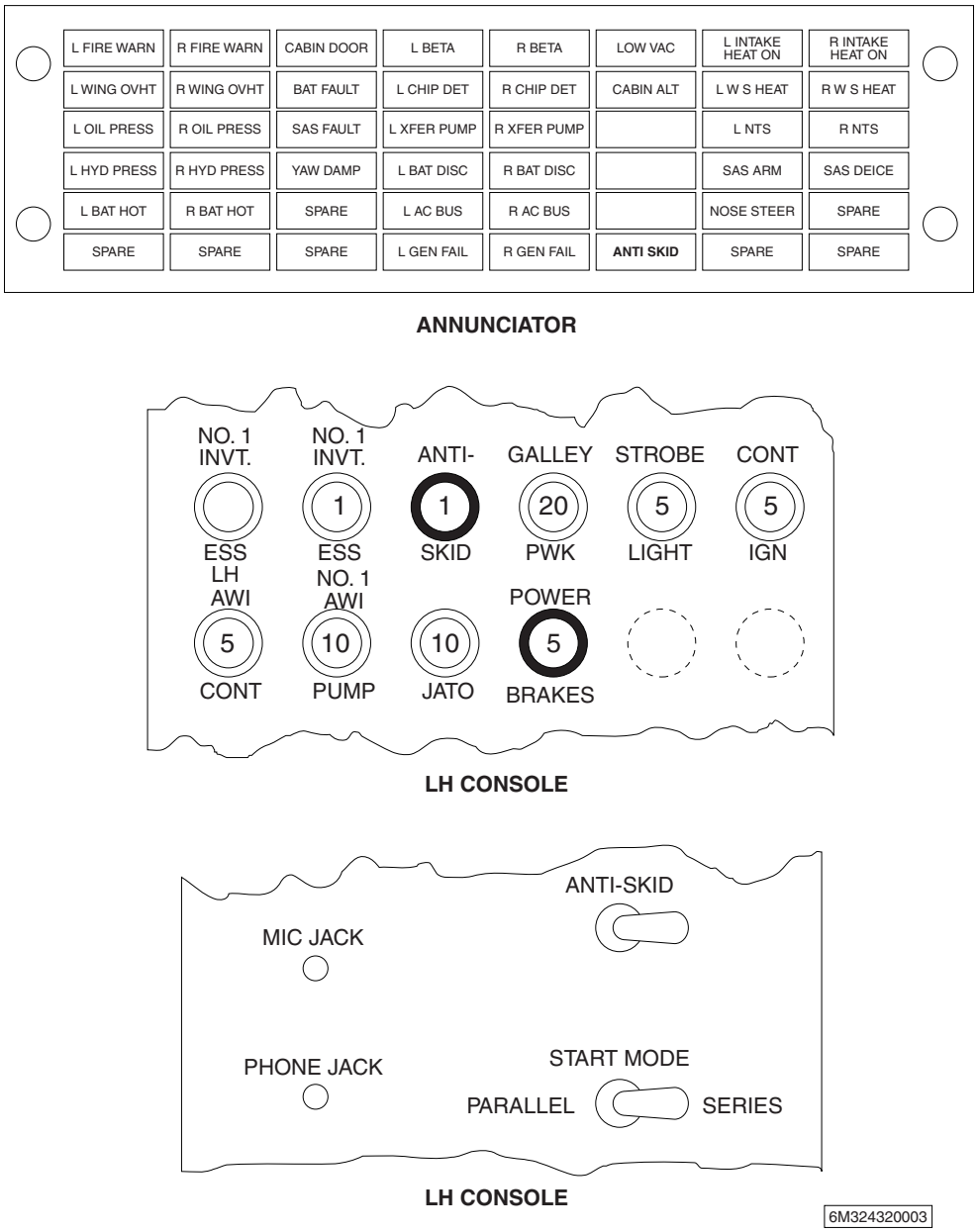


Hydraulic Schematic
Power Brake Relay
Figure 2

2. Power Mode

A. With normal hydraulic system pressure available, battery ON, anti-skid and power brake circuit breakers energized (left console) and anti-skid ON-OFF switch (left console) moved to the ON position, the anti-skid system will be in the power mode (operational) condition.

B. Moving the anti-skid ON-OFF switch to the ON position directs 28 VDC electrical power to the anti-skid control module (see Figure 1) and directs power to the hydraulic pressure supply shutoff valve. The shutoff valve moves to an ON condition (supplying hydraulic pressure) when electrical power is applied.



Anti-Skid Power Brake Power
Figure 3

C. Within the power brake relay valve, hydraulic pressure is routed to the left and right mode valves, the servo valve and the left and right brake metering valves. (See Figure 2).

D. Hydraulic pressure supplied to the mode valves causes the valves to move and block the master cylinder and brake hydraulic fluid supply passage ways. With hydraulic pressure applied to the mode valve in this manner, brake operation will always be in the power mode configuration.

E. When in the power mode configuration, hydraulic pressure routed to the metering valve is blocked from reaching the respective brake by the metering valve.

(1) Pressure enters the cavity beneath the metering valve, causing the metering valve to move towards the control piston.

(2) Pressure supplied to the servo valve is, in turn, routed from the servo valve to the cavity at the top of the metering valve.

(3) This pressure, opposing the pressure in the cavity at the bottom of the metering valve, results in a balancing position of the metering valve.

F. When the pressure is applied to the brake/rudder pedal, hydraulic pressure from the master cylinder pushes the control piston downward which compresses the control and metering springs. This causes the metering valve to move downward. As the metering valve moves downward, the pressure port at the bottom of the valve is unrestricted which allows fluid travel through the metering valve, past the mode valve, and then to the respective brake.

G. Because this is a power brake system, any amount of brake pedal depression causes a larger proportional amount of hydraulic pressure applied to brake.

(1) When the brake pedal is released the control spring in the power brake relay valve will reposition the control piston while the return spring displaces the metering valve upward.

(2) As the metering valve moves upward, the pressure port at the bottom of the valve is blocked and the return port at the top of the valve is open.

(3) Pressure in the brake is then returned through the metering valve and the system return line.

H. Power brake operation exist as long as all of the wheels are rotating. Should one (or more) wheels begin to slow abnormally, the anti-skid electrical control module will sense a wheel skid onset and will apply electrical power to the servo valve.

I. The application of electrical power to the servo valve causes the servo to reposition to block hydraulic pressure input and to provide a return for hydraulic pressure in the cavity at the top of each metering valve. Without pressure in the upper cavity, the pressure in the lower cavity moves metering valves upward, thus reducing brake pressure.

J. As the anti-skid control module senses that the wheel rotation has increased, the control module will in turn remove electrical power to the servo valve thus allowing power pressure to the brakes again.

K. The anti-skid control module senses not only wheel speed but also the entire anti-skid electrical circuitry for faults. Should a fault develop, the anti-skid control module will cause an amber anti-skid light on the annunciator panel to illuminate.

3. Manual Mode

A. Moving the anti-skid switch to the off position removes power from the anti-skid circuit. This action removes electrical power to the hydraulic pressure supply shutoff valve causing the valve to close. This blocks hydraulic pressure to the power brake relay valve.

B. Within the power brake relay valve, and with the lack of hydraulic pressure to the mode valves, when the brake pedals are depressed, fluid pressure displaces the mode valves and opens the passage way between the master cylinder and the respective brake. Pressure supplied by the master cylinders will then travel directly to the brakes. System operation at this time will be the same as that of non-power equipment aircraft.

4. System Filters

A. There are six system filters (Mectron Ind., 330 Turnbull Canyon Rd. City of industry, California 91745) within the power brake relay valve. Four filters (20011-4-10W) are installed at the following locations: one at each of the two master cylinder ports and one at each of the two brake line ports. A fifth filter (20011-6-10W) is installed at the system inlet high pressure port. All five of these Mectron filters should be replaced or ultrasonically cleaned every 1,000 flight hours.

B. A sixth filter (20011-6-10W) is installed within the Hydro-Aire servo valve and is changed only by Hydro-Aire when the entire power brake relay valve is overhauled. The anti-skid valve should be bench tested at five year/5,000 hours intervals, whichever occurs first.

ANTI-SKID POWER BRAKE – MAINTENANCE PRACTICES

1. Servicing – Anti-Skid Power Brake

A. Power Brake Bleeding

(1) Ensure availability of 28 VDC power source, hydraulic power source, and that the hydraulic system reservoir is serviced.

(2) Fabricate a bleeding fixture (see 32-SERVC).

NOTE: It is possible that air can become trapped within the piston bore wells and the hydraulic fluid channels leading between each piston bore within the piston housing of the brake assembly. Refer to Bench Bleeding procedures in the Brake Servicing section in this section to remove aerated hydraulic fluid from the piston housing when the brake assembly is not on the aircraft.

(3) Remove existing bleed port fitting from a left brake and install a number four union.

(4) Install the bleeding fixture between the left number four union and the return line service tee located on the reservoir at the front side of the forward pressure bulkhead.

(5) Select aircraft 28 VDC power ON. (Battery switches ON).

(6) Turn anti-skid selector switch to Off position.

(7) Depress the pilot's left-hand pedal slowly through the full stroke and release completely for five seconds. Repeat this step ten times for the pilot and copilot's left-hand rudder pedal.

(8) Turn anti-skid selector switch to ON position.

(9) Apply light to moderate brake pressure on pilot's left pedal and hold for two minutes. Allow system pressure to purge all remaining air from the left-hand brake system.

(10) Repeat steps **1.**, A., (3) through (9) for the right-hand brakes.

(11) Check all lines for leaks and correct as required.

2. Inspection/Check – Anti-skid Power Brake

A. Anti-skid Response Test

NOTE: The following functional test requires two 0 to 1000 psi (70.30 kg/cm^2) gauges (MIL-H-83282 fluid type); drill motor; and a wheel speed generator drive clip (see SERV-32). The wheel speed generator drive clip can be made from basic shop hardware.

(1) Install gauges at brakes.

(2) Supply hydraulic pressure to system.

(3) Verify that all circuit breakers are energized.

(4) Turn battery switch to ON position.

(5) Turn anti-skid switch to ON position.

(6) Verify the parking brake is OFF.

(7) Remove hub caps to wheels.

CAUTION: MORE THAN 800 PSI (56.24 KG/CM^2) CAN CAUSE DAMAGE TO BRAKE ASSEMBLY.

(8) Fully depress left brake pedal and verify pressure on left brake psi gauge (725, + 25, - 30 psi [50.97 +1.75, -2.10 kg/cm^2]).

(9) Fully depress right brake pedal and verify pressure on right brake gauge.

(10) Install wheel speed drive clip in drill motor.

(11) Fully depress pilots left and right brake pedals and hold.

(12) Rotate the left outboard wheel speed transducer 800-1200 rpm (brake pedal pressure still applied).

(13) Abruptly cease transducer rotation and listen for actuation of power brake servo valve (nose baggage compartment) and note release of pressure at left and right brake to 0-50 psi (0 to 3.51 kg/cm^2). Normally 31 psig is applied to the brake assembly with brake pedal pressure still applied.

(14) Actuation of the power brake servo valve and pressure release at brakes indicate trouble free operation.

(15) Repeat steps **2.**, A., 11 through 13 for remaining wheels.

B. Anti-Skid Functional Check, using 99-577 Field Test Unit (see 32-SERVC)

NOTE: The Hydro-Aire field test unit is designed to facilitate functional tests and/or isolation of anti-skid malfunctions. It may also be used for bench check of the control box, as described further in this section.

Functional check/troubleshooting requirements for aircraft configuration are identical to those specified for Anti-Skid Response Test, Section 32-43-20, except that hubcaps need not be removed.

(1) Remove screws securing aircraft connector to control box and connect J1 and J2 test cables to control box and aircraft connector respectively (see 32-SERVC).

(2) Select L at LO (left outboard) on Dynamic Skid Test side of test panel.

(3) Momentarily depress test switch. As test switch is released verify left and right brake pressures drop to zero (0), followed by a gradual reapplication of brakes (brake pressure must be applied and held during this check).

(4) Repeat previous test selecting at L1 (left inboard) on Dynamic Skid Test side of panel.

(5) Repeat test by selecting R testing RI (right inboard) and RO (right outboard) on Dynamic Skid Test side of panel.

NOTE: If the above tests are unsuccessful and the Anti-Skid light does not illuminate, the fault is located in the control box or anti-skid servo valve.

(6) To evaluate anti-skid servo valve integrity, select SERVO VALVE on tester panel and rotate SERVO VOLTAGE knob until meter indicates 8.0 volts.

(7) With the brake pedals fully depressed, press the Test switch on tester panel.

(8) Verify that brake pressure is released (0 to 50 psi [0 to 3.51 kg/cm^2]) when the test switch is depressed.

(9) Release test switch and verify that brake pressure is reapplied.

(10) Depress the Test switch and slowly rotate SERVO VOLTAGE knob counterclockwise to zero (0). Verify that brake pressure is smoothly reapplied.

NOTE: If the anti-skid valve test was satisfactory, the control box or associated wiring is defective. If unsatisfactory, the anti-skid servo valve or associated wiring is defective.

(11) Check the integrity of wheel speed generators (transducers) and associated wiring integrity by selecting L for both LO and LI and R for both RI and RO respectively in WHEEL SPEED GENERATOR position on test panel.

NOTE: If meter reading is low (approximately 4.0 volts), a short in the respective circuit is indicated. If meter reading is high (approximately 15.0 volts), an open circuit is indicated.

C. Control Box Bench Check

(1) Use Field Test Unit (part number 99-577) to bench check the control box when used with an external 28 VDC power source.

(2) Connect the longer cable connector (J1) to the control box and join together the two remaining connectors on the test cable.

NOTE: This cable configuration provides a simulation of the aircraft system parameters so that a valid test of the box is possible.

D. Skid Response Test

NOTE: This test provides verification of the basic control loop of the control box.

(1) Select left wheel on Dynamic Skid Test side of test panel.

(2) Depress Test switch. As Test switch is released, verify meter gives an indication of 6.0 to 8.0 volts (servo valve command) and decays to zero (0) volts within 15 to 20 seconds.

(3) Repeat test for right wheel with same results.

E. Fault Monitor Test

(1) The fault monitor circuit integrity can be checked by selecting Wheel Speed Generator (left or right) or Servo Valve and inserting a short or open fault and noting fault light illumination.

NOTE: Selecting the toggle switch to OFF simulates a control box power fault and causes illumination of the anti-skid fault lamp.

2-1203-1 ANTI-SKID POWER BRAKE – DESCRIPTION AND OPERATION

1. General

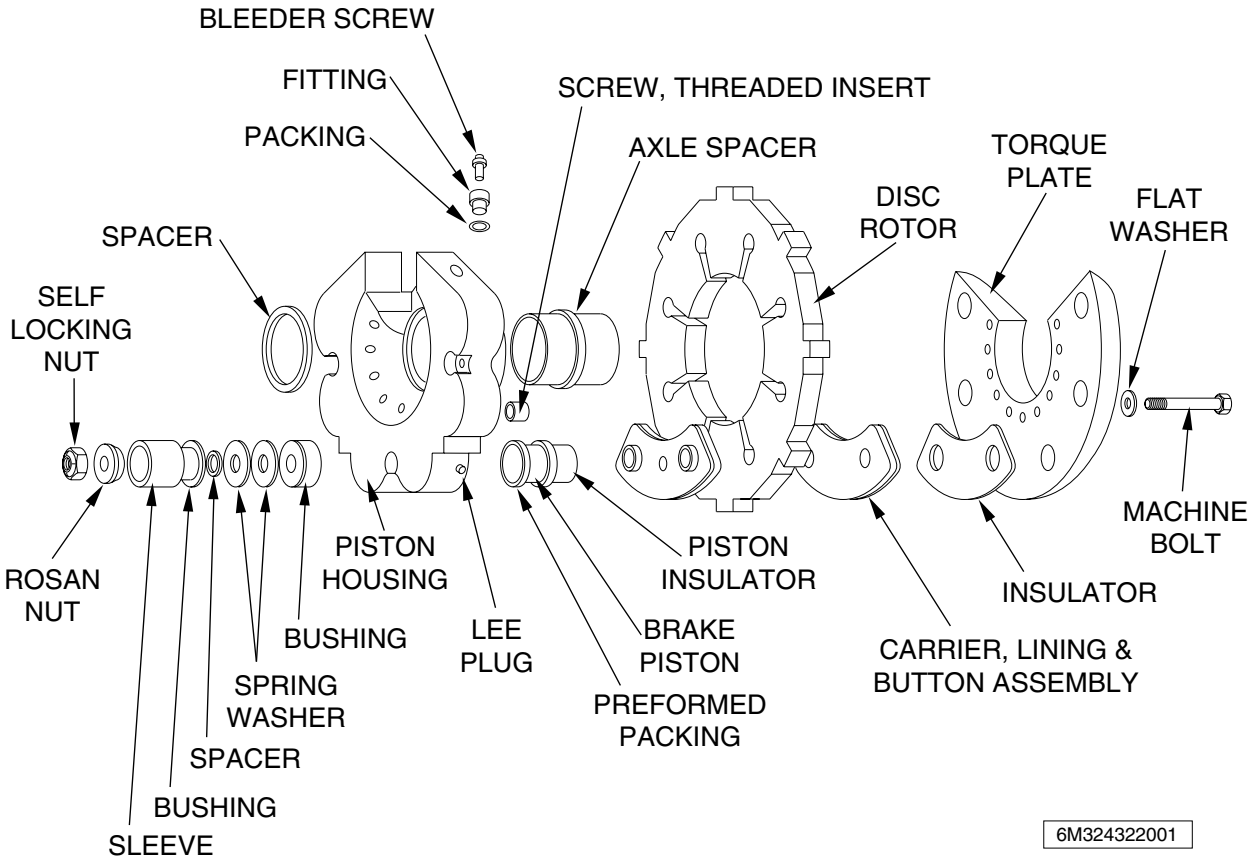
A. The 2-1203-1 single disc brake is designed to withstand 800 psi (56.24 kg/cm^2) operating pressure with 31 psi (2.18 kg/cm^2) back pressure using MIL-H-83282 hydraulic fluid.

B. It is composed of a forged aluminum torque plate fastened to a forged aluminum piston housing assembly with nine washers and bolts. Three segmented carrier, lining, and torque button assemblies are installed on each side of the disc.

NOTE: Current segmented carrier, lining, and torque button assemblies feature a riveted construction. Those manufactured beginning in 1979 will feature a welded construction and improved wear life lining material. Both types are completely interchangeable and may be used in mixed sets.

C. Automatic adjustment is provided by three adjuster assemblies installed in equally spaced wells in the piston housing. Each adjuster consists of a nut, pin, bushing, two spring washers, spacer, and a serrated nut press-fitted into a sleeve.

D. Six insulators, packings, and pistons are housed in piston housing bores. An insulator is installed between each carrier lining and torque button assembly and the torque plate. The insulators shield the torque plate and piston housing from heat generated during brake operation.



2-1203-1 Anti-Skid/Power Brake
Figure 1

E. Spacers mounted on the inside diameter flange of the piston housing position the brake on the axle. A bleeder assembly installed in one of the ports in the piston housing is used to bleed the brake.

F. Hydraulic pressure is applied from the hydraulic master cylinder of the aircraft to the brakes through hydraulic tubing. The pressure forces the pistons to press the three carrier, lining, and button assemblies against the rotor disc and against the three carrier, lining, and button assemblies installed on the torque plate.

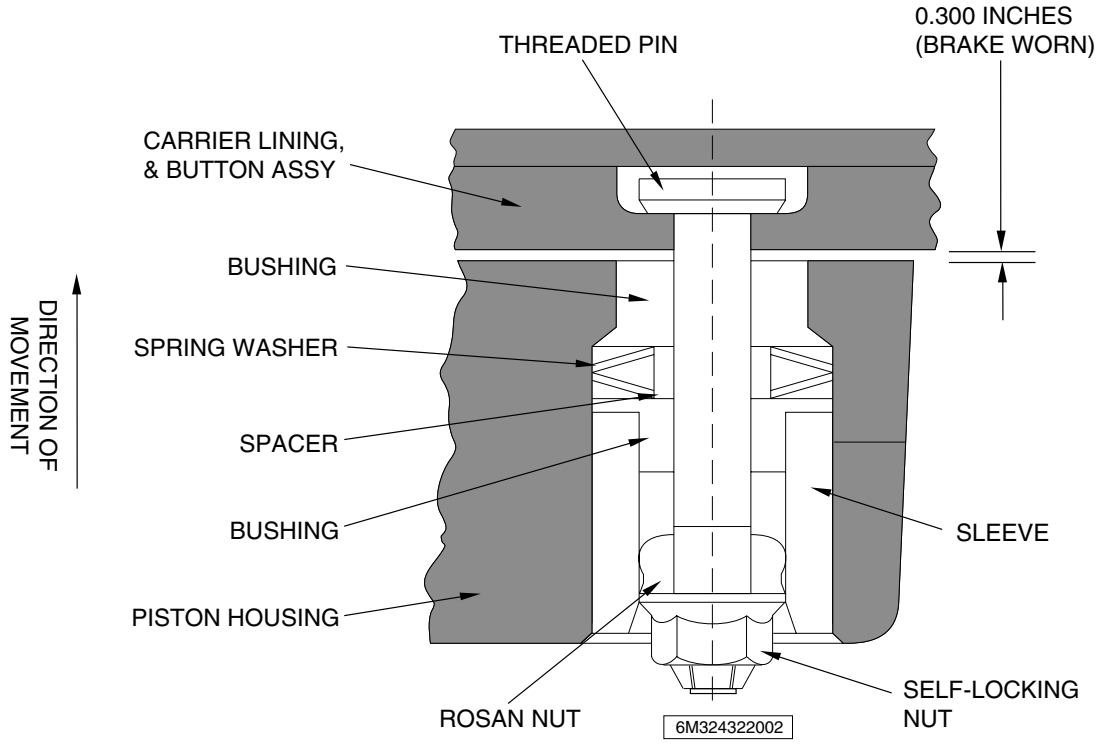
2. Brake Adjuster

NOTE: Figure 2 shows the brake adjuster retracted to full brake clearance with system back pressure applied to the brake.

A. Hydraulic pressure forces the pistons against the carrier, lining, and button assemblies. The threaded pins are pulled in the same direction by the movement of the carrier, lining, and button assemblies, compressing the spring washer. At this point, all brake clearance is taken up and braking follows in proportion to the brake pressure applied.

B. As the lining wears, the carrier, lining, and button assemblies will under pressure of braking, pull the press-fitted serrated nuts forward through the sleeves by means of the threaded pins inserted through the serrated nuts and self-locking nuts.

C. Release of brake pressure allows the compressed spring washers to expand between the bushings, thereby retracting the sleeves, serrated nuts, and self-locking nuts. The press-fitted serrated nuts and sleeves will remain in their advance position with the respect to each other, while the carrier, lining, and torque button assemblies are returned to their preset clearance by the threaded pins.



Brake Wear and Adjustment
Figure 2

2-1203-1 ANTI-SKID POWER BRAKE – MAINTENANCE PRACTICES

1. Removal/Installation – 2-1203-1 Anti-Skid Power Brake

A. Removal – Brake

(1) Jack the aircraft.

(2) Deflate tires (see wheel removal procedures, Section 32-40-00).

(3) Remove the screws and washers securing the hub cap assembly to the wheel and remove the hub cap. (Two screws and washers on magnesium wheel and three screws and washers on aluminum wheel installations). Remove the spacers (NAS43HT3-3) from between the hub cap and wheel axle.

(4) Remove the screws and washers securing the transducer assembly to the wheel assembly and slide the transducer assembly out from the wheel assembly in order to gain access to the transducer electrical connector.

(5) Disconnect the electrical connector to the transducer and remove assembly.

(6) Remove the axle nut and keyed washer and dismount the wheel and tire assembly from the axle of the strut.

CAUTION: PLACE IDENTIFICATION TAGS ON LINES AS THEY ARE REMOVED.

(7) After removing the main wheel and tire assembly, remove hydraulic lines and cap open lines and ports.

(8) Remove the brake assembly by sliding it off over the axle.

B. Installation – Brake

NOTE: The brakes are normally shipped from factory completely assembled and ready for installation.

(1) Prior to installation, check for burrs or rough threads on the axle and axle nut.

(2) Apply thread compound (MIL-T-5544) to threads and all bearing surfaces of washers and nuts prior to installation.

(3) Install the axle spacer into the piston housing of the brake assembly.

(4) Mount one brake in place onto the axle making certain that the brake spacer bottoms onto the strut.

NOTE: Apply a thin coat of bearing grease (MIL-G-3545, or MIL-G-81322 on axle nut treads.

(5) Mount the wheel, tire, and bearing assembly IAW 32-40-00 **2.**, B. Installation – Main Wheel.

(6) Remove the shipping plugs from the brake inlet ports and connect the brake hydraulic fluid line to the inlet port of the piston housing.

(7) Connect the electrical connector to the transducer assembly and insert the assembly into the wheel.

NOTE: For new transducer installations, the corners of the face plate of the electrical connector may need to be rounded off with a file for installation.

(8) Secure the transducer assembly to the wheel assembly with screws and washers

(9) Safety wire the screws.

(10) Install the spacers between the hub cap and wheel axle prior to installing hub cap.

(11) Secure the hub cap assembly to the wheel with screws and washers (two each for magnesium and three each for aluminum wheel installations)

(12) Torque the magnesium wheel hub screws 100 to 110 inch-pounds (11.30 to 12.43 N·m). Torque the aluminum wheel hub cap screws 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(13) Safety wire hub cap screws.

(14) Repeat previous procedures to install remaining brake and wheel assemblies.

(15) Bleed the brakes (see brake servicing procedures, Section 32-43-20).

(16) Perform operational check of system (see Anti-Skid Response Test, Section 32-43-20).

C. Removal – Automatic Adjuster (P/N 107-305)

NOTE: Reference Illustrated Parts Catalog for part numbers and assembly/disassembly, also see Section 32-43-22 (Figure 1) of this Maintenance Manual.

CAUTION: ADJUSTMENTS, REPAIRS, OR REPLACEMENT OF AUTOMATIC ADJUSTERS IS NOT RECOMMENDED WHILE EQUIPMENT IS ON AIRCRAFT. SEE B.F. GOODRICH OVERHAUL PROCEDURES FOR DISASSEMBLY PROCEDURES OF BRAKE.

(1) Remove the locknut from the adjuster pin.

CAUTION: THESE NUTS ARE UNDER A SPRING LOAD; CARE SHOULD BE EXERCISED DURING THEIR REMOVAL TO PREVENT EJECTION OF THE SPRING LOADED PARTS.

(2) Remove Rosan clinch nut, aluminum sleeve (P/N 54-360), and bushing (P/N 26-292) assembly from adjuster pin.

(3) Remove adjuster pin from lining carrier.

(4) Remove spacer (P/N 60-629) and Belville springs P/N 70472-B0750-040-S).

(5) Remove bushing (P/N 26-293) from piston housing adjuster capsule hole.

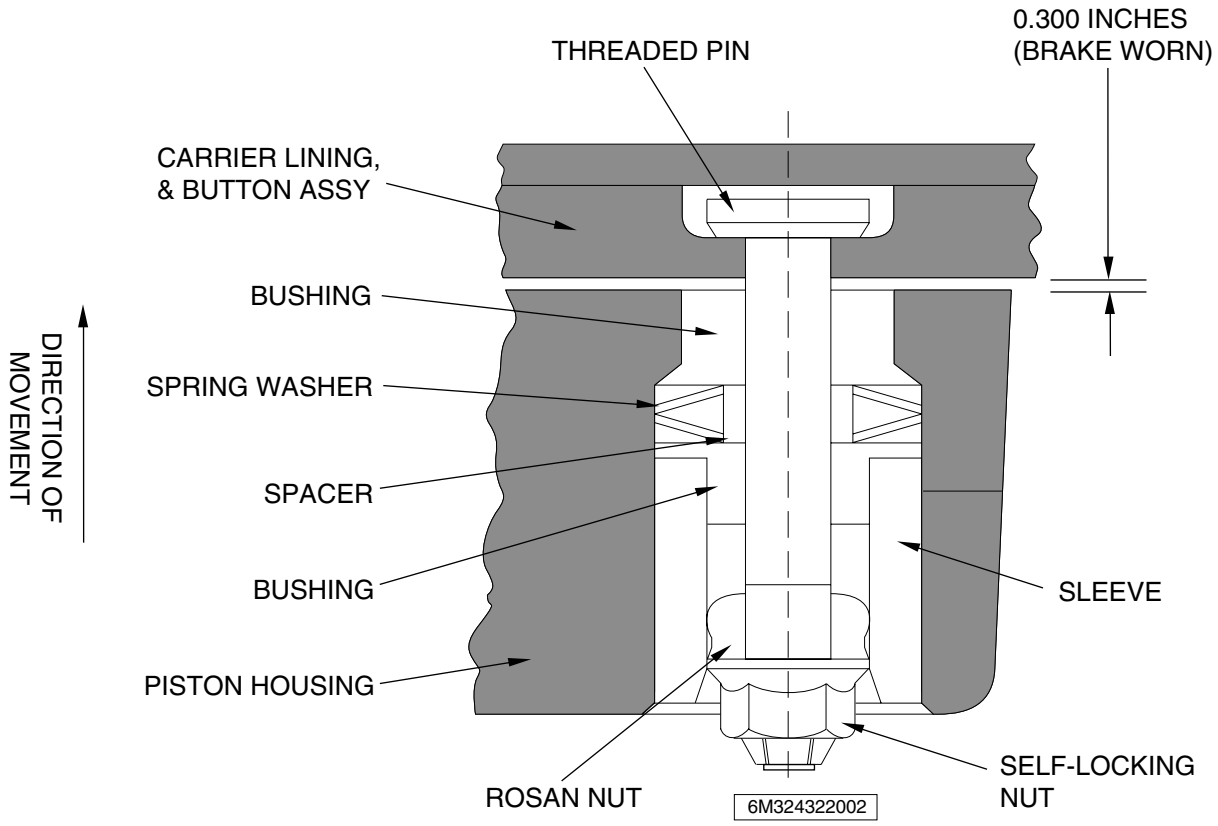
D. Installation – Automatic Adjuster (P/N 107-305)

NOTE: (Reference Chapter 32 of Illustrated Parts Catalog for part numbers and assembly/disassembly).

(1) Bottom the bushing (P/N 26-293) into the piston housing adjuster capsule hole. The bushing may protrude through the piston housing up to 0.010 inches (0.254 mm) but should be recessed no more than 0.040 inches (1.016 mm).

(2) Unplug the hydraulic inlet ports and install a brake lining segment on the piston housing being certain that the pistons are bottomed (reference B.F. Goodrich Overhaul Procedures for assembly).

(3) Install pin (P/N 20-486) through the carrier. Install spacer (P/N 60-629) and Belville springs being certain that the springs are facing in opposite directions, refer to the Illustrated Parts Catalog, Chapter 32.



Brake Adjuster
Figure 201

(4) Press Rosan nut flush into the aluminum sleeve (P/N 54-360) with the chamfered end facing into the sleeve. Install bushing (P/N 26-292).

(5) Install the three piece (Rosan nut, aluminum sleeve P/N 54-360 and P/N 26-292 bushing) assembly over the adjuster pin, and rotate the pin while holding the sleeve until a definite drag is noticed.

(6) Tighten the pin one-half (1/2) turn.

(7) While holding the pin from rotating, install and snug the locknut.

(8) Complete assembly of brake and check clearances.

2. Inspection/Check – Brake (Non Anti-Skid)

A. Brake Wear Measurement Check

(1) Visually check the brake for hydraulic leakage.

(2) If brake pedal is not firm, bleed the brakes (see brake servicing procedures, Section 32-34-20).

CAUTION: BRAKES WITH LOOSE OR MISSING MOUNTING BOLTS SHOULD BE REMOVED FROM AIRCRAFT, AND THE PISTON HOUSINGS SHOULD BE PENETRANT INSPECTED (SEE INSPECTION PROCEDURE). DO NOT TIGHTEN BRAKE ADJUSTER SETTING. NO ADJUSTMENTS, REPAIRS, OR REPLACEMENT ARE RECOMMENDED WHILE THE EQUIPMENT IS ON THE AIRCRAFT.

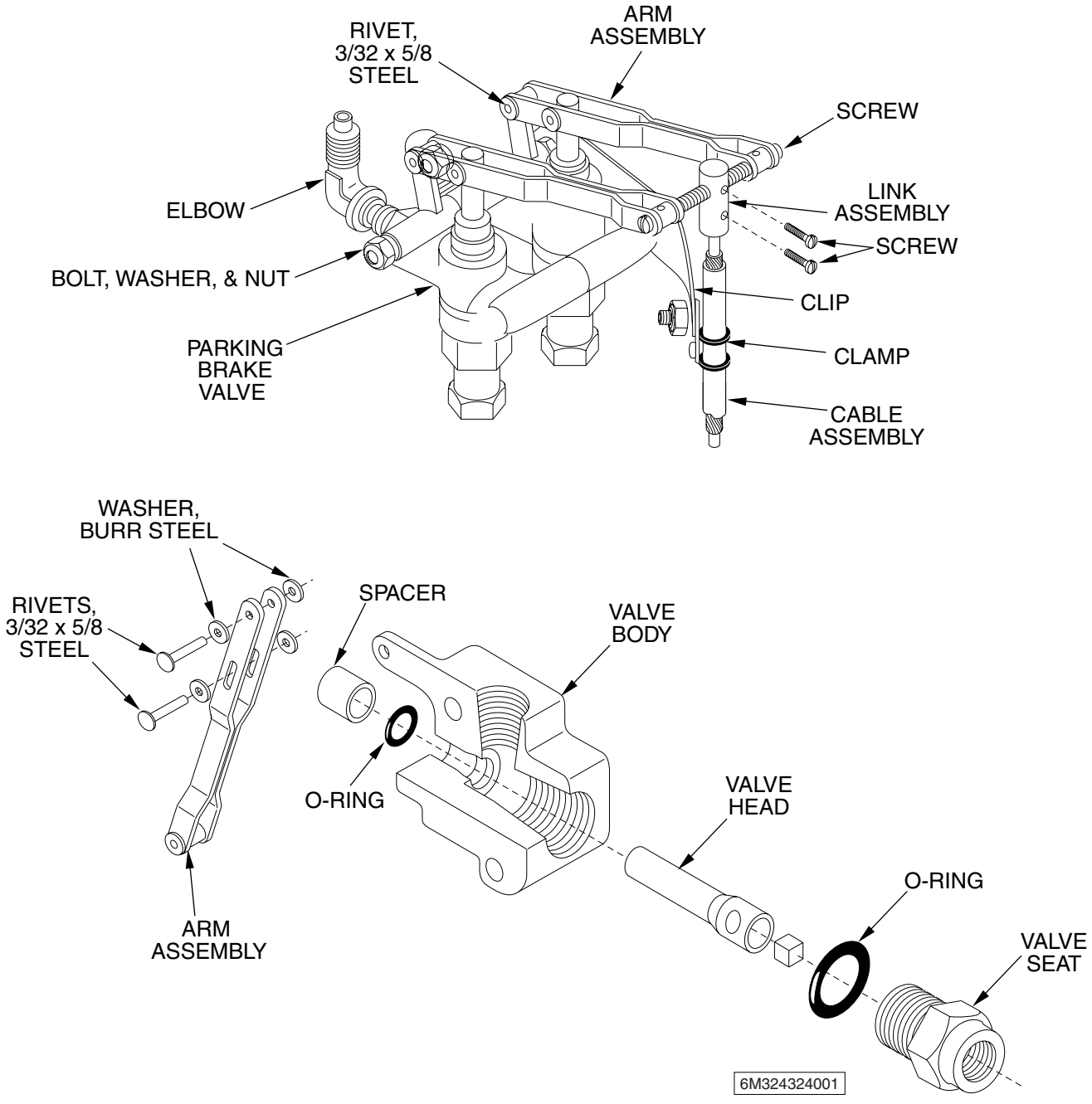
(3) Check for loose bolts and tighten as necessary.

(4) Check for worn brake condition. When dimension is 0.300 inches (7.62 mm) or more with the parking brake set or brake pressure applied the brake is due for a lining inspection. If segmented carrier, lining, and torque button assembly thickness is below 0.198 inch (5.03 mm), the brake is need of overhaul or replacement (see Figure 201).

PARKING BRAKE VALVE – DESCRIPTION AND OPERATION

1. General

A. The parking brake valve assembly is installed forward of the cockpit pedestal on the forward pressure bulkhead. It consists of two parking brake valves mounted as an assembly. Each valve provides a separate operation (see Figure 1).



Parking Brake Valve (Anti-Skid Brake) Figure 1

B. When actuated, these valves trap pressure to the wheel brakes to hold the brakes in the braked (parked) position.

(1) The parking brake control knob is located on the cockpit pedestal and is used to control both parking brake valves simultaneously.

(2) When set to the park or on position (parking brake control knob out), each valve acts as a one way check valve to trap whatever pressure is applied to the respective brake line.

C. The parking brake control can be set first and then pressure applied via either the pilot's brake pedals or the copilot's brake pedals. The valves are released by depressing the button in center of parking brake control and moving the control fully to the forward position.

NOTE: Actuate the brake pedals momentarily after releasing the parking brake control to unseat any component that may have stuck in the brakes on position.

PARKING BRAKE VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – Parking Brake Valve

A. Removal – Parking Brake Valve

(1) Gain access to the shuttle and parking brake valve assembly mounted forward of cockpit pedestal on forward pressure bulkhead.

CAUTION: PLACE IDENTIFICATION TAGS ON LINES AS THEY ARE REMOVED.

(2) Remove and plug hydraulic brake lines.

(3) Loosen the set screws on the link assembly which secures the parking brake control cable.

(4) Remove the clamp securing the parking brake control cable assembly.

(5) Remove the bolts, washers, and nuts securing the valve assembly to the pressure bulkhead.

(6) Remove the valve assembly.

B. Installation – Parking Brake Valve

(1) Position the valve assembly to the pressure bulkhead and secure with bolts, washers, and nuts.

(2) Install the parking brake control cable into the link assembly and tighten the link assembly set screws.

(3) Secure the parking brake control cable assembly to the valve assembly clip with the clamp.

NOTE: Ensure that the control cable assembly does not bottom out at link assembly. Otherwise, allow enough clearance to ensure full travel of the valve's arm assembly so that positive opening and closing actuation of the valve is achieved when operating the parking brake control knob.

(4) Remove plugs and install the hydraulic brake lines.

(5) Bleed the brake system (see brake servicing procedures, Section 32-43-20).

(6) Check to ensure that no fluid leaks.

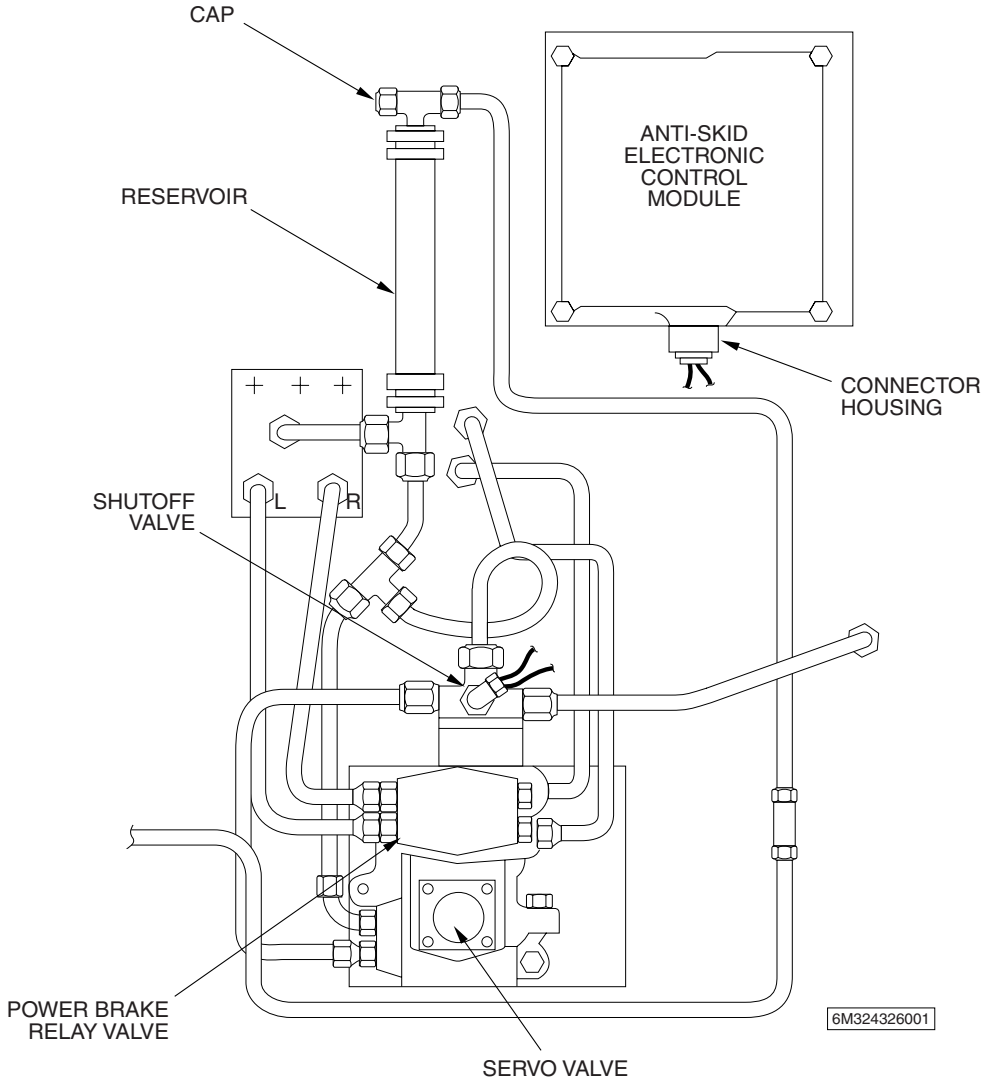
ANTI-SKID CONTROL – DESCRIPTION AND OPERATION

1. General

A. The Anti-skid control module consists of a single printed circuit board mounted in a cast aluminum box. The aircraft connector is attached directly to the circuit board, oriented by a key slot, and held in position by screws threaded into the control box (see Figure 1).

B. The control module accepts the output of the left and right wheel speed generators and converts these signals to a DC voltage directly proportional to wheel speed. An accurate simulation of aircraft velocity is thus available in addition to instantaneous wheel velocity.

C. Left and right wheel speed velocity voltages are averaged to provide a composite signal used through the remainder of the control circuit functions. The wheel velocity signal, in conjunction with the reference deceleration control, continuously updates the reference velocity circuit.



LOOKING AFT – NOSE BAGGAGE COMPARTMENT

Anti-Skid Brake
Figure 1

2. Wheel Speed Generator (Transducer)

A. The wheel speed generator consists of:

(1) A rotor shaft.

(2) A coil assembly.

(3) A bobbin assembly.

(4) Bearings in a housing assembly.

B. A drive coupling is attached to the rotor shaft, which is driven directly by wheel hubcap rotation.

C. A magnetic flux variation of 36 Hz per wheel revolution is supplied to the control box as the necessary wheel speed data input.

ANTI-SKID CONTROL – MAINTENANCE PRACTICES

1. Removal/Installation – Anti-Skid Control

A. Removal – Module

(1) Remove the bolts securing the electrical connector housing at the bottom of the module.

(2) Disconnect the electrical connector housing by carefully pulling it free from its keyed contacts.

(3) Remove the bolts securing the module to its attach plate and remove the module.

B. Installation – Module

(1) Position the module into place and secure to the attach plate with bolts.

(2) Connect the electrical connector housing by inserting it into the keyed contacts on the module.

(3) Secure the electrical connector housing to the module with bolts.

(4) Perform operational check of system.

C. Removal – Reservoir (See Figure 1)

(1) Remove the bolts, nuts, and clamps securing the reservoir to the bulkhead.

(2) Remove the tube assembly from the tee at the top of the reservoir.

(3) Remove the tube assemblies from the tee at the bottom of the reservoir and remove the reservoir.

D. Installation – Reservoir

(1) Position reservoir into place and secure with clamps, bolts, and nuts.

(2) Install the tube assembly to the tee at the top of the reservoir.

(3) Install the tube assemblies to the tee at the bottom of the reservoir.

NOTE: When hydraulic components have been disconnected, the system must be bled of air. See bleeding instructions outlined for the Anti-Skid/Power Brake, Section 32-43-20.

E. Removal – Power Brake Relay and Servo Valve

NOTE: The power brake relay and servo valve are removed as one unit.

(1) Disconnect the electrical wire at its quick disconnect from the servo valve.

(2) Remove the tube assemblies (two on each side) at the power brake valve.

(3) Remove the tube assemblies at the servo valve.

(4) Remove the bolts, washers, and nuts (one nutplate is secured to the attach plate) securing the valve assembly to the attaching plate.

(5) Remove valve assembly.

F. Installation – Power Brake Relay and Servo Valve

(1) Position valve assembly into place and secure with bolts, washers, and nuts.

(2) Install the tube assemblies at the servo valve.

(3) Install the tube assemblies at the power brake valve.

(4) Tighten all hydraulic connections.

(5) Connect the electrical wire to the servo valve.

NOTE: When hydraulic components have been disconnected, the system must be bled of air. See bleeding instructions outlined for the Anti-Skid/Power Brake, Section 32-43-20.

(6) Perform operational check of system.

G. Removal – Solenoid Operated Shutoff Valve

(1) Disconnect the electrical connector from the valve.

(2) Remove the hydraulic tube assemblies at each valve port.

(3) Remove the bolts, washers, and nuts securing the valve to the attach bracket and remove the valve.

H. Installation – Solenoid Operated Shutoff Valve

(1) Position the shutoff valve into place and secure it to the attach bracket with bolts, washers, and nuts.

(2) Install and tighten the hydraulic tube assemblies to the valve ports.

(3) Connect the electrical connector to the valve.

NOTE: When hydraulic components have been disconnected, the system must be bled of air. See bleeding instructions outlined for the Anti-Skid/Power Brake, Section 32-43-20.

(4) Perform operational check of system.

I. Removal – Wheel Speed Generator (Transducer) (Figures 201 and 202)

(1) Remove the screws and washers securing the hub cap assembly to the wheel and remove the hub cap. (Two screws and washers on magnesium wheel and three screws and washers on aluminum wheel installations.) Remove the spacers (NAS43HT3-3) from between the hub cap and wheel axle.

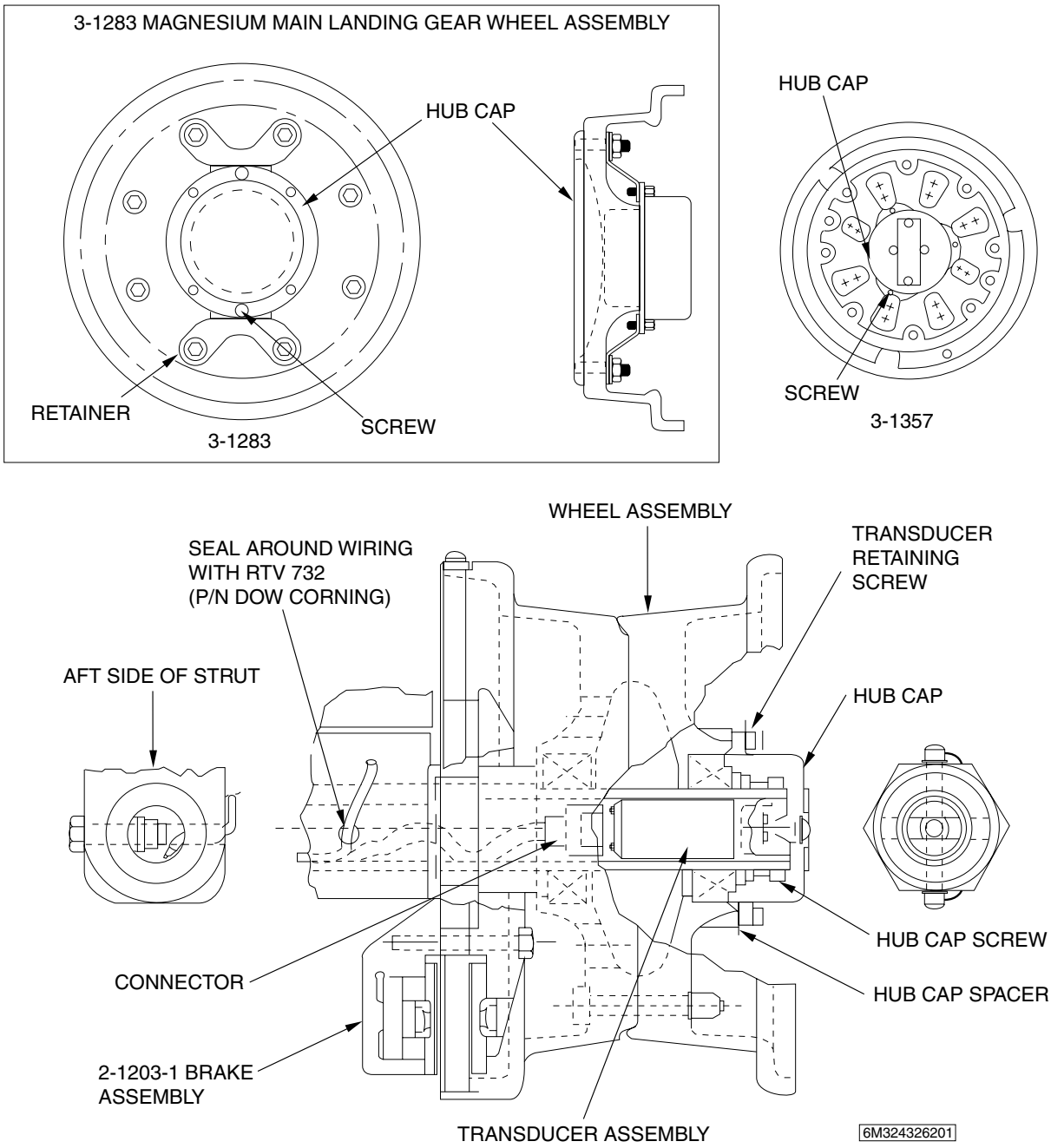
(2) Remove the screws and washers securing the transducer assembly to the wheel assembly and slide the transducer assembly out from the wheel assembly in order to gain access to the transducer electrical connector.

(3) Disconnect the electrical connector to the transducer and remove assembly.

J. Installation – Wheel Speed Generator (Transducer) (Figures 201 and 202)

(1) Connect the electrical connector to the transducer assembly and insert the assembly into the wheel.

NOTE: For new transducer installations, the corners of the faceplate of the electrical connector may need to be rounded off with a file for installation.



Wheel Speed Generator (Transducer)
Figure 201

(2) Install the screws and washers securing the transducer assembly to the wheel assembly and safety wire the screws.

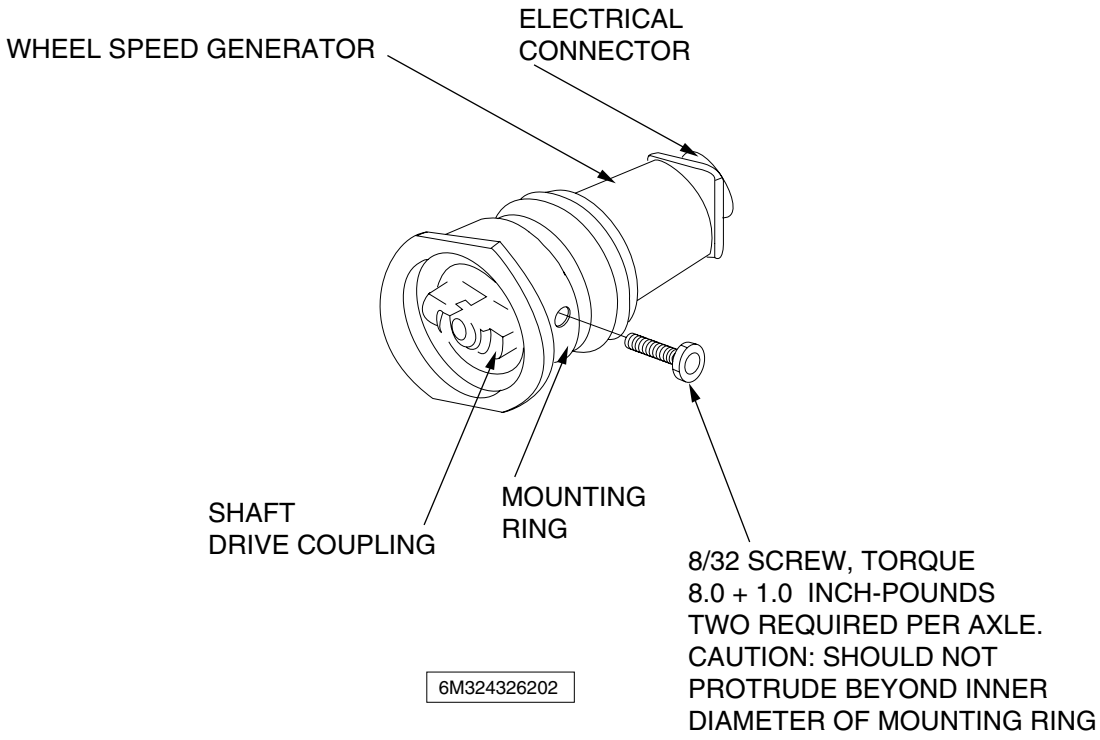
(3) Install the spacers between the hub cap and wheel axle prior to installing hub cap.

(4) Secure the hub cap assembly to the wheel with screws and washers (two each for magnesium and three each for aluminum wheel installations).

(5) Torque the magnesium wheel hub cap screws 100 to 110 inch-pounds (11.30 to 12.43 N·m). Torque the aluminum wheel hub cap screws 20 to 25 inch-pounds (2.26 to 2.82 N·m).

(6) Safety wire hub cap screws.

(7) Perform operational check of system (see Anti-Skid Response Test, Section 32-43-20).



Wheel Speed Generator (Transducer)
Figure 202

COROTATING MAIN GEAR BRAKE – DESCRIPTION AND OPERATION

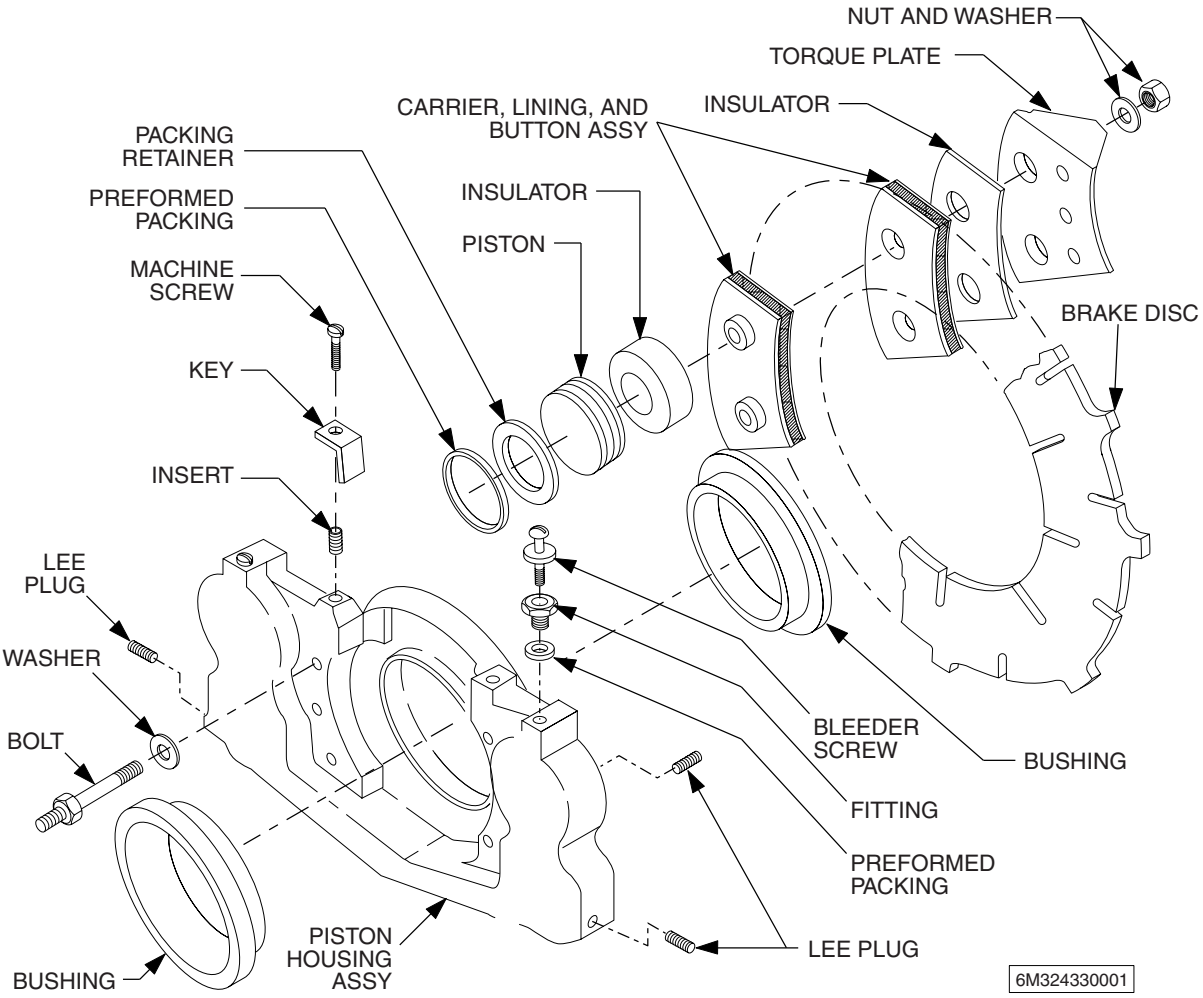
1. General

A. The 2-1268 B.F. Goodrich brake assembly is a customer option item only (Figure 1).

B. The brake is designed for use with MIL-H-83282 hydraulic fluid and to withstand 750 psi operating pressure with zero psi back pressure. It is composed basically of the following parts.

C. The torque plate is fastened to the piston housing with nuts and bolts, washers are used beneath the nuts and bolt heads to prevent galling and stress concentration. Two carrier, lining, and button assemblies are installed on each side of the disc. Heat is shielded from the torque plate and piston housing by insulators. A bleeder screw installed on the piston housing provides bleeding facilities when required. Four pistons, with packings and retainer packings mounted in grooves on the outside diameter are installed in the piston housing, Two bushings, installed opposite each other on the piston housing, provides alignment and resists wear.

D. Hydraulic pressure is applied from the hydraulic master cylinder of the aircraft to the brakes through hydraulic lines. The pressure forces the pistons to press the two carrier, lining, and button assemblies installed on the torque plate.



2-1268 Brake Assembly – for Corotating Wheels
Figure 1

COROTATING MAIN GEAR BRAKE – MAINTENANCE PRACTICES

1. Removal/Installation – Corotating Main Gear Brake

A. Removal – Brake

CAUTION: PLACE IDENTIFICATION TAGS ON LINES AS THEY ARE REMOVED.

(1) After removing the main wheel hub and tire assembly, remove hydraulic lines and cap all open lines and ports.

(2) Remove the bolts from the retainer assembly.

(3) Hold the retainer toward the strut and remove the split ring from the axle housing.

(4) Remove the retainer assembly and the washer from the axle housing.

(5) Slide the brake assembly off over the axle housing.

B. Installation – Brake

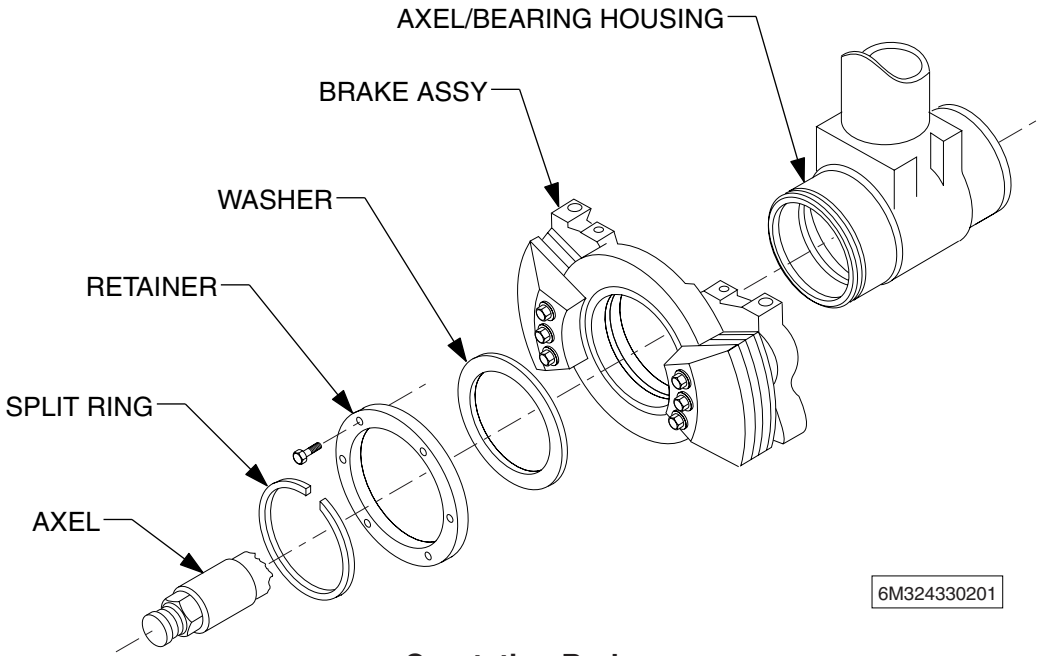
(1) Apply thread compound (MIL-T-5544) to threads and all bearing surfaces of washers and nuts prior to installation.

(2) Mount the brake over the axle housing so that the brake assembly engages the strut.

(3) Mount the washer and the retainer assembly over the axle housing.

NOTE: Notched side of the retainer must be away from the strut.

(4) Slide the retainer assembly, the washer, and the brake assembly against the strut and install the split ring into the groove around the axle housing.



Corotating Brake
Figure 201

(5) Pull the retainer away from the strut so that the notch retains the split ring in its groove.

(6) Install the bolts with washers into the threaded holes in the retainer assembly and tighten to a snug fit using a cross sequence across the diameter.

(7) Using the same cross sequence, lube torque the retainer bolts 25 to 30 inch-pounds (2.82 to 3.39 N·m).

(8) Install lock wire on the retainer bolts.

(9) Install hydraulic lines and bleed the brakes to remove any air from the brake system (see brake servicing procedures, Section 32-43-00).

2. Inspection/Check – Corotating Main Gear brake

A. Brake Wear Measurement Check (While equipment is on the aircraft)

(1) Visually check the brake for hydraulic leakage.

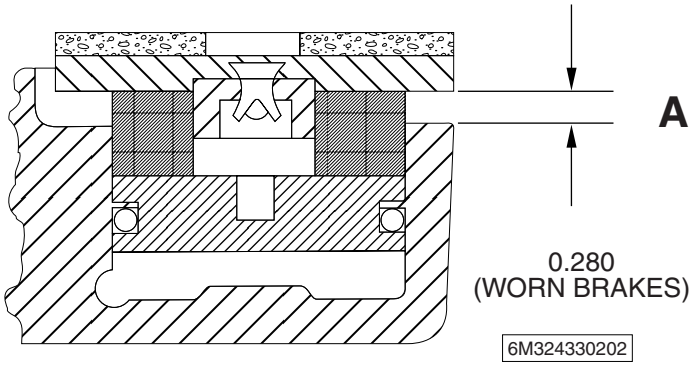
(2) If brake pedal is not firm, bleed the brakes (see brake servicing procedures, Section 32-43-00).

(3) Check for worn brake condition (see Figure 202). When dimension A is 0.280 inches or more, the brake is ready for a lining inspection. If segmented carrier, lining, and torque button assembly thickness is below 0.191 inch, the brake is in need of overhaul (see Repair ad Replacement procedures).

NOTE: Disc wear (grooved rubbing surfaces) is normal and should not be used as the basis for removal of brake as worn. Worn brakes are indicated by worn brake measurement (0.280 inch or more) or disc thickness less than 0.353 inch. Do not remove brakes as worn unless these conditions exist.

(4) Check for loose screws and make sure safety wire is in place.

NOTE: No repairs or replacement are recommended while the equipment is on the aircraft.



Brake Adjustment
Figure 202

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NOSE GEAR STEERING – DESCRIPTION AND OPERATION

1. General

A. All SA226 aircraft use steering systems that turns a shaft (inner cylinder) that extends through the nose gear strut assembly to the wheels. These systems, when used in conjunction with thrust from the engines, provide directional control of the aircraft while taxiing or parking.

B. Aircraft serial numbers T201 through 248, AT001 through 024, and TC201 through 211EEE have an electro-mechanical system that uses a steering wheel and control cables to command an electric servo on the nose gear.

C. Aircraft serial numbers T249 through 999, AT025 through 999, and TC211EEE through 999 have an electrical system that uses the rudder pedals, or as an option, a tiller wheel as a means to command an actuator on the nose gear. Those aircraft not retrofitted by Service Bulletin 32-016 have an electric actuator. Aircraft retrofitted by Service Bulletin 32-016, and aircraft serial numbers T282 through 999, AT062 through 999, and TC244 through 999 have a hydraulic actuator.

D. All Merlin and Metro aircraft are limited in the amount of nose wheel deflection allowed during towing operations. The turn limits in each direction are based on the amount of side load the nose structure can tolerate while the nose gear is being used to pull the aircraft. During taxi, takeoff, and landing the loads are opposite since the aircraft drives the nose gear. Because of the direction the loads are applied during taxiing, the nose wheel steering is allowed to deflect the nose wheels an additional 13° past the tow turn limits.

E. If the tow turn limits are exceeded during towing operation, it is possible to apply excessive loads to the nose gear structure or damage the nose gear steering and centering mechanisms. The turn limits should be clearly marked on the nose gear and should be carefully observed during towing operations (see Chapter 9 for towing turn limits).

F. If the nose wheel towing turn limits are exceeded, a careful inspection should be made of the nose gear supporting structure, nose gear steering system, and nose gear centering mechanism. Particular attention should be paid to the nose steering actuator drive bolts on aircraft equipped with hydraulic powered nose wheel steering.

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STEERING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. On early SA226 series aircraft, the nose wheel steering system consists of a control box, steering wheel, interconnecting cables, an electrical/mechanical servo, and gear box (see Figure 1). The system is also aided by a shimmy damper.

2. Control Box (32-51-10)

A. The control box is located below the cockpit floorboards and forward of the pedestal. It consists of two segmented pulleys (an input pulley and feedback pulley) that pivot on a common bolt. Two adjustable striker bolts are mounted on the input pulley and actuate electrical equipment on the feedback pulley (carbon pile assemblies for SA226-T and switch assemblies for SA226-AT, and TC). This action is used by other electrical equipment within the control box (control amplifier, for the SA226-T and a resistor bank for the SA226-AT and TC to control the servo.

3. Steering Wheel (32-51-20)

A. The steering wheel consists of a hand operated wheel attached to a shaft. On the bottom of the shaft is a cable pulley drum. A spring loaded palm press button is on top of the wheel and is connected by a rod passing through the center of the shaft to a set of ganged switches.

4. Interconnecting Cables (32-51-20)

A. Interconnecting cables and pulley cables from the steering wheel pulley drum connect to the input pulley in the control box. Cables are connected from a pulley on the nose gear to the feedback pulley in the control box. The pulley on the nose gear is mounted to the steerable inner cylinder of the nose landing gear strut.

5. Electrical/Mechanical Servo (32-51-30)

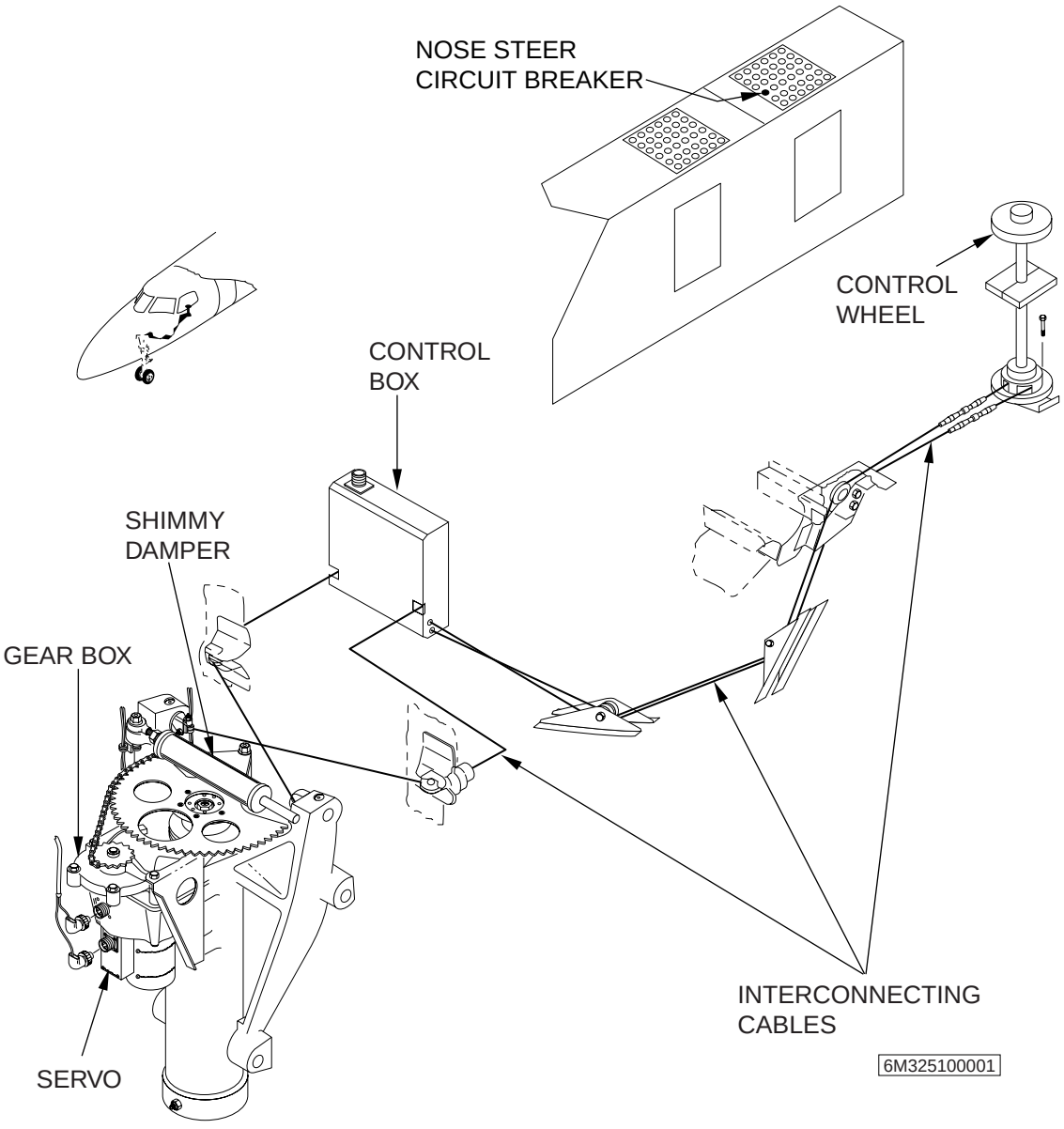
A. The electrical/mechanical servo consists of a DC powered motor and two electrical mechanical clutches. Direction of rotation and torque of the servo's output shaft is determined by the clutches which are controlled by electrical current from the control box.

6. Gear Box (32-51-30)

A. A reduction gear box assembly with a chain driver sprocket is mounted to the front of the nose landing gear. The servo is mounted to the gear box. The sprocket on the gear box is connected by a chain to a large segmented sprocket mounted to the steerable inner cylinder of the nose landing gear.

7. Shimmy Damper (32-52-40)

A. The shimmy damper is mounted to the pulley on the steerable inner cylinder of the gear. The piston of the damper attaches to an ear on the main body of the gear.



Steering System
Figure 1

8. Operation

A. When the palm switch is depressed, power is routed from the NOSE STEER circuit breaker on the nonessential DC bus through a scissor switch on the right main landing gear, to the servo motor, control box, and the amber NOSE STEER light on the annunciator panel. With no command applied at the steering wheel, the input and output pulley are neutral with each other. When electrical control systems in the control box apply current equally to both servo clutches, no movement of the nose gear wheels will occur. A ground for the servo clutches is also provided. Rotating the steering wheel causes the input pulley in the control box to rotate. The striker bolts actuate the electrical equipment on the feedback pulley. This action is sensed as a command by the other electrical equipment within the control box and will increase current to the appropriate servo clutch in the direction of the steering wheel rotation. This increase of current causes the clutch to engage. The servo's output shaft will then drive the gear box. The gear box, through its sprocket and drive chain, will rotate the inner cylinder of the nose gear which will turn the wheels in the command direction. The pulley on the inner cylinder conveys this movement to the feedback pulley in the control box. The feedback pulley moves towards a neutral position with the input pulley, This movement deactivates the electrical equipment on the feedback pulley.

B. This is sensed as nose wheel position by the other electrical equipment within the control box. When the pulleys reach the neutral position, the electrical equipment senses that the nose wheels have moved to the commanded position and the current to both servo clutches is made equal. The shimmy damper works with the steering system to prevent any sudden and uncontrollable movement of the nose gear wheels.

STEERING SYSTEM – FAULT ISOLATION

1. Troubleshooting – Steering System

A. If a defect exists that complies with the items in the DEFECT column, comply with the preliminary and follow the steps in the CORRECTIVE ACTION column. If you suspect an item in the PROBABLE CAUSE column, comply with the preliminary and follow the steps in the CORRECTIVE ACTION column next to the item.

B. When checking for 28 VDC, a voltmeter or a test light may be used. When wiring, wire terminals, or relay contacts are a suspected cause, a test light should be used since it places a slight load on the circuit. The slight load will cause the circuit being checked to open at the trouble point and the light will extinguish. A voltmeter will place almost no load on the circuit and the defective item will be harder to find.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
(1)Steering system inoperative. Annunciator panel light is not illuminated		PRELIMINARY Connect APU to aircraft and turn battery switches ON. Refer to Wiring Diagram Manual Chapter 32-50-01.
	(1)Right main landing gear scissor switch or wiring defective	(1)Remove connector on control box. Hold PRESS button down on control wheel. Check for 28 VDC present at pin D of connector. If 28 VDC is present at pin, proceed to step 4. If 28 VDC is not present at pin D, check for 28 VDC present at terminal 7 and 8 at right wheel well terminal strip. If 28 VDC is not present at terminal 8, proceed to step 2. If 28 VDC is present at terminal 8 but not 7, scissor switch or wiring is defective.
	(2)Circuit Breaker defective	(2)Remove the left forward console door. Check for 28 VDC present at terminal C of power switch on bottom of control wheel assembly. If 28 VDC is present, proceed to step 3. If 28 VDC is not present, circuit breaker or wiring is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	(3)Control wheel power switches (ganged switches) defective or not adjusted properly	(3)Hold PRESS button down. If switch actuators are not completely depressed, adjustement is needed. Check for 28 VDC present at NO terminal of power switch. If 28 VDC is not present, switch is defective.
	(4)Control Box internal wiring defective	(4)Remove top cover of control box and check wiring.
2)Steering system inoperative. Annunciator light illuminates		PRELIMINARY Connect APU to aircraft and turn battery switches ON. Refer to Wiring Manual Chapter 32.
	(1)Servo or wiring defective	(1)Hold PRESS button on control wheel. Listen for servo motor running. If servo motor is running, proceed to step 2. If servo motor is not running, remove motor connector on servo (two pin connector) and check for 28 VDC present at pins A and B. If 28 VDC is present, servo is defective. If 28 VDC is not present, check connector and wiring to nose wheel well terminal strip.
		(2)Remove clutch connector on servo (three pin connector). Using ohmmeter, check pin A to ground. Hold PRESS button down on control wheel. Reading should be 200 ohms to ground for the SA226-T. Direct to ground for SA226-AT and TC. If reading is not the above, check connector and wiring to nose wheel well terminal strip. Check 200 ohm resistor for SA226-T. Check ground through control wheel switch. If reading is the above, check for a voltage at pins B and C. If voltage is not present when control wheel is turned, proceed to step 3. If voltage is present servo is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	(2)Control box defective	(3)Remove control box covers and check for defects within control box (see 32-51-10 Steering Control Box).
(3)Steering overshoots reacts too quickly, sluggish or pulls to the left or right	(1)Cable tension not set properly	(1)Check and set cable tension.
	(2)Control box defective	(2)See 32-51-10 Steering Control Box.
	(3)Servo defective	(3)Check outputs from control box to servo. If outputs are correct (see 32-51-10), servo is defective.
	(4)Chain or gear box defective	(4)Check chain slack. Check gear box (see 32-51-30).
	(5)Shimmy damper defective	(5)Check shimmy damper leakage and damping operation (see 32-52-40).

1. Adjustment/Test – Steering System

A. Operational Check

(1) Connect APU to aircraft.

(2) Turn battery switches ON.

(3) Hold PRESS button down. Verify that servo motor is running and amber NOSE STEER light is illuminated.

(4) Turn control wheel. Verify that nose gear turns smoothly and evenly in that direction and stops at approximately the same angle the control wheel has turned. Holding a constant pressure or turn on the control wheel of a hard over turn, verify that the nose gear turns to its limits.

(5) Turn battery switches OFF.

(6) Disconnect APU from aircraft.

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STEERING CONTROL BOX – DESCRIPTION AND OPERATION

1. General

A. Carbon Pile Control

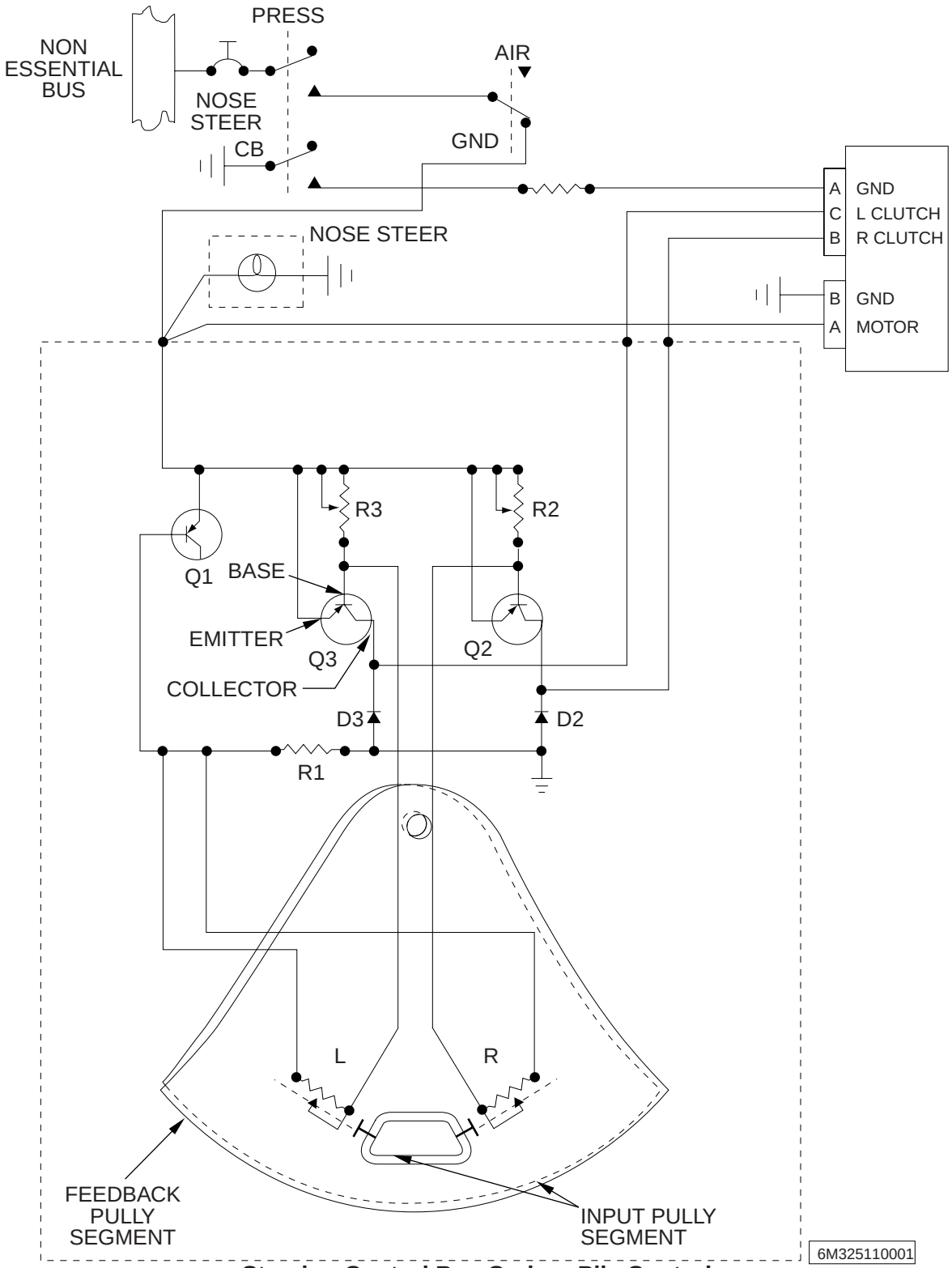
(1) The electrical equipment in the control box for the SA226-T series aircraft consists of two carbon pile assemblies and a transistorized amplifier. Each carbon pile assembly contains two electrical contact boards on each end of a set of carbon washers (discs) stacked on an actuating rod. These assemblies are bracket mounted to the segmented feedback pulley. Two adjustable striker bolts are installed on the segmented input pulley and make contact with the carbon pile actuator rods. The contact boards on each carbon pile assembly are connected to the transistorized amplifier.

(2) When a stack of carbon discs is compressed (loaded), its electrical resistance decreases and, when unloaded, its resistance increases. When the segmented pulleys are in a neutral position with each other, the carbon pile stacks are compressed lightly and evenly by the striker bolts, and their resistance is equal. This is called carbon pile preload. Each carbon pile stack is connected to the base emitter circuit of a transistor (Q2 and Q3) of the amplifier (see Figure 1).

(3) The bias voltages on the bases of these transistors control the amount of current that flows through the emitter collector junction to the servo clutches. Two variable resistors (R2 and R3), also in the base emitter circuit of Q2 and Q3, are adjustable and are used to balance the bias voltages. When the pulleys are in a neutral position, the bias voltages are equal. A low amount of current is allowed to flow to each servo clutch, and no movement of the nose gear results. This low and equal amount of current is referred to as quiescent current.

(4) If the steering wheel is turned left, the input pulley will move left causing the left striker bolt to further compress the left carbon pile stack. The right carbon pile stack will be unloaded. Using Figure 1 as a reference, the stack marked L will decrease its resistance and bleed the bias voltage at the base of Q3 through R1 to ground. This causes Q3 to allow more current through its emitter collector junction to the left clutch. The stack marked R will increase its resistance. The bias voltage at the base of Q2 will increase. The current through the emitter collector of Q2 to the right clutch will decrease. The increase of current to the left clutch causes it to further engage, and the nose gear wheels will turn left. When the nose gear wheels reach the commanded amount of turn, the feedback pulley segment moves to unload the left carbon pile assembly and load the right. This causes the carbon pile stacks to become equally compressed again. The clutch current is then low and equal, and the gear stops moving. When the steering wheel is turned to the right, a similar action takes place and the nose gear wheels turn to the right.

STEERING CONTROL BOX – DESCRIPTION AND OPERATION



Steering Control Box Carbon Pile Control
Figure 1

STEERING CONTROL BOX – FAULT ISOLATION

1. Troubleshooting – Steering Control Box

A. Carbon Pile Control

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Steering system inoperative. Annunciator light illuminates	Carbon pile stacks defective or carbon pile preload not adjusted properly	Adjust carbon pile preload or replace stacks.
	Control amplifier defective or quiescent current not set properly	Adjust quiescent current or replace control amplifier.
	Wiring defective within control box	Repair wiring.

STEERING CONTROL BOX – MAINTENANCE PRACTICES

1. Removal/Installation – Steering Control Box

A. Removal – Control Box

(1) Disconnect electrical connector.

(2) Remove screws holding top cover to box.

(3) Remove nuts on terminal strip. Mark wires so they can be replaced on correct terminals and remove wires from terminal strip.

(4) Remove top cover from area.

(5) Tie control column back securely.

(6) Remove screws in bottom cover and remove bottom cover.

(7) Remove bolt through pulley and back of box (lower right area of box). Observe order of spacer and washer for reinstallation.

(8) Remove bolts through back of box holding box to frames.

(9) Observe cable routing for reinstallation.

(10) Remove bolt center of box through segmented pulleys. Observe order of spacers and washers for reinstallation.

(11) Remove screws in center brace and remove center brace.

(12) Remove safety wire. Remove screws in cable anchors and remove anchors. Mark cables for reinstallation.

(13) Remove segmented pulleys.

(14) Remove cotter pin cable guards in pulley brackets.

(15) Remove box; allow cables to slide through holes in box.

B. Installation – Control Box

(1) Position box and route cable through holes in box.

(2) Route cables over pulleys and install cotter pin cable guards.

(3) Position segmented pulleys in box and route cables on segmented pulleys.

(4) Install cables in cable anchors. Install screws through cable anchors and safety wire.

(5) Position center brace and install screws.

(6) Install bolt through center brace and segmented pulley with the washers and spacers.

(7) Position box and install bolts holding box to frames.

(8) Install bolt through pulley and back of box with the spacer and washer in the lower right area of box.

(9) Check cable tension (see Section 32-51-20).

(10) Install wires from carbon pile assemblies on terminal strip.

(11) Check carbon pile preload setting (see 2.B. Adjustment/Test).

(12) Install wires from top cover on terminal strip and install four nuts on terminal strip.

(13) Check quiescent current (see 2.C.Adjustment/Test).

(14) Install bottom cover, top cover, connector, and safety wire.

(15) Perform operational check.

C. Removal – Carbon Pile Stacks

NOTE: Each carbon pile assembly is made from one-half of the discs in a B-836948 stack. The discs are fragile and should be handled by the edges. Avoid finger prints and contamination of the mating surfaces. When assembling the actuator rod, discs, and contact boards with the keeper, apply a thin film of silicone lubricant (Dow Corning 3450) on the keeper end of the rod. The preload setting may change after installation and initial use. The carbon pile discs tend to seat or wear in. This effect can be minimized by running the adjustment screws tight against the carbon piles before setting the preload, and by setting slightly less than five ohms when the box is first setup (4 to 4.5 ohms suggested).

(1) Remove screws holding top cover of control box. Tie or tape cover out of way of elevator weight.

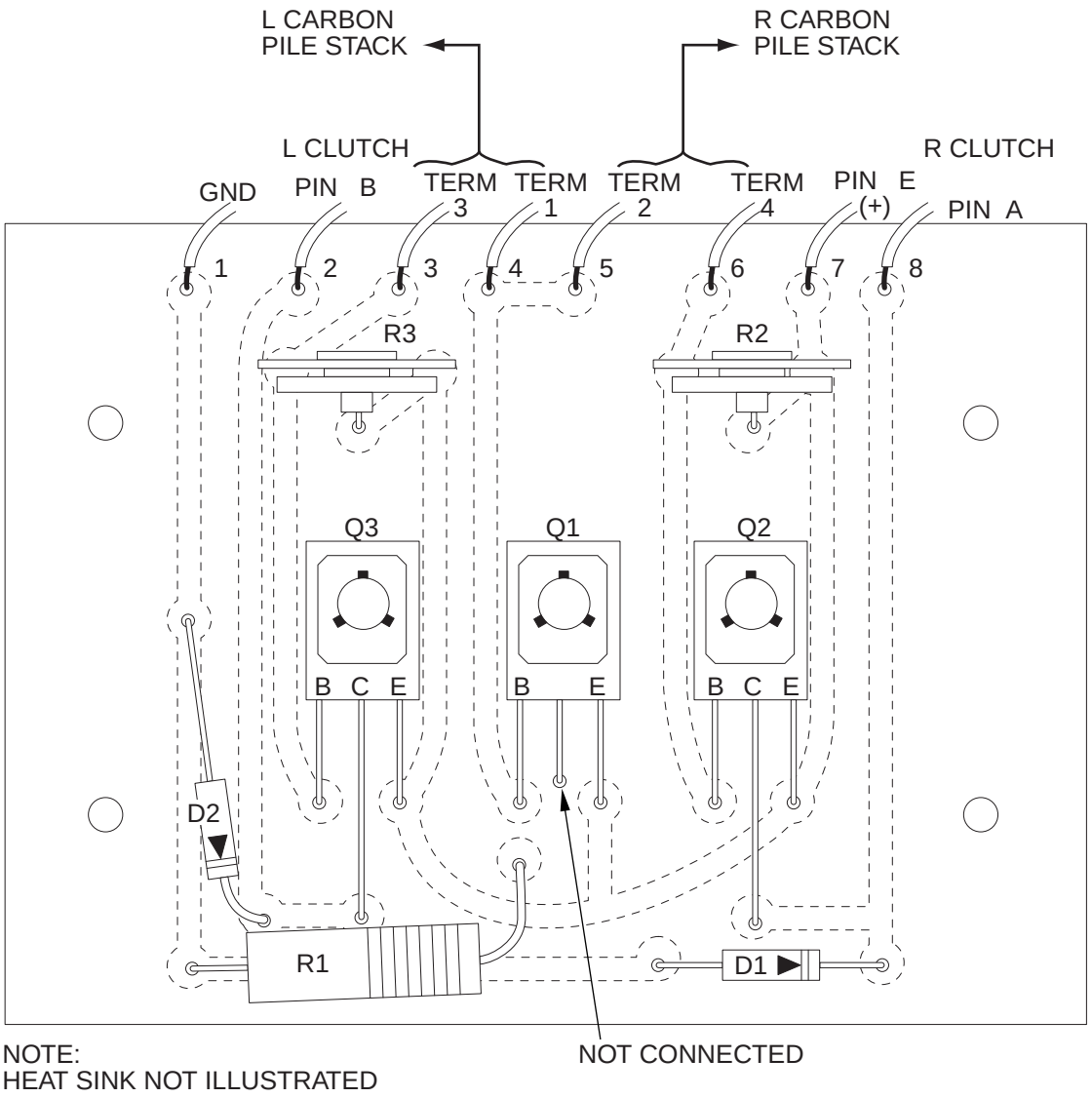
(2) Tie column back securely.

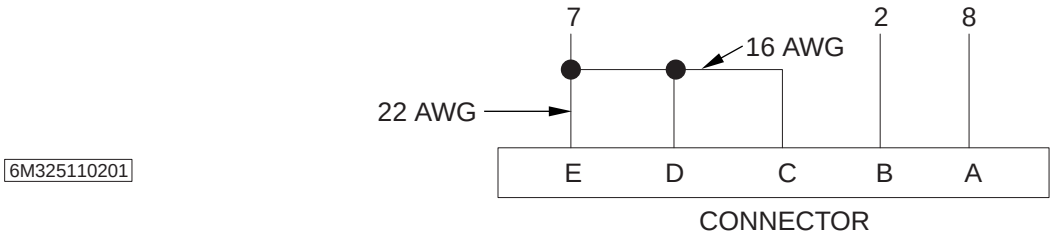
(3) Remove screws in bottom cover. Remove bottom cover.

(4) Loosen locknuts on striker bolts. Screw bolts in.

(5) Remove nuts on terminal strip. Mark wires so they can be replaced on correct terminals.

(6) Removescrews through keepers. Tilt keeper end of rod toward rear of aircraft and remove end of rod from angle. Remove both carbon pile stack assemblies from the aircraft to a suitable work area to replace discs. Mark stack assemblies left and right so they can be replaced in their correct positions. Remove discs and contact boards.





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Control Amplifier
Carbon Pile Control
Figure 201

D. Installation – Carbon Pile Stacks

(1) Inspect teflon heat shrink on rod for roughness, ridges, or wrinkles. Replace if necessary. The major portion of the rod has a piece of teflon shrink tubing applied to provide insulation and a low friction surface for the carbon pile discs. The 500-10 alpha shrink tubing is a higher temperature type than is normally used in electrical work. It requires a special technique for satisfactory application.

(a) Place short end of the rod in a drill chuck; use a propane torch as a heat source. Preheat the rod and position tubing. Starting at the flange end of the rod with the chuck turning, careful application of heat will shrink the tubing in place and give a smooth surface without ridges or wrinkles. The torch flame should not be held directly on the tubing. Instead, the tip of the flame should be kept an inch or inch and a half away.

(2) Inspect contact boards for corrosion or cracks. Replace if necessary. The pad can be cleaned with a pencil eraser if corroded.

(3) Install one contact board on rod. Install 25 discs. Install other contact board. Pads on boards must be toward discs. Tape together with masking tape if necessary. Install keeper on shaft. Installation is the same on the other assembly.

(4) Position carbon pile assemblies on control box.

(5) Install screws in keepers.

(6) Adjust striker bolts. See Maintenance Practices – Carbon Pile Preload Adjustment for procedure.

(8) Install terminals and nuts on terminal strip.

(9) Install covers on control box.

E. Removal – Control Amplifier

(1) Remove connector on control box.

(2) Remove screws in top cover and tie or tape cover out of the way.

(3) Remove nuts and washers from terminal strip.

(4) Remove wires from amplifier and from terminal strip.

(5) Remove top cover to suitable work area.

(6) Remove screw from one corner at connector with ground wire.

(7) Desolder wires from connector pins E, B, and A.

(8) Remove screws, nuts, washers, and spacers holding PC board to cover.

F. Installation – Control Amplifier

(1) Position new amplifier on cover and install four screws, nuts, washers, and spacers.

(2) Solder wires to connector (see Figure 201).

(3) Install ground wire and screw.

(4) Connect wires to terminals 1 and 2 (see Figure 201) of terminal strip in control box.

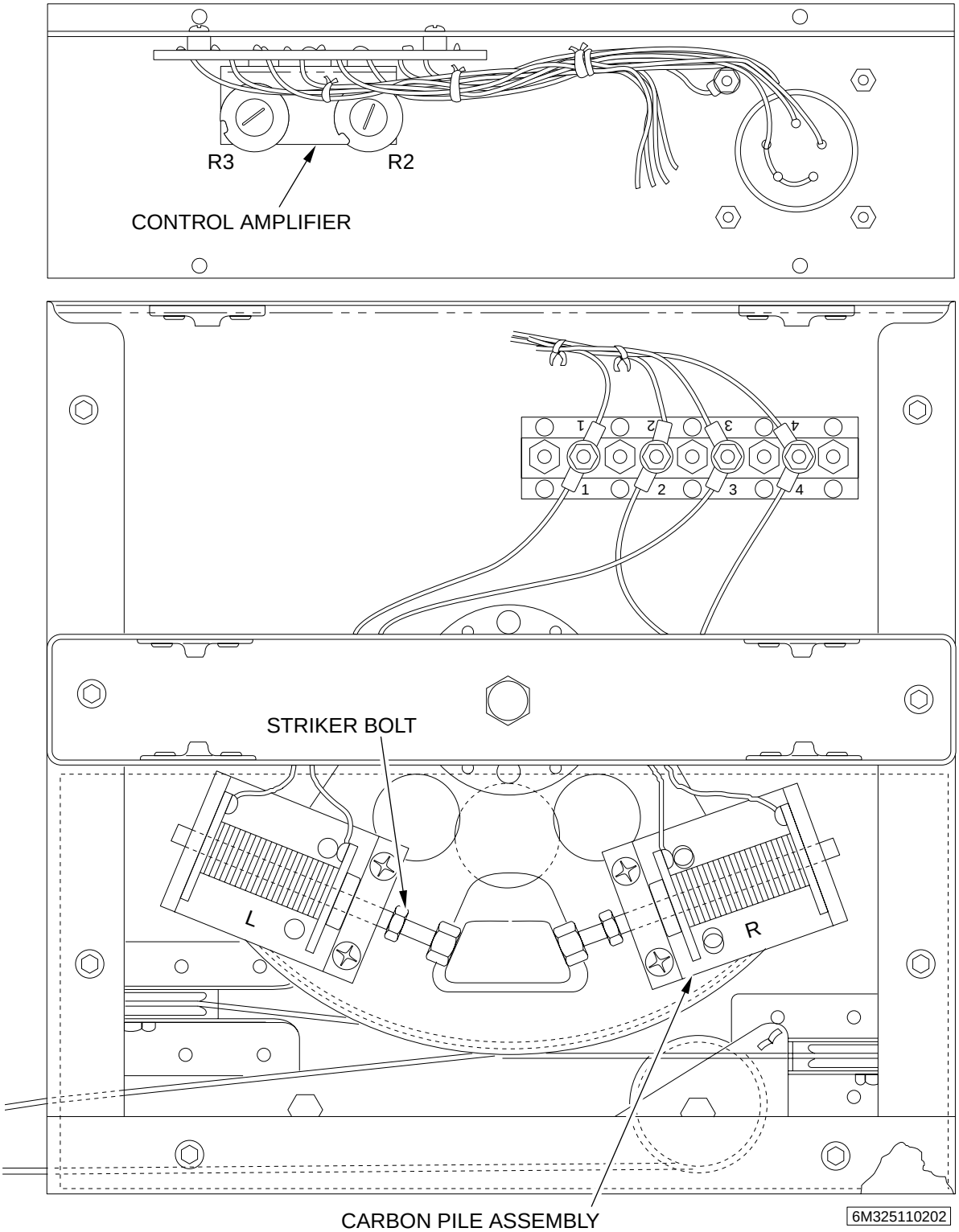
(5) Check carbon pile preload (see 2.B. Adjustment/Test).

(6) Connect wire to terminals 3 and 4.

(7) Check quiescent current (see 2.C. Adjustment/Test).

(8) Install top cover, connector, and safety wire.

(9) Perform operational check.



Steering Control Box Carbon pile Control
Figure 202

2. Adjustment/Test – Steering Control Box

A. Adjustment/Test – Carbon Pile Control Box

(1) Before any adjustment is made to carbon pile stacks or amplifier, be sure cable tension is set properly. (See Section 32-51-20 for tension and proper adjustment techniques).

CAUTION: BEFORE DOING ANY MAINTENANCE OR ADJUSTMENT INSIDE THE BOTTOM OF THE CONTROL BOX, TIE CONTROL COLUMN BACK SECURELY WITH SEAT BELT OR A STRONG TIE STRAP ON PILOT SEAT.

B. Adjustment – Carbon Pile Preload

NOTE: Carbon pile assemblies are very sensitive to slight pressure differences. Some turning and jiggling of the control wheel may be necessary to eliminate bias caused by friction effects.

(1) Be sure steering control wheel and nose gear are centered.

(2) Remove top cover of control box and tie or tape out of the way of elevator weight.

(3) Tie control column back securely. Remove bottom cover.

(4) Remove nuts from terminals 3 and 4 of terminal strip.

(5) Obtain an ohmmeter with an accurate low scale reading of one to ten ohms.

(6) Remove wires connected to carbon piles from terminals 3 and 4 and identify. Clip one lead of ohmmeter to terminals 1 and 2 and obtain a reading of five ohms on terminals 3 and 4 by adjusting the two striker bolts that actuate the carbon pile assemblies. These bolts apply pressure to the carbon pile stacks in the neutral position.

(7) When tightening locknuts, care must be taken not to move adjustment bolt. Tighten locknuts, recheck terminals 3 and 4 with ohmmeter for five ohms reading. Install terminals, nuts, and bottom cover. Before installing top cover, quiescent current should be checked for proper setting. (See 2.C. Adjustment – Quiescent Current).

C. Adjustment – Quiescent Current

NOTE: Be sure carbon pile preload is adjusted before attempting to adjust quiescent current. (See 32-SERVC for test equipment required).

(1) Remove cover of control box.

(2) Remove safety wire on connector. Remove connector.

(3) Connect test equipment.

(4) Tie or tape cover out of way to control box. Ground box top cover to aircraft frame with jumper wire.

(5) Center nose gear and steering wheel.

(6) Turn battery switches ON.

(7) Hold or tape PRESS button down on steering control wheel.

(8) Adjust R2 and R3 to obtain 35 milliamp reading on meters of test box (see Figure 202 for component location). Some turing or jiggling of steering wheel may be necessary to obtain an even reading on both meters.

(9) Verify that a hard over turn of steering wheel obtains a reading of 125 milliamps on meter in direction of turn and a zero (0) reading on the other.

(10) Turn steering wheel left or right. Verify that the milliammeter reading increases in the direction of the turn. Verify that the steering wheel and the nose gear turns in that direction and the meter drops back to zero (0) as the nose gear turns to position. Verify that turning the steering wheel left to a certain degree or angle produces a milliammeter reading of 35 to 125 milliamps on the left meter. Verify that the harder the steering wheel is turned the higher the reading will be. Verify that turning the steering wheel right to same degree or angle as it was turned to the left produces the same reading on the milliammeter as it did on the left.

(11) Turn battery switches to OFF.

(12) Remove test box and jumper.

(13) Install cover.

(14) Reconnect connector and safety wire.

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STEERING CONTROL BOX – DESCRIPTION AND OPERATION

1. General

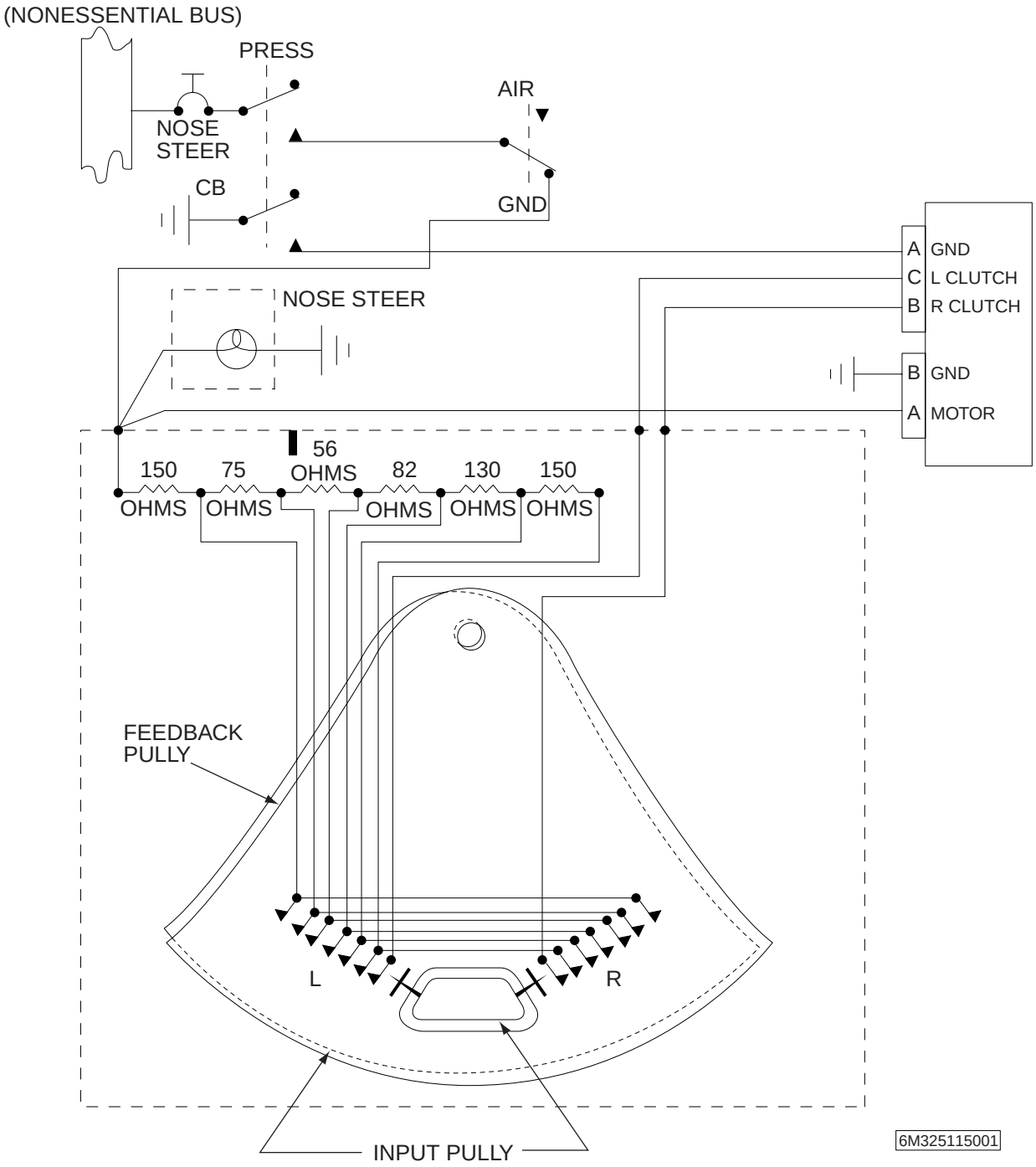
A. Switch Control

(1) The electrical equipment in the control box for the SA226-AT and TC series aircraft consists of two switch assemblies and a resistor bank. Each switch assembly contains seven individual contacts. A barrel at one end of each assembly contains an actuator button. Depressing the actuator button closes the contacts in sequence. These assemblies are mounted to the segmented feedback pulley. two adjustable striker bolts are installed on the segmented input pulley. The striker bolts depress the actuator buttons when the input pulley is rotated. A resistor bank when six resistors is also mounted in the control box.

(2) The first contact that closes in each switch assembly when the actuator is depressed is connected to each servo clutch (see Figure 1). The other six contacts are connected to the six resistors in the resistor bank. The resistors are connected in series (power to resistors to clutch). When the segmented pulleys are in a neutral position with each other, the striker bolts are adjusted so that they do not make contact with the actuator buttons. With no command applied by the steering wheel, none of the switch contacts are closed, no current is applied to the servo clutches, and the nose gear wheels do not turn.

(3) If the steering wheel is turned to the left, the input pulley will move to the left causing the left striker bolt to depress the actuator button. On the left switch assembly, the number of contacts that close depends on the amount the steering wheel is turned. If only the first two contacts are closed, a current flows through all six resistors to the left servo clutch. This produces a small amount of current. The clutch partially engages, and the nose gear wheels turn a small amount. As each of the contacts close, they short out or bypass another resistor in the resistor bank. The current increases as each resistor is depressed. The clutch further engages turning the nose wheels further. When the nose wheels reach the commanded amount of turn the segmented feedback pulley moves causing the left switch assembly to move away from the striker bolt. The contacts open, and the nose wheels stop turning. When the steering wheel is turned to the right, a similar action takes place, and the nose gear wheels turn to the right.

STEERING CONTROL BOX – DESCRIPTION AND OPERATION



Steering Control Box
Switch Control
Figure 1

STEERING CONTROL BOX – FAULT ISOLATION

1. Troubleshooting – Steering Control Box

A. Switch Control

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Steering system inoperative. Annunciator panel light illuminates	Switch assemblies defective or striker bolts not adjusted properly	Adjust striker bolts. Remove and repair switch assemblies.
Steering handling problems	One or more resistors defective in resistor bank	Remove and check resistors.
	Wiring defective within control box	Repair wiring.

STEERING CONTROL BOX – MAINTENANCE PRACTICES

1. Removal/Installation – Steering Control Box

A. Removal/Installation – Switch Control

B. Removal – Control Box

(1) Disconnect electrical connector.

(2) Remove screws holding top cover to box.

(3) Remove screws in bottom cover and remove cover.

(4) Remove nuts and lockwashers from switch assembly actuator barrels holding switch assemblies to angle brackets.

(5) Remove switches, top cover, and resistor bank from control box.

(6) Remove bolt through pulley and back of box at lower right area of box. Observe order of spacer and washer for reinstallation.

(7) Remove bolts through back of box holding box to frames.

(8) Observe cable routing for reinstallation.

(9) Remove bolt center of box through segmented pulleys. Observe order of spacers and washers for reinstallion.

(10) Remove screws in center brace and remove center brace.

(11) Remove safety wire. Remove screws in cable anchors and remove anchors. Mark cables for reinstallation.

(12) Remove segmented pulleys.

(13) Remove cotter pin cable guards in pulley brackets.

(14) Remove box allowing cables to slide through holes in box.

C. Installation – Control Box

(1) Position box and route cable through holes in box.

(2) Route cables over pulleys and install cotter pin cable guards.

(3) Position segmented pulleys in box and route cables on segmented pulleys.

(4) Install cables in cable anchors. Install screws through cable anchors and safety wire.

(5) Position center brace and install screws.

(6) Install bolt through center brace and segmented pulleys with the washers and spacers.

(7) Position box and install bolts securing box to frames.

(8) Install bolt through pulley and back of box with the spacer and washer in the lower right area of box.

(9) Check cable tension (see Section 32-51-20).

(10) position top cover.

(11) Position switch assemblies in angle brackets and install lockwasher and nut on actuator barrels.

(12) Check striker bolt adjustment.

(13) Install bottom and top covers.

(14) Connect electrical connector and safety wire.

(14) Perform operational check.

D. Removal – Switch Assembly and Resistor Bank

(1) Remove safety. Disconnect electrical connector from control box.

(2) Remove screws holding top cover to control box.

(3) Remove screws holding bottom cover to control box. Remove bottom cover.

(4) Remove nuts and lockwashers from switch assembly actuator barrels that hold switches in angle brackets.

(5) Remove switches, top cover, and resistor bank from control box.

(6) Place on suitable work area. Remove safety wire and caps. Remove nut and lockwasher from actuator barrels. Remove screw by actuators and remove switches from covers.

E. Installation – Switch Assembly and Resistor Bank

(1) Inspect for broken, bent, or fused contacts. No contacts should be made with the actuator relaxed. All contacts should be made before the actuator is completely depressed. If contacts are broken or fused, replace entire switch assembly.

(2) Inspect resistor bank for broken wires or evidence of overheating. Replace resistors of necessary.

(3) Install switches in covers. Install screws, lockwashers, and nuts. Install caps and safety wire.

(4) Position switch assemblies, top cover, and resistor bank in control box.

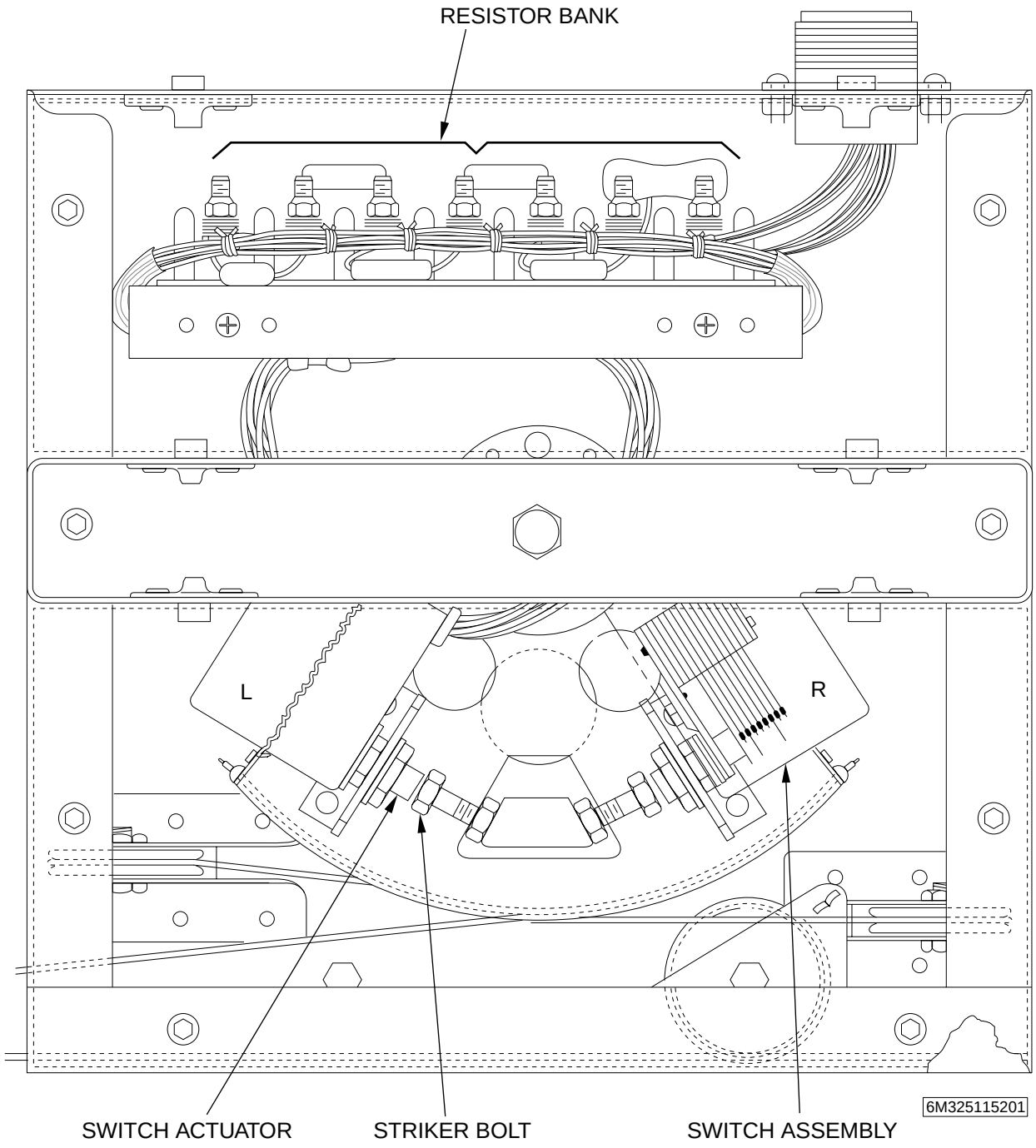
(5) Install nuts and lockwashers that hold switches in angle brackets.

(6) Adjust striker bolts. See Adjustment – Switch Assembly and Striker Bolt.

(7) Install top and bottom covers.

(8) Install connector and safety wire.

(9) Perform operational check.



Steering Control Box
Switch Control
Figure 201

2. Adjustment/Test – Steering Control box

A. Adjustment/Test – Switch Control

(1) Before any adjustment is made to striker bolt, be sure cable tension is properly set. (See Section 32-51-20, Control System maintenance practices, for tension and proper adjustment techniques).

B. Adjustment – Striker Bolt

(1) Be sure cables are adjusted properly and steering wheel and nose gear are centered.

(2) Remove bottom cover of control box.

(3) Adjust the striker bolts to 0.015 plus or minus 0.004 clearance between striker bolts and switch actuators.

(4) Perform operational check.

(5) Install bottom cover.

C. Adjustment – Switch Assembly/Servo Current Check

NOTE: See back of this section, 32-SERVC for test equipment required.

(1) Disconnect electrical connector on control box.

(2) Connect test equipment.

(3) Center nose gear and steering wheel.

(4) Turn battery switches ON.

(5) Hold PRESS button down on steering wheel.

(6) Verify milliammeters read zero (0).

(7) Turn steering wheel left or right. Verify the milliammeter reading goes up in the direction of the turn. Verify the steering wheel and the nose gear turns in that direction and the meter drops back to zero (0) as the nose gear turns to position. Verify turning the steering wheel left to a certain degree or angle produces a milliammeter reading of 35 to 125 milliamps on the left meter. The harder the steering wheel is turned the more the reading will be. Verify turning the steering wheel right to same degree or angle as it was turned to the left produces the same reading on the right milliammeter as it did on the left.

(8) Turn battery switches to OFF.

(9) Disconnect test equipment.

(10) Connect electrical connector on control box.

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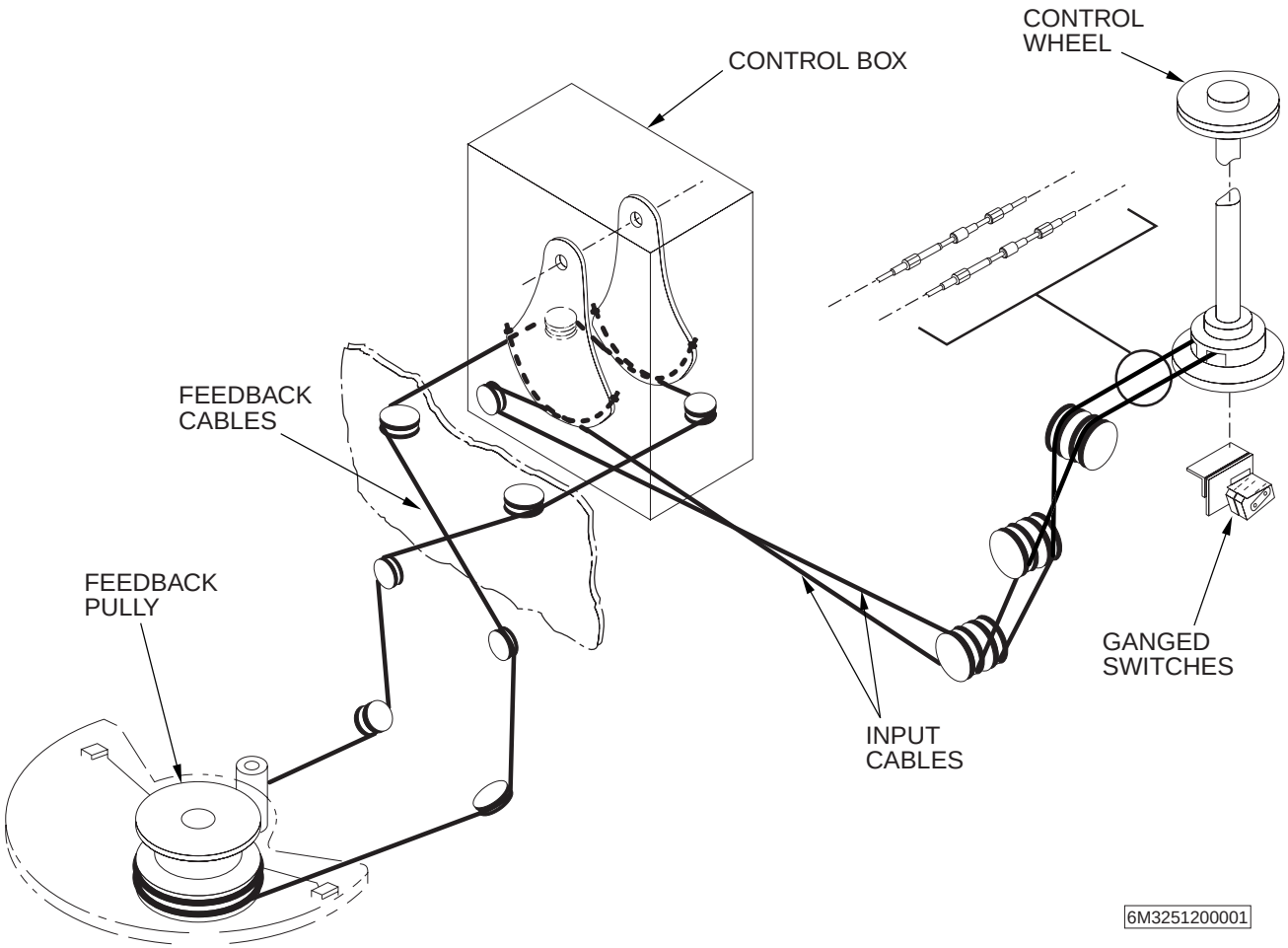
STEERING CONTROL SYSTEM – DESCRIPTION AND OPERATION

1. General

A. Wheel and Cables

(1) The steering control system consists of a steering wheel, palm press switch, shaft, cable pulley drum, and connecting cables. Located forward on the left console, the system gives the pilot a means of steering the aircraft with the left hand while controlling the throttles with the right. The steering wheel is on the upper end of the shaft. Movement is conveyed from the control wheel to the control box by the cable pulley drum and input cables. Movement of the nose gear wheels is conveyed to the control box by the feedback cables. A rod extends through the shaft to a set of ganged switches at the bottom of the assembly. The switches must be made to operate the system. Proper operation of the steering system depends on the inputs to the steering box from the control wheel and the nose gear assembly. Friction should be minimized and cable tensions properly adjusted, particularly from the control wheel to the steering box. Excess cable tension increases friction and may cause binding or rubbing of cables due to deflection of pulley supports, etc. Slack cables cause overshotting steering and undesirable handling of the aircraft.



Steering Control System
Figure 1

STEERING CONTROL SYSTEM – FAULT ISOLATION

1. Troubleshooting – Steering Control System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Steering handling problems	Cable tension not adjusted properly.	Adjust cable tension.
Steering inoperative	Steering on ganged switches not adjusted properly or defective.	Adjust or replace.
Steering servo runs continually, annunciator light illuminated	Switch actuation rod sticking.	Free and lubricate.

STEERING CONTROL SYSTEM – MAINTENANCE PRACTICES

1. Removal/Installation – Steering Control System

A. Removal – Steering Control Wheel

(1) Remove forward and center left console doors.

(2) Mark cables so they can be replaced correctly. Remove safety wire and unscrew turnbuckles.

(3) Loosen allen screws under steering wheel; press wheel down and remove woodruff key.

(4) Remove roll pin in press button with one-eighth (1/8) inch drift punch and remove press button spring and foam rubber washer.

(5) Remove steering wheel.

(6) Remove screws through phenolic block at top of console and screws at bracket bottom of shaft.

(7) Tilt bottom of shaft forward toward console door opening.

(8) Disconnect electrical connection on switches.

(9) Remove shaft through forward console door opening.

B. Installation – Steering Control Wheel

(1) Place top of shaft through forward console door opening and into hole in console.

(2) Reconnect electrical connections.

(3) Installscrews through console phenolic block and bracket.

(4) Place steering wheel over shaft and push down to console (polished side that has engraved NOSE STEERING facing up).

(5) Place foam rubber washer over shaft. Place spring on top of switch actuator rod.

(6) Position press button on top of rod so that the word PRESS can be read from pilot seat. Align hole in button and rod. Install rod pin.

(7) Lift steering wheel up and align woodruff key slots. Install woodruff key.

(8) Lift wheel up until the allen screws align with groove around shaft. Tighten allen screws.

(9) Install turnbuckles. Set tension (see 2.B. Adjustment – Input Cables).

(10) Perform operational test.

(11) Close console doors.

2. Adjustment/Test – Steering Control System

A. Adjustment – Ganged Switch

(1) Remove left forward console door.

(2) Loosen screw extending through switches and slotted hole in mounting bracket.

(3) Hold PRESS button down. Push switch against end of actuator rod until both switches are made tighten screw.

(4) Perform operational test.

(5) Close console doors.

B. Adjustment – Input Cables Adjustment

(1) Remove left forward console door.

(2) Center steering wheel and nose gear wheels.

(3) Remove safety on turnbuckles.

(4) Check tension forward of left console. Tension to be set at ten pounds at 70°F (21.11°C).

(5) Safety turnbuckles and close console door.

C. Adjustment – Feedback Cables

(1) Centering steering wheel and nose gear wheels.

(2) Check tension upper aft section of nose gear where cables cross. Tension to be set at ten pounds at 70°F (21.11°C).

(3) Adjustment is made where cable ends mount to segmented chain sprocket.

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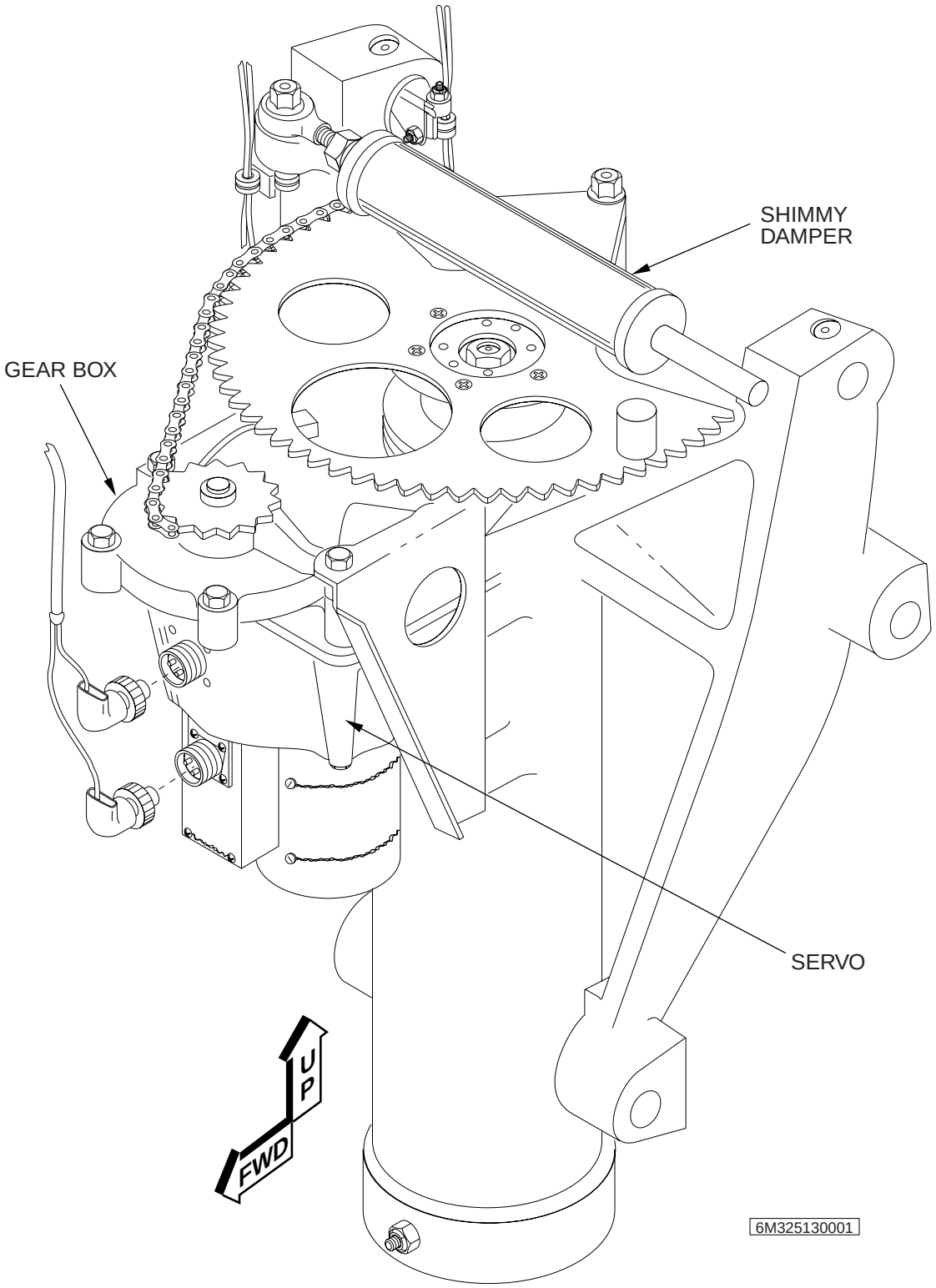
STEERING SERVO AND GEAR BOX – DESCRIPTION AND OPERATION

1. General

A. The steering servo and gear box are mounted on the nose landing gear. A sprocket mounted on the gear box is connected by a chain to a segment sprocket attached to the steerable inner cylinder of the nose gear. Mounted to the gear box is the servo which consists of a DC powered motor and two sets of driving discs enclosed in containers with powdered particles. Torque output of the servo is controlled by current from the control box.

STEERING SERVO AND GEAR BOX – DESCRIPTION AND OPERATION



Servo and Gear Box
Figure 1

STEERING SERVO AND GEAR BOX – FAULT ISOLATION

1. Troubleshooting – Steering Servo and Gear box

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Steering system inoperative. Annunciator light illuminates		PRELIMINARY Connect APU to aircraft. Turn battery switches ON. Refer to Wiring Diagram Manual Section 32-50-01.
	Servo or wiring defective	1.Hold PRESS button down on control wheel. Listen for motor running. If servo motor is running, proceed to step 2. If servo motor is not running, remove motor connector on servo (two pin connector) and check for 28 VDC present at pins A and B. If 28 VDC is present, servo is defective. If 28 VDC is not present, check connector and wiring to nose wheel well terminal strip. 2.Remove clutch connector on servo (three pin connector). Using ohmmeter, check pin A to ground. Hold PRESS button down on control wheel. Reading should be 200 ohm to ground for the SA226–T. Direct to ground for SA226_AT and TC. If reading is not the above, check connector and wiring to nose wheel well terminal strip. Check 200 ohm resistor for SA226–T.
Steering overshoots reacts too quickly, sluggish or pull to the left or right	Servo defective	Check outputs from control box to servo. If outputs are correct above, servo is defective.
	Chain or gear box defective	Check chain slack. Check gear box.
	Shimmy gear damper defective	Check shimmy damper for leakage and damping operation (see 32-50-05).

STEERING SERVO AND GEAR BOX – MAINTENANCE PRACTICES

1. Servicing – Steering Servo and Gear Box

A. Inspect servo and gear box assembly for condition and security. If steering drive chain is dry, lubricate using MIL-L-7870.

2. Removal/Installation – Steering Servo and Gear Box

A. Removal – Servo

(1) Remove safety wire.

(2) Disconnect electrical connectors.

(3) Remove bolts attaching servo to gear box.

(4) Remove servo.

B. Installation – Servo

(1) Position servo in gear box bracket. (To align servo pilot shaft teeth in gear box, manually turn nose gear slightly by wheels).

(2) Secure servo to gear box with bolts.

(3) Connect electrical connectors and safety wire.

(4) Perform operational check.

C. Removal – Gear Box

(1) Remove servo.

(2) Remove chain from one end of segment sprocket.

(3) Remove safety wire from bolts attaching gear box to top of strut.

(4) Remove bolts.

(5) Remove gear box.

(6) Remove bolt through shaft and sprocket.

(7) Remove sprocket.

D. Installation – Gear Box

(1) Install sprocket and bolt on gear box.

(2) Position gear box on brackets.

(3) Secure gear box to brackets with bolts and safety.

(4) Install chain.

(5) Install servo.

(6) Perform operational check.

3. Adjustment/Test – Steering Servo and Gear Box

NOTE: No adjustment can be made to servo or gear box while installed on the aircraft.

A. Adjustment – Gear Box

(1) Remove chain.

(2) Turn sprocket on top of gear box and check for slop left or right.

(3) If there is slop, replace gear box.

B. Adjustment – Chain

(1) Check chain slack.

(2) Chain slack to be 0.06 to 0.50 inch.

(3) If slack is more than 0.50 inch replace chain.

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NOSE GEAR STEERING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. Variable Authority Electric Actuator

(1) This system uses an electronic amplifier and fault protection box that receives a command signal from a set of potentiometers (pots) connected to the rudder pedal linkage and a follow-up signal from a similar set of pots to control an electrically powered actuator (see Figure 1). The actuator is connected to the inner cylinder of the nose gear which turns the wheels, and the follow-up pots follow this action. The shimmy damper is also connected to the inner cylinder and aids the system.

2. Amplifier and Fault Protection Box (32-52-10)

A. The amplifier and fault protection box is located in the cockpit and attached to the inner surface of the aft portion of the left console. The box contains a printed circuit (PC) board. Circuitry on the PC board is used to control the actuator, power control relay, and green NOSE STEER light on the annunciator panel.

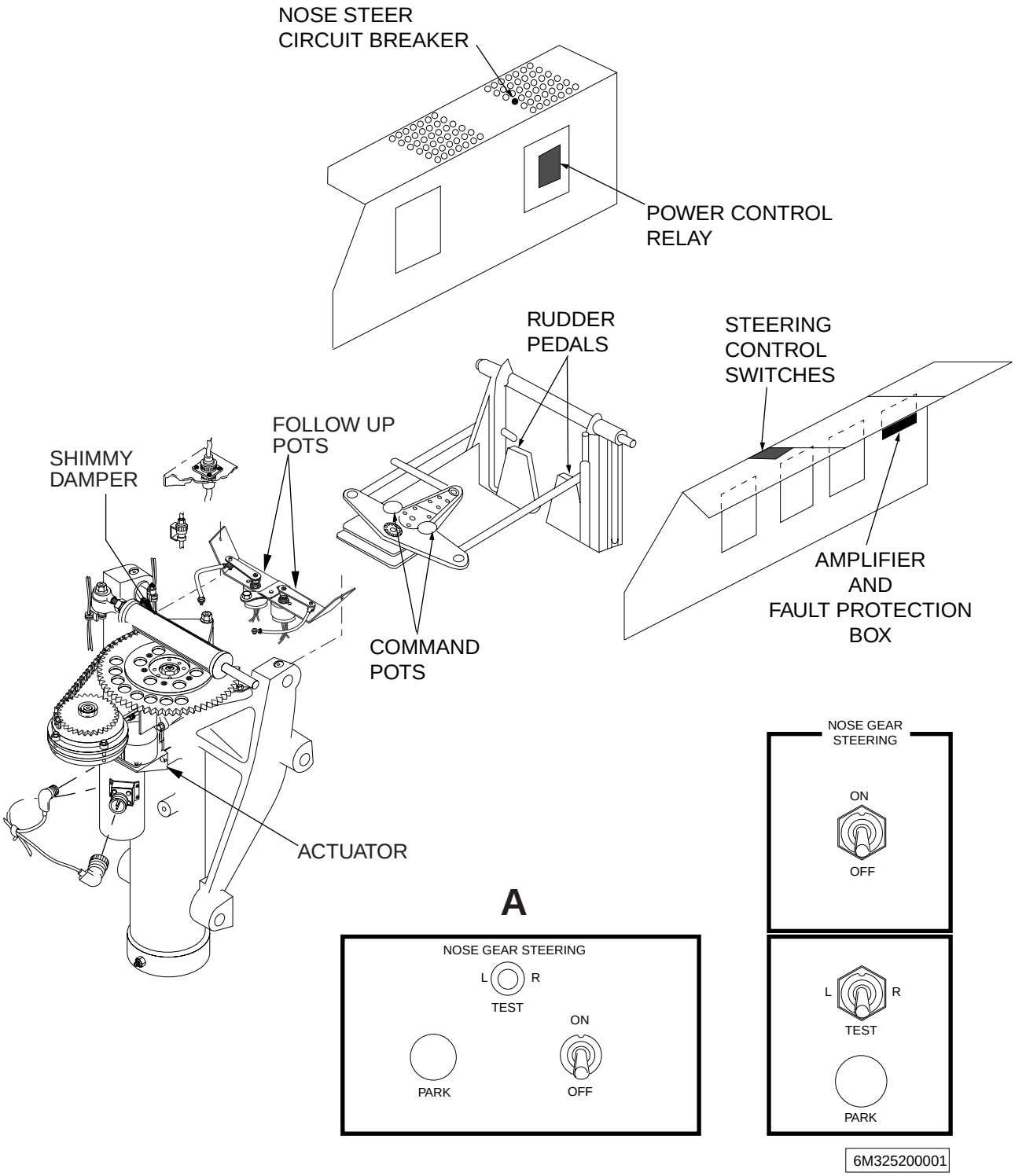
3. Steering Control System (32-52-20)

A. The control system consists of two command pots (one is a control pot and the other is a monitor) on the pilot rudder pedal bellcrank, three switches on the left console, and the actuator power control relay located behind the rear door of the right console.

B. A gear on the shaft of each pot is put into motion by a drive gear. A bolt attached to the center of the drive of the drive gear passes through a bearing on the bellcrank. An arm is welded on the head of the bolt. The arm is attached by an adjustable rod to a bracket on the forward pressure bulkhead. Later models have two rods and a double arm.

C. The bellcrank pivots on a bolt in the same bracket. The pilot rudder pedals are connected to the bellcrank. The pilot and copilot rudder pedal bellcranks are interconnected. When the left or the right rudder pedals are pushed, the bellcranks pivot.

NOSE GEAR STEERING SYSTEM – DESCRIPTION AND OPERATION



Nose Gear Steering System
Figure 1

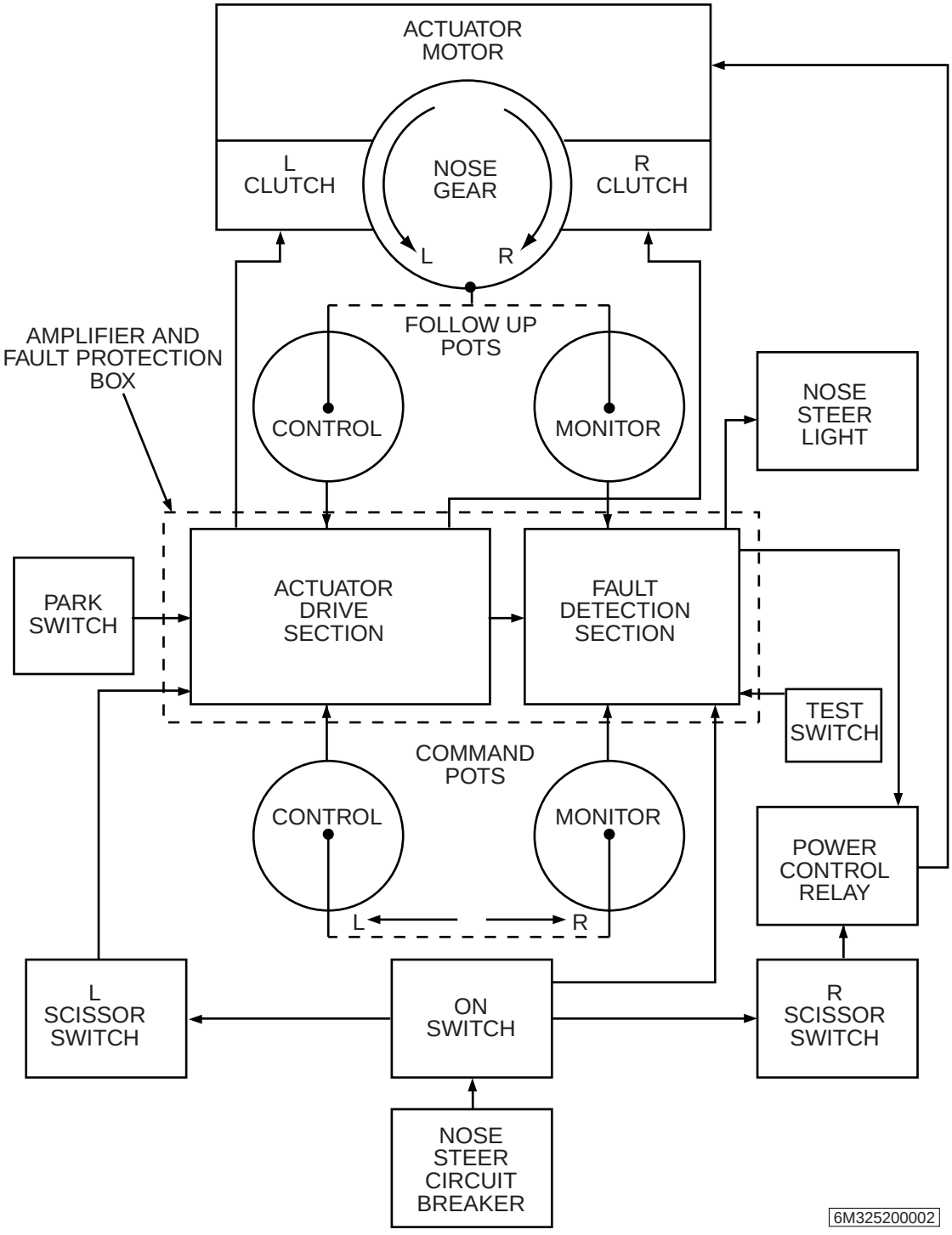
D. The NOSE STEERING ON switch routes power from the NOSE STEER circuit breaker on the nonessential bus through scissor switches on both main gears to the power control relay and amplifier and fault protection box. The PARK switch and TEST switch will be discussed later on under Operation. The power control relay connects power to the actuator and is controlled by the amplifier and fault protection box.

4. Electric Actuator and Follow-Up System (32-52-30)

A. Located on the nose gear, the actuator consists of a DC powered motor, two powdered iron magnetic clutches, and a gear reduction unit. The motor provides the initial torque. The clutches are controlled by current for the amplifier and fault protection box. The clutches couple the torque in a clockwise or counterclockwise direction to the gear reduction unit. A sprocket attached to the output shaft of the gear reduction unit is connected by a chain to a large segmented sprocket mounted on the steerable inner cylinder of the nose gear. The inner cylinder extends through the main body of the nose gear and is connected to the wheels. The follow-up system consists of a control and monitor pot mounted on a bracket on the upper aft portion of the nose gear. An arm on the shaft of each pot is connected by an adjustable rod to a pulley on the inner cylinder.

5. Shimmy Damper (32-52-40)

A. The shimmy damper is mounted to the pulley on the inner cylinder of the nose gear. The piston of the damper attaches to an ear on the main body of the gear.



Nose Gear Steering System Diagram
Figure 2

6. Operation

A. When the pilot rudder bellcrank pivots, the shafts of the command pots rotate. Similarly, when the pulley on the inner cylinder of the nose gear turns, the shafts of the follow-up pots rotate. This causes the wiper within the pots to pass across the resistance winding. A voltage is applied across the resistance winding of each of the pots by the amplifier and fault protection box (see Figure 2). The amplifier and fault protection box receives delta voltage as the wiper passes across the resistance windings. This delta voltage is used to reference rudder pedal and nose gear position.

B. The amplifier and fault protection box contains two main sections, the actuator drive section and the fault detection section. The actuator drive section receives the delta voltage from the command and follow-up control pots and drives the actuator clutches. The fault detection section receives the delta voltage from the command and follow-up monitor pots. Logic circuitry compares the monitor delta voltage to the control pot delta voltage and controls the NOSE STEER light and power control relay. When the steering system is operating normally, the fault detection circuitry provides continuous illumination of the NOSE STEER light and closes the power control relay connecting motor power and clutch drive current to the actuator. Should the fault detection section sense a difference between the delta voltage of the command control and monitor port, or the follow-up control and monitor pots, it will cause the NOSE STEER light to blink and open the power control relay. The TEST switch duplicates this fault condition and produces the same effect. When the rudder pedals are pushed either left or right, the command control pot delta voltage changes. This causes an imbalance in the actuator drive section. The actuator drive section then provides clutch drive current for the appropriate clutch. This allows the actuator to turn the nose gear to bring the follow-up control pot to the position necessary to bring it back to a balanced condition. The steering system operates in the normal authority and maximum (MAX) authority modes. Normal authority is used for high speed taxiing, and the nose wheels will turn approximately eight degrees left or right for the corresponding maximum rudder pedal travel. MAX authority is used for slow speed taxiing and parking maneuvers. In MAX authority, the wheels will turn nearly to the limits of the centering cam off the nose gear. The PARK button is used to shift the system into MAX authority. A light within the button illuminates slowly over a period of a few seconds as the system shifts into MAX authority. When the button is pushed, circuitry in the actuator drive section increases the delta voltage of the command pots to make the nose gear turn further. This action brings the follow-up pots to the position to balance the actuator drive section. The shimmy damper works with the steering system to prevent any sudden and uncontrollable movement of the nose wheels.

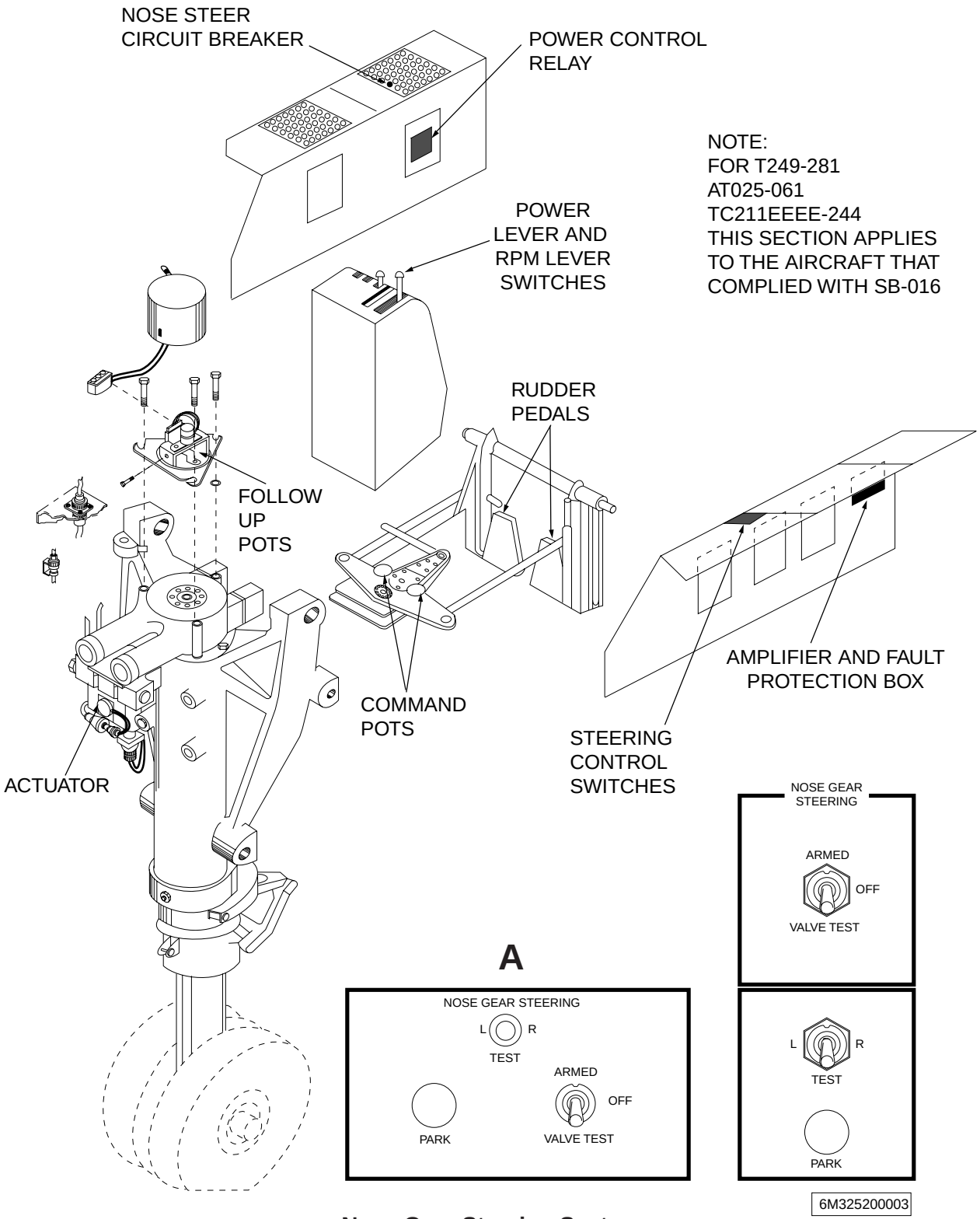
7. Variable Authority – Hydraulic Actuator

A. This system uses an electronic amplifier and fault protection box that receives a command and follow-up signal to control the hydraulically powered actuator. The commanded signal comes from a set of pots on the rudder pedal bellcrank. The follow-up signals comes from a similar set of pots on the nose gear. The actuator is mounted directly to the inner cylinder of the nose gear which turns the wheels. The follow-up pots track this action.

8. Amplifier and Fault Protection Box (32-52-10)

A. The amplifier and fault protection box is located in the cockpit and attached to the inner surface of the forward portion of the left console. The box contains a printed circuit (PC) board. Circuitry on the PC board is used to control the actuator, power control relay, and green NOSE STEER light on the annunciator panel.

NOSE GEAR STEERING SYSTEM – DESCRIPTION AND OPERATION



Nose Gear Steering System
Figure 3

9. Steering Control System (32-52-20)

A. The control system consists of two command pots (one is control, the other is a monitor) on the pilot rudder pedal bellcrank, a switch on the left POWER lever, and a switch on the right RPM lever. It also consists of three switches on the left console, and the actuator power control relay located behind the rear door of the right console.

B. A gear on the shaft of each pot is put into motion by a drive gear. A bolt attached to the center of the drive of the drive gear passes through a bearing on the bellcrank. An arm is welded on the head of the bolt. The arm is attached by an adjustable rod to a bracket on the forward pressure bulkhead. Later models have two rods and a double arm.

C. The bellcrank pivots on a bolt in the same bracket. The pilot rudder pedals are connected to the bellcrank. The pilot and copilot rudder pedal bellcranks are interconnected. The bellcranks pivot when the left or the right rudder pedal is pushed.

D. The NOSE STEERING ARMED switch routes power from the NOSE STEER circuit breaker on the nonessential bus through scissor switches on both main gears to the power control relay and amplifier and fault protection box. The VALVE TEST function checks the operation of the actuator solenoid valve. The PARK switch and TEST switch will be discussed later on under Operation.

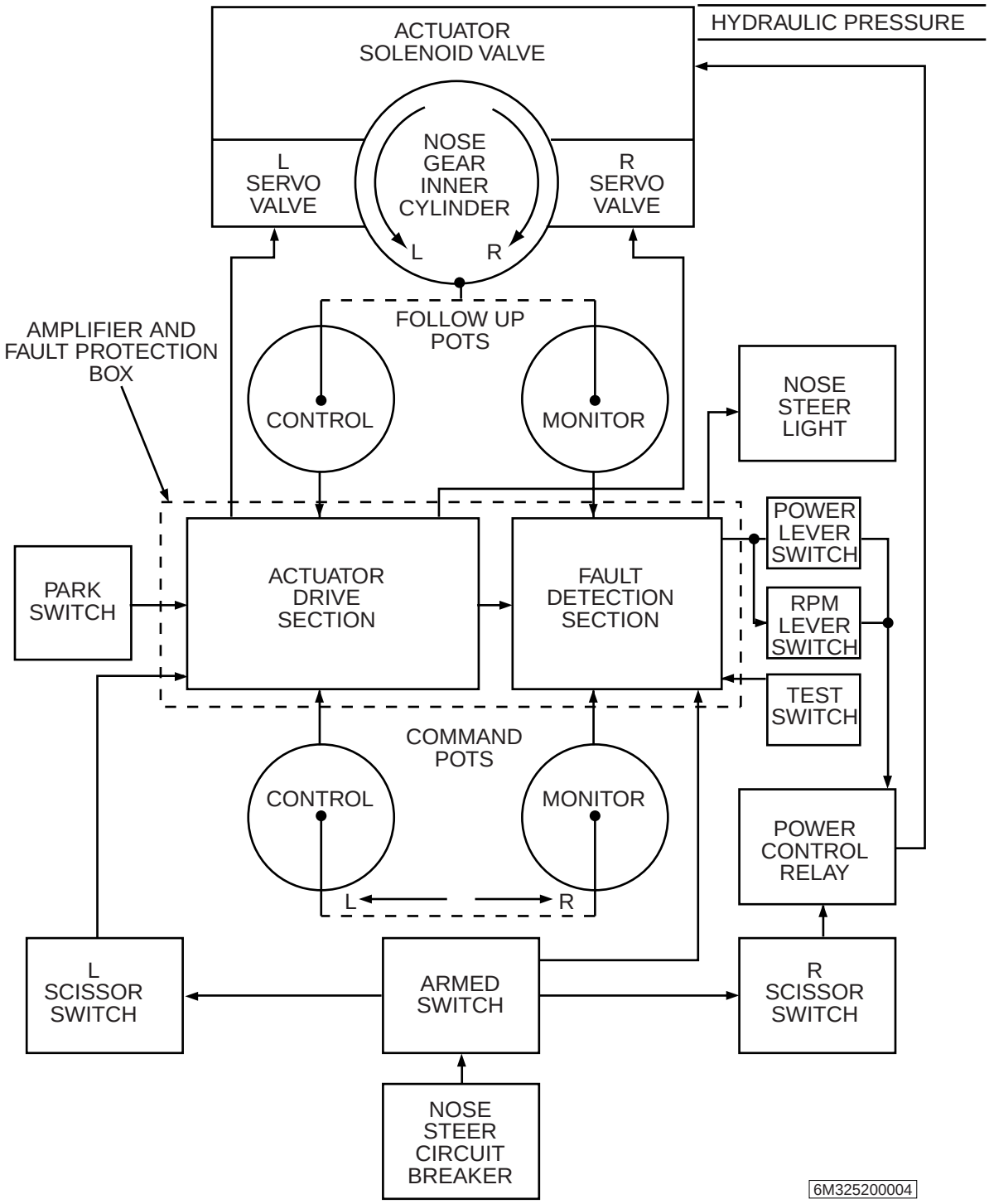
E. The POWER lever switch is thumb operated and must be actuated to close the power control relay. The RPM lever switch is wired parallel to the POWER lever switch and is actuated when the right RPM lever is in full LOW position. The power control relay connects power to the actuator and is also controlled by the amplifier and fault protection box.

10. Electric Actuator and Follow-Up System (32-52-30)

A. Located on the nose gear, the actuator houses a solenoid valve, two servo valves, and two pistons connected to a rack and pinion gear arrangement. Hydraulic pressure from the normal gear down line of the left landing gear actuator is allowed to pass to the servo valves by the solenoid valve. The servo valves are controlled by voltage from the amplifier and fault protection box. The servo valves meter hydraulic pressure to, the pistons. The pistons are connected to two racks that turn the pinion gear.

B. The actuator housing is bolted to the main body of the nose landing gear. The pinion gear is attached directly to the inner cylinder which extends through the main body of the nose gear to the wheels.

C. The follow-up system consists of a control and monitor pot piggyback mounted on a triangular bracket. A single shaft turns both pot wipers. A bracket on the shaft is attached to the actuator pinion gear. The same bolts that attach the actuator housing to the nose gear also attach the triangular bracket.



Nose Gear Steering System Diagram
Figure 4

11. Nose Wheel Steering Operation

A. When the pilot rudder bellcrank pivots, the shafts of the command pots rotate. Similarly, when the pulley on the inner cylinder of the nose gear turns, the shaft of the follow-up pots rotate. This causes the wiper within the pots to pass across the resistance winding. A voltage is applied across the resistance winding of each of the pots by the amplifier and fault protection box (see Figure 4). The amplifier and fault protection box receives a delta voltage as the wiper passes across the resistance windings. This delta voltage is used to reference rudder pedal and nose gear position.

B. The amplifier and fault protection box contains two main sections, the actuator drive section and the fault detection section.

C. The actuator drive section receives the delta voltage from the command and follow-up control pots and drives the actuator servo valves.

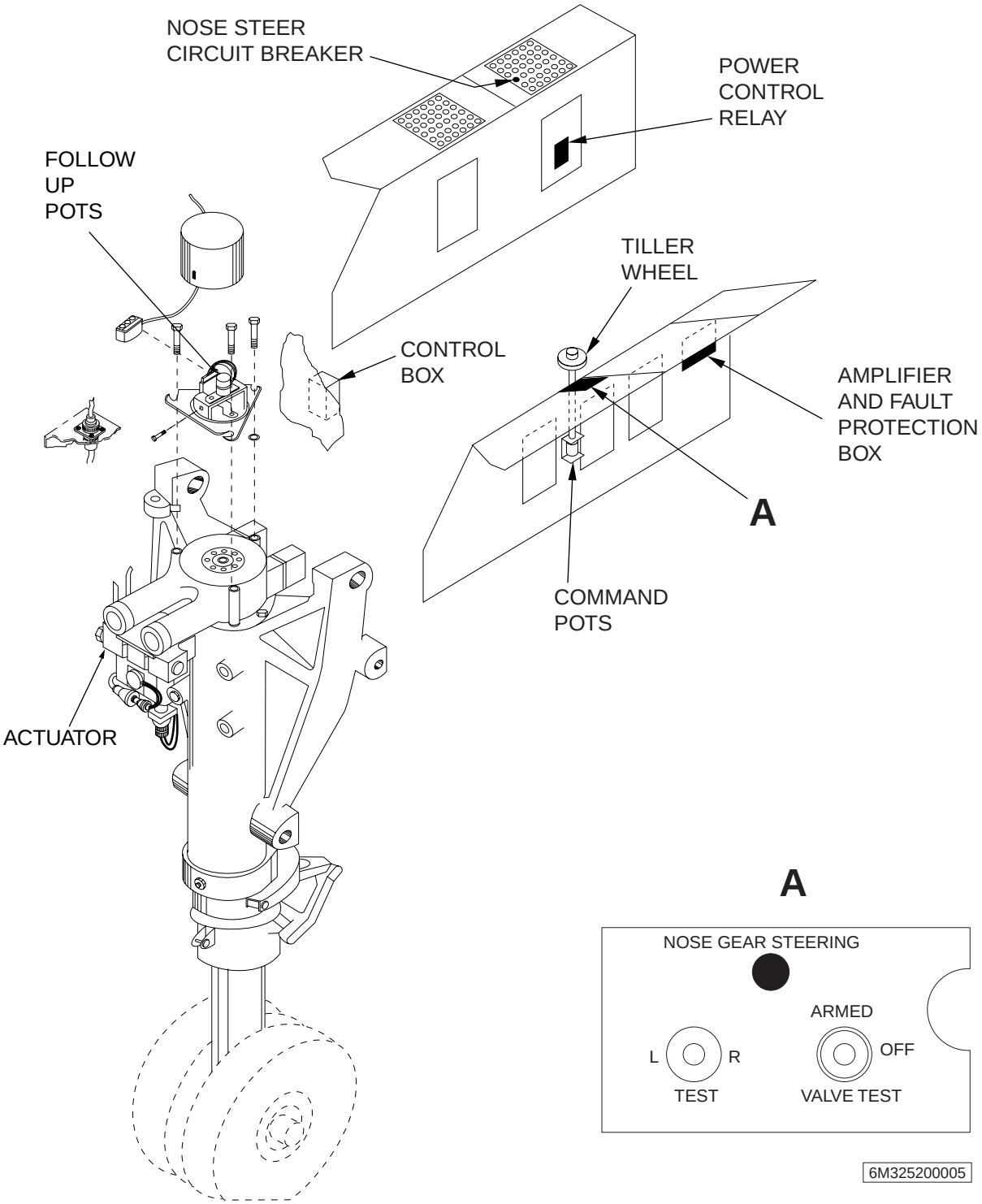
D. The fault detection section receives the delta voltage from the command and follow-up monitor pots. Logic circuitry compares the monitor delta voltage to the control pot delta voltage and controls the NOSE STEER light and power control relay.

E. When the steering system is operating normally, the fault detection circuitry provides con tinuous illumination of the NOSE STEER light and closes the power control relay connecting solenoid valve power and servo valve voltage to the actuator. Should the fault detection section sense a difference between the delta voltage of the command control and monitor pots, or the follow-up control and monitor pots, it will cause the NOSE STEER light to blink and open the power control relay. The TEST switch duplicates this fault condition and produces the same effect.

F. When the rudder pedals are pushed either left or right, the command control pot delta voltage changes this causes an imbalance in the actuator drive section. The actuator drive section then provides the voltage necessary to the appropriate servo valve to allow the actuator to turn the nose gear. This will bring the follow-up control pot to the position neces sary to bring it back to a balanced condition.

G. The steering system operates in the normal authority and maximum (MAX) authority modes. Normal authority is used for high speed taxiing, and the nose wheels will turn approximately 11° left or right for the corresponding maximum rudder pedal travel. MAX authority is used for slow speed taxiing and parking maneuvers. In MAX authority, the wheels will turn to the limits of the actuator.

H. The PARK button is used to shift the system into MAX authority. A light within the button illuminates slowly over a period of a few seconds as the system shifts into MAX authority. When the button is pushed, circuitry in the actuator drive section increases the delta voltage of the command pots to make the nose gear turn further to bring the follow-up pots to the position to balance the actuator drive section.



Tiller Wheel Steering
Figure 5

12. Optional Tiller Wheel

A. General

(1) A tiller wheel steering system is provided as a customer option. The system uses a steering or tiller wheel installed in the forward portion of the left console. The tiller wheel replaces, in function the rudder pedal bellcrank of the variable authority system. A standard bellcrank is installed. The system uses the same amplifier and fault protection box, actuator, and follow-up pots as the variable authority system, but operates in a single or fixed mode of authority. This fixed authority is adequate for high and slow speed taxiing and parking maneuvers.

(2) The amplifier and fault protection box receives a command signal from a set of potentiometers (pots) connected to the tiller wheel and a follow-up signal from a similar set of pots to control a hydraulically powered actuator (see Figure 5). The actuator is mounted directly to the inner cylinder of the nose gear which turns the wheels, and the follow-up pots follow this action.

13. Amplifier and Fault Protection Box (32-52-10)

A. The amplifier and fault protection box is located in the cockpit and attached to the inner surface of the forward portion of the left console. The box contains a printed circuit (PC) board. Circuitry on the PC board is used to control the actuator, power control relay, and green NOSE STEER light on the annunciator panel.

14. Steering Control System (32-52-20)

A. The control system consists of a tiller wheel assembly on the left console, two switches near it, a control box just to the right of the pedestal on the forward pressure bulkhead, and an actuator power control relay located behind the rear door of the right console. The tiller wheel assembly contains a switch to activate the steering system and two command potentiometers, one is a control, the other is a monitor.

B. A steering wheel at the top of the tiller wheel assembly is attached to a shaft that extends through the assembly to its lower end. The two pots are piggyback mounted on a bracket at the bottom of the assembly. A single shaft turns both pot wipers. A slotted arm on the pot shaft is put into motion by an arm attached to the steering shaft. This arm has a pin that rides in the slot of the other arm. A spring on the steering shaft automatically centers the mechanism when the steering wheel is released. A press button in the center of the steering wheel is connected to a rod that passes through the center of the steering shaft to a switch. The press button is spring loaded and must be depressed to operate the steering system.

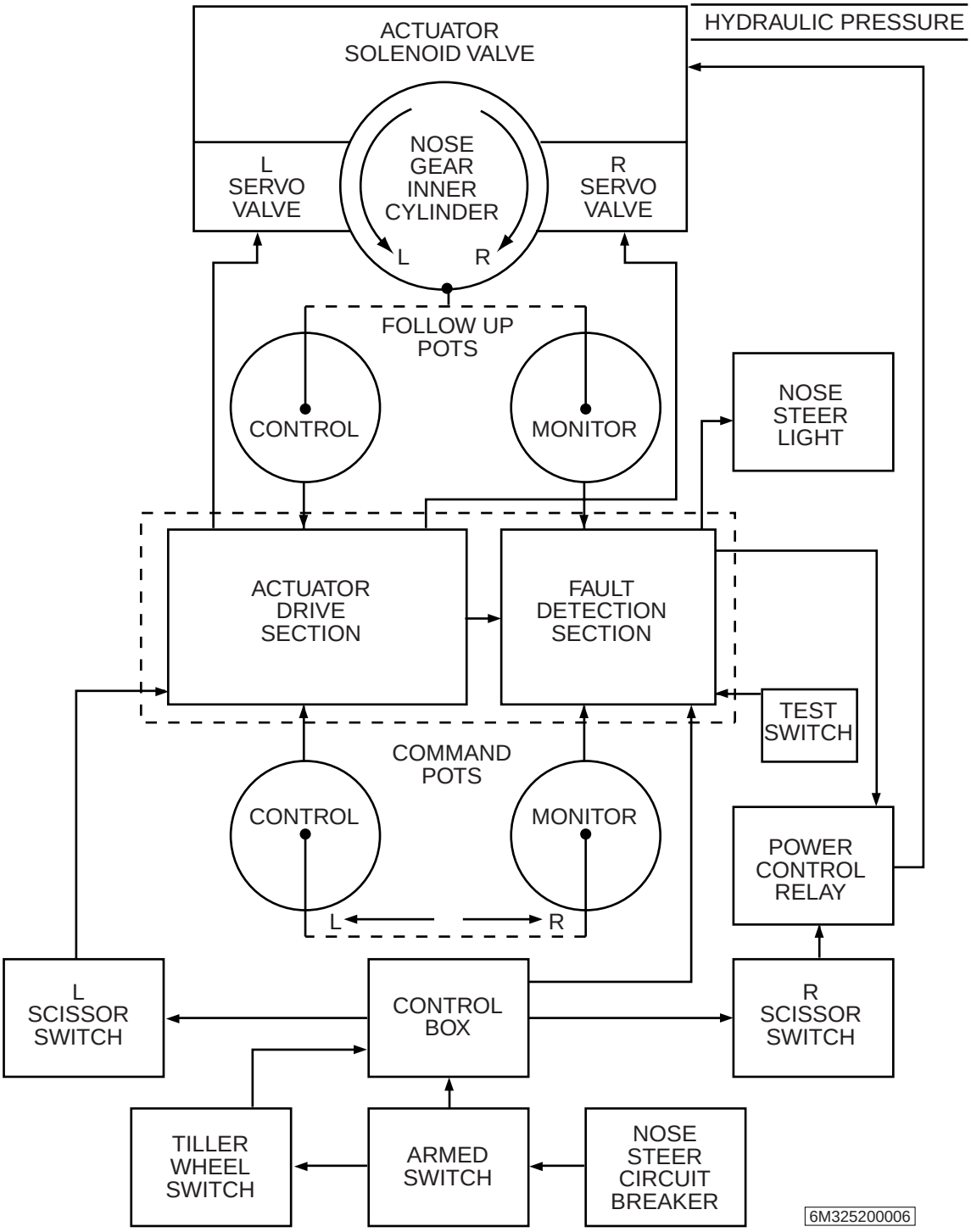
C. The NOSE STEERING ARMED switch routes power from the NOSE STEER circuit breaker on the nonessential bus, through scissor switches on both main gears to the power control relay and amplifier and fault protection box. The VALVE TEST function checks the operation of the actuator solenoid valve.

D. The TEST switch will be discussed later under Operation. The power control relay connects power to the actuator and is controlled by the amplifier and fault protection box.

15. Electric Actuator and Follow-Up System (32-52-30)

A. Located on the nose gear, the actuator houses a solenoid valve, two servo valves, and two pistons connected to a rack and pinion gear arrangement. Hydraulic pressure from the normal gear down line of the left landing gear actuator is allowed to pass to the servo valves by the solenoid valve. The servo valves are controlled by voltage from the amplifier and fault protection box. The servo valves meter hydraulic pressure to, the pistons. The pistons are connected to two racks that turn the pinion gear. The pinion gear is attached directly to the inner cylinder which extends through the main body of the nose gear to the wheels.

B. The follow-up system consists of a control and monitor pot piggyback mounted on a triangular bracket. A single shaft turns both pot wipers. A bracket on the shaft is attached to the actuator pinion gear. The same bolts that attach the actuator housing to the nose gear also attach the triangular bracket.



Tiller Wheel Steering Diagram
Figure 6

16. Nose Wheel Steering Operation

A. When the tiller wheel is turned, the shaft of the command pots rotate. Similarly, when the inner cylinder of the nose gear turns, the shaft of the follow-up pots rotate. This causes the wiper within the pots to pass across the resistance winding. A voltage is applied across the resistance winding of each of the pots by the amplifier and fault protection box (see Figure 4). The amplifier and fault protection box receives a delta voltage from the wiper as it passes across the resistance windings. This delta voltage is used to reference tiller wheel and nose gear position.

B. The amplifier and fault protection box contains two main sections, the actuator drive section and the fault detection section. The actuator drive section receives the delta voltage from the command and follow-up control pots and drives the actuator servo valves. The fault detection section receives the delta voltage from the command and follow-up monitor pots. Logic circuitry compares the monitor delta voltage to the control pot delta voltage and controls the NOSE STEER light and power control relay. When the steering system is operating normally, the fault detection circuitry provides continuous illumination of the NOSE STEER light and closes the power control relay connecting solenoid valve power and servo valve voltage to the actuator. Should the fault detection section measure a difference between the delta voltage of the command control and monitor pots, or the follow-up control and monitor pots, it will cause the NOSE STEER light to blink and open the power control relay. The TEST switch duplicates this fault condition and produces the same effect. When the tiller wheel is turned left or right, the command control pot delta voltage changes. This causes an imbalance in the actuator drive section. The actuator drive section then provides the voltage necessary to the appropriate servo valve to allow the actuator to turn the nose gear. This will bring the follow-up control pot to the position necessary to bring it back to a balanced condition.

NOSE GEAR STEERING SYSTEM – FAULT ISOLATION

1. Troubleshooting – Nose Gear Steering

A. If a defect exists that complies with the items in the DEFECT column, comply with the preliminary and follow the steps in the CORRECTIVE ACTION column. If you suspect an item in the PROBABLE CAUSE column, comply with the preliminary and follow the steps in the CORRECTIVE ACTION column next to the item.

B. When checking for 28 VDC, a voltmeter or a test light may be used. When wiring, wire terminals, or relay contacts are a suspected cause, a test light should be used since it places a slight load on the circuit. The slight load will cause the circuit being checked to open at the trouble point and the light will extinguish. A voltmeter will place almost no load on the circuit and the defective item will be harder to find.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
1.Steering system inoperative. Annunciator panel light is not illuminated		PRELIMINARY Connect APU to aircraft. Refer to Wiring Manual Chapter 32. Remove left aft console door. Remove amplifier and fault protection box connector (P29), turn battery switches ON. Place NOSE STEERING switch to ARMED or hold PRESS button down on tiller wheel. Check for 28 VDC at (P29) pins 20 and 21.
	1.Circuit breaker or left main gear scissor switch or ARMED switch defective	1.If 28 VDC is not present at either pin check ARMED switch and circuit breaker. If 28 VDC is present at 21 but not at 20, check left main gear scissor switch.
	2.Tiller wheel steering only a.Control box defective b.Armed switch defective c.Tiller wheel On switch not adjusted properly or defective	2.Tiller wheel steering only: Remove connector (P518) from control box and check pins H, N, and A to pin K. If 28 VDC is present at H, N, and A pins, control box is defective. If 28 VDC is not present at H, or N, check ARMED switch. If 28 VDC is not present at A, check tiller wheel ON switch.
	3.Amplifier and fault protection box defective	2.Jumper pin 22 to pin 2. If NOSE STEER light on annunciator panel illuminates, amplifier and fault protection box is defective.

NOSE GEAR STEERING SYSTEM – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
2.Steering system inoperative. Annunciator panel light is illuminated		PRELIMINARY Connect APU to aircraft. Refer to Wiring Manual Chapter 32. Remove left aft console door. Remove amplifier and fault protection box connector (P29). Turn battery switches ON. Place NOSE STEERING switch to ARMED or hold PRESS button down on tiller wheel. Jumper pin 1 to pin 22 of (P29).
	1.Power lever switch defective or speed lever not adjusted properly or wiring defective	1.Listen for power connector relay (K20) to close or relay should close when power lever switch is depressed. If speed lever switch is installed, relay should close when right speed lever is pulled completely aft. If relay closes proceed to step 4. If relay does not close, check power lever switch or check speed lever switch to actuate as right speed lever is pulled to LOW and check wiring.
	2.Power control relay (K20) or wiring defective	2.If relay will not close under procedure of step 1, remove right rear console door. Early model aircraft relay is located between copilot seat tracks under floorboard. Check for 28 VDC present at X1 of relay (K20).
	3.Right main gear scissor switch or wiring defective	3.If 28 VDC is not present at X1 of relay (K20) check right scissor switch and wiring. If 28 VDC is present, check wiring to X2 of relay K20. Check diode X1 to X2 of K20 if relay chatters. If relay will not close if X2 is grounded, relay is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	4.Actuator or wiring defective	4.Remove (motor) solenoid valve connector on actuator (larger two pin connector), check for 28 VDC present between pins A and B. If 28 VDC is not present, check wiring to relay (K20) and check ground on pin B. If 28 VDC is present at A and B, remove servo valve connector or check for ground present at pins A and D. Turn Battery switches OFF. Remove jumper and install connector on amplifier and fault protection box. Turn Battery switch ON. If power connector relay is closed, a voltage should be present from pins B and C to pins A and D of servo valve connector. If voltage is present at both actuator connectors, actuator is defective.
3.Steering pulls left or right or makes tighter turn in one direction from the other, or aircraft turns in direction TEST switch is pushed	1.Centering voltage not set properly	1.Adjust centering voltage (see 32-52-30 or 32-52-35).
	2.Rudder pedal or tiller wheel pots not nulled	2.Null pots (see 32-52-20).
	3.Plugged filters in actuator	3.Clean or replace filters (see 32-52-35).
	4.Diodes defective at actuator wire harness	4.Check and replace diodes.
4.Aircraft steers left or right and faults annunciator panel light blinks	1.Wires broken to rudder pedal pots, tiller wheel pots or follow-up pots	1.Check wiring to pots.
	2.Rudder pedal pots or follow-up pots not nulled	2.Null pots (see 32-52-20 and 32-52-35).
	3.Diodes defective at actuator wire harness	3.Check and replace diodes.

NOSE GEAR STEERING SYSTEM – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	4.Amplifier and fault protection box defective	4.After checking 1, 2, and 3 above. replace amplifier and fault protection box.
5.Steering sluggish or over reacts	1.Plugged filters in actuator	1.Clean or replace filters (see 32-52-35).
	2.Shimmy damper defective	2.Check shimmy damper (see 32-52-40).

1. Operational/Test – Nose Gear Steering System

A. Taxi Check – Variable Authority Electric Actuator

(1) Move aircraft to a ramp or taxiway of sufficient length (about 1,000 feet) and width to turn the aircraft around.

(2) With the aircraft moving, place nose steering switch to ON. Observe NOSE STEER light on steady. Apply rudder pedal. Observe the aircraft turns in that direction.

(3) Press PARK button. Observe button lights up and aircraft makes a tighter turn.

(4) Going straight, Observe the rudder pedals are aligned. If it takes left or right rudder pedal to go straight, try this check in the opposite direction to eliminate yaw from the wind.

(a) Observe that moving the test switch left or right produces a kick of the aircraft in that direction, a blinking of the NOSE STEER light, then auto disengagement, and a castering of nose wheels back to straight. Observe both kicks to the left and right are of the same magnitude.

B. Static or Hangar Check

(1) Connect APU to aircraft.

(2) Turn battery switches ON.

(3) Turn nose steering switch ON. Observe NOSE STEER light on steady and servo motor running.

(4) Push left rudder pedal. observe nose gear wheels turn left.

(5) Press PARK button. Observe wheel turn further to the left.

NOTE: The light within the PARK button when pressed, brightens slowly over a period of a few seconds and when released, dims slowly.

(6) Repeat steps **1.**, B., (4) and (5) for right rudder pedal.

(7) Center rudder pedals and nose wheels.

(8) Push TEST switch left. Observe that wheels kick left and stop, and NOSE STEER light blinks.

NOTE: A slight movement of the rudder pedals to the right may be necessary.

(9) Repeat steps **1.**, B., (7) and (8) for right TEST.

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

(10) Turn battery switches OFF.

(11) Disconnect APU.

C. Taxi Check – Variable Authority Hydraulic Actuator

(1) Move aircraft to a ramp or taxiway of sufficient length (about 1,000 feet) and width to turn the aircraft around.

(2) With the aircraft moving, place nose steering switch to ARMED and power lever switch depressed on those aircraft with RPM lever switch, power lever switch need not be depressed with RPM levers in full LOW position. Observe NOSE STEER light on steady. Apply rudder pedal. Observe the aircraft turns in that direction.

(3) Press PARK button. Observe button lights up, and aircraft makes a tighter turn.

(4) Going straight, observe the rudder pedals are aligned. If it takes left or right rudder pedal to go straight, try this check in the opposite direction to eliminate yaw from the wind.

(a) Observe that moving the test switch left or right produces a kick of the aircraft in that direction, a blinking of the NOSE STEER light, then auto disengagement, and a castering of nose wheels back to straight. Observe both kicks to the left and right are of the same magnitude.

D. Static or Hangar Check

NOTE: Nose steering and landing gear control circuit breakers must be in, landing gear handle must be down, and aircraft must be on the ground to perform this check.

(1) Connect hydraulic cart to aircraft.

(2) Connect APU to aircraft.

(3) Turn battery switches ON.

(4) Apply pressure with hydraulic cart.

(5) With nose steering switch ARMED. Observe NOSE STEER light on steady, and hydraulic pressure indicator reads 2,000 psi.

(6) Depress power lever switch and hold. On those aircraft with RPM lever switch, pull right engine RPM lever back to full LOW.

(7) Push left rudder pedal. Observe nose gear wheels turn left.

(8) Press PARK button. Observe wheel turn further to the left.

NOTE: The light within the PARK button when pressed, brightens slowly over a period of a few seconds and when released, dims slowly.

(9) Repeat steps **1.**, D., (7) and (8) for right rudder pedal.

(10) Center rudder pedals and nose wheels.

(11) Push TEST switch left. Observe wheels kick left and stop, and NOSE STEER light blinks.

NOTE: A slight movement of the rudder pedals to the right may be necessary.

(12) Repeat steps **1.**, D., (10) and (11) for right TEST.

(13) Move nose steering arm switch to VALVE TEST. Observe the nose wheels do not turn with movement of rudder pedals.

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

(14) Turn battery switches OFF.

(15) Fill hydraulic reservoir to full mark with flaps up.

(16) Disconnect hydraulic cart.

E. Taxi Check – Optional Tiller Wheel

(1) Move aircraft to a ramp or taxiway of sufficient length (about 1,000 feet) and width to turn the aircraft around.

(2) With the aircraft moving, place nose steering switch to ARMED, observe NOSE STEER light on steady.

(3) Depress and hold PRESS button on top of tiller wheel.

(4) Turn tiller wheel. Observe Aircraft turns in that direction.

(5) With tiller wheel centered, observe aircraft goes straight, if it does not, try this check in the opposite direction to eliminate yaw from the wind.

(a) Observe that moving the TEST switch left or right produces a kick of the aircraft in that direction, a blinking of the NOSE STEER light, then auto disengagement, and a castering of the nose wheels back to straight. Observe both kicks are the same magnitude.

F. Static or Hangar Check

NOTE: Nose steering and landing gear control circuit breakers must be in, landing gear handle must be down, and aircraft must be on the ground to perform this check.

(1) Connect hydraulic cart to aircraft.

(2) Connect APU to aircraft.

(3) Turn battery switches ON.

(4) Apply pressure with hydraulic cart.

(5) Move nose steering arm switch to ARMED.

(6) Depress and hold PRESS button on tiller wheel.

(7) Turn tiller wheel. Observe that nose wheels turn in that direction and are approximately equal to the amount the wheel is turned.

(8) Release tiller wheel (observe It centers).

(9) Center nose wheels.

(10) Push TEST switch left. Observe that wheels kick left and stop, and NOSE STEER light blinks.

NOTE: A slight movement of the rudder pedals to the right may be necessary.

(11) Repeat steps **1.**, F., (9) and (10) with test switch right.

(12) Move nose steering arm switch to VALVE TEST. Observe the nose wheels do not turn with movement of rudder pedals.

(13) Turn battery switches OFF.

(14) Fill hydraulic reservoir to full mark with flaps up.

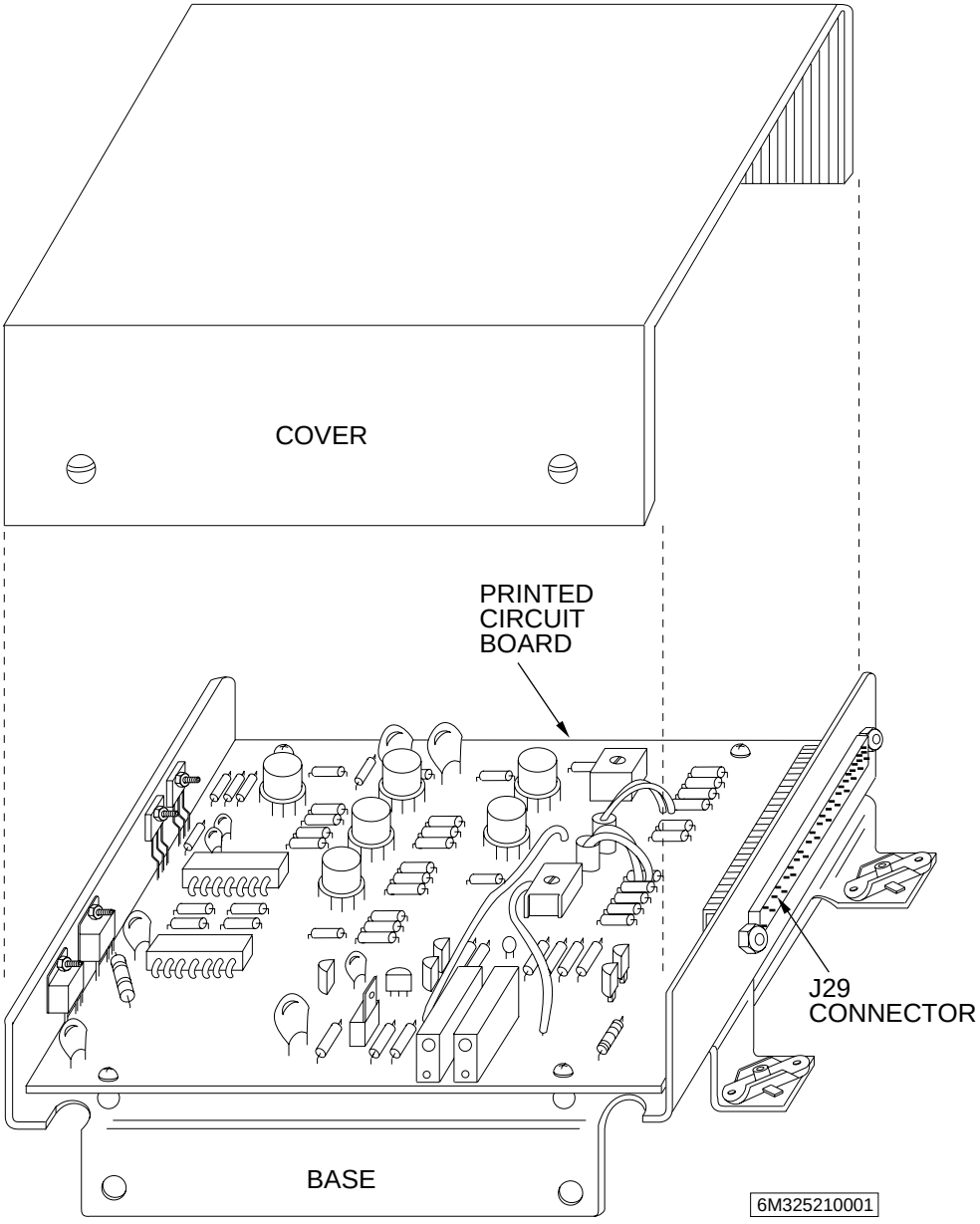
(15) Disconnect hydraulic cart.

(16) Disconnect APU.

AMPLIFIER AND FAULT PROTECTION BOX– DESCRIPTION AND OPERATION

1. General

A. The amplifier and fault protection box is located in the cockpit and attached to the inner surface of the left console just behind the aft door. The box contains a printed circuit (PC) board circuitry on PC board is comprised of the actuator drive section and the fault detection section. The actuator drive section receives delta voltage from the command and follow-up control pots, processes this information, and controls the actuator clutches or servo valves.



Amplifier and Fault Protection Box
Figure 1

AMPLIFIER AND FAULT PROTECTION BOX– DESCRIPTION AND OPERATION

B. The fault detection section receives delta voltage from the command and follow-up control and monitor pots. It controls the NOSE STEER light and the actuator power control relay. When the system is operating normally. the NOSE STEER light is illuminated steady and the power control relay is closed. Should the fault detection section senses a difference between the control and monitor pot delta voltages, the NOSE STEER light blinks and the opens the power control relay.

C. The actuator drive section contains circuitry to shift the system to MAX authority for the variable authority system. A light within the PARK button is illuminated slowly for a few seconds as the system shifts into MAX authority.

AMPLIFIER AND FAULT PROTECTION BOX– MAINTENANCE PRACTICES

1. Removal/Installation – Amplifier and Fault Protection Box

A. Removal – Amplifier

(1) Remove pilots seat in accordance with Chapter 25.

(2) Remove left rear console door.

(3) Remove screws holding amplifier to inner side of console.

(4) Disconnect electrical connector.

(5) Remove amplifier from console.

B. Installation – Amplifier

(1) Place amplifier inside console.

(2) Connect electrical connector.

(3) Position amplifier, secure with screws.

(4) Install console door.

(5) Install pilots seat in accordance with Chapter 25.

(6) Perform operational check.

2. Adjustment/Test – Amplifier and Fault Protection Box

A. All adjustments to the system can and should be made with the command and follow-up pots (see 32-52-20, 32-52-30, and 32-52-35 for procedures). The amplifier and fault protection box should be removed from the aircraft and adjusted in the shop (see Component Maintenance Manual Chapter 32).

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STEERING CONTROL– DESCRIPTION AND OPERATION

1. General

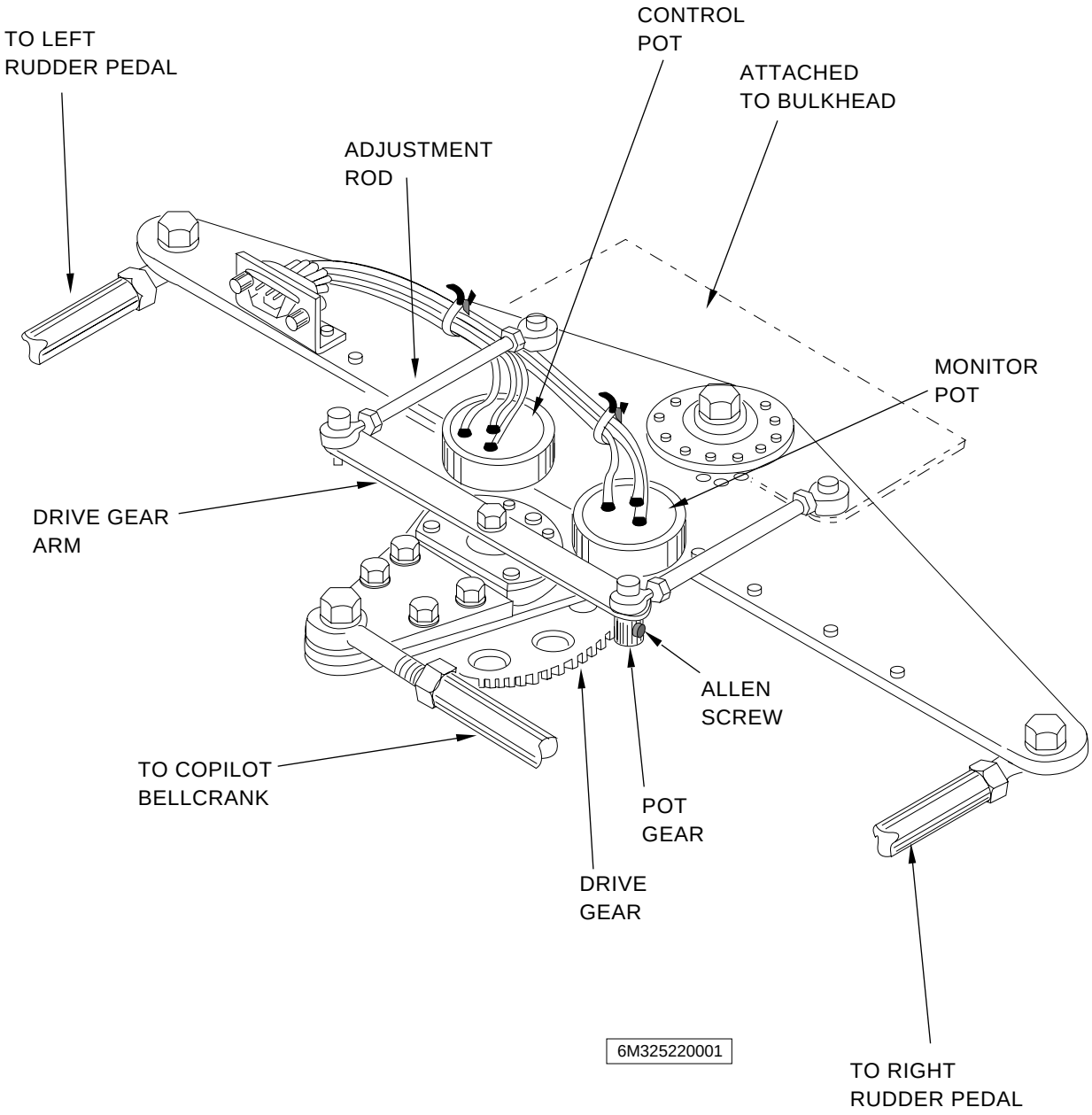
A. Rudder Pedal Control

(1) Steering control is accomplished by potentiometers (pots) located on the rudder pedal bellcrank forward of the pilot rudder pedals; one is a control pot and the other is a monitor pot. Both pots are known as command pots.

(2) A gear on the shaft of each pot is put into motion by a drive gear. A bolt attached to the center of the drive gear passes through a bearing on the bellcrank. An arm is welded on the head of the bolt. This arm is attached by an adjustable rod to a bracket on he forward pressure bulkhead. Later models have two rods and a double arm (T279-999, AT061-999, and TC238-999). The bellcrank pivots on a bolt in the same bracket. The pilot rudder pedals are connected to the bellcrank. When the left or right rudder pedals are pushed, the bellcranks pivot.

(3) When the pilot rudder bellcrank pivots, the shafts of the command pots rotate. This causes the wiper within the pots to pass across the resistance winding. A voltage is applied across the resistance winding of each of the pots by the amplifier and fault protection box. The amplifier and fault protection box receives a delta voltage as the wiper passes across the resistance windings. This delta voltage is used to reference rudder pedal position.

(4) The amplifier and fault protection box uses the delta voltage from the control pot to drive the actuator and the delta voltage from the monitor pot for fault protection. Proper operation of the steering system depends first upon the proper adjustment of these pots. On aircraft T291, AT066, and TC253 the pot shafts can be adjusted separately. When the pots are adjusted properly they are said to be nulled; that is, with the rudder pedals even or centered, both pot wipers area at the same place on their respective resistance windings.



Rudder Pedal Control
Figure 1

STEERING CONTROL– FAULT ISOLATION

1. Troubleshooting – Steering Control

A. Rudder Pedal Control

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Steering handling problems	Wires broken at pots	Repair wiring.
System faults annunciator , panel light blinking	Roll pin bent at drive gear	Replace roll pin in drive gear and drive gear arm bolt.
	Pots not nulled	Null pots.

1. Removal/Installation – Steering Control

A. Removal – Rudder Pedal Bellcrank

(1) Remove pilot seat in accordance with Chapter 25.

(2) Tie pilot and copilot inboard rudder pedals together around bottom aft of pedestal with cargo tie down strap or strong rope.

CAUTION: THE PILOT AND COPILOT INBOARD RUDDER PEDAL ASSEMBLIES HAVE CABLES ATTACHED TO THEM THAT TRAVEL AFT TO THE INTERCONNECT BELLCRANK, AND ATTACH TO A HEAVY SET OF SPRINGS. WHEN THE INBOARD RODS ON THE RUDDER PEDAL BELLCRANK ARE REMOVED, THE PEDAL CAN SNAP FORWARD WITH ENOUGH FORCE TO CAUSE INJURY OR DAMAGE. CARE SHOULD BE TAKEN DURING THIS REMOVAL.

(3) Disconnect electrical connector from bellcrank.

(4) Observe hardware installation upon removal for installation.

(5) Remove nut, washer, and bolt from outboard rod end on the bellcrank.

(6) Remove nut and washer from the inboard rod end. Have an assistant hold and push copilot rudder pedals as necessary to help in removing bolt. Hold inboard rudder pedal back with right hand and remove bolt with left hand.

(7) Remove nut, washer, and bolt from interconnecting rod at center of bellcrank. You may find it easier to remove this rod end before removing inboard rod end.

(8) Remove nut, washer, and allen headscrew from pot adjustment rod ends at pot drive gear arms.

(9) Remove nut, washers, and bolt from center attach point bulkhead bracket.

(10) Remove bellcrank.

B. Installation – Rudder Pedal Bellcrank

(1) Position bellcrank and install center bolt, washers, and nut.

(2) Inspect pot adjustment rods and rod ends. Replace if necessary, (New pot adjustment rods come with new bellcrank assembly).

(3) Install allen headscrews through pot adjustment rod ends and pot drive gear arms. Install washers and nuts. When installing new bellcrank on aircraft with serial numbers before T279, AT061, and TC238 without double pot adjustment rods, the following steps apply:

(a) Install outboard rod on pot drive gear arm.

(b) Install inboard rod on pot drive gear arm.

(c) Center bellcrank. The outboard rod and pot drive gear arm should form a 90° angle.

(d) Place inboard rod on bracket on bulkhead. Position rod so it and the pot drive gear arm form a 90° angle.

(e) Mark bracket through hole in rod end and drill a No. 30 hole at this location. Deburr hole.

(f) Install allen head screw through rod end and bracket. Install washer and nut.

(4) Have assistant push copilot inboard rudder pedal. This should pull pilots inboard pedal back via the tie strap or rope.

(5) Install bolt, washer, and nut on inboard rod on bellcrank.

(6) Install bolt, washer, and nut on interconnecting rod on bellcrank.

(7) Install bolt, washer, and nut on outboard rod on bellcrank.

(8) Remove tie strap or rope.

(9) Check and adjust pots.

(10) Reconnect electrical connector.

(11) Install pilot seat in accordance with Chapter 25.

(12) Perform operational check.

2. Adjustment/Test – Steering Control

A. Adjustment – Rudder Pedal Pot

(1) Two basic types of null meters are available. One type uses batteries or 115 VAC, 60 Hz, and a DC power supply to produce a small DC current used by a meter similar to an ohmmeter to measure the resistance of the pots. Another type of tester uses voltage from the aircraft and the steering system itself and the actual delta voltage from the pots is read on two voltmeters. This type of tester can also be used to measure the actual centering drive voltage the actuator servo valves or clutches are receiving. The drive voltage is set by adjusting the nose gear follow-up pots simultaneously (see Section 32-52-30 and 32-52-40 for procedure). Before this is done it must be verified that the rudder pedal pots and the follow-up pots are nulled.

(2) When using the voltmeter type tester, voltage at the wiper to ground of each pot should be 12 volts with rudder pedals centered. Full left rudder pedal both pots should read approximately five volts and full right approximately 19 volts. Some testers have the capability of measuring the tracking voltage of the pots. This can be done by measuring with a voltmeter from wiper to wiper of each pot. If pots are tracking each other correctly, no voltage will be read. Pots are adjusted using the same procedure as the nullmeter type tester step e (see page 204) with electric power on the aircraft and steering system. See SERVICE INFORMATION in this chapter for more information on testers.

NOTE: The following procedures are for the ohmmeter type nullmeter.

(a) Disconnect electrical connector on pilot rudder pedal bellcrank. ordisconnect amplifier connector when using amplifier plug test harness.

(b) Connect nullmeter test plug.

(c) Center rudder pedals. Pull both rudder pedals back to be sure pilot leg length adjustments are the same.

(d) Nullmeter – some nullmeters have a green band at center scale. On these meters, pots should be adjusted to center of green band.

1 Sensitivity adjustment, high.

2 Select switch – rudder pedal – L control.

3 If needle is not in center of scale, adjustment is required.

4 Select switch – rudder pedal – R monitor.

5 If needle is not in center of scale, adjustment is required.

NOTE: Push left and right rudder pedal slightly. Needle should move to the left and right respectively. If needle does not move, check for broken wires.

(e) To adjust pots.

1 Loosen jam nuts on adjustment rods.

2 Remove screws and attaching rods to drive gear arms.

3 Adjust rods as needed.

4 Install rods, screws, and nuts on drive gear pot arms.

NOTE: Aircraft T291-999, AT066-999, and TC253-999, pots can be adjusted separately by loosening set screw in pot gears and turning pot shaft with screwdriver.

CAUTION: IF POTS ARE ADJUSTED SEPARATELY, ENSURE THAT THE POT GEARS ARE PUSHED ALL THE WAY UP ON THE POT SHAFTS. THE ALLEN SET SCREWS ARE TIGHT, AND THAT THERE IS A CLEARANCE OF 0.030 INCH BETWEEN ALLEN SET SCREWS AND DRIVE GEAR.

(f) Center rudder pedals. Repeat step 2.A.(2)(d).

(g) Tighten jam nuts.

(h) Remove test plug and connect connector.

(i) Perform operational check.

STEERING CONTROL– DESCRIPTION AND OPERATION

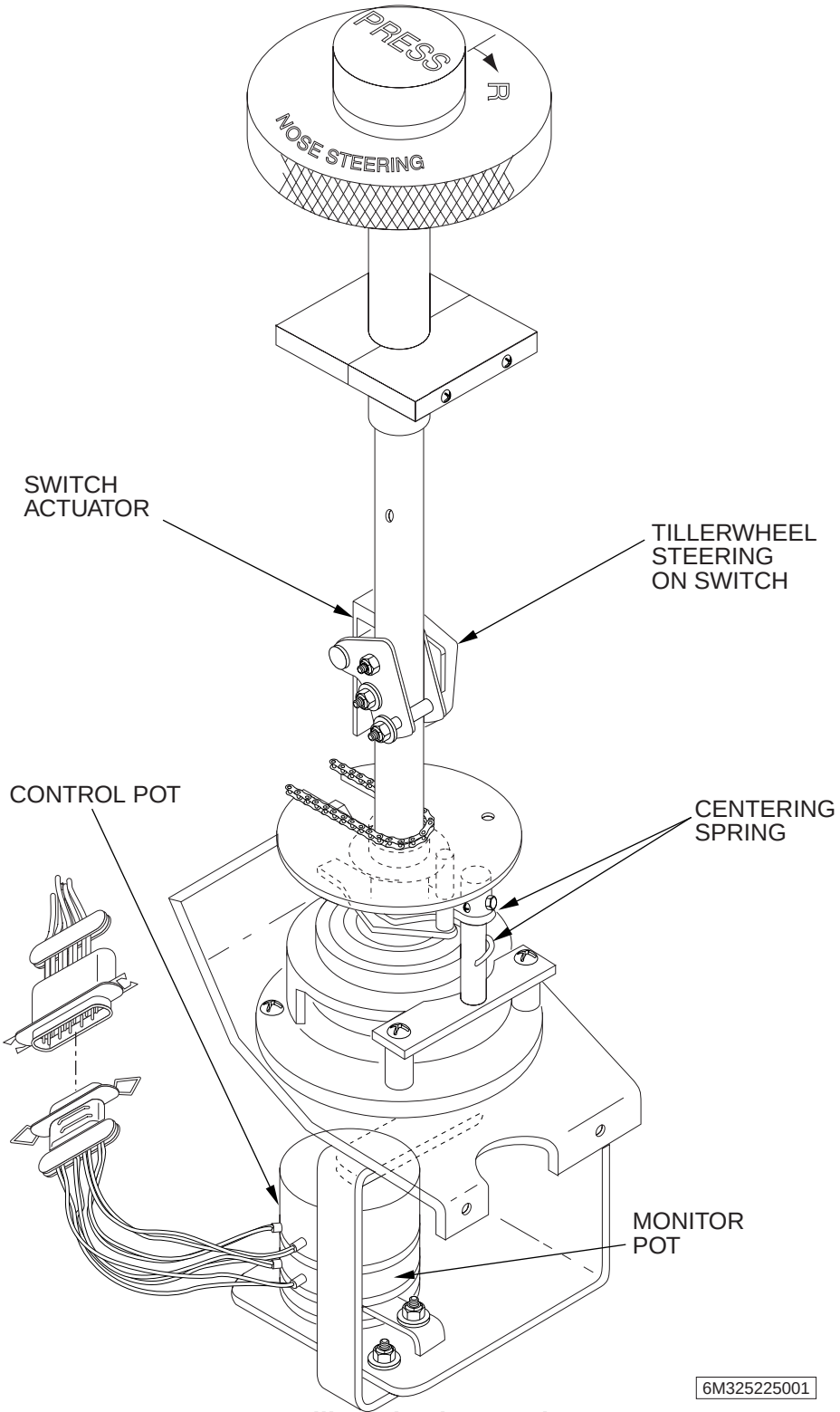
1. General

A. Tiller Wheel Control

(1) The control system consists of a tiller wheel assembly on the left console and a control box just to the right of the pedestal on the pressure bulkhead. The tiller wheel assembly contains a switch to activate the steering system and two command potentiometers; one is a control, the other is a monitor.

(2) A steering wheel at the top of the tiller wheel assembly is attached to a shaft that extends through the assembly to its lower end. The two pots are piggyback mounted on a bracket at the bottom of the assembly. A single shaft turns both pot wipers. A slotted arm on the pot shaft is put into motion by an arm attached to the steering shaft. This arm has a pin that rides in the slot of the other arm. A spring on the steering shaft automatically centers the mechanism when the steering wheel is released. A press button in the center of the steering wheel is connected to a rod that passes through the center of the steering shaft to a switch. The press button is spring loaded and must be depressed to operate the steering system. A relay in the control box closes when the press button is depressed, routing power to other components in the steering system.

(3) When the tiller wheel is turned, the shaft of the command pots rotate. This causes the wiper within the pots to pass across the resistance winding. A voltage is applied across the resistance winding of each of the pots by the amplifier and fault protection box. The amplifier and fault protection box receives a delta voltage as the wiper passes across the resistance windings. This delta voltage is used to reference tiller wheel position. The amplifier and fault protection box uses the delta voltage from the control pot to drive the actuator and the delta voltage from the monitor pot for fault protection. Proper operation of the steering system depends first upon the proper adjustment of these pots. When the pots are adjusted properly they are said to be nulled; that is, with the rudder pedals even or centered, both pot wipers are at the same place on their respective resistance windings.



Tiller Wheel Control
Figure 1

STEERING CONTROL– FAULT ISOLATION

1. Troubleshooting – Steering Control

A. Tiller Wheel Control

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Steering handling problems	Wires broken at pots	Repair wiring.
System faults annunciator, light blinking	Centering spring defective	Check or replace spring.
	Slotted arm slipping on pot shaft	Replace roll pin.
	Pots not nulled	Null pots.
Steering system inoperative	Steering on switch defective or not adjusted properly	Adjust or replace switch.
	Control box defective	Replace wiring or replace box.

STEERING CONTROL– MAINTENANCE PRACTICES

1. Removal/installation – Steering Control

A. Removal – Tiller Wheel

(1) Remove left forward and center console doors.

(2) Disconnect electrical connector to pots.

(3) Loosen allen screws under steering wheel. Press wheel down and remove woodruff key.

(4) Remove roll pin in press button with one-eighth (1/8) inch drift punch and remove press button spring and foam rubber washer.

(5) Remove steering control.

(6) Remove screws through phenolic block at top of console and bracket at bottom of shaft.

(7) Tilt bottom of shaft forward toward console door opening.

(8) Disconnect electrical connections on switches.

(9) Remove shaft through forward console door opening.

B. Installation – Tiller Wheel

(1) Place top of shaft through forward console door opening and into hole in console.

(2) Connect electrical connections on switches.

(3) Install screws through console and through phenolic block and bracket.

(4) Place steering wheel over shaft and push down to console. (Polished side has engraved NOSE STEERING facing up).

(5) Place foam rubber washer over shaft. Place spring on top at switch actuator rod.

(6) Place press button on top of rod so that the word PRESS can be read from pilot seat. Align hole in button and rod. Install roll pin.

(7) Lift steering wheel up and align woodruff key slots. Install woodruff key.

(8) Lift wheel up until the allen screws align with groove around shaft. Tighten allen screws.

(9) Null pots in accordance step 2.A..

(10) Perform operational check.

(11) Close console doors.

C. Removal – Tiller Wheel Pots

(1) Remove tiller wheel in accordance 1.A..

(2) Remove screws and clips at base of pots.

(3) Lift pots out of tiller wheel base and remove.

(4) Remove roll pin attaching slotted arm to pot shaft.

(5) Remove slotted arm.

D. Installation – Tiller Wheel Pots

(1) Install slotted arm and roll pin on pot shaft.

(2) Position pots in base of tiller wheel with pin of arm in slot of slotted arm.

(3) Install screw and clips at base of pots.

(4) Pots can be nulled with tiller wheel not installed.

(5) Install tiller wheel in accordance with 1.B ..

E. Removal – Control Box

(1) Disconnect connector.

(2) Remove screws securing the cover that contains circuitry to base.

(3) Remove case.

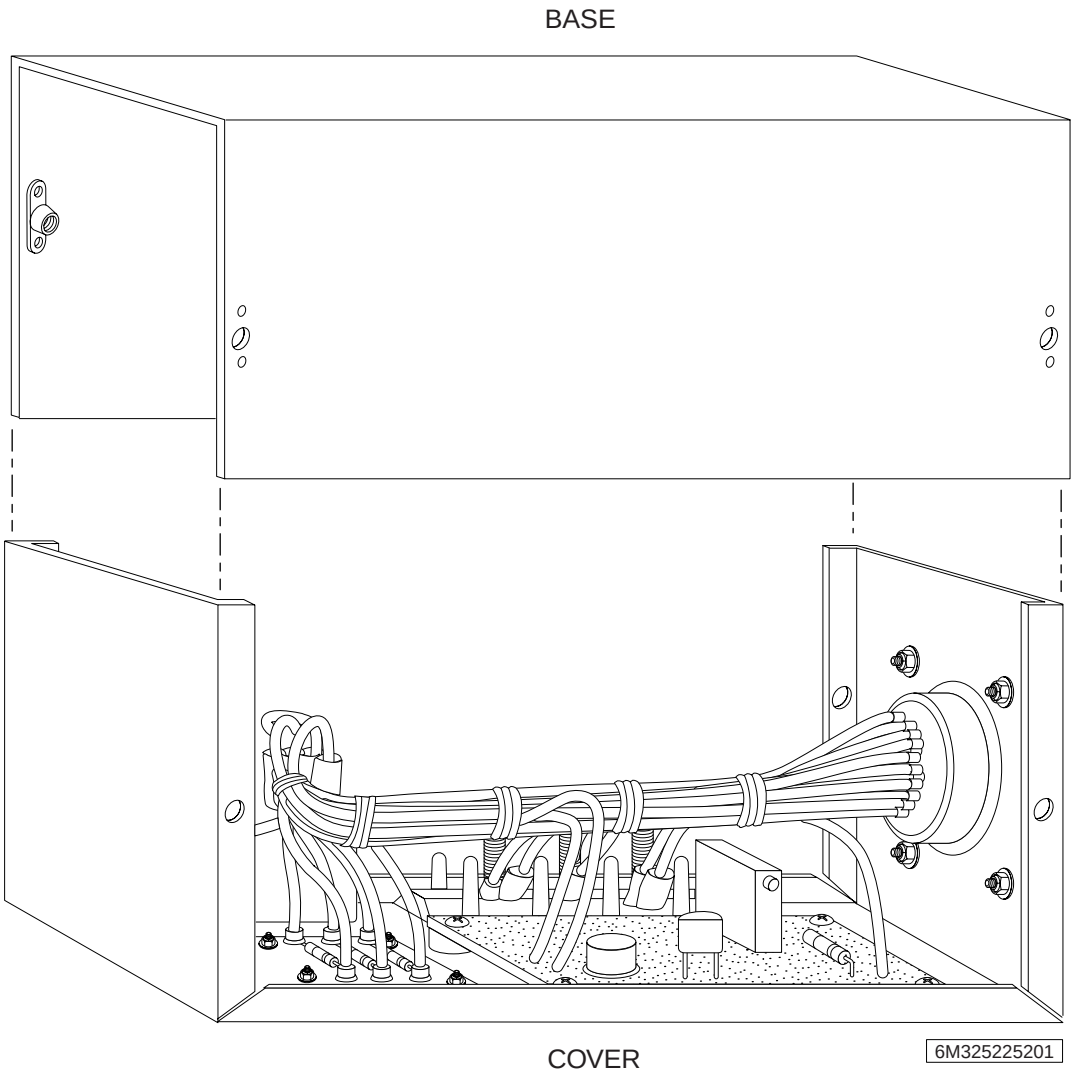
F. Installation – Control Box

(1) Position cover on base.

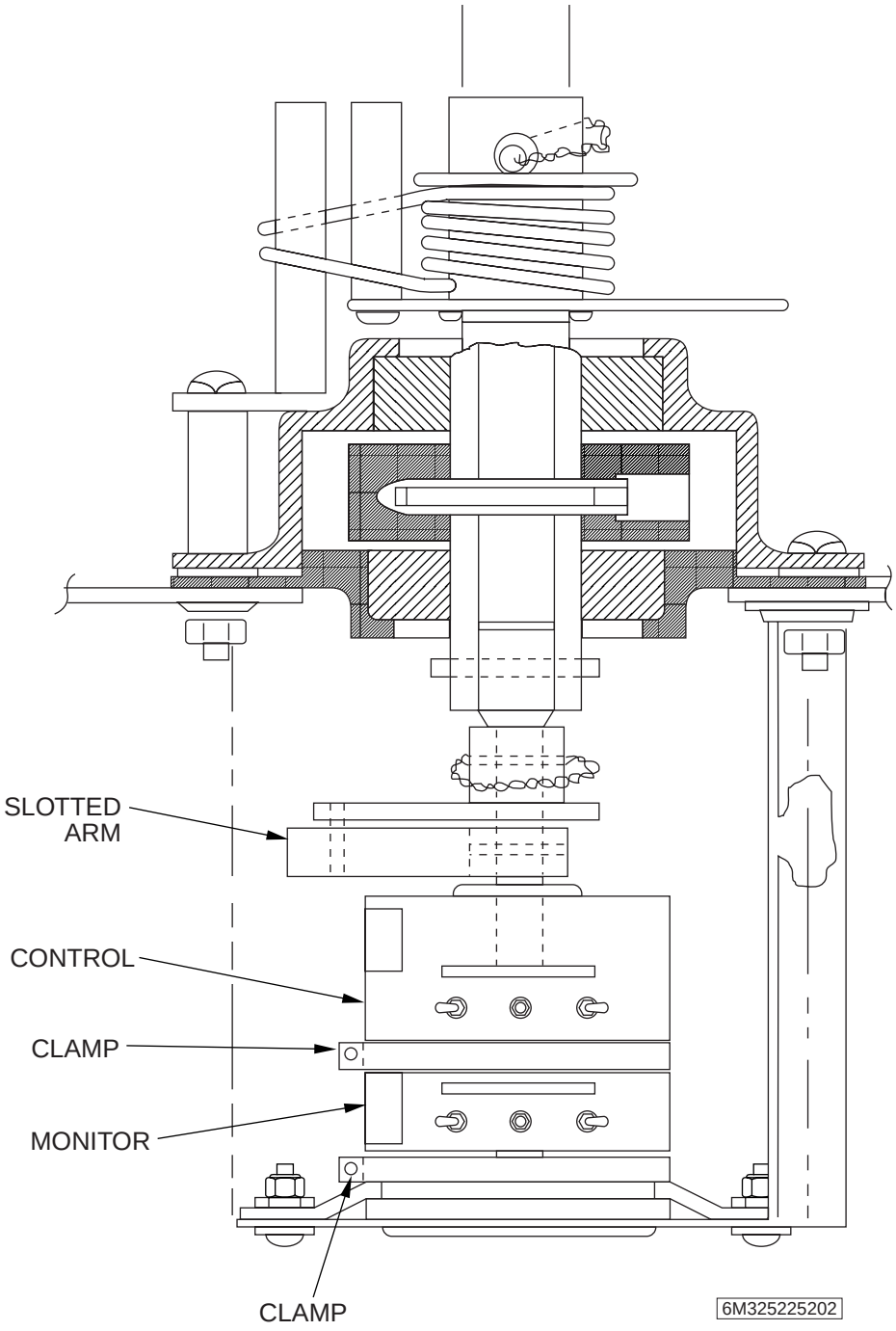
(2) Secure cover with screws.

(3) Reconnect connector.

(4) Perform operational check.



Tiller Wheel Control Box
Figure 201



Tiller Wheel Control Pots
Figure 202

2. Adjustment/Test – Steering Control

A. Adjustment – Tiller Wheel Pot

(1) Two basic types of null meters are available. One type uses batteries or 115 VAC, 60 Hz, and a DC power supply to produce a small DC current used by a meter similar to an ohmmeter to measure the resistance of the pots. Another type of tester uses voltage from the aircraft and the steering system itself and the actual delta voltage from the pots is read on two voltmeters. This type of tester can also be used to measure the actual centering drive voltage the actuator servo valves are receiving. The drive voltage is set by adjusting the nose gear follow-up pots simultaneously (see Section 32-52-35 for procedure). Before this is done it must be verified that the tiller wheel pots and the follow-up pots are nulled.

(2) When using the voltmeter type tester, voltage at the wiper to ground of each pot should be 12 volts with rudder pedals centered. Full left turn on the tiller wheel both pots should read approximately zero volts and full right approximately 24 volts. If tester is connected at amplifier and fault protection box, connector reading should be 12 volts centered, approximately 10 volts left, and approximately 14 volts right. Some testers have the capability of measuring the tracking voltage of the pots. This can be done by measuring with a voltmeter from wiper to wiper of each pot. If pots are tracking each other correctly, no voltage will be read. Pots are adjusted using the same procedure as the nullmeter type tester step (f) with electrical power on the aircraft and steering system. See 32-SERVC in this chapter for more information on testers.

NOTE: The following procedures are for the ohmmeter type nullmeter.

(a) Remove left forward console door.

(b) Disconnect electrical connector on lower portion of tiller wheel; or remove rear console door and disconnect amplifier connector when using amplifier plug test harness.

(c) Connect nullmeter test plug.

(d) Make certain tiller wheel is centered.

(e) Nullmeter – some nullmeters have a green band at center scale. On these meters, pots should be adjusted to center of green band.

1 Sensitivity adjustment, high.

2 Select switch – rudder pedal (tiller wheel) – L control (upper).

3 If needle is not in center of scale, adjustment is required.

4 Select switch – rudder pedal (tiller wheel) – R monitor (lower).

5 If needle is not in center of scale, adjustment is required.

NOTE: Turn tiller wheel slightly to left and right. Needle should move to the left and right respectively. If needle does not move, check for broken wires.

(f) To adjust pots.

1 Center tiller wheel, nullmeter to L control (upper pot).

2 Loosen screw in clamp around center of pots and turn upper pot until needle is in center of scale.

3 Tighten screw in clamp. Nullmeter to R monitor (lower pot).

4 Loosen screw in clamp at bottom of pots and turn lower pot until needle is in center of scale.

5 Tighten screw in clamp.

(g) Repeat step 2.A.(e).

(h) Remove test plug and connect connector.

(i) Perform operational check.

(j) Install console door.

B. Adjustment – Tiller Wheel Switch

(1) Remove left forward console door.

(2) Loosen screw extending through switch and slotted hole in mounting bracket.

(3) Hold PRESS button down. Push switch against end of actuator tab until both switches are made. Tighten screw.

(4) Perform operational check.

(5) Reinstall console door.

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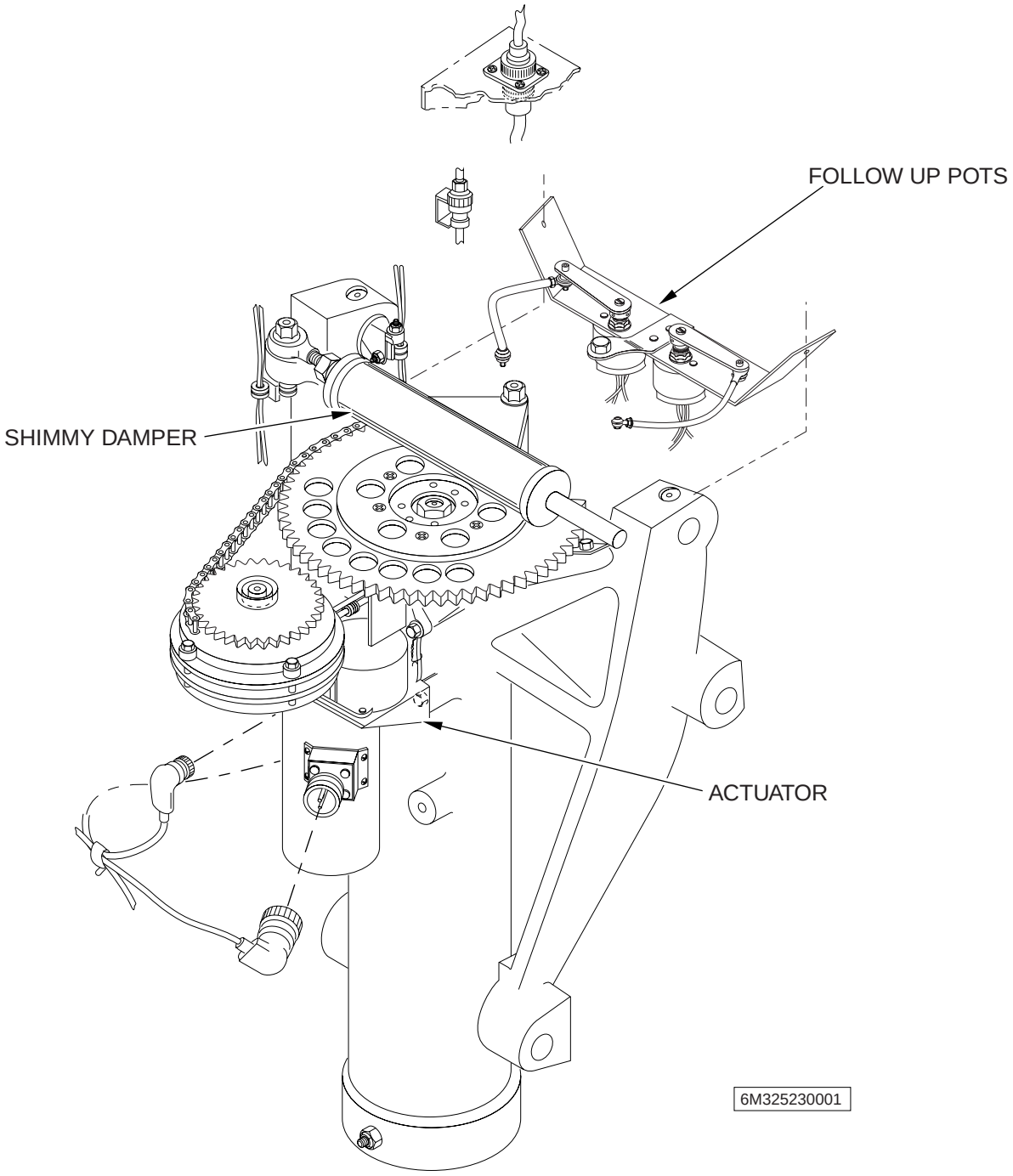
ELECTRIC ACTUATOR AND FOLLOW-UP SYSTEM – DESCRIPTION AND OPERATION

1. General

A. Located on the nose gear, the actuator consists of a DC powered motor, two powdered iron magnetic clutches and a gear reduction unit. The motor provides the initial torque. The clutches are controlled by current from the amplifier and fault protection box. The clutches couple the torque in a clockwise or counterclockwise direction to the gear reduction unit. A sprocket attached to the ouput shaft of the gear reduction unit is connected by a chain to a large segmented sprocket mounted on the steerable inner cylinder of the nose gear. The inner cylinder extends through the main body of the nose gear and is connected to the wheels. The follow-up system consists of a control and monitor pot mounted on a bracket on the upper aft portion of the nose gear. An arm on the shaft of each pot is connected by an adjustable rod to a pulley on the inner cylinder.

B. When the nose gear turns, the shafts of the follow-up pots rotate. This causes the wiper within the pots to pass across the resistance winding. A voltage is applied across the resistance winding of each of the pots by the amplifier and fault protection box. The amplifier and fault protection box receives a delta voltage as the wiper passes across the resistance windings. This delta voltage is used to reference nose gear position.

C. The amplifier and fault protection box uses the delta voltage from the control pot to drive the actuator and the delta voltage from the monitor pot for fault protection. Proper operation of the steering system depends upon the proper adjustment of these pots. The adjustable rods are used to adjust the pots. When the pots are adjusted properly they are said to be nulled; that is with the nose gear centered, both pot wipers are at the same place on their respective resistance windings.



Electric Actuator and Follow-Up System
Figure 1

ELECTRIC ACTUATOR AND FOLLOW-UP SYSTEM – FAULT ISOLATION

1. Troubleshooting – Electric Actuator and Follow-Up System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Steering system inoperative. Annunciator panel light illuminated		PRELIMINARY Connect APU to aircraft. Turn battery switches on. Refer to Wiring Manual, Chapter 32.
	Actuator or wiring defective	Place NOSE STEERING switch to ON. Remove servo motor connector on actuator (larger two pin connector), check for 28 VDC present between pins A and B. If 28 VDC is not present, check wiring to relay (K20) and check ground on pin B. If 28 VDC is present at A and B, remove clutch connector or check for ground present at pin A. If power relay is closed, a voltage should be present from pins B and C to pin A of clutch connector. If voltage is present at both actuator connectors, actuator is defective.
Steering pulls left or right or makes tighter turn in one direction from the other, or aircraft turns in direction TEST switch is pushed	Centering voltage not set properly	Adjust centering voltage.
Aircraft steers left or right and system faults. Annunciator panel light blinks	Wires broken to follow-up pots	Check wiring to pots.
	Follow-up pots not nulled	Null pots.
	Diodes defective at actuator wire harness	Check and replace diodes.
Steering sluggish or over reacts	Actuator defective	Replace actuator.
	Shimmy damper defective	Check shimmy damper (see 32-52-40).

ELECTRIC ACTUATOR AND FOLLOW-UP SYSTEM – MAINTENANCE PRACTICES

1. Servicing – Electric Actuator and follow-Up System

A. Inspect actuator, sprockets, and drive chain for condition and security. If chain is dry, lubricate using MIL-L-7870. No adjustment can be made to actuator while installed on the aircraft.

2. Removal/Installation – Electric Actuator and follow-Up System

A. Removal – Actuator

(1) Remove safety wire and disconnect electrical connectors.

(2) Remove safety wire and four mounting bolts.

(3) Remove actuator by tilting bottom forward and slipping chain off sprocket.

(4) Remove safety wire, allen screw, and washer holding sprocket to actuator.

(5) Remove sprocket.

B. Installation – Actuator

(1) Position sprocket on new actuator and install washer, allen screw, and safety wire.

(2) Slip chain over sprocket. Position actuator. Turn wheels to align. Install bolts.

(3) Measure chain slack at 0.06 to 0.50 inches.

(4) Install washers (shims) between actuator and mounts AN960-416, AN960-416L, and AN960-416D. Put one washer on each mount bolt. All four shims are the same thickness.

(5) To install shims, loosen bolts. Remove bolts one at a time.

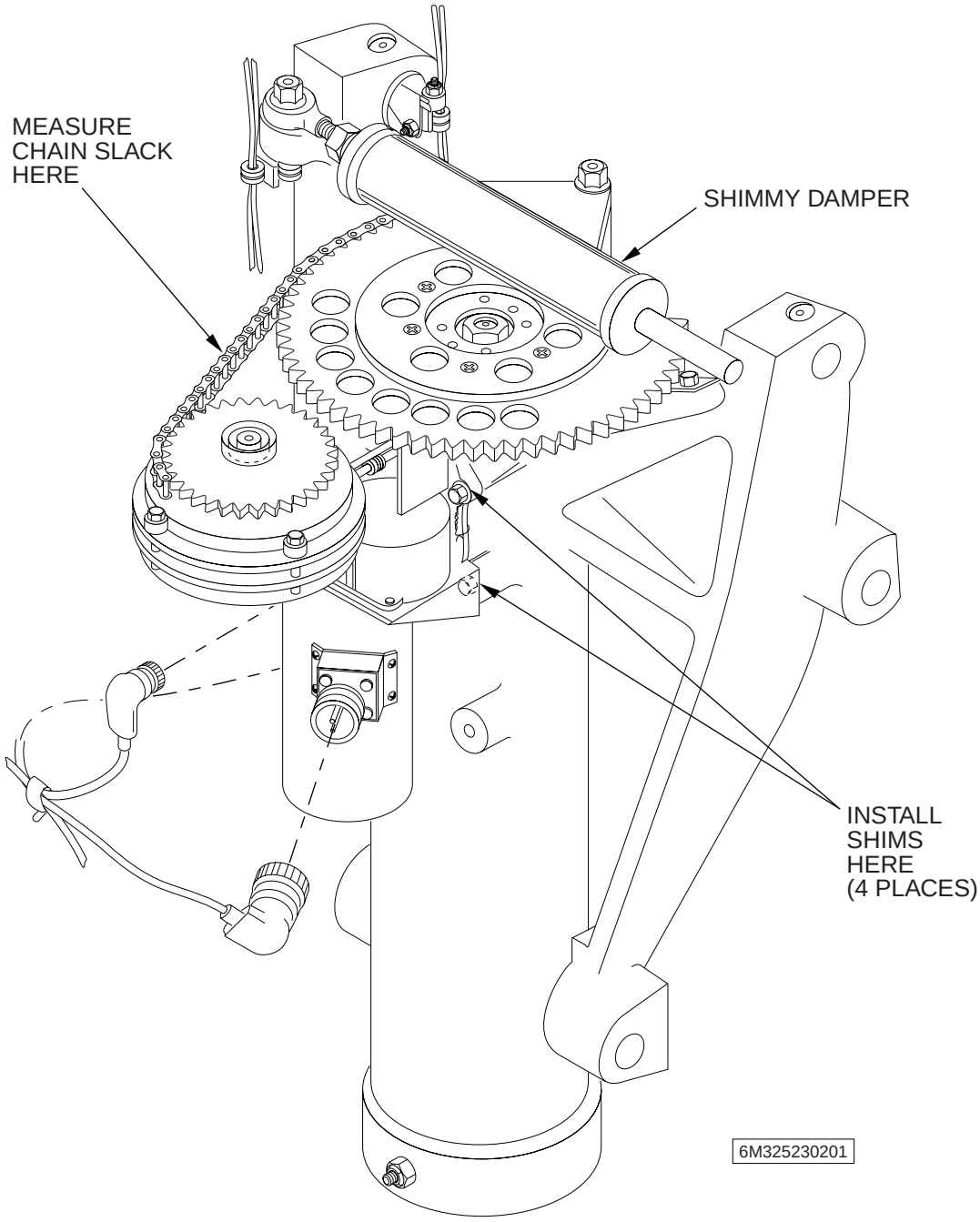
(6) Install shim. Install bolt.

NOTE: Top two bolts to be NAS1304-7H and AN960-416L washer under head. Bottom right bolt through clamp to be NAS1304-5H and AN960-416L washer under head. Bottom left bolt to be NAS1304-4H and AN960-416L washer under head.

(7) Tighten bolts and safety wire.

(8) Connect electrical connectors and safety wire.

(9) Perform operational check.



Electric Steering Actuator
Figure 201

C. Removal – Follow-Up Pot Assembly

(1) For aircraft T245, 249-257; AT003E, 025-042; and TC211EEEE-221 cut six wires to pots at splices. For aircraft T258-271; AT043-053; TC222-226 disconnect connector on right side of wheel well (see inset A, Figure 202). For aircraft T272-281 (Exc 276), AT054-061; and TC227-244 (Exc 228 and 238) disconnect connector right upper rear corner of wheel well (see inset B on Figure 202). Remove pins from connector.

(2) Remove ties and clamps as necessary.

(3) Install tow bar. Turn gear to clear shimmy damper mount from head of large bolt in center of pot bracket and remove bolt.

(4) Turn gear to limit. Remove screw and rod end from pot arm.

(5) Turn gear to other limit and remove rod end.

(6) Remove bolts on each end of pot bracket.

(7) Remove pot bracket.

D. Installation – Follow-Up Pot Assembly

(1) Position bracket and install end bolts.

(2) With gear turned to limit, install rod end and screw on pot arm.

(3) Hold other rod so it does not catch underneath bracket. Turn gear to other limit.

(4) Install rod end on pot arm.

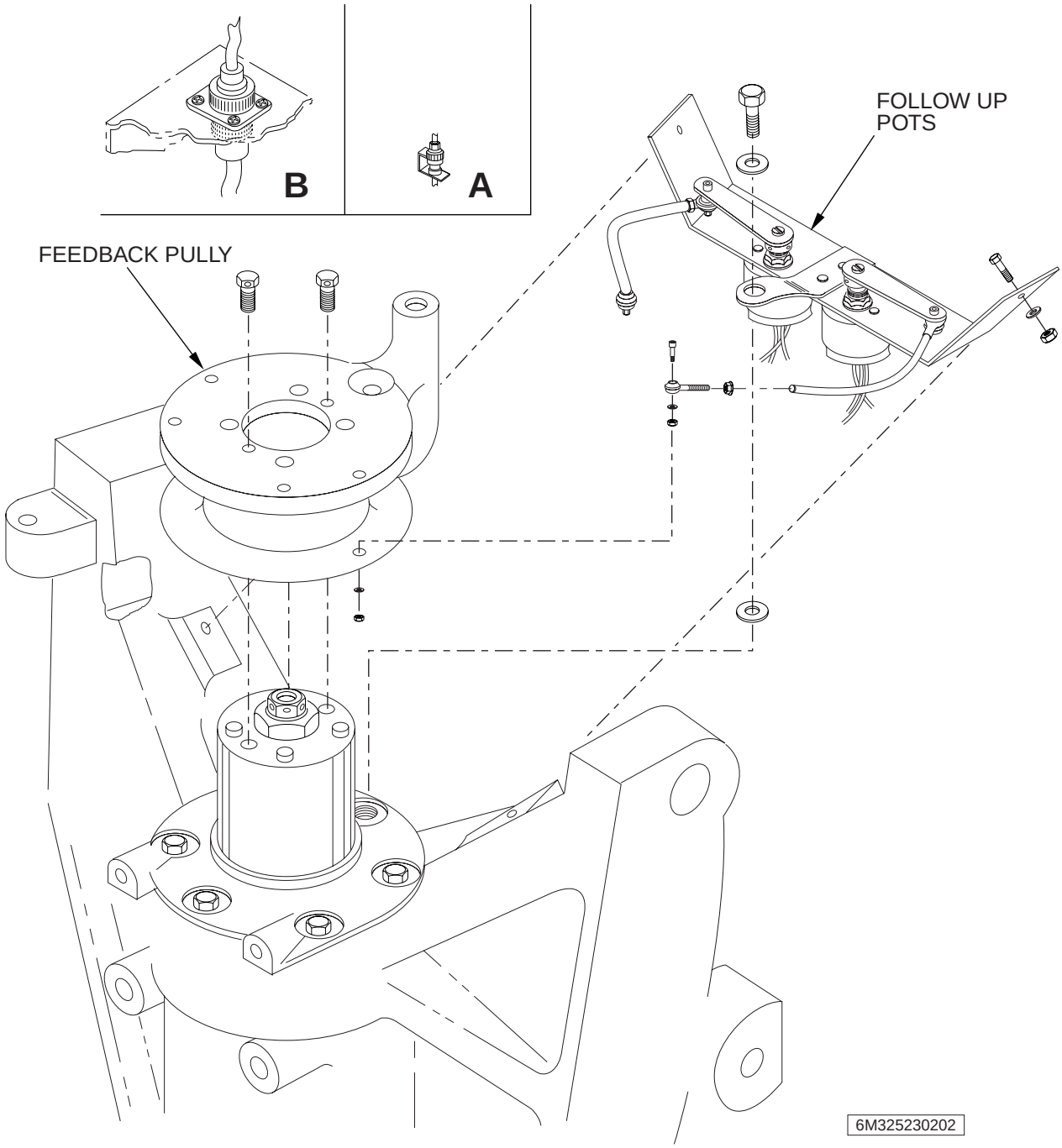
(5) Install large bolt in center of bracket and torque to 220 to 300 inch-pounds (24.86 to 33.90 nm). Shim washers if necessary so as not to warp bracket.

(6) Splice wires or install pins in connector and/or connect connector.

(7) Tie and clamp as necessary.

(8) Null pots (see Adjustment/Test – Nose Gear Follow-Up Pots).

(9) Perform operational check.



Actuator and Follow-Up
Pot Assembly
Figure 202

3. Adjustment/Test – Electric Actuator and Follow-Up System

A. Adjustment – Follow-Up Pot

(1) Two basic types of nullmeters are available. One uses batteries or 115 VAC, 60 Hz, and a DC power supply to produce a small DC current used by a meter similar to an ohmmeter to measure the resistance of the pots. Another type of tester uses voltage from the aircraft and the steering system itself and the actual delta voltage from the pots is read on two voltmeters. This type of tester can also be used to measure the actual centering drive voltage the actuator clutches are receiving. The drive voltage is set by adjusting the nose gear follow-up pots. Before this is done it must be verified that the rudder pedal pots and the follow-up pots are nulled.

(2) When using the voltmeter type tester, voltage at the wiper to ground of each pot should be 12 volts with nose gear centered. Connecting a two bar and turning the gear full left, both pots should read approximately 16 volts and full right approximately eight volts. Some testers have the capability of measuring the tracking voltage of the pots. This can be done by measuring with a voltmeter from wiper to wiper of each pot. If pots are tracking each other correctly, no voltage will be read. Pots are adjusted using the same procedure as the nullmeter type tester step (e) with electrical power on the aircraft and the steering system. See 32-SERVC in this chapter for more information on testers.

NOTE: The following procedures are for the ohmmeter type nullmeter.

(a) Disconnect connector (upper right rear corner of wheel well) or disconnect amplifier connector when using amplifier plug test harness.

(b) Install nullmeter test plug.

(c) Center nose gear.

(d) Nullmeter – Some nullmeters have a green band at center scale. On these meters, pots should be adjusted to center of green band.

1 Sensitivity adjustment – high.

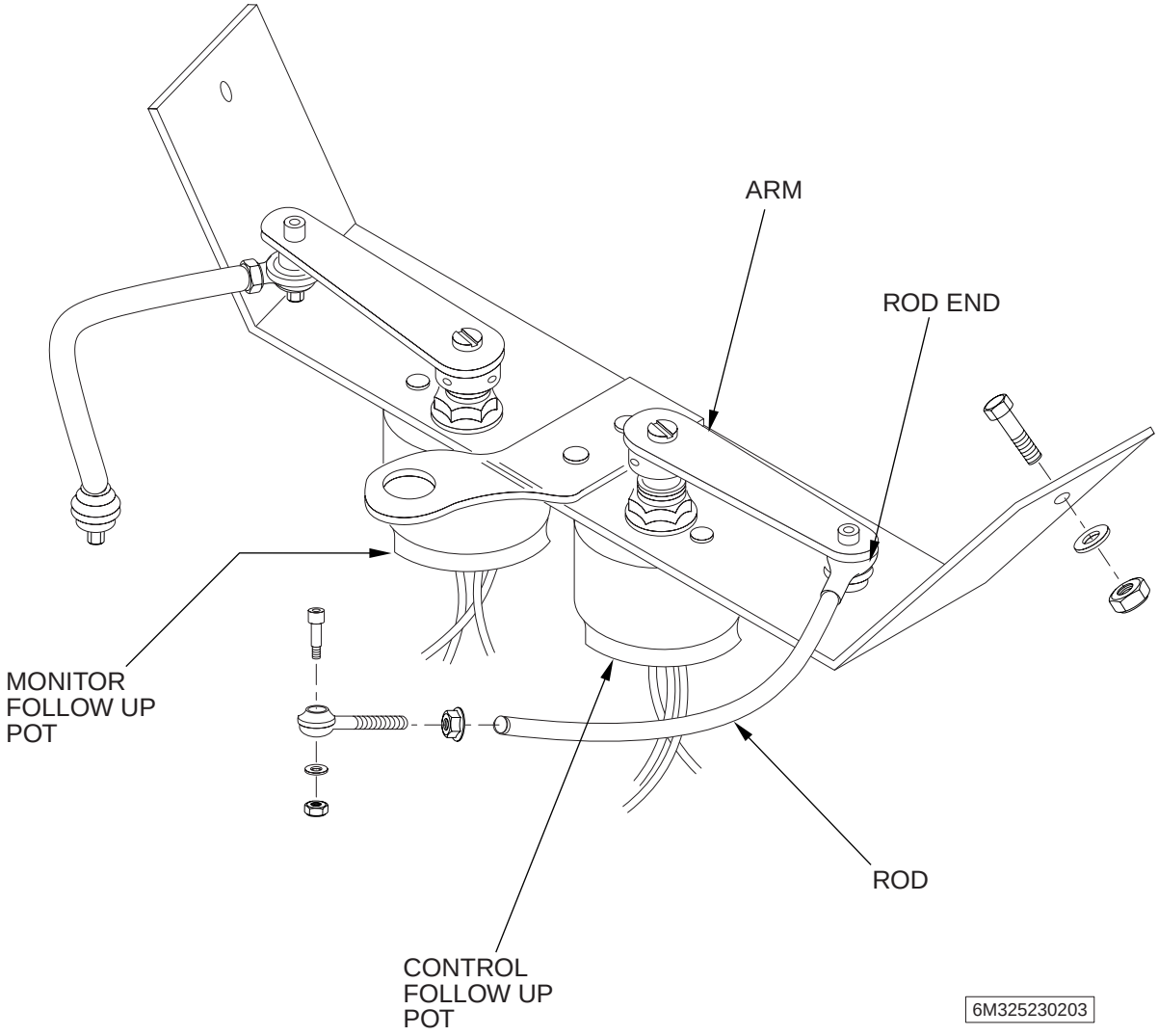
2 Select switch – nose gear – L control.

3 If needle is not in center of scale, adjustment is required.

4 Select switch – nose gear – R control.

5 If needle is not in center of scale, adjustment is required.

NOTE: Moving nose gear slightly to the left or right, nullmeter's needle should move left or right respectively. If needle does not move, check for broken wires.



Follow-Up Pots
Figure 203

(e) To adjust pots.

1 Install tow bar and turn gear in direction to bring left or right rod end and pot arm in position for easy access to screw through arm and rod end.

2 Loosen jam nuts on rod.

3 Remove screw and nut. Adjust as needed.

4 Install screw and nut.

(f) Center gear. Repeat step 3.A.(2)(d).

(g) Tighten jam nuts.

(h) Remove test plug and reconnect connector.

(i) Perform operational check.

B. Adjustment – Centering Drive Voltage

NOTE: Be sure rudder pedal pots and nose gear pots are nulled before following this procedure. When centering rudder pedals, pull both rudder pedals back to be sure pilot leg length adjustments are the same.

(1) Center rudder pedals.

(2) Install tow bar and center nose gear.

(3) Remove clutch connector (upper three pin connector) on actuator.

(4) Obtain a voltmeter with an accurate low scale one to five volts.

(5) Turn battery switches ON.

(6) Turn nose steering switch ON.

(7) Check pins B and C to ground. Readings should be approximately one volt and both the same.

(8) Adjust pots using same procedure as step (e) of nullmeter pot adjustment procedure.

(9) Center nose and check readings.

(10) Turn nose steering switch OFF.

(11) Turn battery switches OFF.

(10) Reconnect connector and safety wire.

(11) Perform operational check.

HYDRAULIC ACTUATOR AND FOLLOW-UP SYSTEM – DESCRIPTION AND OPERATION

1. General

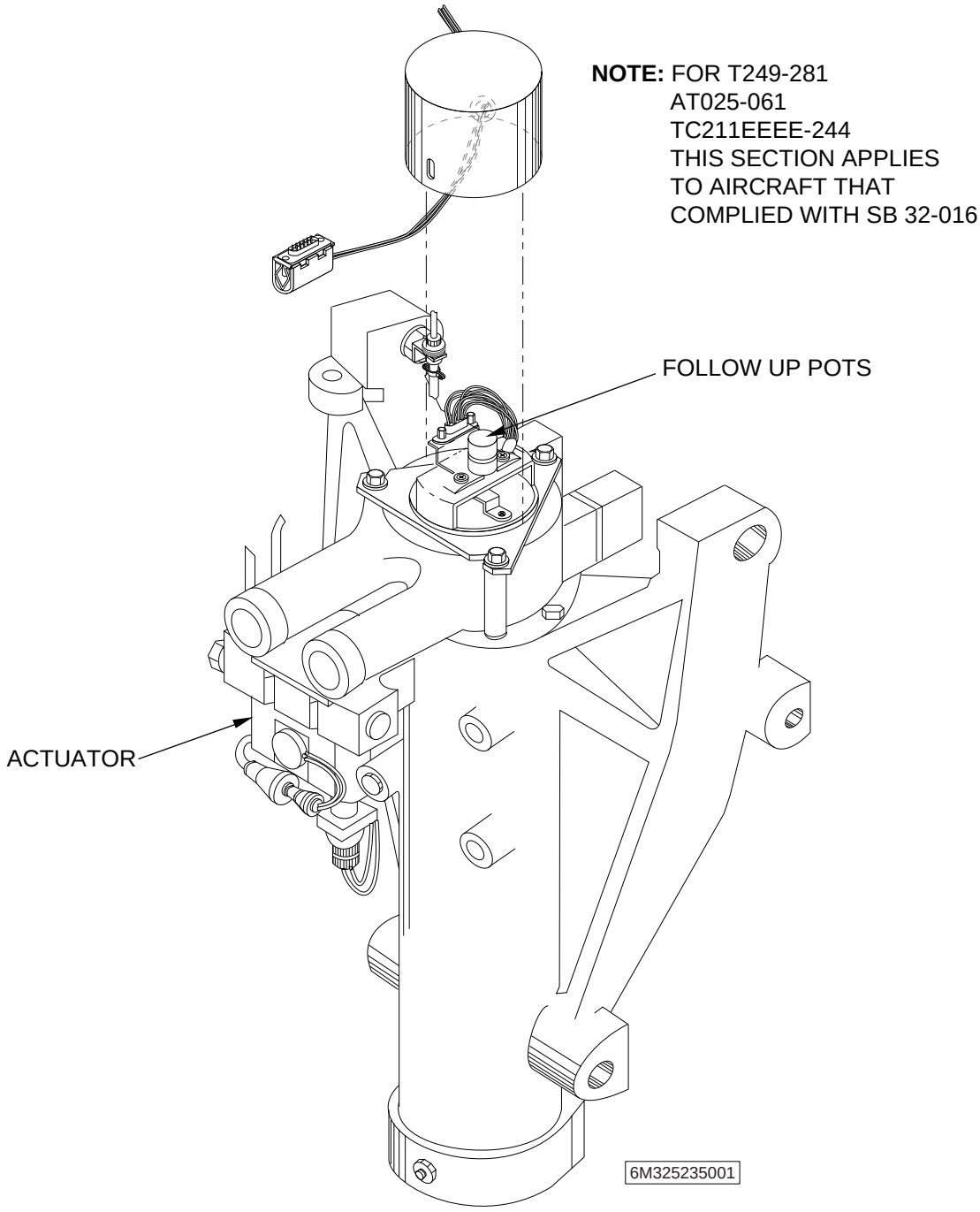
A. The actuator houses a solenoid valve, two servo valves, and two pistons connected to a rack and pinion gear arrangement. Hydraulic pressure from the normal gear down line of the left landing gear actuator is allowed to pass to servo valves by the solenoid valve. The servo valves are controlled by voltage from the amplifier and fault protection box. The servo valves meter hydraulic pressure to pistons. The pistons are connected to two racks that turn the pinion gear. The pinion gear is attached directly to the inner cylinder which extends through the main body of the nose gear to the wheels.

B. The follow-up system consists of control and monitor pot piggyback mounted on a triangular bracket. A single shaft turns both pot wipers. A bracket on the shaft is attached to the actuator pinion gear. The same bolts that attach the actuator housing to the nose gear also attach the triangular bracket.

C. When the nose gear turns, the shafts of the follow-up pots rotate. This causes the wiper within the pots to pass across the resistance winding. A voltage is applied across the resistance winding of each of the pots by the amplifier and fault protection box. The amplifier and fault protection box receives a delta voltage as the wiper passes across the resistance windings. This delta voltage is used to reference nose gear position.

D. The amplifier and fault protection box uses the delta voltage from the control pot to drive the actuator and the delta voltage from the monitor pot for fault protection. Proper operation of the steering system depends on the proper adjustment of these pots. When the pots are adjusted properly they are said to be nulled; that is with the nose gear centered, both pot wipers are at the same place on their respective resistance windings.

HYDRAULIC ACTUATOR AND FOLLOW-UP SYSTEM – DESCRIPTION AND OPERATION



Hydraulic Actuator and Follow-Up Pots
Figure 1

HYDRAULIC ACTUATOR AND FOLLOW-UP SYSTEM – FAULT ISOLATION

1. Troubleshooting – Hydrauliic Actuator and Follow-Up System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Steering system inoperative. Annunciator panel light is illuminated		PRELIMINARY Connect APU to aircraft, Refer to Wiring Diagram Manual Chapter 32-50-00. Turn battery switches on. Place NOSE STEERING switch to ARMED or hold PRESS buton down on tiller wheel.
	Actuator or wiring defective	Remove solenoid valve connector on actuator (larger two pin connector), check for 28 VDC present between pins A and B. If 28 VDC is not present, check wiring to relay (K20) and check ground on pin B. If 28 VDC is present at A and B, remove servo valve connector or check for ground present at pins A and D. If power relay is closed, a voltage should be present from pins B and C to pins A and D of servo valve connector. If voltage is present at both actuator connectors, actuator is defective
Steering pulls left or right or makes tighter turn in one direction from the other, or aircraft turns in direction TEST switch is pushed	Centering voltage not set properly	Adjust centering voltage.
	Plugged filters in actuator	Clean or replace filters.
	Diodes defective at actuator wire harness	Check and rerplace diodes.
Aircraft steers left or right and faults. Annunciator panel light blinks	Wires broken to follow-up pots	Check wiring to pots.
	Diodes defective at actuator wire harness	Check and replace diodes.
Steering sluggish or over reacts	Plugged filters in actuator	Clean or replace filters.

HYDRAULIC ACTUATOR AND FOLLOW-UP SYSTEM – MAINTENANCE PRACTICES

1. Removal/Installation – Hydraulic Actuator and Follow-Up System

A. Removal – Follow-up Pot Assembly

(1) For aircraft T258-271, AT043-053, TC222-226, disconnect connector on right side of wheel well (see insert A, Figure 201). For aircraft T272-291, AT054-070. and TC227-261, disconnect connector right upper corner of wheel well (see insert B, Figure 201). Remove pins from connector. For aircraft T299-999, AT071-999, and TC262-999, remove pot cover on top of actuator and disconnect connector.

(2) Remove ties and clamps as necessary and remove pot cover on top of actuator.

(3) Remove no. 8 screw (right) and no. 6 screw (left) attaching pot shaft bracket to actuator.

(4) Remove large bolts attaching actuator and pot assembly to gear.

NOTE: Notice position of hose clamp brackets and hardware for installation of new pot bracket.

(5) Remove pot bracket assembly.

B. Installation – Follow-up Assembly

(1) Position pot bracket and install large bolts with shims (AN960-616L washers) between pot bracket and actuator, and washers and hose clamp brackets.

(2) Torque bolts 220 to 300 inch-pounds (24.86 to 33.90 nm).

(3) Install no. 8 screw (right) and no. 6 screw (left) in pots shaft bracket.

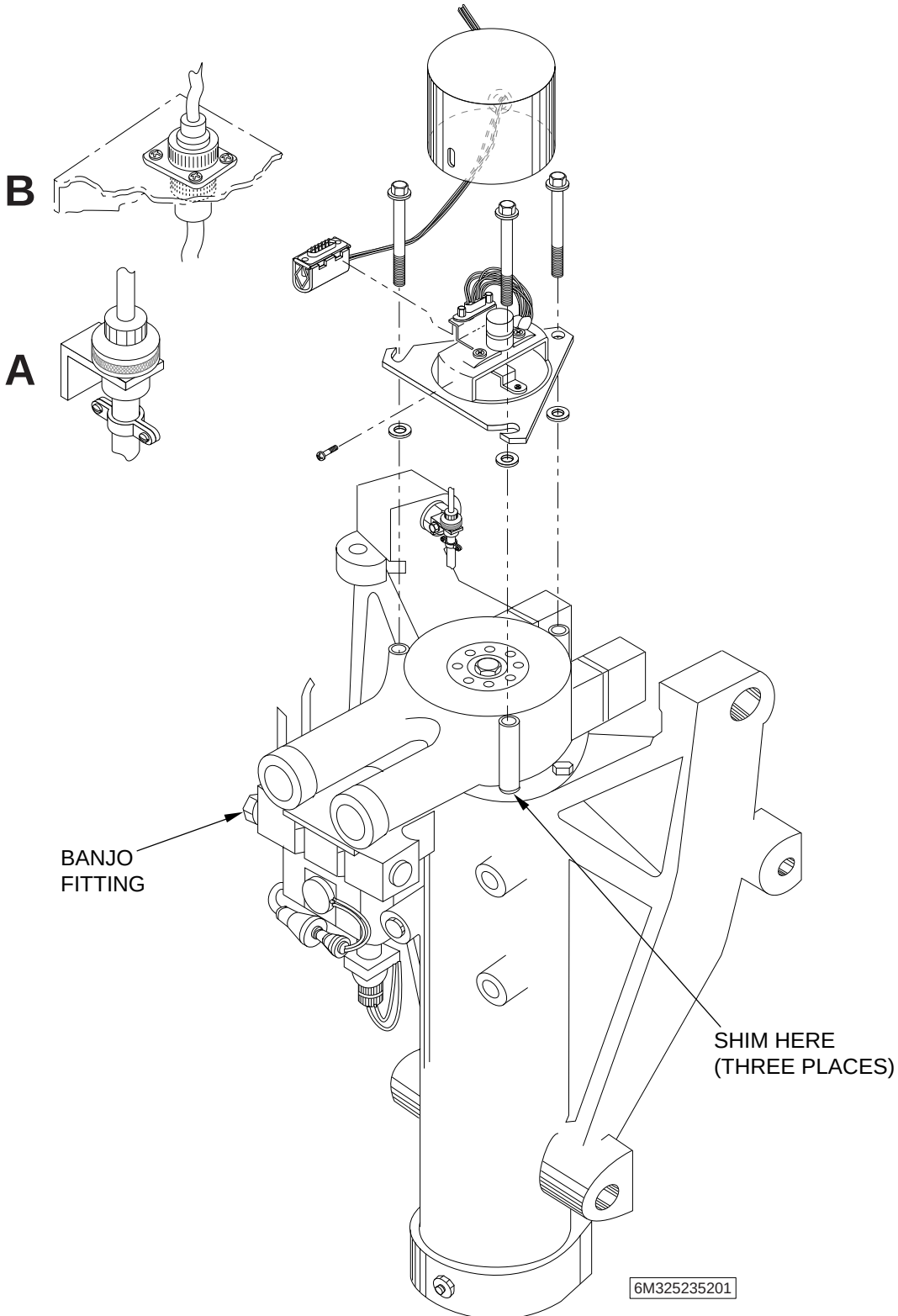
(4) Splice wires or install pins in connector and/or reconnect connector.

(5) Tie and clamp wires as necessary.

(6) Null pots. See nose gear follow-up pots adjustment and test.

(7) Install cover.

(8) Perform operational check (see 32-52-00, this chapter).



Actuator and Follow-Up Pots
Figure 201

C. Removal – Nose Steering Actuator (Pre SB32-037/039) (Figure 202)

NOTE: This procedure applies to earlier aircraft that are not equipped with a cannister filter and anti-cavitation system, and on which SB32-037 or SB32-039 have not been accomplished.

(1) Bleed off hydraulic pressure in accumulator by moving flap handle up and down several times with electrical power on (battery switches on) until hydraulic pressure indicator reads zero (0). Close hydraulic shutoffs.

(2) Bleed off air pressure on top of hydraulic reservoir by removing spring lock and loosening filler cap on outboard side of left nacelle, allowing air pressure to escape. Tighten cap and secure with spring lock.

(3) Remove and cap line on return line at banjo fitting (right forward side of actuator).

(4) Remove and cap line on pressure line at restrictor fitting (aft of banjo fitting).

(5) Disconnect electrical connectors (P49 and P50) on actuator.

(6) Remove screws and cover of pot bracket assembly on top of actuator.

(7) Remove screw No. 8 (right) and screw No. 6 (left) that attaches pot shaft bracket to actuator.

(8) Remove large bolts holding actuator and pot bracket to top of gear.

NOTE: Observe position of hose clamp brackets and hardware for installation of new actuator.

(9) Lift pot bracket from top of actuator and stow to facilitate removal of actuator.

CAUTION: DO NOT REMOVE CAP IN CENTER OF INNER SLEEVE (TO PREVENT FLUID UNDER HIGH PRESSURE FROM LEAKING).

(10) Remove safety wire and drive bolts attaching actuator drive to inner sleeve of gear.(11) Remove actuator.

(12) Remove banjo and restrictor fittings from actuator.

D. Installation – Nose Steering Actuator Pre SB32-037/039 (Figure 202)

(1) Inspect O-rings on banjo fitting. Replace if necessary and install fitting on new actuator.

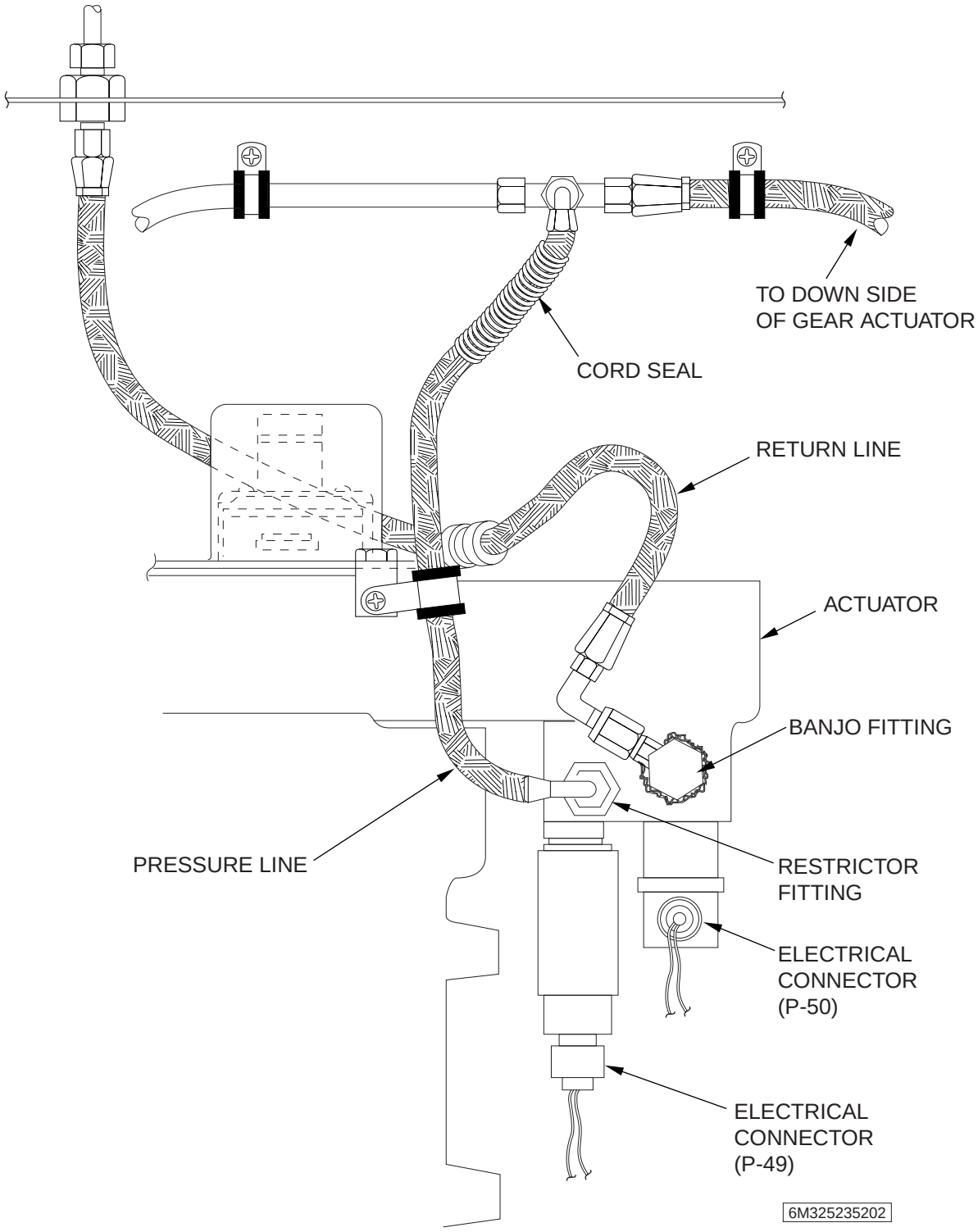
(2) Install restrictor fitting on new actuator (fitting should be brown for AT and TC, gold for T only).

(3) Apply a thin coating of landing gear grease (MIL-G-23827) to exposed end of inner cylinder.

(4) Position actuator over inner sleeve on top of strut.

(5) Install drive bolts and tighten down until pins extend through actuator drive.

(6) If actuator housing mounting pads are not seated on top of strut, remove drive bolts and tap actuator up.



Nose Steering Actuator
Installation Pre SB32-037/039
Figure 202

(7) Install one shim under each actuator housing mounting pad and strut. Tap actuator down and insert large bolts to align shims.

(8) Install drive bolts, torque 50 to 70 inch-pounds (5.65 to 7.91 nm), and safety wire to cap in center of inner sleeve.

(9) Install No. 8 screw (right) and No. 6 screw (left) attaching pot shaft bracket to actuator.

(10) Remove large bolts and position follow-up pot assembly on top of actuator.

(11) Install shims (AN960-616L washers) between pot assembly and actuator.

(12) Install large bolts with washers, hose clamp brackets, and torque 220 to 300 inch-pounds (24.86 to 33.90 nm).

(13) Remove cap and install pressure line on restrictor fitting.

NOTE: Pressure line is connected to a Tee fitting on the landing gear actuator down line.

(14) Remove cap and install return line on banjo fitting and safety wire banjo fitting.

NOTE: Return line connects to a fitting in the upper left rear corner of wheel well.

(15) Connect and safety wire two electrical connectors (P49 and P50) on actuator.

(16) Check null of pots in accordance with 2.B.(5).

(17) Install cover on pot assembly.

(18) Connect hydraulic cart and fill hydraulic reservoir to full mark. (Make sure flaps are up).

(19) Remove B nut and line on pack pressurization line at elbow fitting in left wheel well inboard keelson (first fitting aft of main bleed air line) and apply approximately 30 psi (2.11 Kg/cm^2) of air pressure to B nut end of line.

(20) Disconnect air line and reinstall B nut and line on elbow fitting. Torque B nut 40 to 65 inch-pounds (4.52 to 7.34 nm).

(21) Replace B nut and line on elbow fitting.

(22) Connect GPU to aircraft.

(23) Turn battery switches on.

(24) Apply hydraulic pressure with hydraulic cart.

(25) Move nose steering armed switch to ARM.

(26) Depress and hold nose steering PARK button and power lever switch. (on those aircraft equipped with tiller wheel, depress PRESS button and hold).

(27) Operate system through to left and right limits approximately six times To bleed the actuator. If necessary, loosen B nut on banjo fitting and let fluid flow until free of air bubbles and retighten B nut.

(28) Check for hydraulic leaks.

(29) Perform operational check (see 32-52-00, this chapter).

(30) Fill hydraulic reservoir to full mark with flaps up and disconnect hydraulic cart.

(31) Disconnect GPU.

E. Removal – Nose Steering Actuator (Post SB32-037/039) (Figure 203)

NOTE: This procedure applies to all aircraft retrofitted (SB32-037, or SB32-039 to provide an inlet cannister type filter in the hydraulic pressure line to the actuator and an anti-cavitation system on the actuator cylinders that prevents shimmy.

CAUTION: IF MAINTENANCE IS REQUIRED ON THE NOSE GEAR STEERING UNIT 35050 ARMING VALVE ASSEMBLY, THE 35021 MANIFOLD MUST ALSO BE REPLACED. REPLACEMENT OF THE ARMING VALVE, WITHOUT CHANGING THE MANIFOLD, CAN RESULT IN SYSTEM POPPET BYPASS LEAKAGE CAUSING A SLOW, UNCOMMANDED TRANSITION FROM CASTOR MODE TO STEERING MODE. THE ARMING VALVE ASSEMBLY IS NOT FIELD REPLACEABLE.

(1) Bleed off hydraulic pressure in accumulator by moving flap handle up and down several times with electrical power on (battery switches on) until hydraulic pressure indicator reads zero (0). Close hydraulic shutoffs.

(2) Bleed off air pressure on top of hydraulic reservoir by removing spring lock and loosening filler cap on outboard side of left nacelle allowing air pressure to escape. Tighten cap and secure with spring lock.

(3) Remove and cap return line (item 13) on left forward side of actuator.

(4) Remove and cap pressure line (item 1) on right side of actuator.

(5) Disconnect electrical connectors (P49 and P50) on actuator.

(6) Remove screws and potentiometer cover (item 10) on top of actuator.

(7) Remove screws attaching potentiometer shaft bracket to actuator.

(8) Remove large bolts (item 11) securing actuator (item 19) and potentiometer bracket to top of gear.

(9) Lift potentiometer bracket from top of actuator and stow to facilitate removal of actuator.

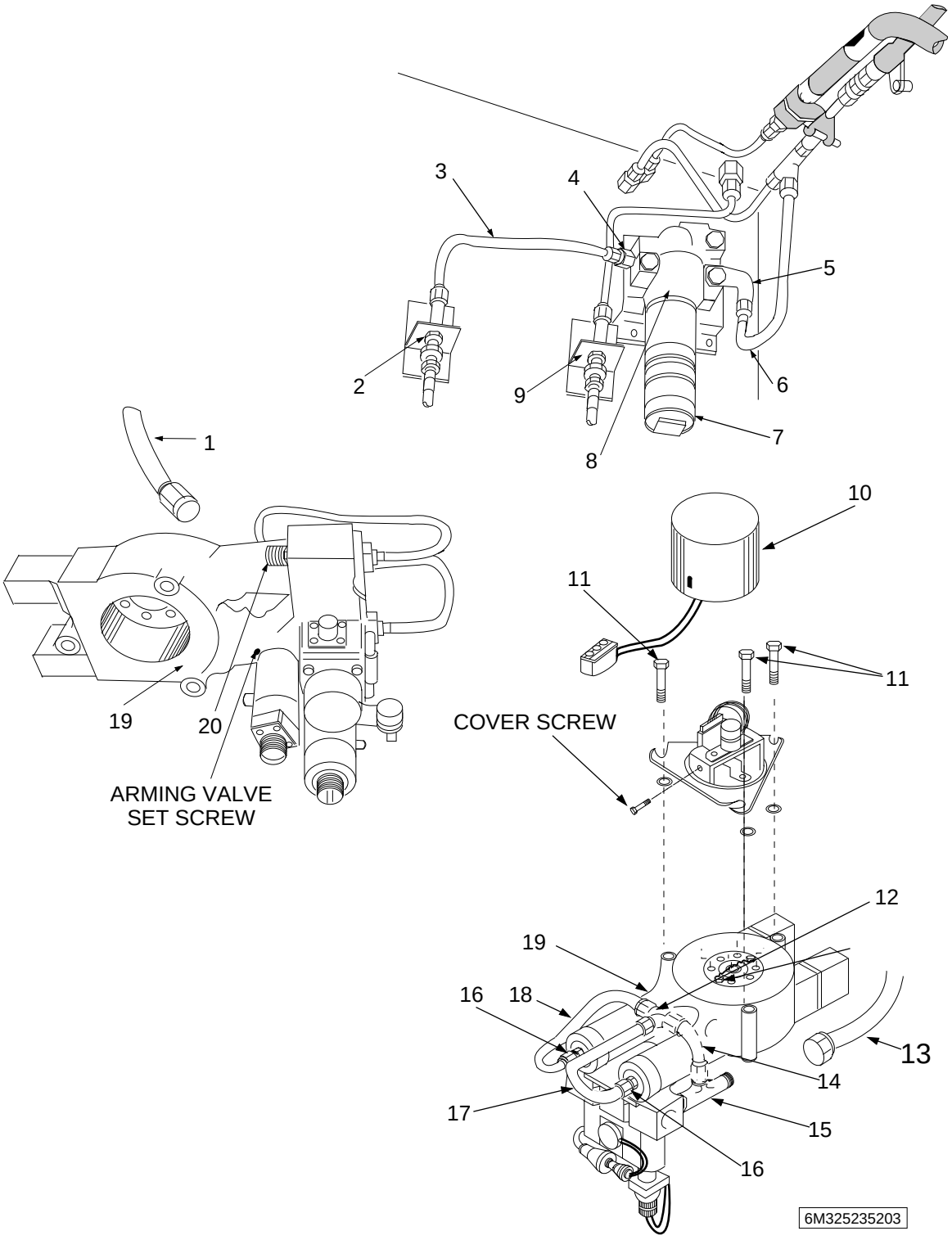
CAUTION: TO PREVENT FLUID UNDER HIGH PRESSURE FROM LEAKING, DO NOT REMOVE CAP IN CENTER OF INNER SLEEVE.

(10) Remove safety wire and drive bolts attaching actuator drive to inner sleeve of gear.

(11) Remove actuator.

(12) Remove tube (item 14), tube (item 17), and tube (item 18).

(13) Remove Tee (item 15). Install protective cover on opening in actuator.



Nose Steering Actuator
Installation (Post SB32-037/039)
Figure 203

(14) Remove check valves (item 16). Install protective cover on openings.

(15) Remove union (item 20) from pressure port. Install protective cover on open port.

F. Installation – Nose Steering Actuator (Post SB32-037/039) (Figure 203)

(1) Inspect O-rings on check valves (item 16). Replace if necessary and install check valves in new actuator.

(2) Install union (item 20) and O-ring into pressure port.

(3) Install Tee fitting (item 15), jam nut, and O-ring into return port of actuator.

(4) Install tube (item 14) between Tee fitting (item 15) and Tee fitting (item 12). Torque fittings 40 to 65 inch-pounds (4.52 to 7.34 nm).

(5) Install tube (item 18) between Tee fitting (item 12) and check valve. Torque end fitting 40 to 65 inch-pounds (4.52 to 7.34 nm).

(6) Install tube (item 17) between Tee fitting (item 12) and check valve. Torque end fitting 40 to 65 inch-pounds (4.52 to 7.34 nm).

(7) Apply a thin coating of landing gear grease (MIL-G-23827) to exposed end of inner cylinder.

(8) Position actuator (item 19) over inner sleeve on top of strut.

(9) Install drive bolts and tighten until four pins extend through actuator drive.

(10) If actuator housing mounting pads are not seated on top of strut, remove drive bolts and tap actuator up.

(11) Install one shim under each actuator housing mounting pad and strut. Tap actuator down and insert three large bolts to align shims.

(12) Install drive bolts, torque 50 to 70 inch-pounds (5.65 to 7.91 nm), and safety wire to cap in center of inner sleeve.

(13) Install screws attaching potentiometer shaft bracket to actuator.

(14) Position follow-up potentiometer assembly on top of actuator.

(15) Install shims (washers) between potentiometer assembly and actuator.

(16) Install large bolts (item 11) with washers and torque 220 to 300 inch-pounds (24.86 to 33.90 nm).

(17) Install pressure hose (item 1) union. Torque hose end fitting 70 to 120 inch-pounds (7.91 to 13.56 nm).

NOTE: Pressure line connects to union (item 9) at aft wheel well bulkhead.

(18) Install return line onto Tee fitting.

NOTE: Return line connects to union (item 2) at aft wheel well bulkhead.

(19) Connectl and safety wire electrical connectors (P49 and P50) on actuator.

(20) Check null of potentiometers.

(21) Install cover (item 10) over potentiometers.

(22) Connect hydraulic cart and fill hydraulic reservoir to full mark. Make sure flaps are up.

(23) Remove B nut on power pack pressurization line at elbow fitting in left wheel well inboard Keelson (first fitting aft of main bleed air line). Apply approximately 30 psi (2.11 Kg/cm²) air (head) pressure to power pack.

(24) Disconnect air line and reinstall B nut and line on elbow fitting. Torque B nut 40 to 65 inch-pounds (4.52 to 7.34 nm).

(25) Connect GPU to aircraft.

(26) Turn battery switches on.

(27) Apply hydraulic pressure with hydraulic cart.

(28) Move nose steering armed switch to ARM.

(29) Depress and hold nose steering PARK button and power lever switch.

(30) Operate system through to left and right limits approximately six times to bleed the actuator. If necessary, loosen return hose end fitting and let fluid flow until free of air bubbles. Torque hose end fitting 70 to 120 inch-pounds (7.91 to 13.56 nm).

(31) Check for hydraulic leaks.

(32) Perform operational check (see 32-50-00 this chapter).

(33) Fill hydraulic reservoir to full mark with flaps up and disconnect hydraulic cart.

(34) Disconnect GPU.

G. Removal – Nose Steering Filter Element (Figure 203)

NOTE: Removal/Installation procedures for the Nose Steering Filter Element are for inlet cannister type filter assemblies installed in the hydraulic pressure line as a result of compliance with SB32-037, and SB32-039. Replace the filter element P/N AC3255F-8Y14 whenever the by-pass indicator pin “pops” out or every 1,000 hours whichever occurs first. The "red" by-pass indicator pin must be manually reset after changing the filter element.

(1) Bleed off hydraulic pressure in accumulator by moving flap handle up and down several times with electrical power on (battery switches on) until hydraulic pressure indicator reads zero (0). Close hydraulic shutoffs.

(2) Bleed off air pressure on top of hydraulic reservoir by removing spring lock and loosening filler cap on outboard side of left nacelle allowing air pressure to escape. Tighten cap and secure with spring lock.

(3) Cut and remove safety wire from filter bowl (item 7).

(4) Unscrew filter bowl from filter head (item 8).

(5) Remove snap ring from upper groove of filter bowl.

(6) Remove and discard filter element.

H. Installation – Nose Steering Filter Element (Figure 203)

(1) Check condition of O-ring at top of filter bowl. Replace as necessary.

(2) Insert new filter element into filter bowl. Install snap ring.

(3) Screw filter bowl with new element into filter head. Torque filter bowl 250 to 300 inch-pounds (28.25 to 33.90 nm).

(4) Safety wire filter bowl to filter head in accordance with MS33540.

I. Removal – Nose Gear Steering Actuator Manifold

CAUTION: IF MAINTENANCE IS REQUIRED ON THE NOSE GEAR STEERING ARMING VALVE, THE MANIFOLD AND VALVE MUST BE REPLACED AS AN ASSEMBLY. REPLACEMENT OF THE ARMING VALVE, WITHOUT CHANGING THE MANIFOLD, CAN RESULT IN SYSTEM POPPET BYPASS LEAKAGE CAUSING A SLOW, UNCOMMANDED TURNING OF THE NOSEWHEEL. THE ARMING VALVE ASSEMBLY IS NOT FIELD REPAIRABLE.

(1) Bleed off hydraulic pressure in accumulator by moving flap handle up and down several times with battery switches ON until hydraulic pressure indicator reads zero (0). Close hydraulic shutoffs.

(2) Bleed off air pressure on top of hydraulic reservoir by removing spring lock and loosening filler cap on outboard side of left nacelle allowing air pressure to escape. Tighten cap and secure with spring lock.

NOTE: The servo valves and pressure switch, attached to the manifold, may be replaced without manifold change out.

(3) Relieve strut pressure.

(4) Cut safety wire on electrical connectors and disconnect.

(5) Cover wheels and strut below steering actuator manifold.

(6) Cut safety wire on set screws securing solenoid valve. Remove set screws.

(7) Remove solenoid valve.

NOTE: Ensure O-rings are removed.

(8) Remove set screws on bottom of manifold.

(9) Remove set screws on front of manifold.

(10) Remove nose gear steering manifold.

J. Installation – Nose Gear Steering Actuator Manifold

(1) Install nose gear steering actuator manifold after inspecting the unit.

(2) Install set screws into the manifold.

(3) Install O-rings.

(4) Install solenoid turning it clockwise until it seats firmly against manifold.

(5) Install set screws and tighten.

(6) Replace O-rings and install solenoid valve.

(7) Install set screws, tighten, and safety wire.

(8) Connect electrical connectors and secure with safety wire.

(9) Service strut IAW Landing Gear Strut Replenishing, chapter 12.

(10) Service hydraulic reservoir.

(11) Unmask nose gear strut and remove gear down lock.

(12) Perform nose gear steering taxi check.

2. Adjustment/Test – Hydraulic Actuator and Follow-Up System

A. Follow-Up Pot Adjustment

(1) Two basic types of nullmeters are available. One uses batteries or 115 VAC, 60 Hz, and a DC power supply to produce a small DC current used by a meter similar to an ohmmeter to measure the resistance of the pots. Another type of tester uses voltage fro the aircraft and the steering system itself, and the actual delta voltage from the pots is read on two voltmeters. This type of tester can also be used to measure the actual centering drive voltage the actuator servo valves are receiving. The drive voltage is set by adjusting the nose gear follow-up pots simultaneously. Before this is done it must be verified that the rudder pedal pots and the follow-up pots are nulled.

(2) When using the voltmeter type tester, voltage at the wiper to ground of each pot should be 12 volts with the nose gear centered. Connecting a tow bar and turning the gear full left, both pots should read approximately 16 volts and full right approximately eight volts. Some testers have the capability of measuring the tracking voltage of the pots. This can be done by measuring with a voltmeter from wiper to wiper of each pot. If pots are tracking each other correctly no voltage will be read. Pots are adjusted using the same procedure as the nullmeter type tester, with electrical power on the aircraft and the steering system. See back of this chapter, 32-SERVC, for more information on testers.

B. Adjustment and Test of Nose Gear Follow-Up Pots

NOTE: The following procedures are for the ohmmeter type nullmeter.

(1) Remove connector in upper right corner of wheel well (see Figure 204) or remove amplifier connector when using amplifier plug test harness.

(2) Install nullmeter test plug.

(3) Center nose gear.

(4) Nullmeter – some nullmeters have a green band at center scale. On these meters, pots should be adjusted to center of green band.

(a) Sensitivity adjustment – high.

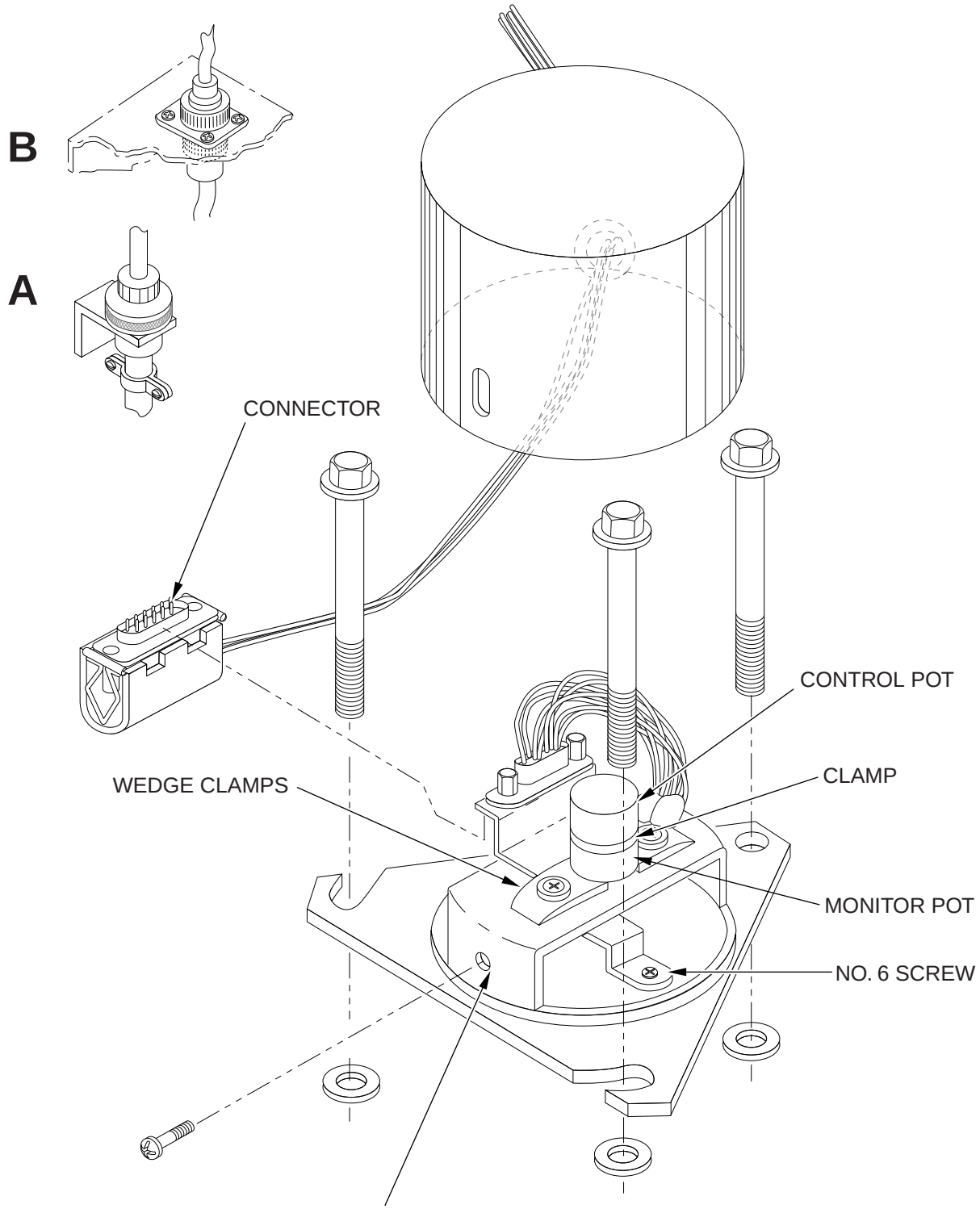
(b) Select switch – nose gear – L control (upper pot).

(c) If needle is not in center of scale, adjustment is required.

(d) Select switch – nose gear – R control (lower pot).

(e) If needle is not in center of scale, adjustment is required.

NOTE: Move nose wheels slightly to the left and right. Nullmeter needle should move left and right respectively. If needle does not move, check for broken wires.



NOTE: FOR ACCESS TO ALLEN SCREWS THAT HOLD POT SHAFT TO POT SHAFT BRACKETS, USE HOLES FORE AND AFT.

Nose Gear Follow-UP Pots
Figure 204

(5) To adjust pots:

(a) Remove screws in pot assembly cover on top of actuator. Remove cover.

(b) Loosen screws in clips holding pots to bracket.

(c) Nullmeter to R monitor (lower pot). Turn both pots simultaneously until needle is in center of scale. This adjusts lower pot.

(d) If upper pot needs adjustment, loosen screw in clamp at center of pots. Nullmeter to L control (upper pot). Turn upper pot until needle is in center of scale. Tighten screw in clamp.

(6) Center gear. Repeat step 2.B.(4).

(7) Position cover and install screws.

(8) Remove test plug and reconnect connector.

(9) Perform operational check.

NOTE: Pots should be adjusted slightly. If more adjustment is needed, loosen allen screw in pot shaft bracket and adjust pot shaft.

D. Centering Drive Voltage Adjustment

NOTE: Be sure rudder pedal pots or tiller wheel pots and nose gear pots are nulled before following this procedure. When centering rudder pedals, pull both rudder pedals back to be sure pilot leg length adjustments are the same.

(1) Center rudder pedals or tiller control.

(2) Center nose gear.

(3) Remove servo valve connector (four pin connector) on actuator.

(4) Obtain a voltmeter with an accurate low scale one to five volts.

(5) Remove pot assembly cover.

(6) Have an assistant turn battery switches on and depress power lever switch or tiller wheel press button.

(7) Loosen screws in wedge clamps at nose gear pots.

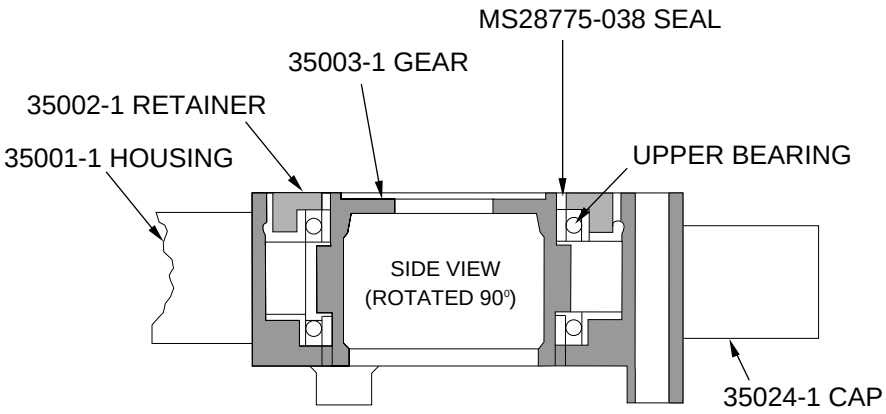
(8) Turn both pots simultaneously to obtain a reading of less than one volt on pins B and C to ground of servo valve connector. Both readings should be the same. Tighten screws in clamps.

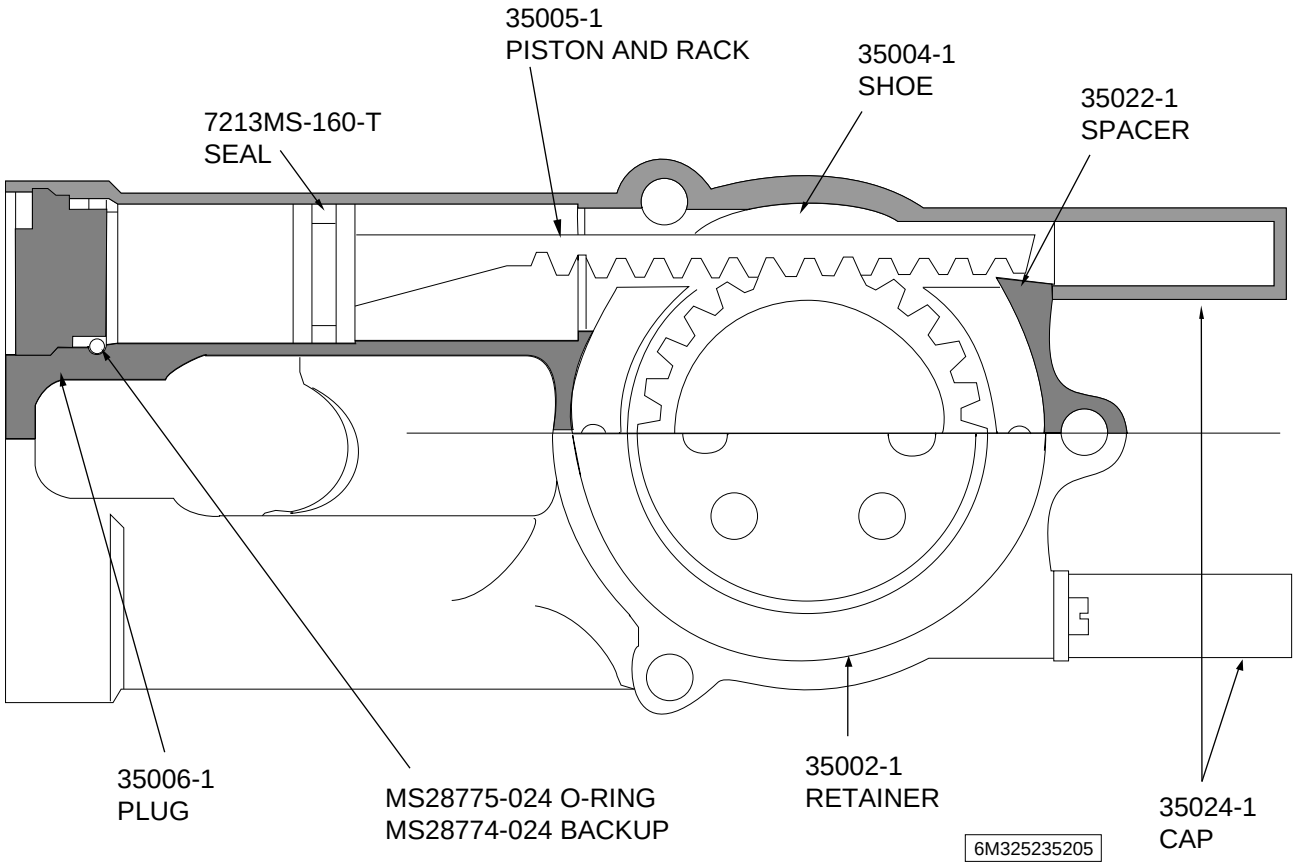
NOTE: Pots should be adjusted slightly. If more adjustment is needed, loosen allen screw in pot shaft bracket and adjust pot shaft.

(9) Battery switches OFF.

(10) Install cover on pot assembly.

(11) Install servo valve connector and safety wire.





Steering Actuator
Figure 205

3. Approved Repairs – Hydraulic Actuator and Follow-Up System

A. Nose Gear Steering Actuator Seal Replacement

(1) Remove follow-up pot assembly and actuator (see Removal – Actuator).

(2) Place actuator in suitable work area.

(3) Disassemble actuator.

NOTE: The manufacturer of the steering actuator (Airight Sales) has provided the following procedure for resealing the actuator.

NOTE: These parts to be reinstalled in their original positions.

(a) Loosen lockscrews and remove 35006-1 plug (2 places).

(b) Loosen lockscrew and remove the 35002-1 bearing retainer and upper pinion bearing.

(c) Remove bearing spacers 35022-1 and pinion gear 35003-1.

(d) Remove shoes, 35004-1 (2 places).

(e) Remove 35024-1 caps (2 places).

(f) Remove piston and rack 35005-1 (2 places).

(g) Remove piston seals and discard.

(h) Clean all parts.

(4) Reassemble as follows:

(a) Install seals (part no. 7213MS-160-T Greene, Tweed & Co.) on both pistons. The radius on the back-up rings are to mate with the internal radius on the T part of the seals.

(b) Lubricate cylinder bores and seals with Dow Corning No. 55M grease (MIL-G-4343).

(c) Lubricate the rack, pinion gear, bearings, and shoes with grease (MIL-G-10924C, or MIL-G-23827A).

(d) Install the piston and rack, being careful not to damage the new seals. Place pistons in neutral position by pushing them into the cylinders until the end of the rack is even with the outside edge of the housing.

(e) Install the shoes 35004-1 and the bearing spacers 35022-1.

(f) Install the pinion gear with the No. 8-32 screw thread and the No. 6-32 screw threads perpendicular to the centerline of the actuator, with the No. 8-32 screw thread on the right side (both pistons should be in neutral position).

(g) Install upper pinion bearing and 35002-1 bearing retainer; torque retainer to 200 inch-pounds and tighten lockscrew.

(h) Install 35024-1 caps (2 places).

(i) Lubricate O-rings with Dow Corning No. 55M grease (MIL-G-4343). Install 35006-1 plugs (2 places). If the actuator is a -1, screw the plugs in until the plugs are even with the housing, and then tighten lockscrews. If the actuator is a -2, screw the plugs in all the way, back out one half turn, and then tighten lockscrews.

(j) Add a dot of green paint near the nameplate to identify the modification.

(5) Install actuator and follow-up pot assembly (see Installation – Nose Steering Actuator-Post SB32-037/039, this section).

B. Actuator Filter Cleaning and Replacement

NOTE: Several problems with hydraulic nose wheel steering systems have been traced to plugged filters within the servo valve installed in the steering actuator. Symptoms of plugged filters are more steering authority in one direction or sluggish steering in general. Replacement filters are available through AIRIGHT SALES, 1544 S. Washington, Wichita, Kansas 67211.

A procedure to clean the filters in lieu of replacement is provided.

(1) Filter removal:

(a) Follow steps 1.E.(1) and (2) of Actuator Removal.

(b) Thoroughly clean exterior of Moog valve around filter retaining screws.

(c) Remove safety wire from filter plug.

(d) Remove plug.

(e) Carefully insert a 3/32 inch screw into the inner filter assembly. Turn the screw clockwise to engage inner filter. Remove the filter. Re-install outer part to prevent foreign material from entering the housing.

(2) Filter cleaning:

(a) Clean filter in an ultrsonic cleaner for a minimum of 30 minutes. The recommended cleaning fluid is Stoddard Solvent.

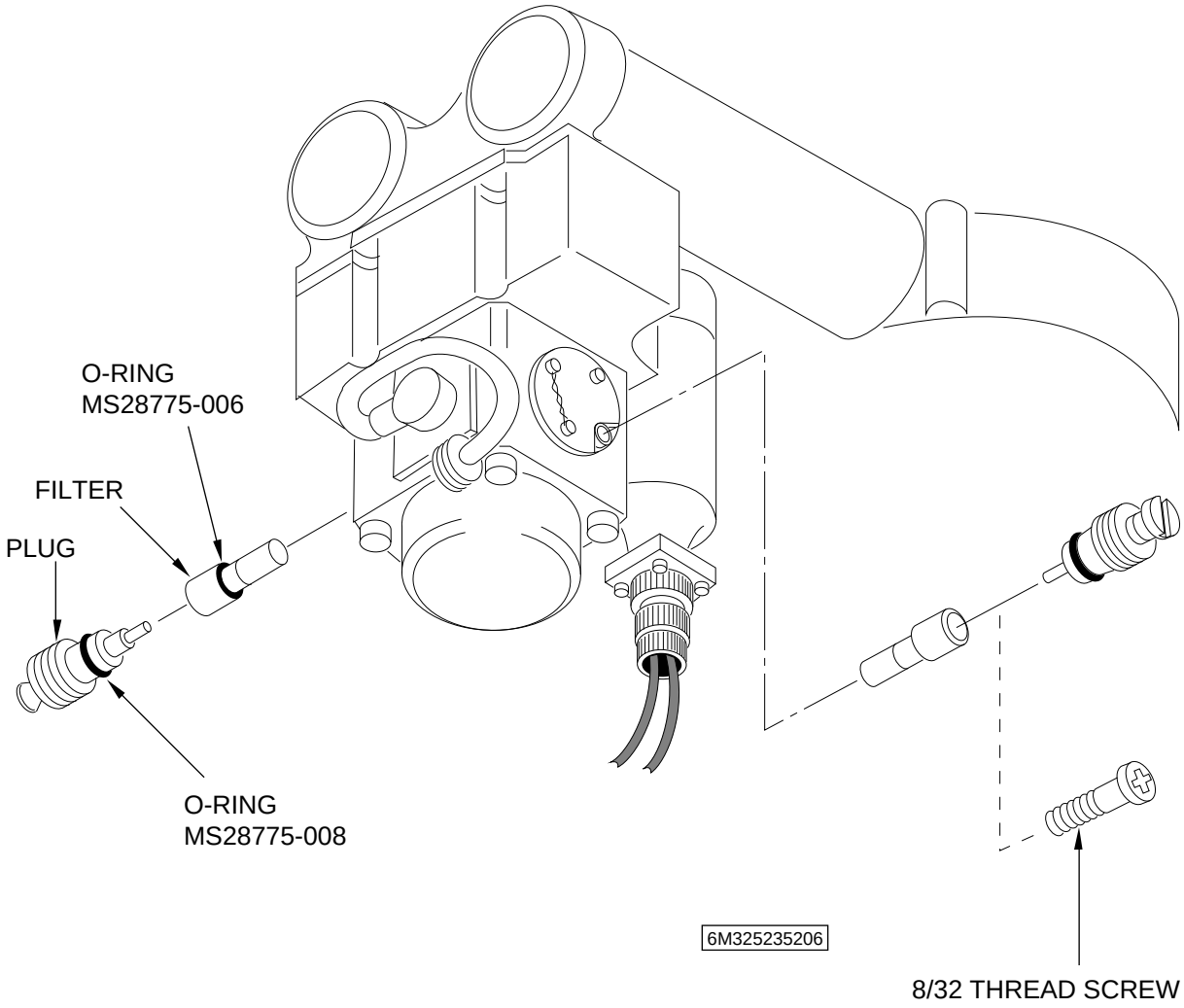
(b) Dry filter with shop air and check filter for free flow. Lubricate O-ring with MIL-H-83282.

(3) Installation:

(a) Install filters and plug. Safety the plugs.

(b) Follow steps 1.F.(18) through (30) of Actuator Installation.

FILTER PART NUMBERS: 350506 AIRIGHT SALES
020125047-12/15H MOOG



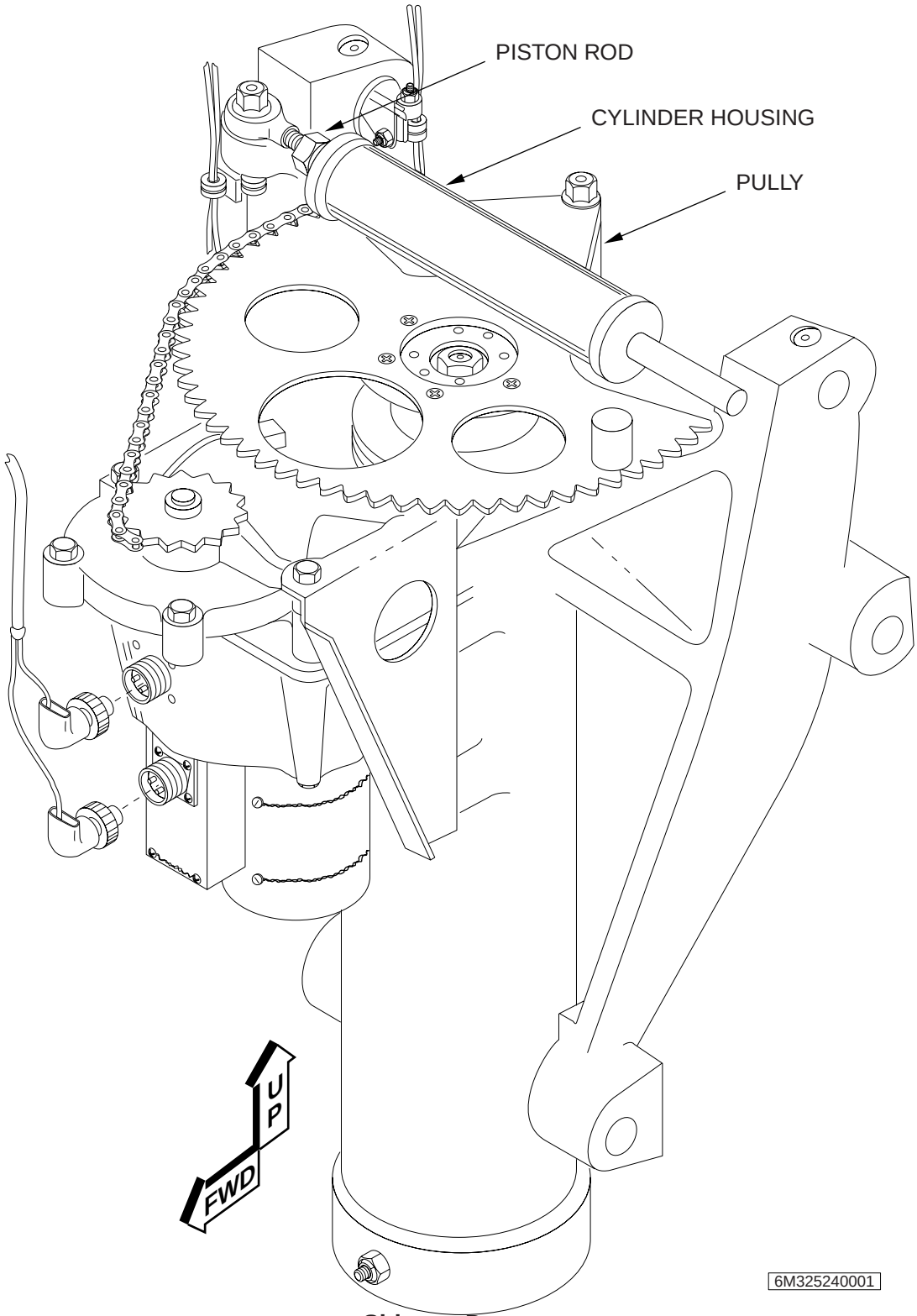
Filter / Plug Actuator
Figure 206

SHIMMY DAMPER – DESCRIPTION AND OPERATION

1. General

A. A nose landing gear shimmy damper unit is installed on production model SA226 series aircraft using an electric nose wheel steering actuator. (Not installed on those aircraft with hydraulic actuator (see Service Bulletin 32-016). The shimmy damper is located at the top of the nose gear strut (see Figure 1). The piston rod is attached to an ear on the stationary upper right section of the strut, and the cylinder is attached to the pulley on the steerable inner cylinder at the nose gear. See Component Maintenance Manual. Section 32-50-00 for disassembly, assembly, and servicing instructions.

SHIMMY DAMPER – DESCRIPTION AND OPERATION



Shimmy Damper
Figure 1

SHIMMY DAMPER – MAINTENANCE PRACTICES

1. Removal/Installation – Shimmy Damper

A. Removal – Shimmy Damper

(1) Remove cotter pin, nut, washer, and bolt attaching the piston rod of the damper unit to the stationary upper section of the nose landing gear strut assembly.

(2) Remove the cotter pin, nut, washer, and bolt attaching the cylinder housing of the damper unit to the pulley on top of the nose gear inner cylinder.

(3) Remove damper unit (see Component Maintenance Manual for disassembly, assembly, and servicing instructions).

B. Installation – Shimmy Damper

(1) Position damper unit into mounting location.

(2) Install bolt, washer, nut, and cotter pin securing the cylinder housing of the damper unit to the pulley.

(3) Install bolt, washer, nut, and cotter pin in securing piston rod of damper unit to upper section of nose gear strut assembly.

2. Adjustment/Test – Shimmy Damper

A. Test/Check

(1) Remove cotter pin, nut, and bolt in piston rod end.

(2) Check for damping operation and leakage.

(3) Replace shimmy damper if leaking or not damping.

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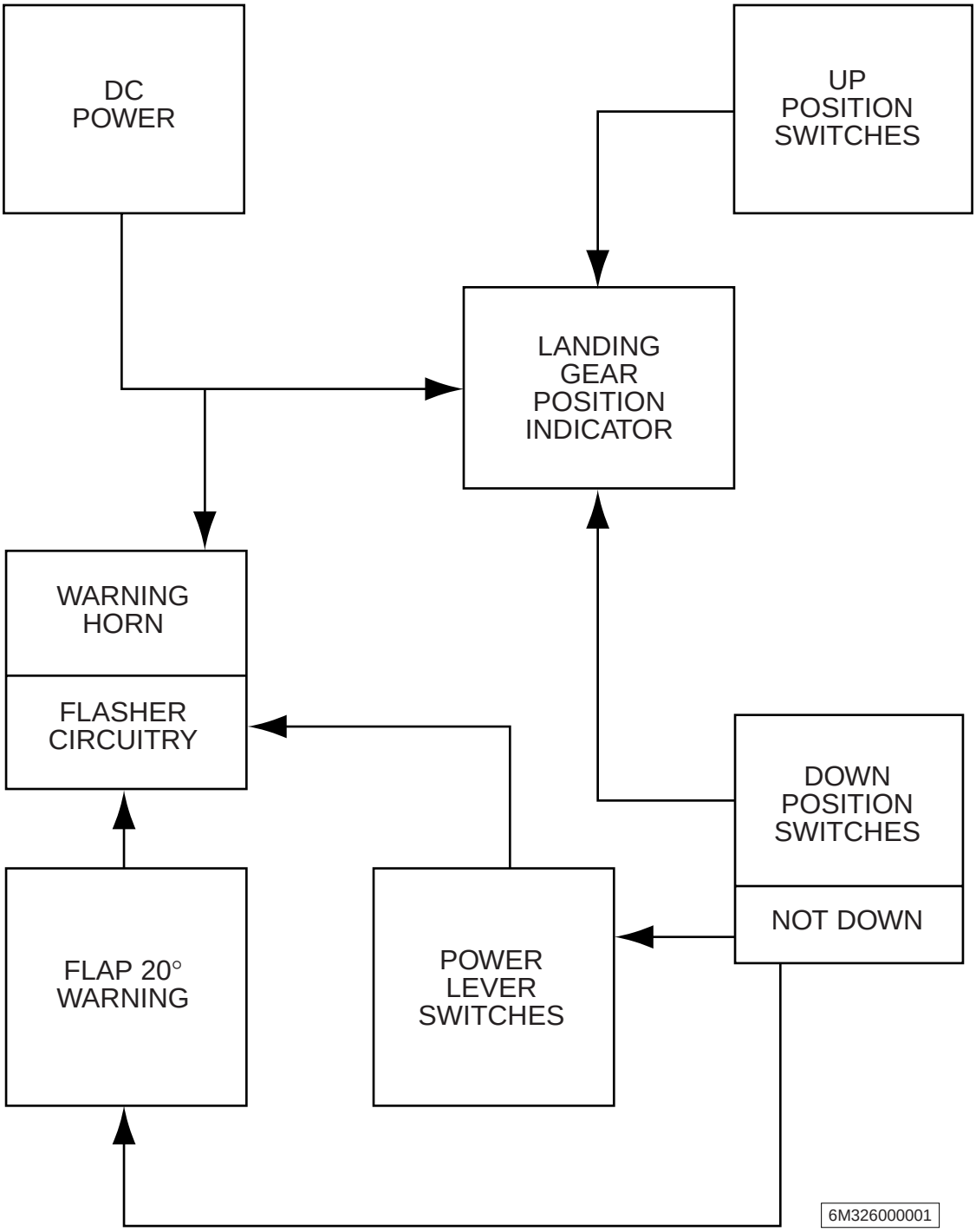
LANDING GEAR POSITION AND WARNING – DESCRIPTION AND OPERATION

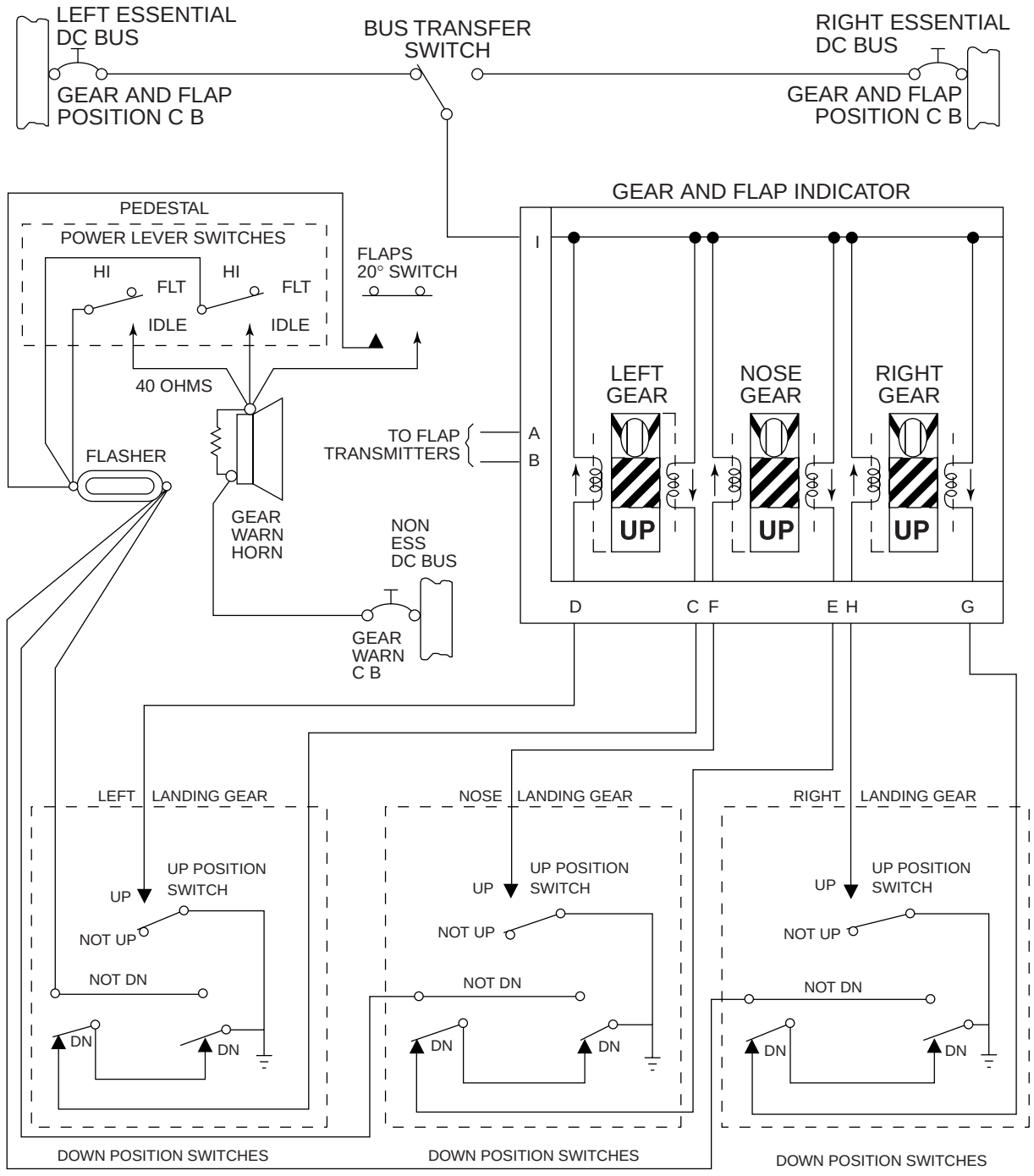
1. General

A. The landing gear position and warning system consists of a position indicator, three position switches on each landing gear, and a warning horn and its associated circuitry. Two types of indicators are used on SA226 aircraft T201-275, 277, and 278; AT001-061; TC201-231, 233, 234, and 236 use an indicator that has three symbol indicators within the same case as the flap position indicator (see Section 32-60-20). T276, 279-999; AT062-999; and TC232, 235, 237-999 use an indicator that has three green and three red lights to indicate gear position (see Section 32-60-25). Three switches on each landing gear are used for position indication and warning. On aircraft not complying with SB32-057, an up position switch along with an uplimit switch are mounted on the uplock hook assembly. When the gear is retracted, the uplock hook catches a roller on the strut and the up switches are actuated. On aircraft complying with SB32-057, the up limit and position switches are mounted on the left and right drag brace mount fittings. The down position switches are attached to the side structure of the wheel well and are actuated when the landing gears are down and locked. Aural warning is provided on SA226 aircraft if the landing gear or the flaps are in an unsafe condition. T201-275, 277, and 278; AT001-062, TC201-231, 233, 234, and 236 use a warning horn and flasher unit that are separate units (see 32-60-35). T276, 279-999; AT062-999; TC232, 235, 237-999 use a warning horn and flasher circuitry that are in the same case as the flap position circuitry (see 32-60-35).



Landing Gear Position and Warning
Figure 1

LANDING GEAR POSITION AND WARNING – FAULT ISOLATION



NOTE: LANDING GEAR POSITION SWITCHES SHOWN WITH LANDING GEAR DOWN AND LOCKED. DOWN POSITION SWITCHES SHOWN WIRED IN PARALLEL IN COMPLIANCE WITH SB32-017.

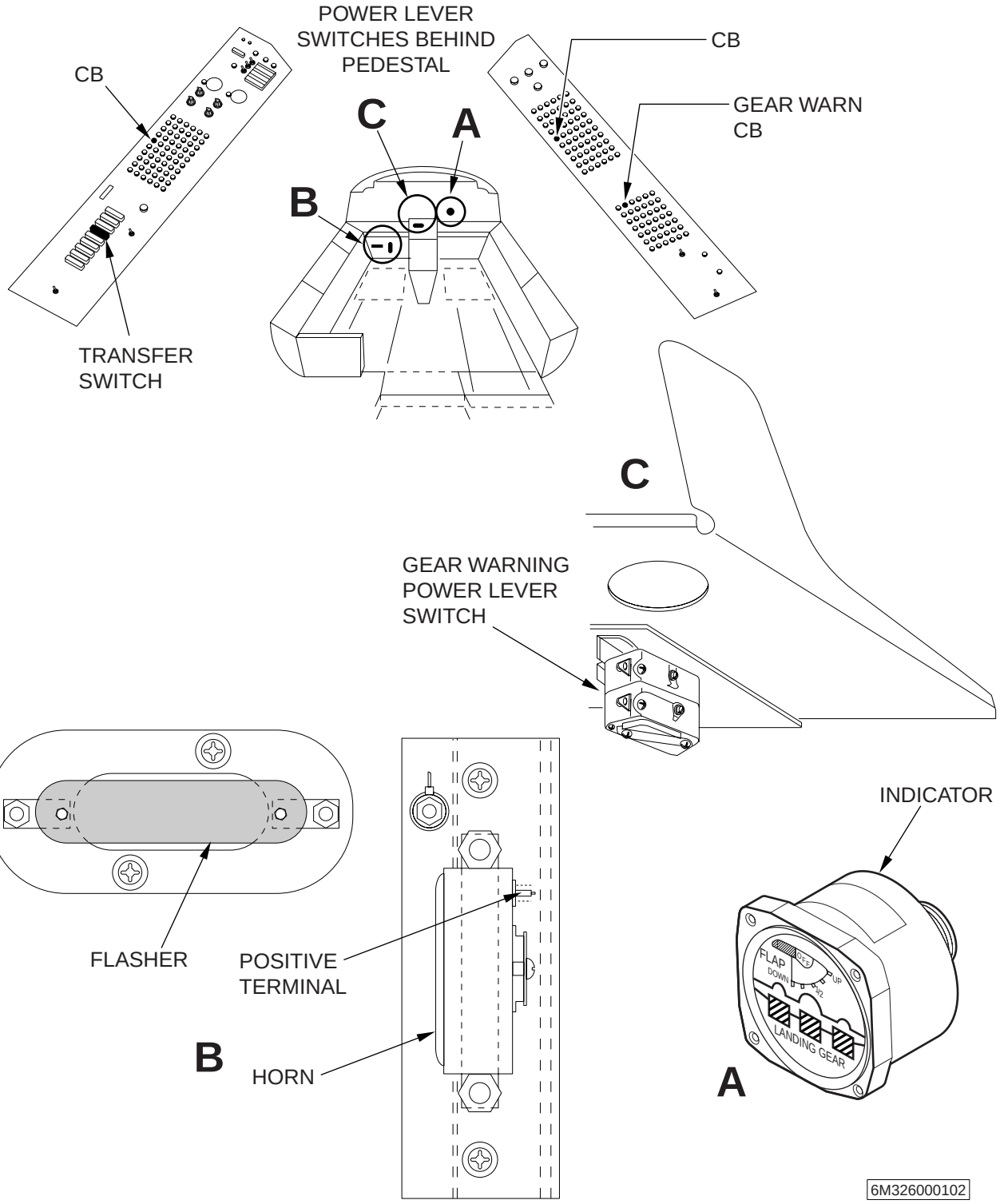
6M326000101

Landing Gear Position and Warning
Figure 2

LANDING GEAR POSITION AND WARNING – FAULT ISOLATION

1. Troubleshooting –Landing Gear Position and Warning

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
One or more landing gear fail to indicate down and locked. Landing gear are actually down and locked		PRELIMINARY Refer to Wiring Manual, Chapter 32.
	1. GEAR & FLAP POS circuit breaker, wiring, or transfer switch defective	1. If all three landing gear show unsafe, check pin 1 for 28 VDC. If 28 VDC is present, proceed to step 2. If 28 VDC is not present, circuit breaker, wiring, or transfer switch is defective.
	2. Down position switch adjusted improperly. Switch or wiring defective. For position switch adjustment, see 32-60-10	2. Check for ground at pins C (left), G (right) or E (nose). If ground is not present, down position switches are defective or not adjusted properly or wiring is defective.
	3. Indicator defective see 32-60-20	3. If ground is present at all three pins, indicator is defective.
One or more landing gear fail to indicate up and locked, with landing gear retracted	1. Up position switch not adjust ed properly. For position switch adjustment, see 32-60-10	1. Retract landing gear and check rigging and switch adjustment, (see 32-30-00).



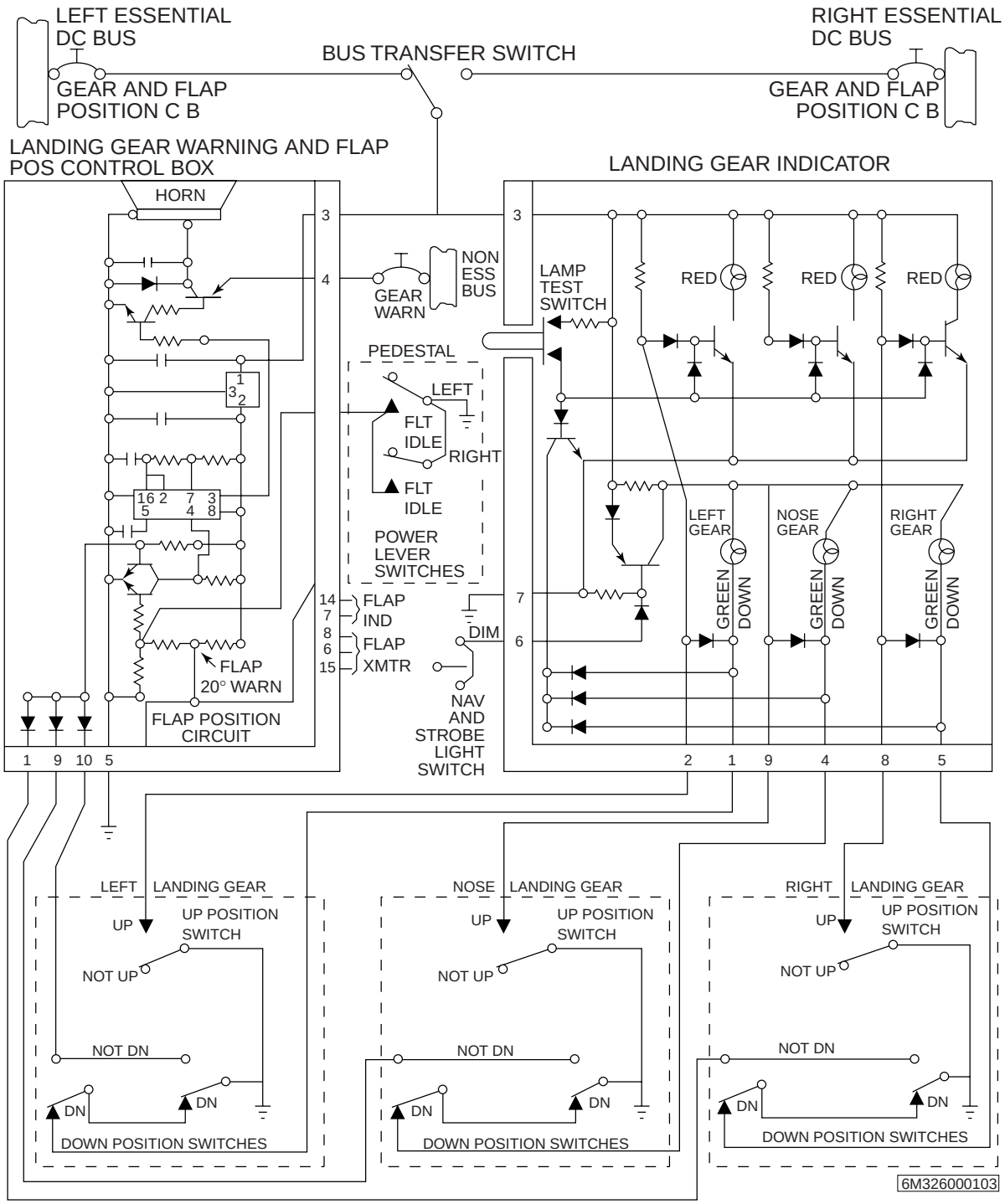
Component Locator
Figure 3

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	2. Switch or wiring defective	2. Remove connector from landing gear indicator. Refer to Wiring Manual, Chapter 32. Check for ground at pins D (left), H (right, or F (nose). Actuate appropriate switch when checking. If ground is not present when switch is actuated, switch wiring is defective.
	3. Indicator defective	3. If ground is present at all three pins when switches are actuated, indicator is defective.

LANDING GEAR POSITION AND WARNING – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Warning horn fails to sound		PRELIMINARY Refer to Wiring Manual Chapter 32.
	1. GEAR WARN circuit breaker or wiring defective	1. Check for 28 VDC present at positive terminal of horn, if 28 VDC is not present, circuit breaker or wiring is defective.
	2. Horn defective, see 32-60-30	2. If 28 VDC is present connect negative terminal of horn to ground. If horn sounds steady, proceed to step 3. If horn does not sound steady, horn is defective.
	3. Flasher defective see 32-60-30	3. Pull both power levers back to flight idle gate. Ground one side at flasher, then the other. When one side of the flasher is grounded, horn should sound steady.
	4. Power lever switches or wiring defective. For power lever switch adjustment, see 32-60-30	4. If it does not, power lever switches or wiring are defective. When the other side of the flasher is grounded, the horn should beep. If it does not, flasher is defective.
	5. Flap 20° warning switch defective or not adjusted properly. For switch adjustment, see 32-60-30	5. Check wiring to switch or switch adjustment.

LANDING GEAR POSITION AND WARNING – FAULT ISOLATION

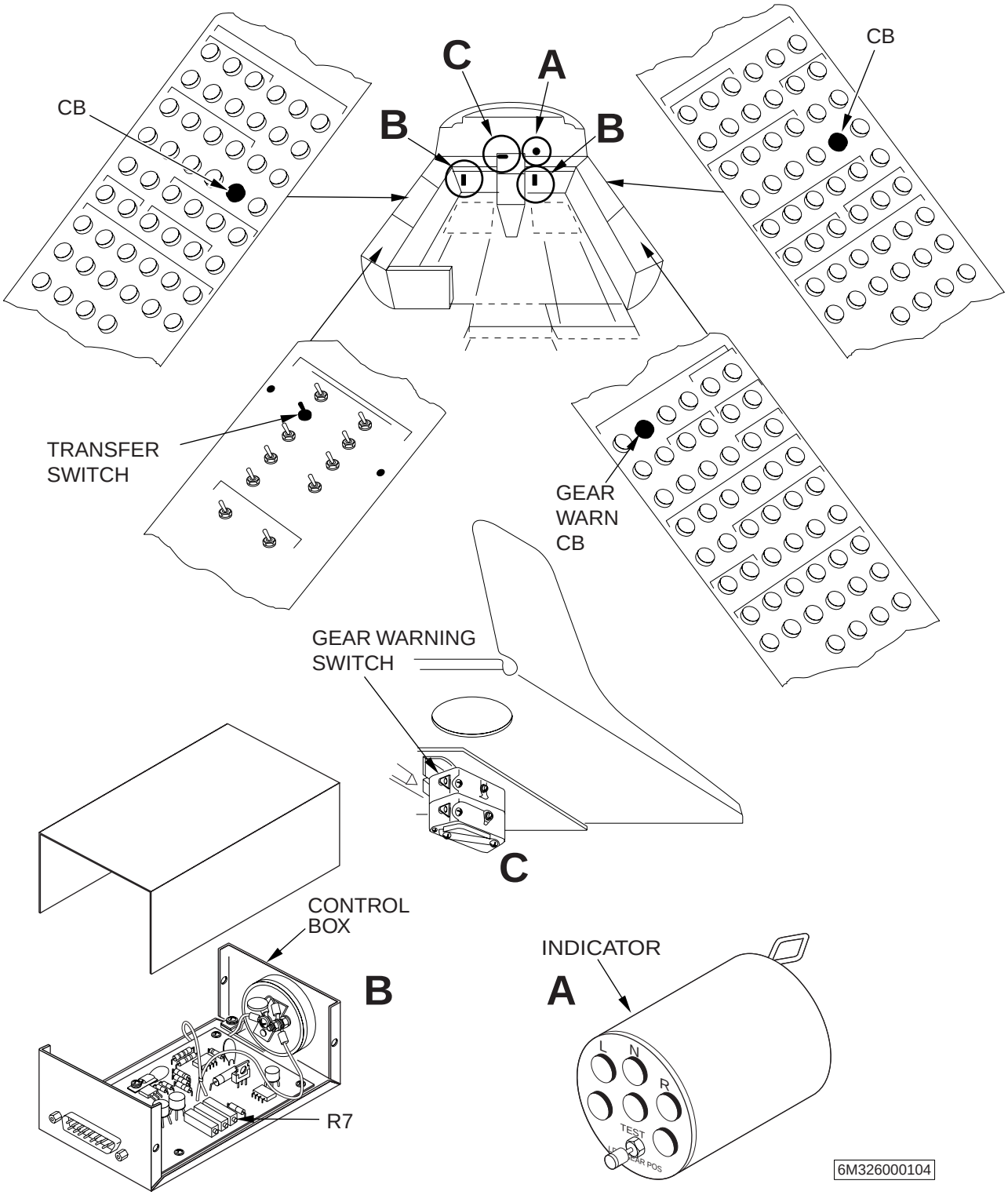


NOTE: LANDING GEAR POSITION SWITCHES SHOWN WITH LANDING GEAR DOWN AND LOCKED. DOWN POSITION SWITCHES SHOWN WIRED IN PARALLEL IN COMPLIANCE WITH SB32-017.

Landing Gear Position and Warning
Figure 4

LANDING GEAR POSITION AND WARNING – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
One or more landing gear fail to indicate down and locked. Landing gear are actually down and locked		PRELIMINARY Refer to Wiring Manual Chapter 32.
	1. GEAR & FLAPS POS circuit breaker, wiring, or transfer switch defective	1. If all six lights on landing gear indicator fail to illuminate when lamp test switch is depressed. remove connector and check 28 VDC at pin 3 and ground at pin 7. If not present, circuit breaker, wiring, or transfer switch is defective.
	2. Landing gear indicator defective see 32-60-25	2. If present, indicator is defective. If one or more lights fail to illuminate,indicator is defective.
	3. Down position switch adjusted improperly. Switch or wiring defective. For position switch adjustment, see 32-60-10	3. Remove indicator connector. Check for ground at pin 1 (left), 5 (right), 4 (nose). If ground is not present, down position switches are defective or not set properly or wiring is defective.



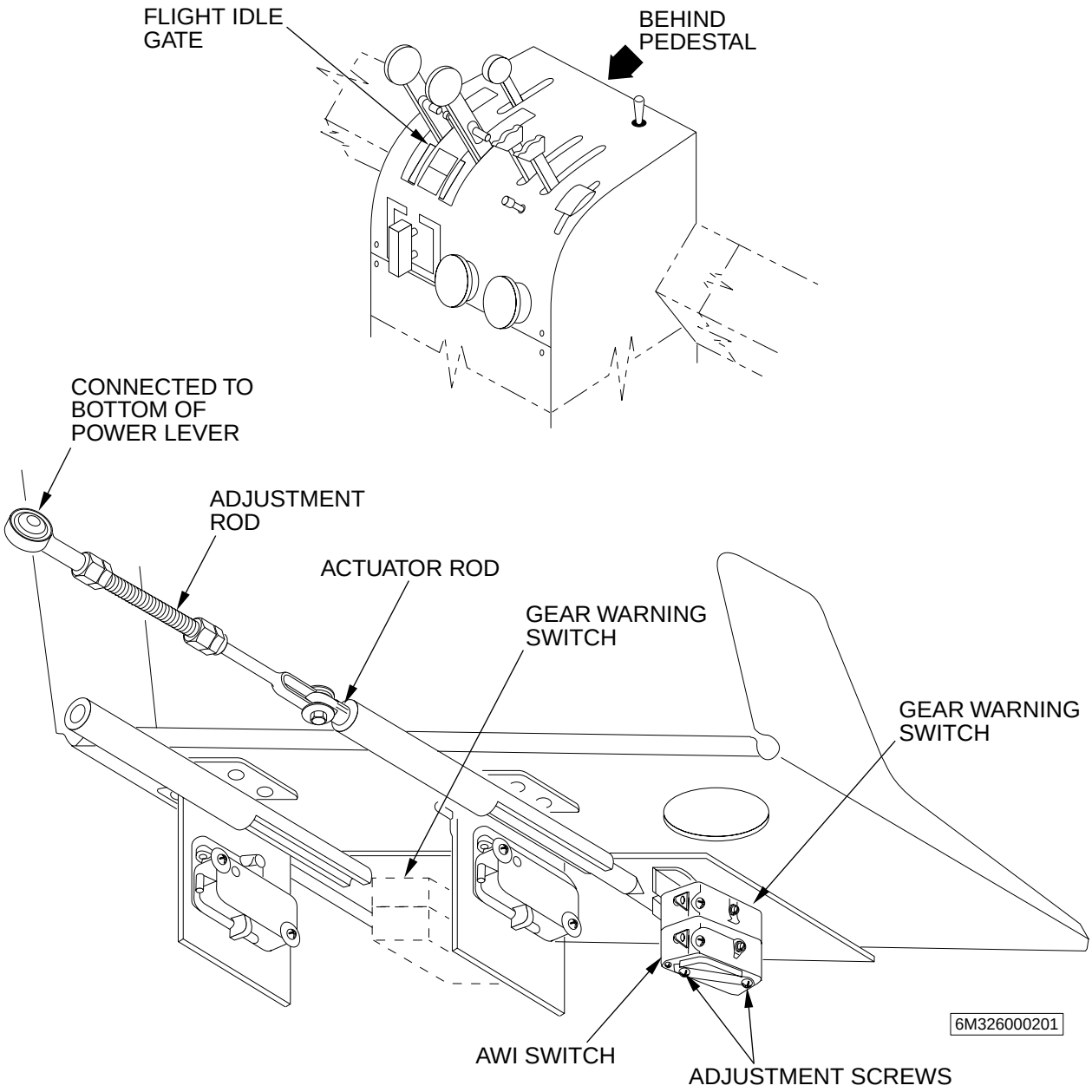
Component Locator
Figure 5

LANDING GEAR POSITION AND WARNING – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
One or more landing gear show unsafe with landing gear retracted	1. Up position switch not adjusted properly. For position switch adjustment, see 32-60-10	1. Retract landing gear and check rigging and switch adjustment, (see 32-30-00).
	2. Switch or wiring defective	2. Remove connector from landing gear indicator. Refer to Wiring Manual, Chapter 32. Check for ground at pins 2 (left), 8 (right, or 9 (nose). Actuate appropriate switch when checking. If ground is not present when switch is actuated, indicator is defective.
Warning horn fails to sound		PRELIMINARY Refer to Wiring Manual, Chapter 32.
	1. GEAR WARN circuit breaker or wiring defective	1. Remove connector from gear warning and flap position control box. Check pin 4 for 28 VDC. If 28 VDC is not present, circuit breaker is defective.
	2. GEAR & FLAP POS circuit breaker, transfer switch, or wiring defective	2. Check pin 3 for 28 VDC. If 28 VDC is not present, circuit breaker, transfer switch, or wiring defective. Check pin 5 for ground.
	3. Power lever switches or wiring defective. For power lever switch adjustment, see 32-60-00	3. Check pin 2 for ground when power levers are at the flight idle gate.

LANDING GEAR POSITION AND WARNING – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	4. Gear warning and flap control box defective, see 32-60-35	4. If the above voltage and grounds are confirmed, box is defective.
Flap 20° warning inoperative	1. Flap position circuitry not adjusted properly, see 32-60-35 and 27-51-06	1. If flap position indicator is adjusted properly and horn does not sound at 20°, but does sound remove control box and adjust R7 (pot closest to horn) until horn sounds at 20° about a needles width below half.
	2. Gear warning and flap position control box defective, see 32-60-35	2. If horn cannot be adjusted to sound, control box is defective.



Power Lever Position Switch Adjusment
Figure 201

LANDING GEAR POSITION AND WARNING – MAINTENANCE PRACTICES

1. Adjustment/Test – Landing Gear Position and Warning

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP, REPEAT UNTIL HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

NOTE: Every 200 hours as part of Letter Check C, the landing gear position and warning system must be checked as part of landing gear extension and retraction check. For lubrication and sealing gear switches, see Section 32-60-10.

A. Power Lever Position Switch Adjustment

(1) Remove pedestal center cover.

(2) Before adjusting rods attached to bottom of power levers.

(a) Pull power levers back behind flight idle gate.

(b) Loosen screws in slotted ends of switches and push switches against rods until actuators are fully depressed.

(c) Tighten screws.

(3) Loosen lock nuts on rods and adjust rods to actuate switches 1/8 of an inch before the flight idle gate when power levers are pulled back. This is measured from the gate to the lift latch. Switches must stay actuated when power levers are at flight idle gate. Some aircraft also have AWI arming switches at the same location.

(4) Tighten lock nuts.

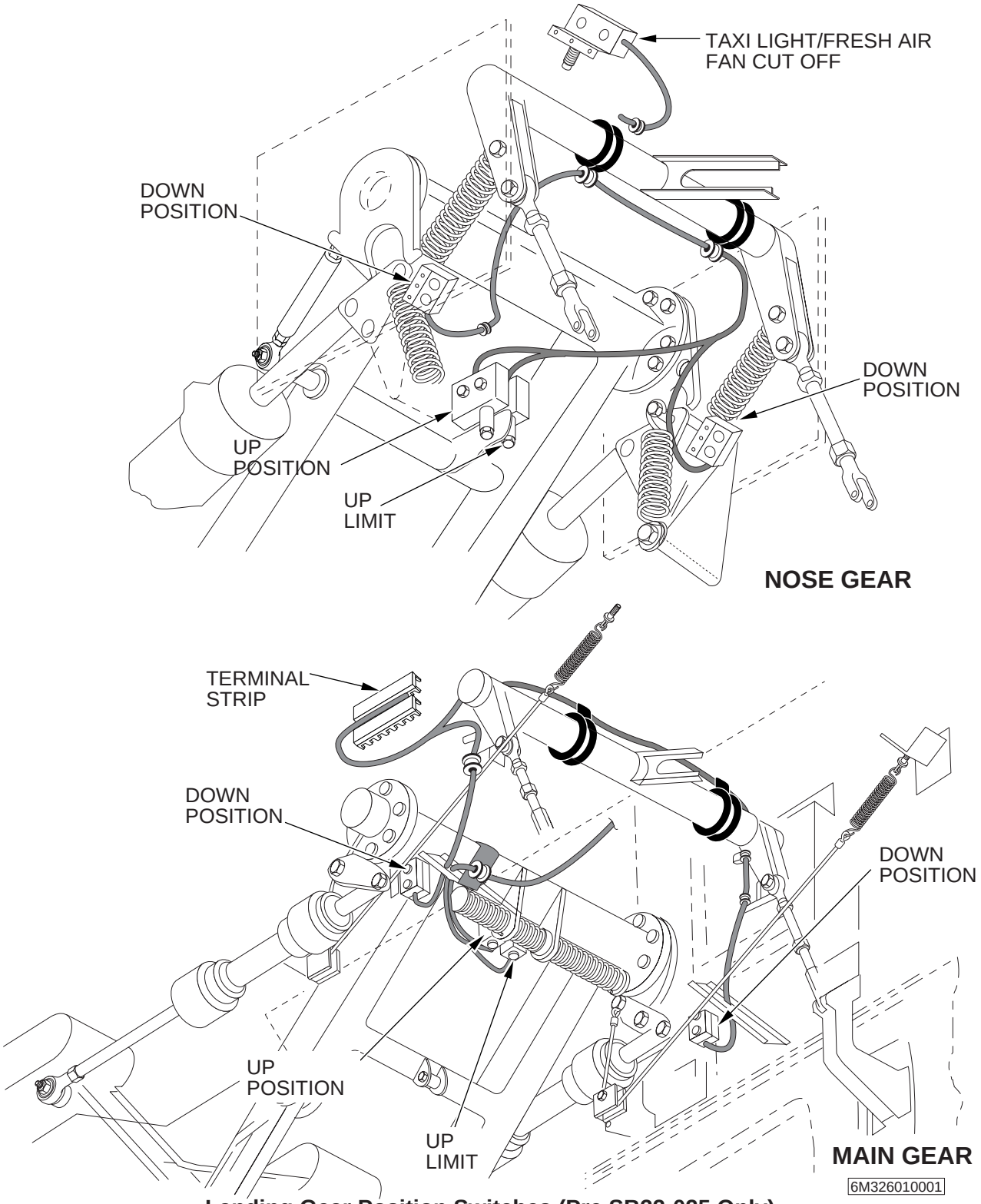
(5) Perform landing gear warning functional check (see Section 32-60-30 and 32-60-35).

(6) Replace pedestal center cover.

LANDING GEAR POSITION SWITCHES – DESCRIPTION AND OPERATION

1. General

A. Three switches on each landing gear are used for the position indicating system. One switch is for up and locked indication, and two are for down and locked indication. The up position switch along with the up limit switch (see 32-30-00) are attached to the uplock hook and are actuated by the uplock roller. The down position switches are bracket mounted to the wheel well structure on the left and right sides and actuated by a bellcrank cam arrangement attached to the gear actuator. An additional switch is bracket mounted to the right side of the nose wheel well and is actuated by the door closure hook assembly. This switch turns off the taxi light and fresh air fan when the gear is retracted. All four switch assemblies on each gear are the same assembly, the only difference is the way they are mounted. The switches are spring-loaded open. The position system uses the normally open side of the switches, and the warning system uses the normally closed side of the down switches. Each switch assembly consists of a microswitch, an actuator, two cover halves, and attaching hardware. The actuator is spring-loaded away from the button on the microswitch. An additional spring between the actuator plunger and the microswitch prevents the actuator from sticking closed in cold weather. The spring was added on early model SA226 aircraft by Service Bulletin 32-013. Aircraft T284-999, AT063-999, and TC248-999 the spring was installed at the factory. To prevent the possibility of a false down indication on the landing gear indicator, Service Bulletin 32-017 was issued to change the wiring of the down position switches from series to parallel. Aircraft T283-999, AT062-999, and TC247-999 the down position switches were wired in parallel at the factory. If the down position switches are wired in parallel, both switches on each landing gear must be actuated for the indicator to display a down and locked indication.



Landing Gear Position Switches (Pre SB32-025 Only)
Figure 1

LANDING GEAR POSITION SWITCHES – MAINTENANCE PRACTICES

1. Servicing – Landing Gear Position Switches

NOTE: Every 200 hours as part of Letter Check C, switch assemblies should be removed, disassembled, lubricated with Dow Corning DC-4 silicone grease, reassembled, installed, and sealed with RR-1422-A2 or equivalent. Also as part of Letter Check C, switches must be checked as part of landing gear extension and retraction check.

2. Adjustment/Test – Landing Gear Position Switches

A. Adjustment – Position and limit switches

(1) Jack aircraft.

(2) Remove rod ends from door closure hook assemblies. Tie or tape doors open. Tie or tape rod ends to side of wheel well (see Section 32-30-00).

(3) Up position and limit switches.

(a) Retract landing gear.

(b) Loosen screws holding switches to uplock hook.

(c) Adjust switches so that they actuate approximately 7/32 inch before uplocks are fully engaged.

(d) Tighten screws.

(4) Down position switches.

(a) Extend landing gear.

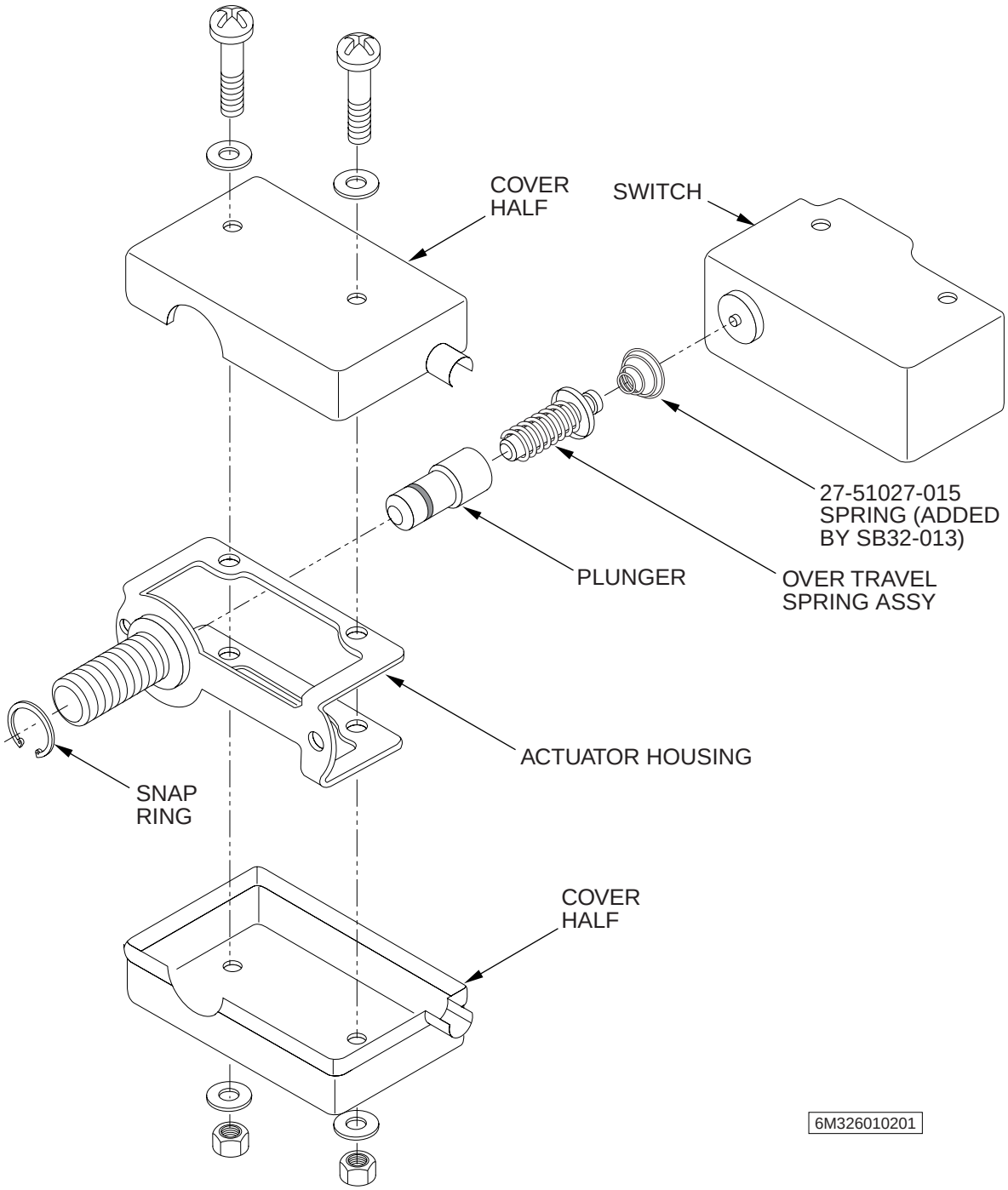
(b) Loosen nuts on actuator barrels.

(c) Adjust switches so that they actuate approximately 7/32 inch before downlocks are fully engaged.

(d) Tighten nut.

(5) Connect rod ends to door closure hook assembly.

(6) Perform extension and retraction check (see Section 32-30-00).



Position Switch Assembly
Pre SB32-025 Only
Figure 201

3. Removal/Installation – Landing Gear Position Switches

A. Removal – Switch Assembly

(1) Up position and up limit switches.

(a) Remove sealer from head of screws and nuts attaching switch assemblies to uplock hook.

(b) Remove screws, nuts, and washers. Drop switch assemblies down. Separate and remove covers.

(c) Separate actuators from switches.

(2) Down position switches.

(a) Remove sealer from head of screws and nuts attaching covers to switches.

(b) Remove screws, nuts, and washers.

(c) Separate and remove covers.

(d) Separate switches from actuators. Leave actuators installed unless replacement is warranted.

(3) Leave wires connected to switches unless replacement is warranted.

(4) Remove and discard snap ring from plungers. Remove plungers, over-travel spring assemblies, and 27-51027-015 spring and clean with MPK and brush.

(5) Clean inside of actuator barrel with MPK and pipe cleaners or small round brush.

(6) Inspect plunger for scoring. Smooth plunger as necessary using fine sandpaper or crocus cloth.

(7) Scrape old sealer from switch covers. Do not clean switch covers with MPK. Alcohol may be used to remove oil and grease from covers.

B. Lubrication, Assembly, Installation, and sealing – Switch Assembly

(1) Fill and coat plungers with Dow Corning DC-4 silicone grease.

(2) Install over-travel springs in plungers. Install plungers in barrel of actuator housings.

(3) Install 27-51027-015 spring on button of over-travel spring assemblies.

(4) Position one up switch in actuator. Position covers and install screws through assembly. Position switch assembly on uplock hook. Assemble other up switch on uplock hook and install washers and nuts on screws.

(5) Position down position switches in actuators. Position covers and install screws through assemblies (screw heads inboard). Install washer and nuts on screws. Switch assemblies should not require adjustment if actuators were not removed.

(6) Adjust landing gear switches.

(7) Perform landing gear extension and retraction check (see Section 32-00-00).

(8) Using PR-1422-A2 or equivalent, seal seams in covers, screw heads, nuts, around holes where wires exit assembly, and around actuator barrels at cover.

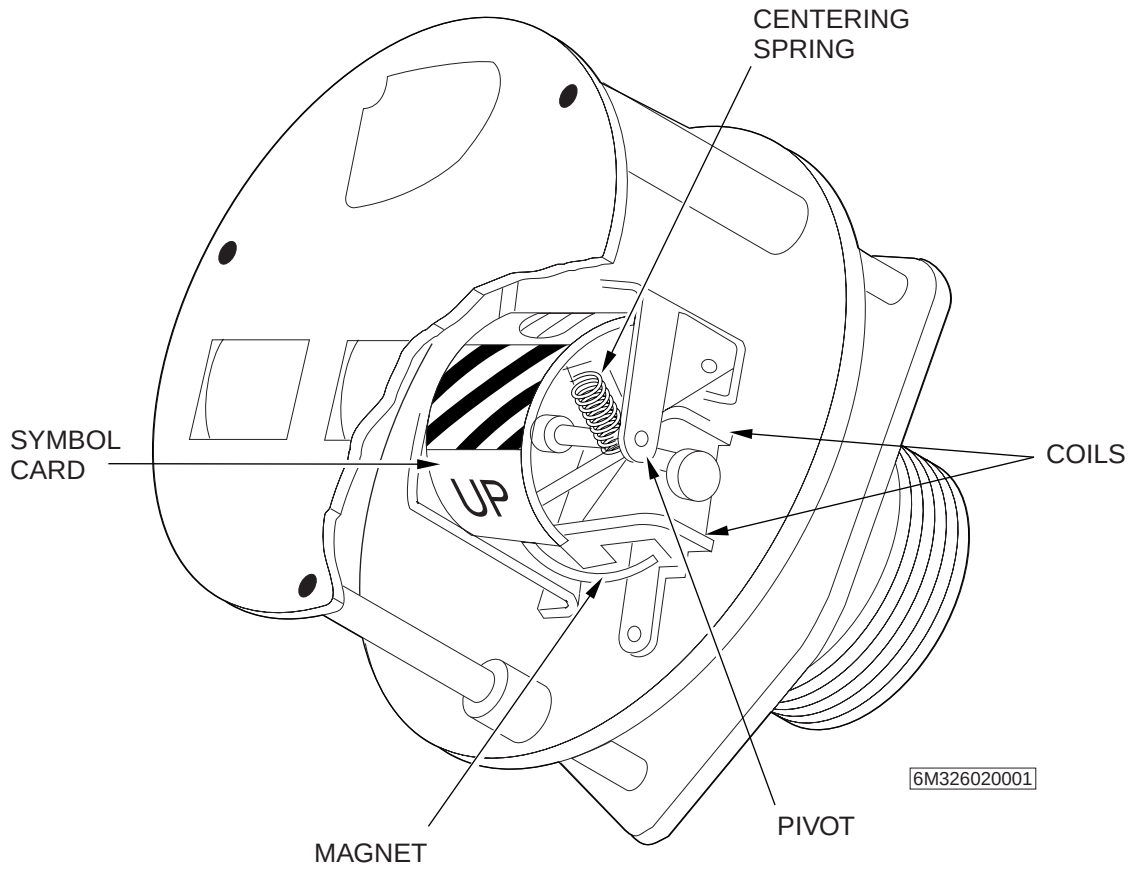
LANDING GEAR POSITION INDICATOR – DESCRIPTION AND OPERATION

1. General

A. The landing gear position indicator uses three DC solenoid type symbol indicators within the same case as the flap position indicator. The three symbol indicators consists of two coils, a horseshoe magnet, and a symbol card. A pivot in the center of the magnet allows it to rotate in such a way that the open ends of magnet can enter into the open space of the coils. The symbol card is attached to the magnet and displays three symbols: a miniature landing gear, which appears when the gear is down and locked; a red and white cross hatch warning symbol is displayed when the gear is in transit or unsafe; and the word UP is visible when the gear is up and locked.

B. The coils receive power from the GEAR & FLAP POS circuit breakers on the left or right essential DC buses. The power can be connected to either bus by the transfer switch on the rear portion of the left console. The position switches on the landing gear are spring-loaded open. One switch is for up and locked indication, and two are for down and locked indication.

C. When the gear is up and locked, the up switch closes completing the ground path to one coil. One end of the magnet is pulled into the coil and the UP symbol appears. Similarly, when the gear is down and locked the down switches close. The other coil is energized, and the down symbol is brought into view. The magnet is spring-loaded to the center position. When either the indicator received no power or the landing gear switches are not closed, the crosshatch symbol is displayed.



Landing Gear and Flap Indicator
Figure 1

LANDING GEAR POSITION INDICATOR – MAINTENANCE PRACTICES

1. Servicing – Landing Gear Position Indicator

A. Every 200 hours as part of Letter Check C, the landing gear position indicator must be checked as part of landing gear extension and retraction check. No adjustment can be made to indicator while installed on the aircraft.

2. Removal/Installation – Landing Gear Position Indicator

A. Removal – Indicator

(1) Remove four screws attaching indicator to instrument panel.

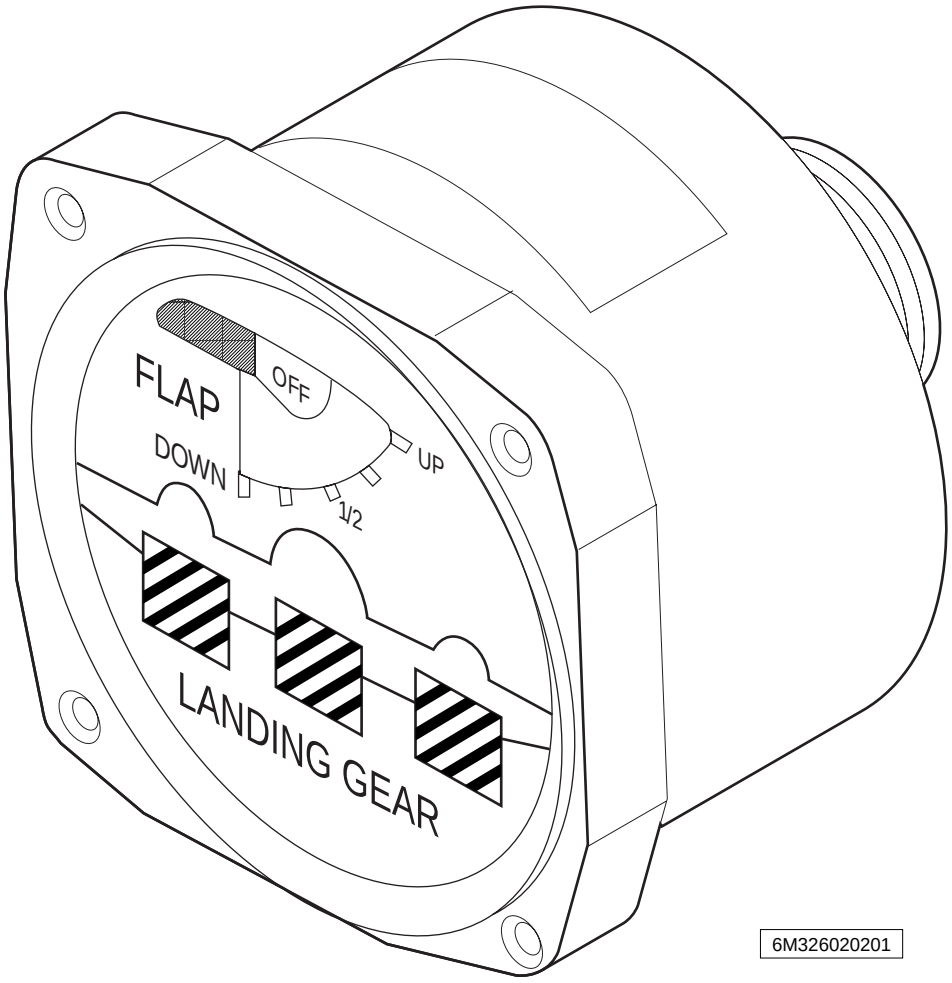
(2) Pull indicator out of instrument panel and remove connector. For rear mounted instruments, reach under and behind instrument panel and bring indicator down either between wire bundle and rudder pedal hangar rod or behind rod. Remove connector.

B. Installation – Indicator

(1) Install connector.

(2) Position indicator and install four screws.

(3) Perform landing gear extension and retraction check (see Section 32-00-00) and operational check of flap position indicating system (see Section 27-50-06).

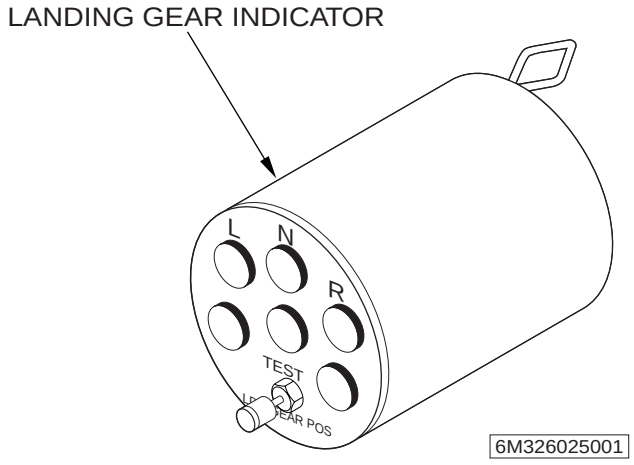


Gear and Flap Indicator
Figure 201

LANDING GEAR INDICATOR – DESCRIPTION AND OPERATION

1. General

A. The indicator uses three green lights, one for each gear, to indicate down and locked. Three red lights are illuminated when the gears are unsafe or in transit. No indication of up and locked is provided. The lights receive power from GEAR POS circuit breakers on the left and right essential DC buses. The power can be connected to either bus by the transfer switch on the rear portion of the left console. The position switches on each gear are spring-loaded open. One switch is for up and locked indication and two are for down and locked indication. Both switches must be closed to illuminate the green light. When the gear is up and locked the red light is extinguished. The lamp test switch on the indicator illuminates all six lights when pressed. The three green lights on the indicator dim when the NAV & STROBE lights switch is turned on.



Landing Gear Indicator
Figure 1

LANDING GEAR INDICATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Landing Gear Indicator

A. Removal – Indicator

(1) Loosen screw in clamp (lower left corner of indicator).

(2) Reach up behind instrument panel and push indicator out front of panel.

(3) Disconnect connector.

B. Installation – Indicator

(1) Reconnectl connector on indicator.

(2) Position indicator in instrument panel and tighten screw in clamp (lower left corner of indicator).

(3) Perform landing gear extension and retraction check (see 32-00-00).

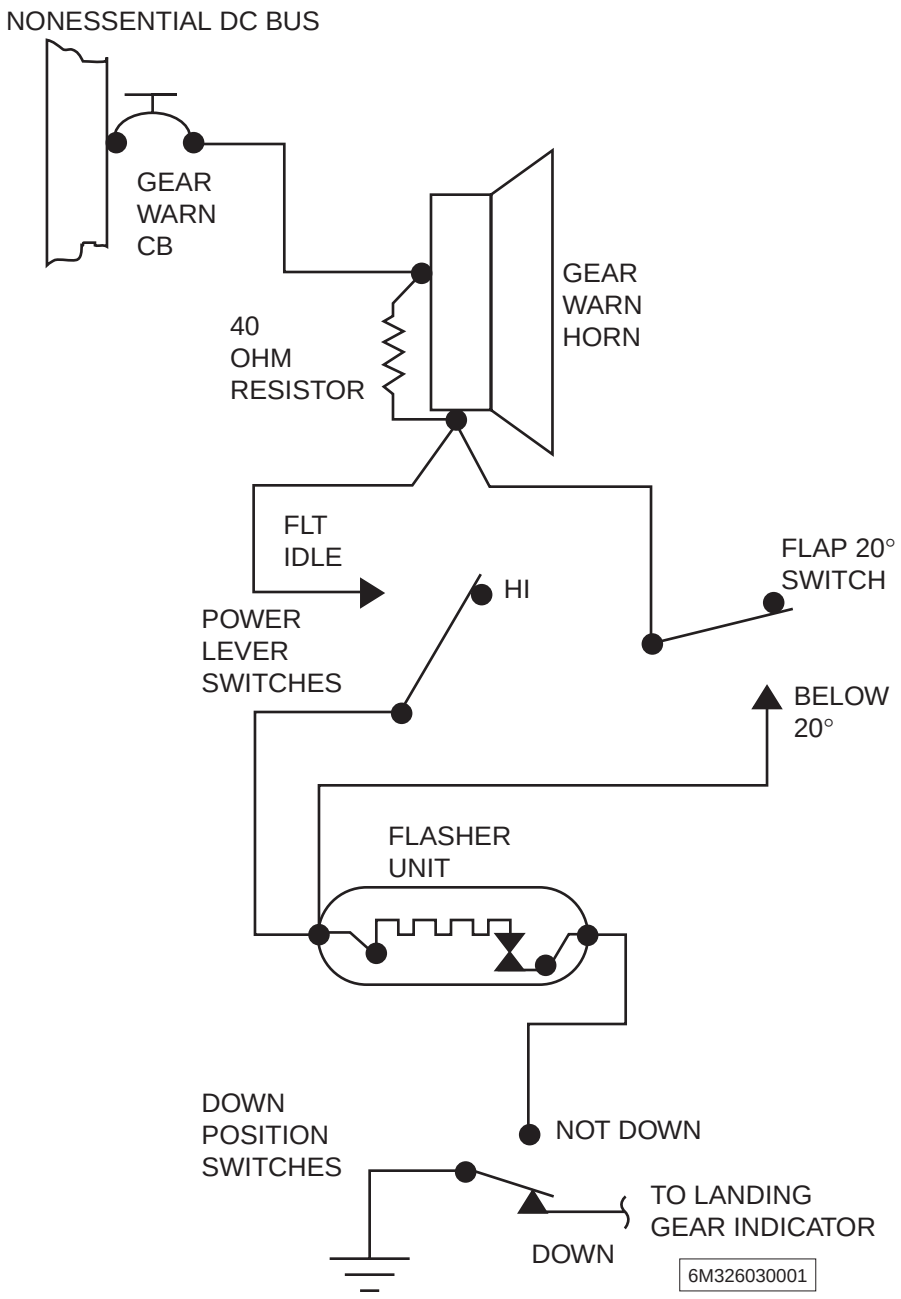
LANDING GEAR WARNING – DESCRIPTION AND OPERATION

1. General

A. The landing gear warning system uses a horn located on the forward pressure bulkhead just forward of the pilot's left rudder pedal. A flasher unit located adjacent to the horn makes the horn beep or flash. The flap 20° warning switch is actuated by a cam attached to the flap position transmitter.

B. The gear warning horn receives power from the GEAR WARN circuit breaker on the nonessential DC bus. The horns ground path to the flasher is completed by switches actuated by either power lever or by the flap 20° warning switch (if installed). The flasher's ground path is completed by the down position switches. Should any of the landing gears not be down and locked, and either power lever be at or behind the flight idle gate, or the flaps be extended below 20°, the horn will sound.

NOTE: Aircraft T201-208 and AT001-003 do not incorporate the 20° warning function.



Landing Gear Warning Circuit
Figure 1

1. Removal/Installation – Landing gear Warning

A. Removal – Landing Gear Warning Horn

(1) Remove nut and remove wires from positive terminal on horn.

(2) Remove grounding screw from bracket at center of horn.

(3) Loosen bolt in clamp at bottom of horn.

(4) Remove horn.

B. Installation – Landing Gear Warning Horn

(1) Position horn in clamp.

(2) Install grounding screw in horn with terminals.

(3) Tighten bolt in clamp at bottom of horn.

(4) Perform functional check.

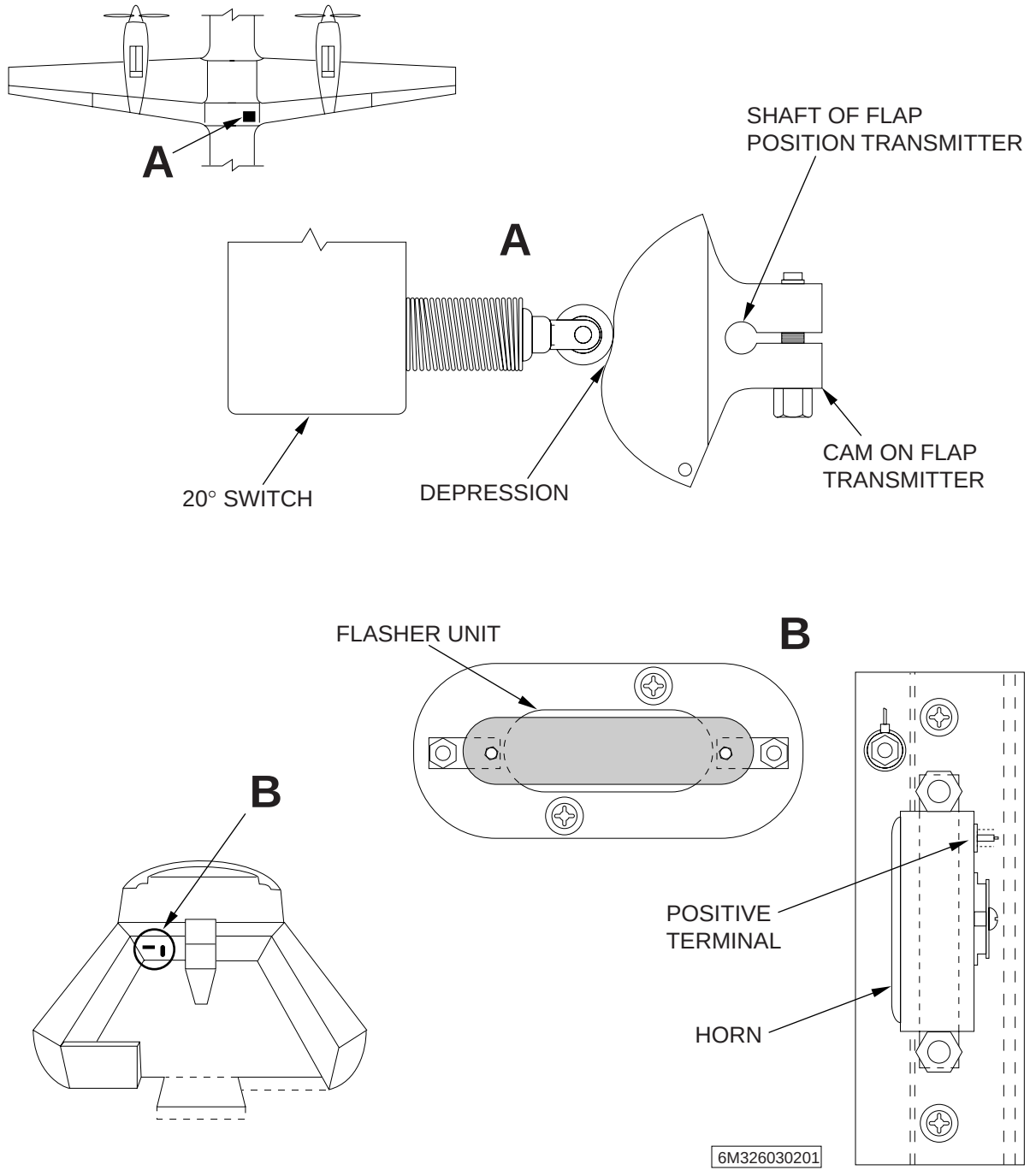
C. Removal – Flasher

(1) Grasp flasher and pull straight aft.

D. Installation – Flasher

(1) Position flasher and push into pins. Care should be taken not to break pins.

(2) Perform functional check.



Landing Gear Warning
Figure 201

2. Inspection/Check – Landing gear Warning

A. Functional Check – Landing Gear Warning

NOTE: After replacement or adjustment of components in landing gear warning system this check should be performed.

(1) Jumper terminals six and seven on left wheel well terminal strip.

(2) Place power levers at flight idle gate.

(3) Turn battery switches on.

(4) Observe landing gear warning horn flashes or beeps (sound).

(5) Push power levers forward approximately 1/8 inch. Observe horn stops sounding.

(6) If flap 20° switch was replaced or adjusted, lower flaps. As flaps pass 20°, about 5/8 on indicator,observe landing gear horn sounds.

(7) Turn battery switches off and remove jumper from left wheel well terminal strip.

B. Adjustment – Flap 20° Warning Switch

(1) Remove left walking beam access panel.

(2) Move flaps to 1/2 (18°).

(3) Adjust switch and cam until switch roller just makes contact with depression in cam.

(4) Move flaps to 20°, switch should actuate. If not, adjust switch in toward cam until it does.

(5) Perform landing gear warning functional check.

NOTE: For more information on switch adjustment, see Maintenance Manual, Section 27-50-06. Switch is also part of SAS system, see Maintenance manual, Chapter 27.

(6) Install access panel.

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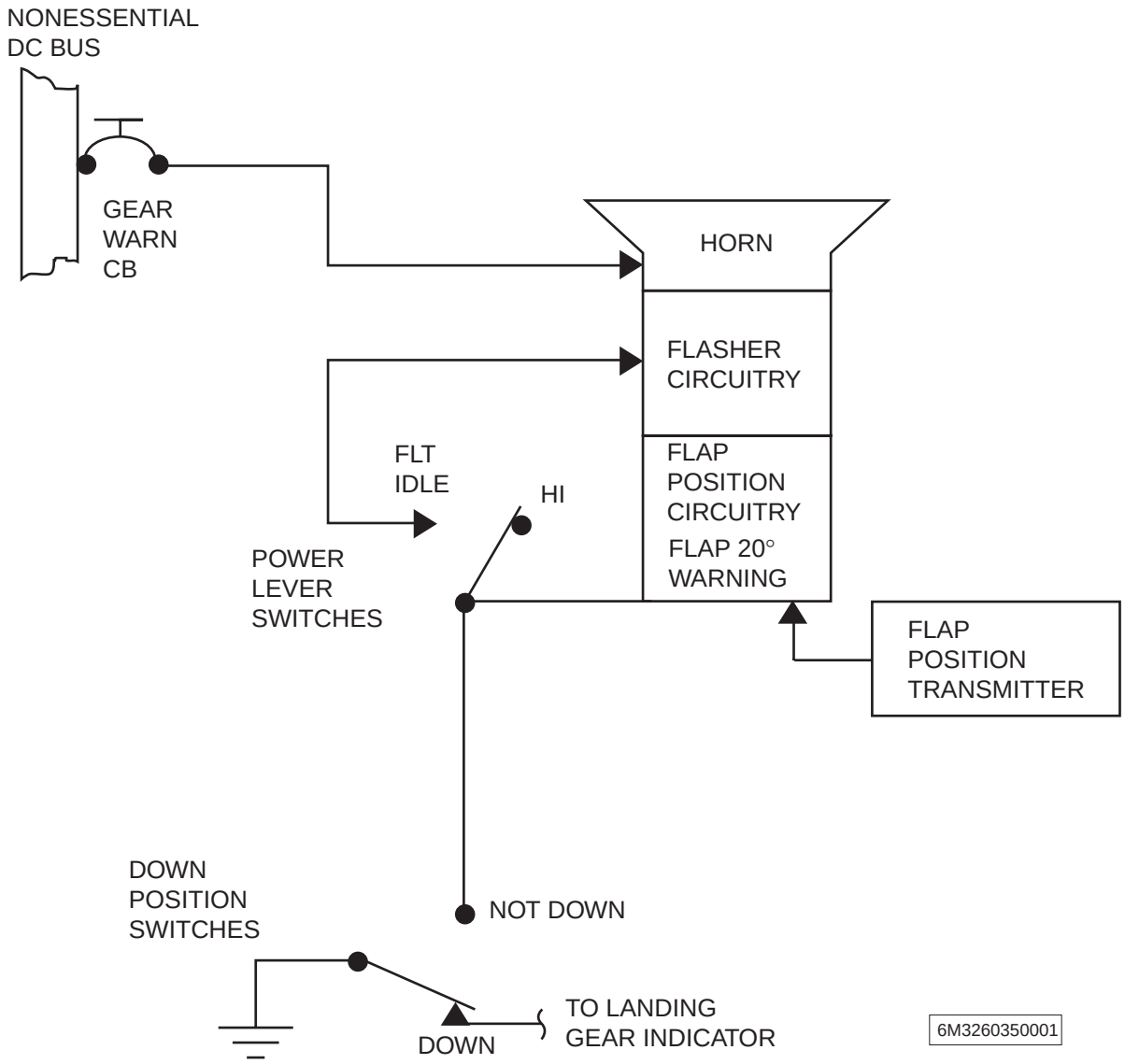
LANDING GEAR WARNING CONTROL BOX – DESCRIPTION AND OPERATION

1. General

A. On aircraft T279-291; AT062-999; and TC232, 235, 237-999 the landing gear warning and flap position control box is located on the forward pressure bulkhead just forward of the pilot's left rudder pedal. On aircraft T292-999 it is located on the forward pressure bulkhead just forward of the copilot's left rudder pedal.

B. The landing gear and flap position control box contains the warning horn, circuitry to make the horn sound or beep, and the flap position indication circuitry. The warning horn circuitry receives power from the GEAR WARN circuit breaker on the nonessential DC bus. The circuitry is controlled by the down position switches. Should any of the landing gear not be down and locked, and either power lever be at or behind the flight idle gate, or the flaps be extended below 20° the horn will sound.

LANDING GEAR WARNING CONTROL BOX – DESCRIPTION AND OPERATION



Landing Gear Warning Circuit
Figure 1

LANDING GEAR WARNING CONTROL BOX – MAINTENANCE PRACTICES

1. Removal/Installation – Landing Gear Warning Control Box

A. Removal – Landing gear Warning and Flap Position Control box

(1) Remove screws securing control box.

(2) Disconnect connector.

(3) Remove control box.

B. Installation – Landing gear Warning and Flap Position Control box

(1) Reconnect connector to control box.

(2) Adjust flap position indication circuitry (see Maintenance Manual, Chapter 27) and perform operational check.

(3) Perform landing gear warning functional check.

(4) Position cover on base and secure with screws.

2. Adjustment/Test – Landing Gear Warning Control Box

NOTE: Flap 20° warning is part of flap position indication system (see Maintenance Manual, Section 27-50-06). The switch on the flap transmitter is part of the SAS system (see Chapter 27).

A. Landing Gear Warning Functional Check

After replacement or adjustment of components in landing gear warning system this check should be performed.

(1) Jumper terminals six and seven on left wheel well terminal strip.

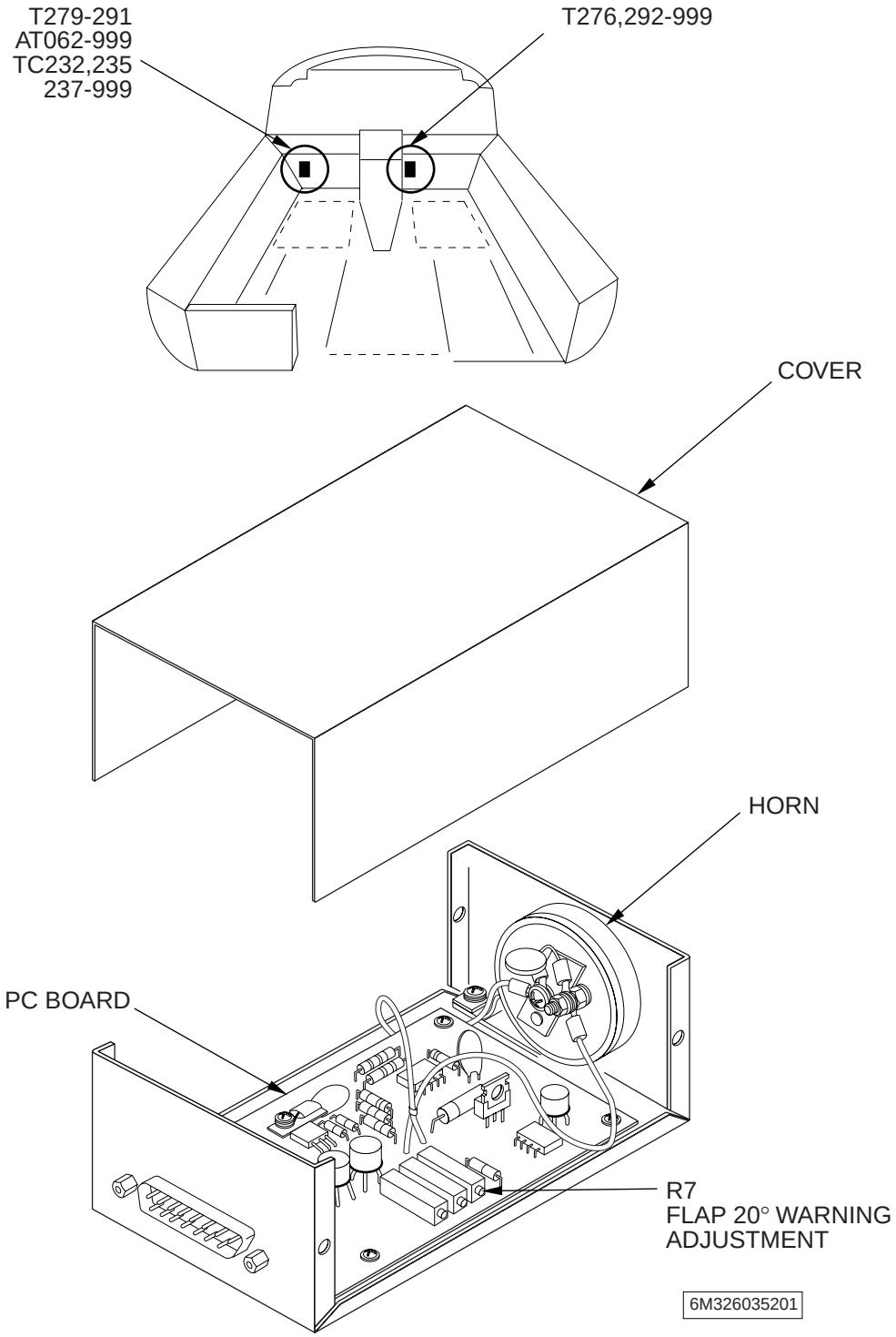
(2) Place power levers at flight idle gate. Turn battery switches on.

(3) Landing gear warning horn should flash or beep (sound).

(4) Push power levers forward approximately 1/8 inch. Horn should stop sounding.

(5) If flap 20° warning was adjusted, lower flaps. As flaps pass 20°, about 5/8 on indicator, landing gear warning horn should sound.

(6) Turn battery switches off and remove jumper from left wheel well terminal strip.



Landing Gear Warning Control Box
Figure 201

AN ELBIT SYSTEMS OF AMERICA, LLC, COMPANY

MAINTENANCE MANUAL

CHAPTER 33 – LIGHTS

LIST OF EFFECTIVE PAGES

SUBJECT	SECTION	PAGE	DATE
33-EFF-PGS	-	1	OCT 01/13
33-EFF-PGS	-	2	OCT 01/13
33-CONTENTS	-	1	NOV 01/03
33-CONTENTS	-	2	NOV 01/03
33-CONTENTS	-	3	NOV 01/03
33-CONTENTS	-	4	NOV 01/03
33-CONTENTS	-	5	NOV 01/03
33-CONTENTS	-	6	NOV 01/03
33-00-00	DO	1	NOV 01/03
33-10-00	DO	1	NOV 01/03
33-10-00	DO	2	NOV 01/03
33-10-01	DO	1	NOV 01/03
33-10-01	DO	2	NOV 01/03
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33-10-01	MP	202	NOV 01/03
33-10-02	DO	1	NOV 01/03
33-10-02	DO	2	NOV 01/03
33-10-02	MP	201	NOV 01/03
33-10-02	MP	202	NOV 01/03
33-10-02	MP	203	NOV 01/03
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33-20-04	MP	203	NOV 01/03
33-20-04	MP	204	NOV 01/03
33-20-04	MP	205	NOV 01/03
33-20-04	MP	206	NOV 01/03
33-20-05	DO	1	NOV 01/03
33-20-05	DO	2	NOV 01/03
33-20-05	DO	3	NOV 01/03
33-20-05	DO	4	NOV 01/03
33-20-05	DO	5	NOV 01/03
33-20-05	MP	201	NOV 01/03
33-20-06	DO	1	NOV 01/03
33-20-06	MP	201	NOV 01/03
33-20-06	MP	202	NOV 01/03
33-30-00	DO	1	NOV 01/03
33-30-01	DO	1	NOV 01/03
33-30-01	MP	201	NOV 01/03
33-30-02	DO	1	NOV 01/03
33-30-02	DO	2	NOV 01/03
33-30-02	MP	201	NOV 01/03
33-30-03	DO	1	NOV 01/03
33-30-04	DO	1	NOV 01/03
33-30-04	DO	2	NOV 01/03
33-30-04	DO	3	NOV 01/03
33-40-00	DO	1	NOV 01/03
33-40-01	DO	1	NOV 01/03
33-40-01	MP	201	NOV 01/03
33-40-01	MP	202	NOV 01/03
33-40-01	MP	203	NOV 01/03
33-40-02	DO	1	OCT 01/13
33-40-02	MP	201	OCT 01/13
33-40-02	MP	202	OCT 01/13
33-40-02	MP	203	OCT 01/13
33-40-03	DO	1	NOV 01/03
33-40-03	MP	201	NOV 01/03
33-40-03	MP	202	NOV 01/03
33-40-03	MP	203	NOV 01/03
33-40-04	DO	1	NOV 01/03
33-40-04	MP	201	NOV 01/03
33-40-04	MP	202	NOV 01/03

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T 201-999, AT 001-999
TC 201-999

MAINTENANCE MANUAL

CHAPTER 33 – LIGHTS

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33-40-05	MP	202	NOV 01/03	33-40-07	DO	1	NOV 01/03
33-40-06	DO	1	NOV 01/03	33-40-07	MP	201	NOV 01/03
33-40-06	MP	201	NOV 01/03	33-40-07	MP	202	NOV 01/03
33-40-06	MP	202	NOV 01/03	33-40-07	MP	203	NOV 01/03
33-40-06	MP	203	NOV 01/03	33-50-01	DO	1	NOV 01/03
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LIGHTS – DESCRIPTION AND OPERATION

1. General

A. This chapter describes in detail those components used on SA226 series aircraft which provides internal and external lighting. The internal lighting covers the flight compartment, passenger compartment, and cargo compartments. The external lighting consists of navigation, strobe, ice detection, landing, taxi, rotating beacon, and logo lights.

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Lamp Replacement Chart
Figure 1

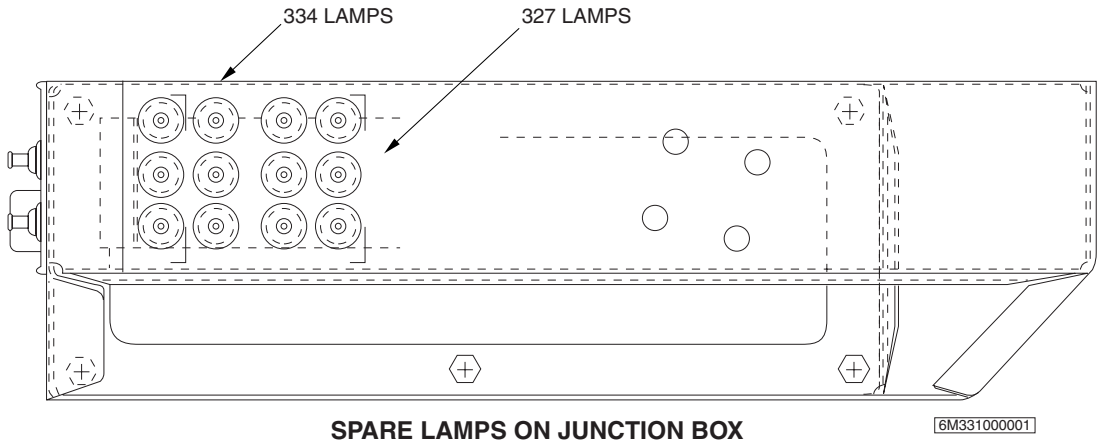
COCKPIT LIGHTING – DESCRIPTION AND OPERATION

1. General

A. The cockpit switch panels, consoles, and glareshield are lighted by post light assemblies and by edge lighted panels. Additional lighting is provided by two map lights flush-mounted in the cockpit overhead paneling.

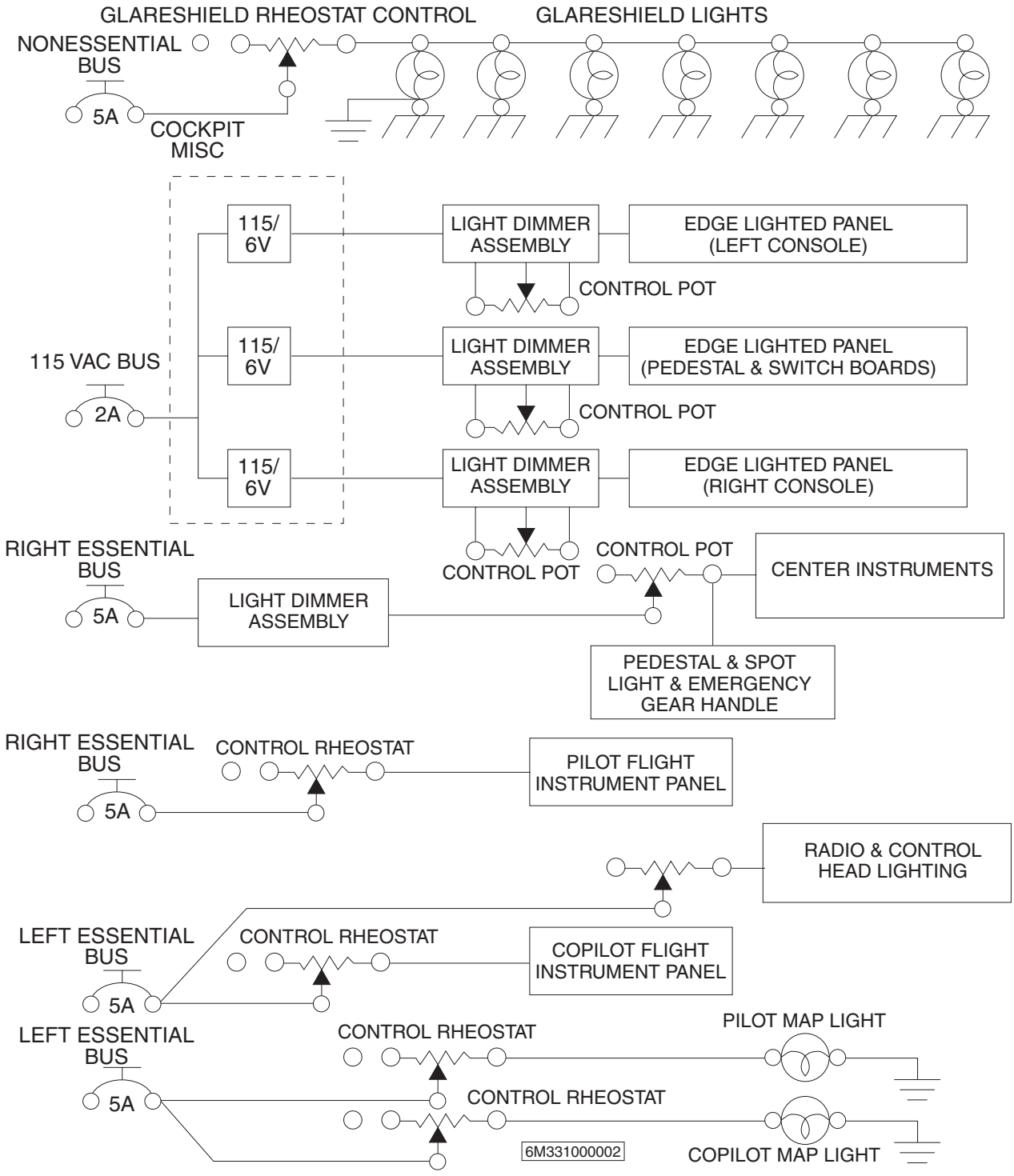
B. Control of cockpit lighting is provided by dimmer assemblies or by rheostats. Energizing current is supplied from the 28 VDC buses except for edge-lighted panels which receive power from the 115 VAC bus through a 115/6.3 VAC step-down transformer.

C. Spare lamps for the cockpit instrument lights are mounted in grommets on the junction box behind the pilot's seat (see Figure 1).



Cockpit Spare Lamps
Figure 1

COCKPIT LIGHTING – DESCRIPTION AND OPERATION



Cockpit Lighting System
Figure 2

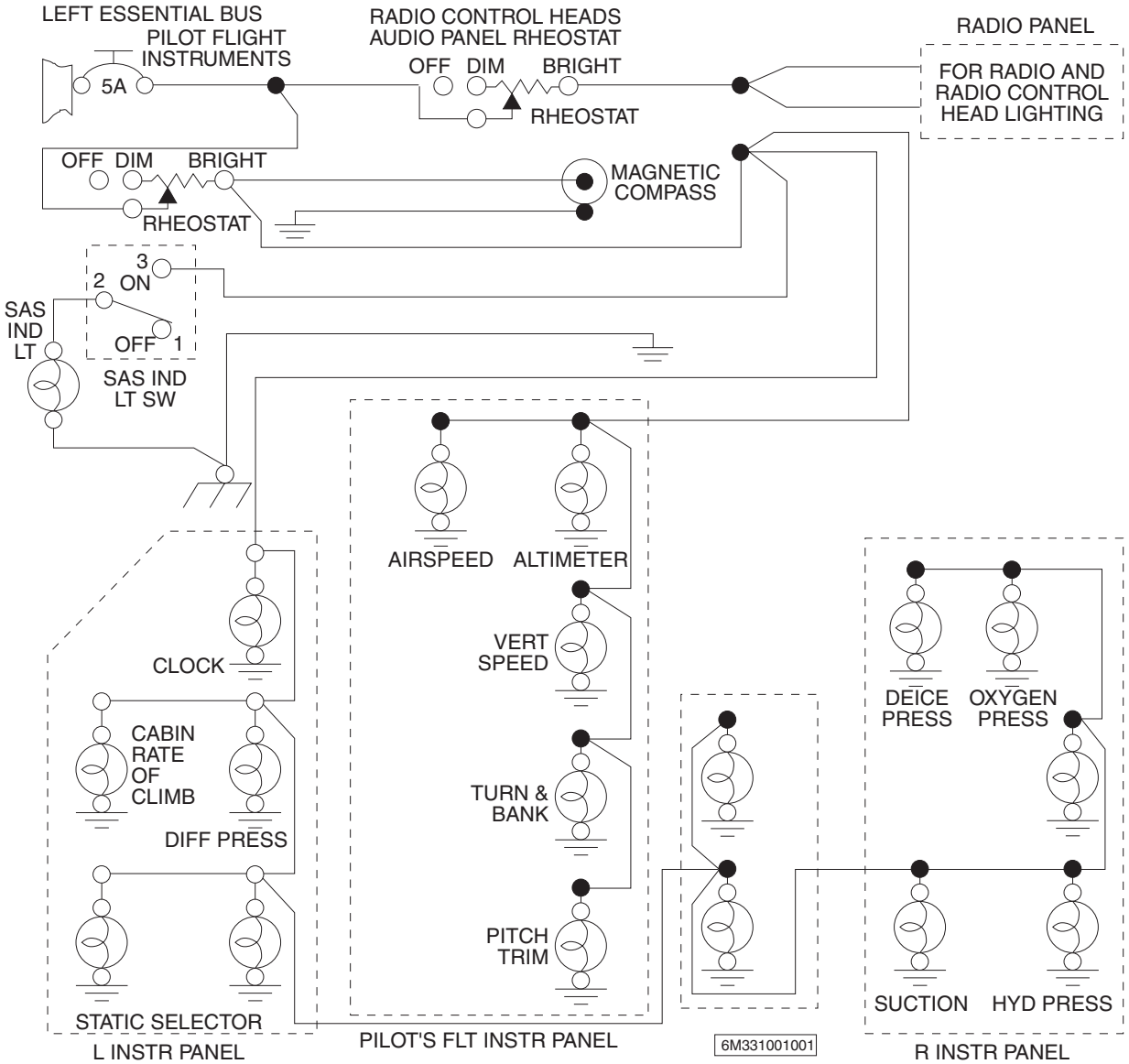
FLIGHT INSTRUMENT LIGHTS – DESCRIPTION AND OPERATION

1. General

A. Pilot's Flight Instrument Lighting

(1) Each pilot's flight instrument is illuminated by a hooded, adjustable post mounted light assembly, which restricts its light beam to the face of the instrument. The post light lamp holder protrudes above the instrument face.

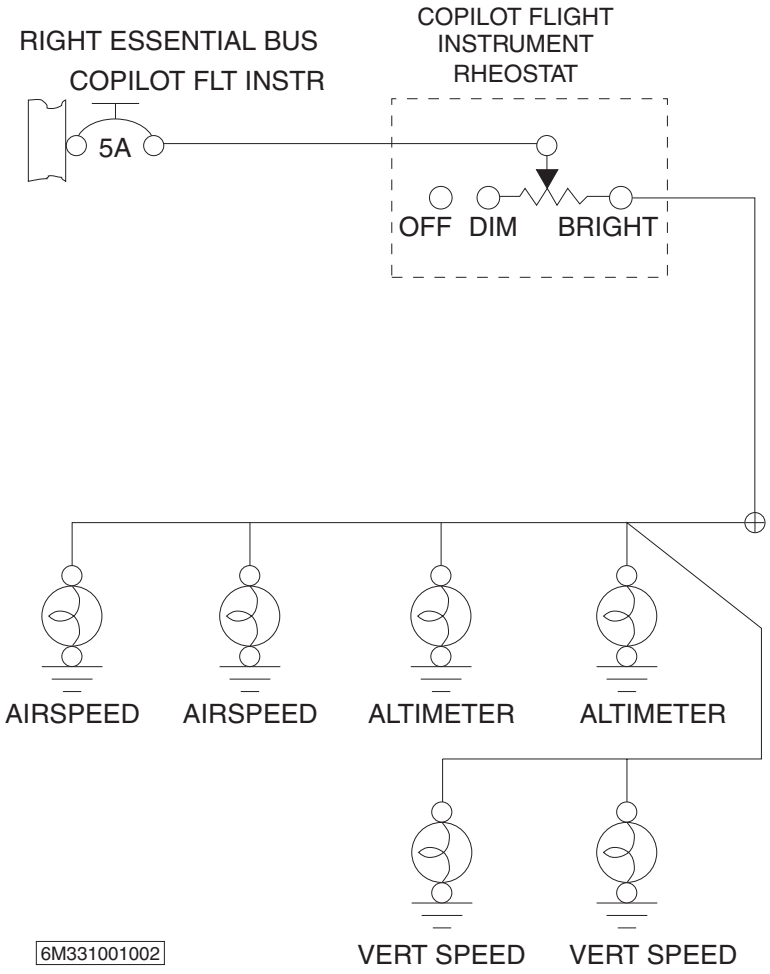
(2) Connections and contact assemblies are mounted behind the instrument panel. A control rheostat located on the forward left console provides ON/OFF switching and dimming control. Circuit protection for the flight instrument lighting system is provided by a five amp circuit breaker on the left essential bus.



Pilot Flight Instrument Lights
Figure 1

B. Copilot's Flight Instrument Lighting

(1) The copilot's flight instruments are similarly lighted as the pilot's instruments. A control rheostat located on the forward right console provides ON/OFF switching and dimming control. A five amp circuit breaker located on the right essential bus provides for circuit protection.



Copilot Flight Instrument Lights
Figure 2

FLIGHT INSTRUMENT LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Flight Instrument Lights

A. Removal – Pilot's Flight Instrument Control Rheostat

CAUTION: ENSURE ELECTRICAL POWER TO PILOT'S FLIGHT INSTRUMENT CONTROL RHEOSTAT IS OFF.

(1) Loosen set screw securing rheostat control knob and remove knob.

(2) Remove nut and washers securing rheostat to console.

(3) Gain access to the rheostat electrical wiring connections by removing screws securing forward left access panel and removing panel.

(4) Mark and cut electrical wiring attached to rheostat.

(5) Remove rheostat.

B. Installation – Pilot's Flight Instrument Control Rheostat

CAUTION: ENSURE ELECTRICAL POWER TO PILOT'S FLIGHT INSTRUMENT CONTROL RHEOSTAT IS OFF.

(1) Solder electrical wiring to control rheostat.

(2) Position rheostat on console and secure with nut and washers.

(3) Position rheostat control knob and secure with screw set.

(4) Position forward left console access panel and secure with attaching screws.

(5) Perform operational check.

C. Removal – Copilot's Flight Instrument Control Rheostat

CAUTION: ENSURE ELECTRICAL POWER TO COPILOT'S FLIGHT INSTRUMENT CONTROL RHEOSTAT IS OFF.

(1) Loosen set screw securing rheostat control knob and remove knob.

(2) Remove nut and washers securing rheostat to console.

(3) Gain access to the rheostat electrical wiring connections by removing screws securing forward left access panel and removing panel.

(4) Mark and cut electrical wiring attached to rheostat.

(5) Remove rheostat.

D. Installation – Copilot's Flight Instrument Control Rheostat

CAUTION: ENSURE ELECTRICAL POWER TO PILOT'S FLIGHT INSTRUMENT CONTROL RHEOSTAT IS OFF.

(1) Solder electrical wiring to control rheostat.

(2) Position rheostat on console and secure with nut and washers.

(3) Position rheostat control knob and secure with screw set.

(4) Position forward left console access panel and secure with attaching screws.

(5) Perform operational check.

E. Removal – Post light Bulb

(1) Pull lamp hood straight out of its retainer.

(2) Remove light bulb.

F. Installation – Post light Bulb

(1) Install light bulb in post light receptacle.

(2) Snap hood of post light assembly in place.

G. Removal – Post light Assembly

(1) Remove screws securing the flight instrument adjacent to post light and withdraw instrument enough to gain access to post light mounting nuts on aft side of panel.

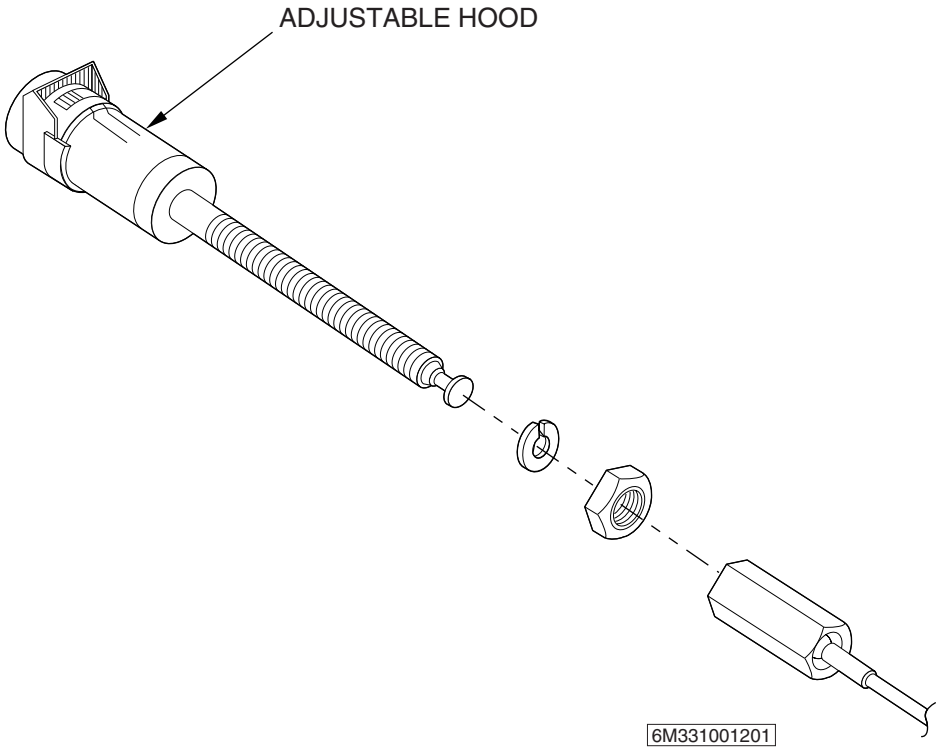
(2) Remove plastic nut and electrical contact assembly from post light. Remove mounting nut and washers.

H. Installation – Post light Assembly

(1) Position post light assembly and secure to panel with washers and mounting nuts.

(2) Connect electrical contact assembly to post light and secure with plastic nut.

(3) Position flight instrument and secure with mounting screws.



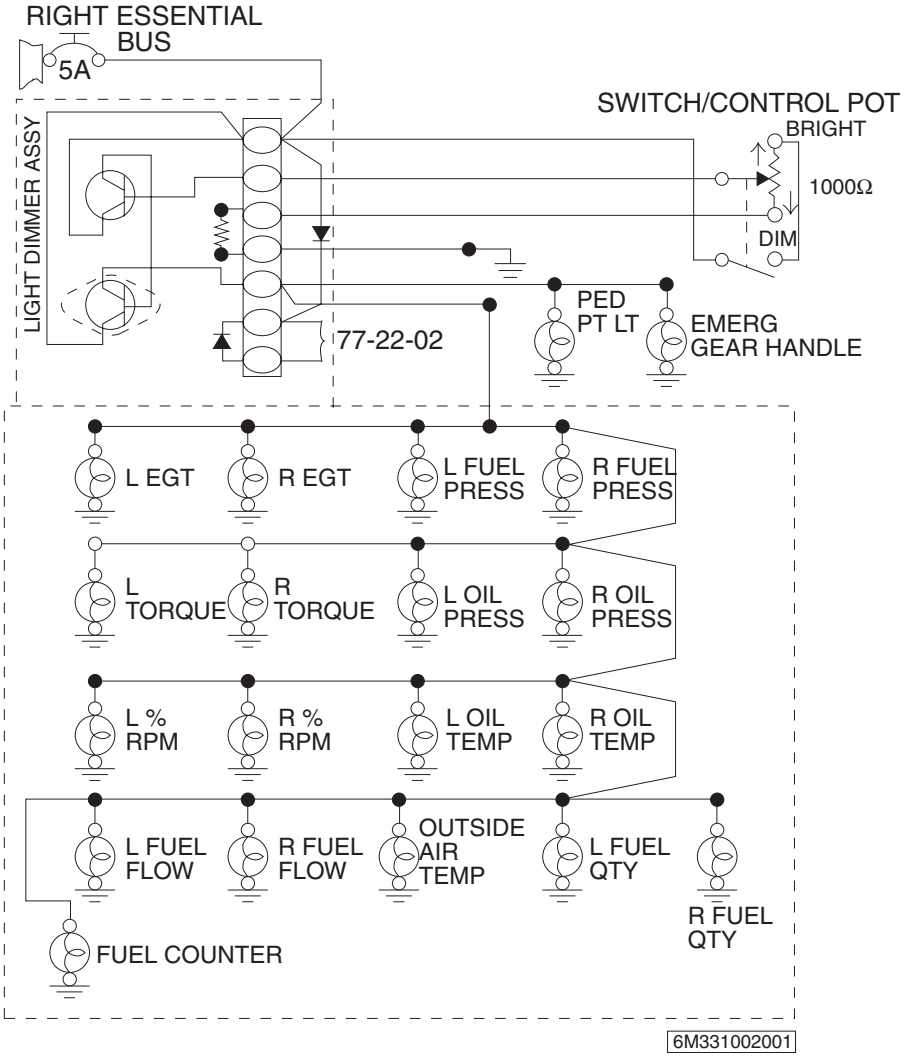
Flight Instrument Adjustable Post Light
Figure 201

CENTER INSTRUMENT LIGHTS – DESCRIPTION AND OPERATION

1. General

A. The center instrument lighting system (see Figure 1) consists of post light assemblies, dimmer control assembly, switch/control potentiometer, and circuit breaker.

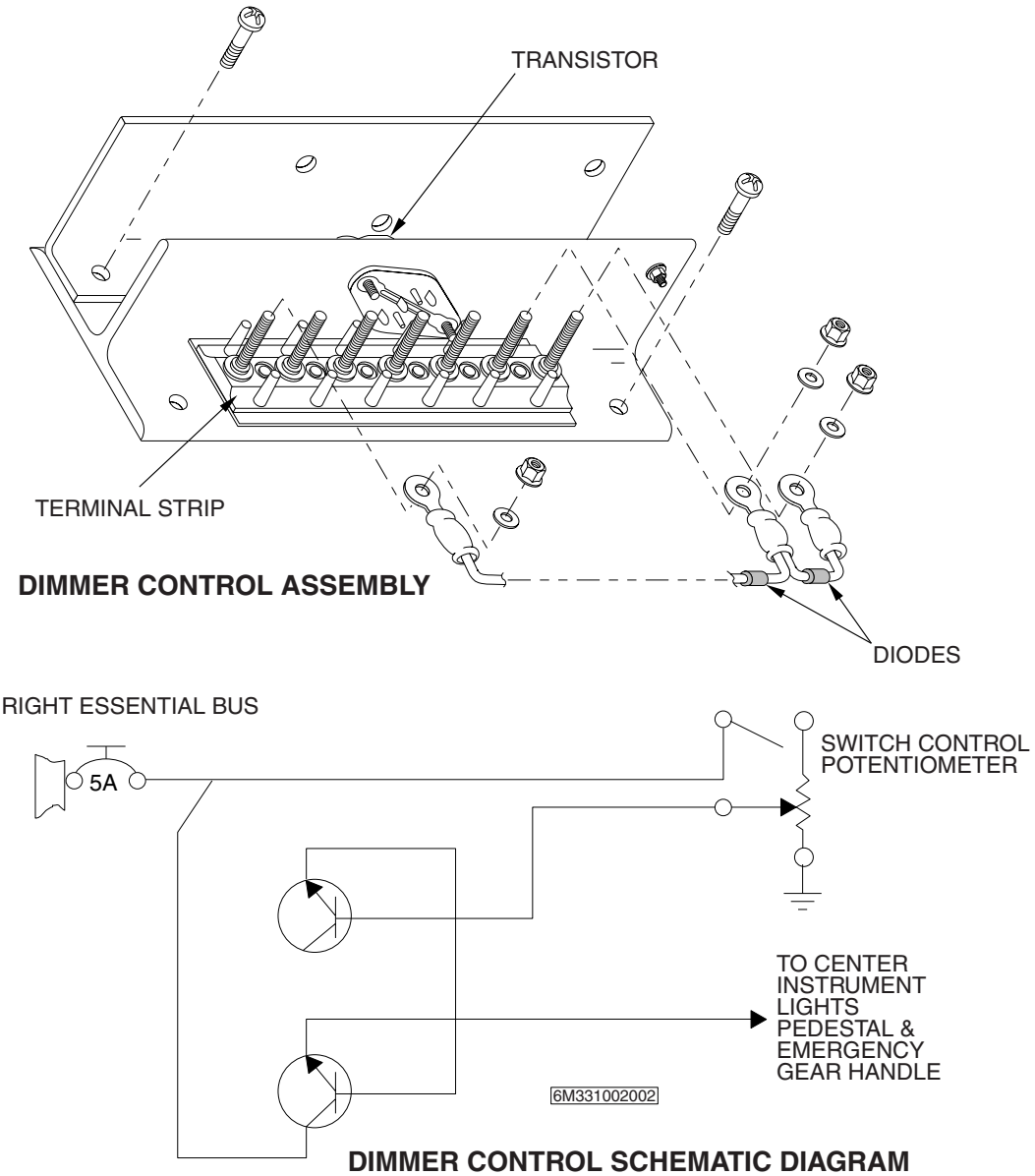
B. The post light assemblies are adjustable, hooded, post mounted lights which are designed to provide illumination downward and inward over the face of the instrument. Terminal connections and contact assemblies for the post lights are mounted behind the instrument panel.



Center Instrument Lights
Figure 1

CENTER INSTRUMENT LIGHTS – DESCRIPTION AND OPERATION

C. A dimmer control assembly located under the forward left console (see Figures 202 and 203) controls center instrument dimming. The light dimmer control assembly incorporates control transistor (2N492), a power transistor (2N305), and a control pot (53C1000). The 1000 ohm control pot increases or decreases current to the base of the control transistor, which feeds current to the base of the power transistor. Current from the power transistor is fed to the center instrument panel post lights. Energizing current for the dimmer control assembly is supplied by a five amp circuit breaker on the 28 VDC essential bus.



Center Instrument Lights
Dimmer Control Assembly
Figure 2

CENTER INSTRUMENT LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Center Instrument Lights

A. Removal – Post Light Assembly

(1) Remove screws securing the center instrument adjacent to post light and withdraw instrument enough to gain access to post light mounting nuts on aft side of panel.

(2) Remove plastic nut and electrical contact assembly from post light. Remove mounting nut and washers.

(3) Remove post light assembly.

B. Installation – Post Light Assembly

(1) Position post light assembly and secure to panel with washers and mounting nut.

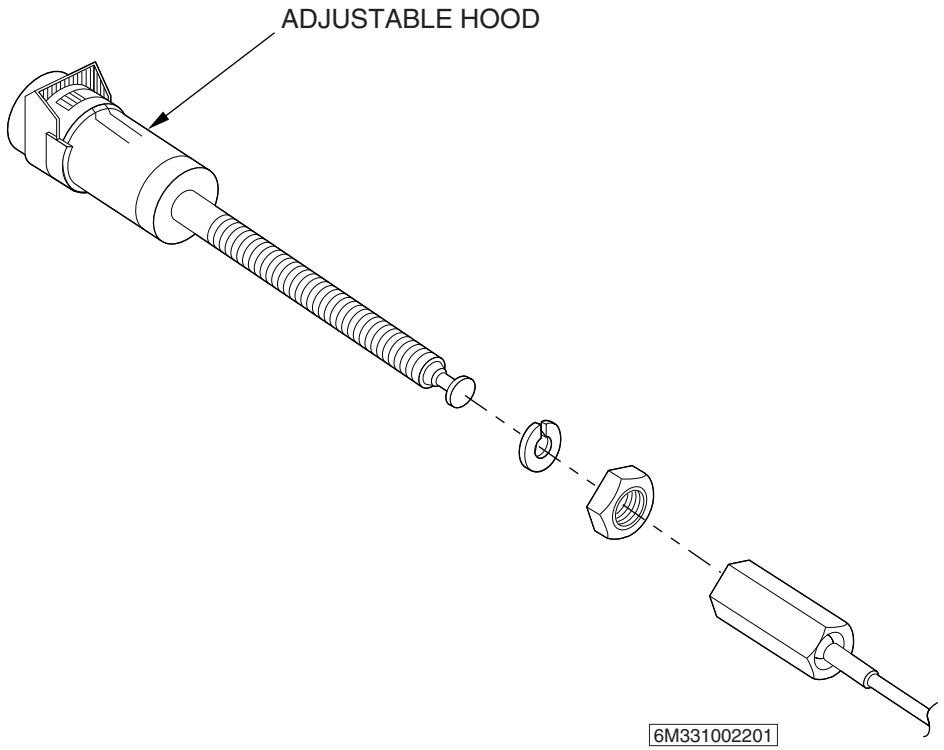
(2) Connect electrical contact assembly to post light and secure with plastic nut.

(3) Position center instrument and secure with mounting screws.

C. Removal – Post Light Bulb

(1) Pull lamp hood straight out of its retainer.

(2) Remove light bulb.



Center Instrument Post Light Assembly Figure 201

D. Installation – Post Light Bulb

(1) Install light bulb in post light receptacle.

(2) Snap hood of post light assembly in place.

E. Removal – Dimmer Assembly

CAUTION: ENSURE ELECTRICAL POWER TO DIMMER ASSEMBLY IS OFF.

(1) Gain access to dimmer assembly by removing attaching screws and removing forward left console access panel.

(2) Mark and remove electrical wiring from dimmer assembly terminal strip.

(3) Remove screws securing dimmer assembly and remove dimmer assembly from console.

F. Installation – Dimmer Assembly

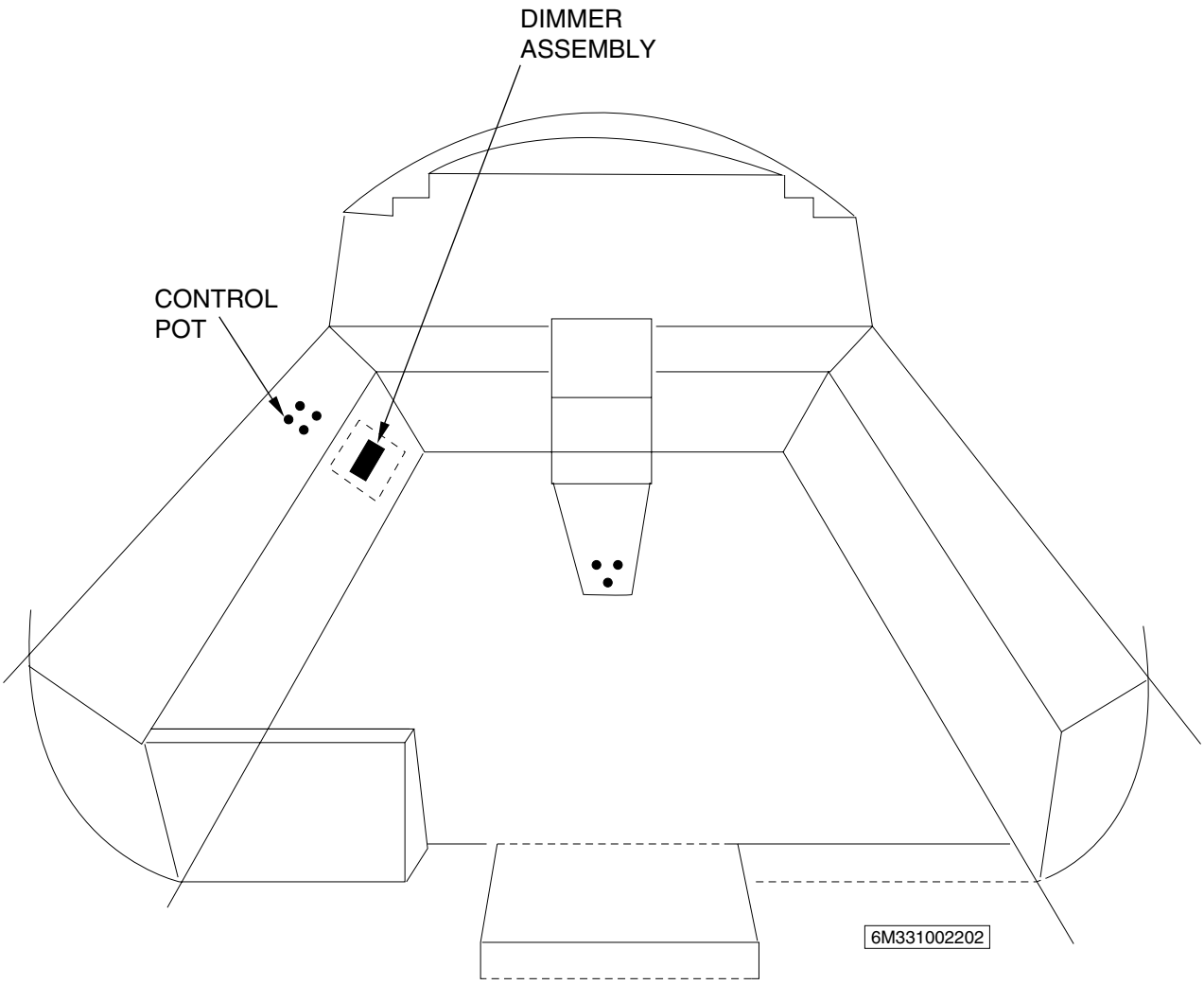
CAUTION: ENSURE ELECTRICAL POWER TO DIMMER ASSEMBLY IS OFF.

(1) Position dimmer assembly and secure with attaching screws.

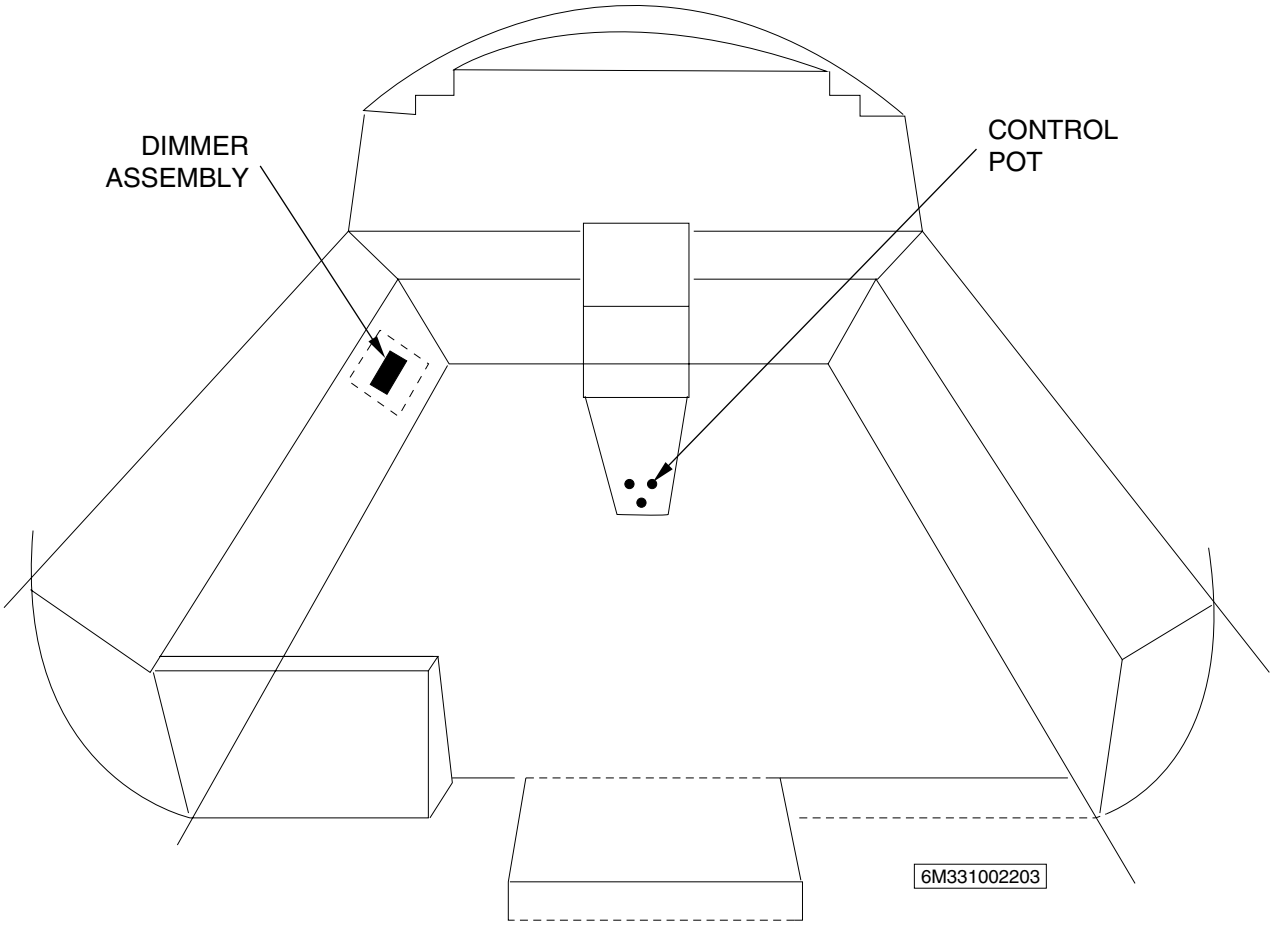
(2) Connect electrical wiring to terminal strip of dimmer assembly.

(3) Position console access panel and secure with attaching screws.

(4) Perform operational check.



Center Instrument Lights Dimmer Assembly
Figure 202



Center Instrument Lights Dimmer Assembly
Figure 203

G. Removal – Center Instrument Control Potentiometer/Switch

CAUTION: ENSURE ELECTRICAL POWER TO CENTER INSTRUMENT POTENTIOMETER/SWITCH IS OFF.

(1) Loosen set screw securing center instrument potentiometer/switch control knob and remove knob.

(2) Remove nut and washers securing potentiometer/switch to console.

(3) Gain access to the potentiometer/switch electrical wiring connections by removing screws securing forward left console access panel and removing panel.

(4) Mark and cut electrical wiring attached to potentiometer/switch.

(5) Remove potentiometer/switch.

H. Installation – Center Instrument Control Potentiometer/Switch

CAUTION: ENSURE ELECTRICAL POWER TO CENTER INSTRUMENT POTENTIOMETER/SWITCH IS OFF.

(1) Solder electrical wiring to potentiometer/switch.

(2) Position and secure potentiometer/switch to console with attaching nut and washers.

(3) Position and secure potentiometer/switch control knob with set screw.

CENTER INSTRUMENT LIGHTS – MAINTENANCE PRACTICES

I. Removal – Center Instrument Potentiometer/Switch

CAUTION: ENSURE ELECTRICAL POWER TO CENTER INSTRUMENT POTENTIOMETER/SWITCH IS OFF.

(1) Loosen set screw securing potentiometer/switch control knob and remove knob.

(2) Remove nut and washers securing potentiometer/switch.

(3) Gain access to the potentiometer/switch electrical wiring connections by removing screws securing pedestal panel and withdrawing panel enough to cut and detach electrical wiring attached to potentiometer/switch.

(4) Mark and cut electrical wiring attached to potentiometer/switch.

(5) Remove potentiometer/switch.

J. Installation – Center Instrument Potentiometer/Switch

CAUTION: ENSURE ELECTRICAL POWER TO CENTER INSTRUMENT POTENTIOMETER/SWITCH IS OFF.

(1) Solder electrical wiring to potentiometer/switch.

(2) Position potentiometer/switch on pedestal panel and secure with attaching nut and washers.

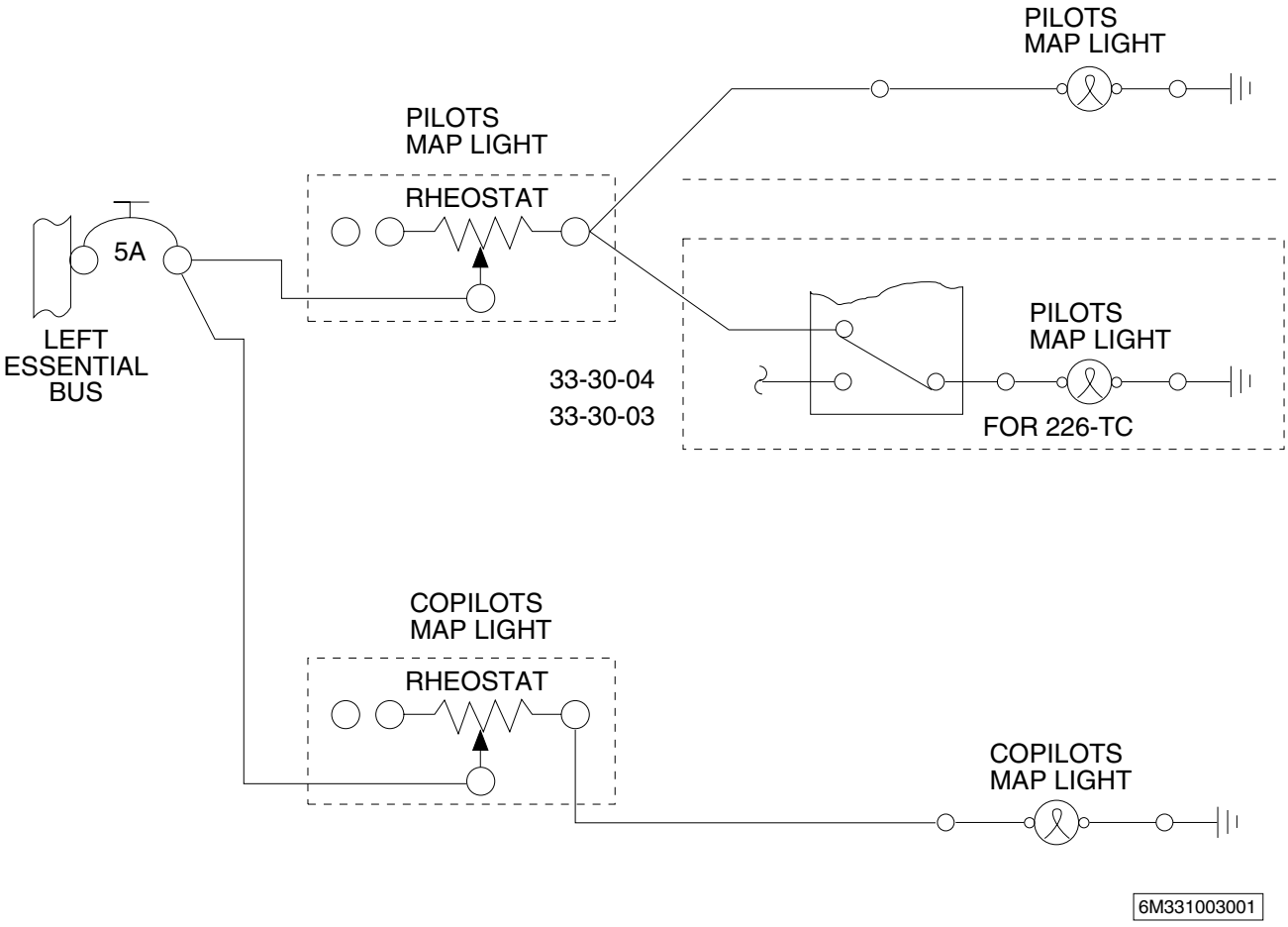
(3) Position control knob on potentiometer/switch and secure with set screw.

(4) Secure pedestal panel with attaching screws.

COCKPIT OVERHEAD MAP LIGHTS – DESCRIPTION AND OPERATION

1. General

A. Cockpit overhead map lighting consists of a flush-mounted light assembly overhead of the pilot's seat and a flush-mounted light assembly overhead of the copilot's seat. The pilot's map light is controlled by a rheostat mounted on the forward left console. A rheostat mounted on the forward right console controls the copilot's map light. Energizing current for the map lights is supplied from a five amp circuit breaker on the left essential bus. For map light operation as part of the auxiliary lighting system, see Section 33-30-04.



Cockpit Overhead Map Lights
Figure 1

COCKPIT OVERHEAD MAP LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Cockpit Overhead Map Lights

A. Removal – Overhead Map Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT IS OFF.

(1) Remove screws securing lens retainer and lens.

(2) Remove lens retainer and lens.

(3) Remove light assembly mounting screws from mounting plate assembly.

(4) Withdraw light assembly and disconnect electrical wiring.

(5) Remove light assembly.

B. Installation – Overhead Map Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT IS OFF.

(1) Connect electrical wiring to light assembly.

(2) Position light assembly and secure to mounting plate with screws.

(3) Position lens. Secure lens and lens retainer with screws.

C. Removal – Pilot's Overhead Map Light Control Rheostat

CAUTION: ENSURE ELECTRICAL POWER TO OVERHEAD MAP LIGHT CONTROL RHEOSTAT IS OFF.

(1) Loosen set screw securing rheostat control knob and remove knob.

(2) Remove nut and washers securing rheostat.

(3) Gain access to the rheostat electrical wiring connections by removing screws securing forward left console access panel and removing panel.

(4) Mark and cut electrical wiring attached to rheostat.

(5) Remove rheostat.

D. Installation – Pilot's Overhead Map Light Control Rheostat

(1) Solder electrical wiring to rheostat.

(2) Position rheostat on console and secure with nut and washers.

(3) Position control knob and secure with set screw.

(4) Position forward left console access panel and secure with attaching screws.

E. Removal – Copilot's Overhead Map Light Control Rheostat

CAUTION: ENSURE ELECTRICAL POWER TO OVERHEAD MAP LIGHT CONTROL RHEOSTAT IS OFF.

(1) Loosen set screw securing rheostat control knob and remove knob.

(2) Remove nut and washers securing rheostat.

(3) Gain access to the rheostat electrical wiring connections by removing screws securing forward right console access panel and removing panel.

(4) Mark and cut electrical wiring attached to rheostat.

(5) Remove rheostat.

F. Installation – Copilot's Overhead Map Light Control Rheostat

(1) Solder electrical wiring to rheostat.

(2) Position rheostat on console and secure with nut and washers.

(3) Position control knob and secure with set screw.

(4) Position forward right console access panel and secure with attaching screws.

G. Removal – Cockpit Overhead Map Light Bulb

(1) Remove screws securing lens retainer.

(2) Remove lens retainer.

(3) Remove lens.

(4) Remove light bulb.

H. Installation – Cockpit Overhead Map Light Bulb

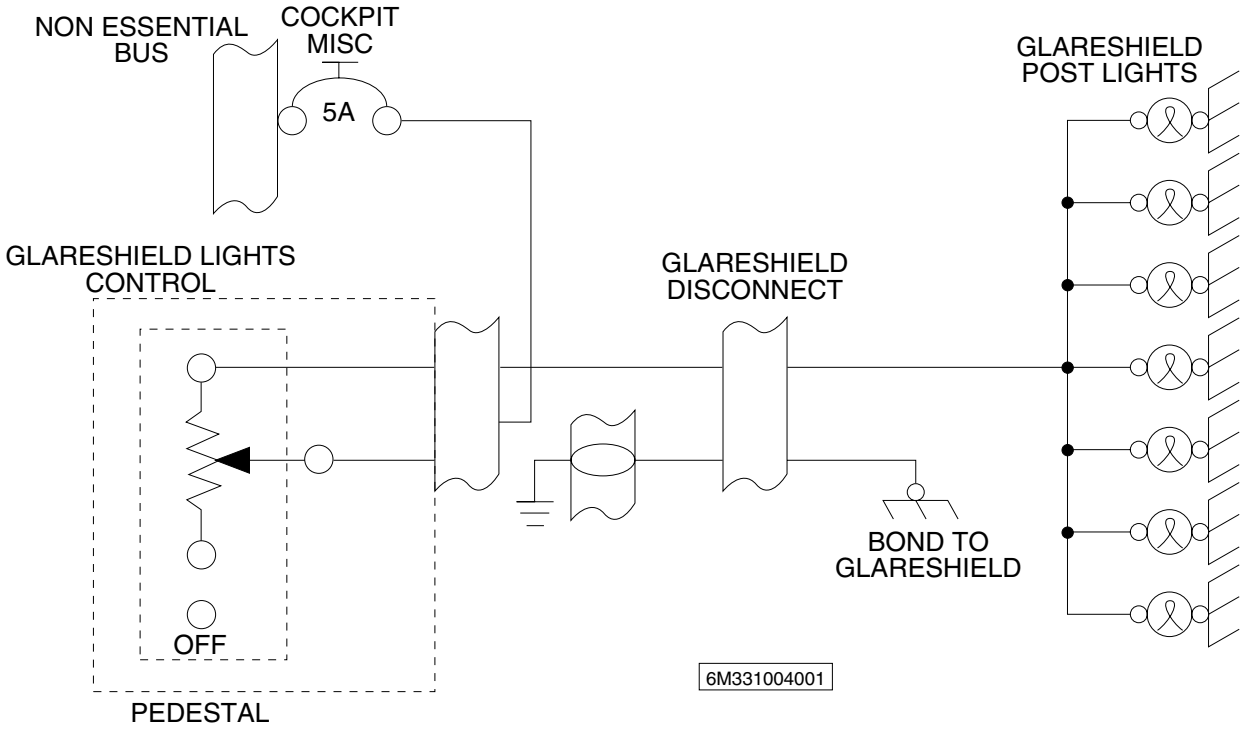
(1) Insert light bulb in receptacle and turn until properly seated.

(2) Position lens retainer and secure with screws.

GLARESHIELD LIGHTS – DESCRIPTION AND OPERATION

1. General

A. The glareshield is illuminated by seven post lights evenly spaced around the upper perimeter of glareshield. On early model SA226 series aircraft, glareshield lights are controlled by a rheostat located on the forward right console. On later model SA226 series aircraft, the control rheostat is located on the lower pedestal. Energizing current for the glareshield lighting is supplied by a five amp circuit breaker on the nonessential bus.



Glareshield Lights
Figure 1

GLARESHIELD LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Glareshield Lights

A. Removal – Glareshield Control Rheostat (T201-248, AT001-033, TC201-213)

CAUTION: ENSURE ELECTRICAL POWER TO GLARESHIELD CONTROL RHEOSTAT IS OFF.

(1) Loosen set screw securing rheostat control knob and remove knob.

(2) Remove nut and washers securing rheostat to console.

(3) Gain access to the rheostat electrical wiring connections by removing screws securing forward left console access panel and removing panel.

(4) Mark and cut electrical wiring attached to rheostat.

(5) Remove rheostat.

B. Installation – Glareshield Control Rheostat (T201-248, AT001-033, TC201-213)

CAUTION: ENSURE ELECTRICAL POWER TO GLARESHIELD CONTROL RHEOSTAT IS OFF.

(1) Solder electrical wiring to rheostat.

(2) Position rheostat on console and secure with nut and washers.

(3) Position control knob on rheostat and secure with set screw.

(4) Position forward left console access panel and secure with attaching screws.

C. Removal – Glareshield Control Rheostat (T249-999, AT034-999, TC214-999)

CAUTION: ENSURE ELECTRICAL POWER TO GLARESHIELD CONTROL RHEOSTAT IS OFF.

(1) Loosen set screw securing rheostat control knob and remove knob.

(2) Remove nut and washers securing rheostat.

(3) Gain access to the rheostat electrical wiring connections by removing screws securing pedestal panel and withdrawing panel enough to cut and detach electrical wiring attached to glareshield control rheostat.

(4) Mark and cut electrical wiring attached to rheostat.

(5) Remove rheostat.

D. Installation – Glareshield Control Rheostat (T249-999, AT034-999, TC214-999)

CAUTION: ENSURE ELECTRICAL POWER TO GLARESHIELD CONTROL RHEOSTAT IS OFF.

(1) Solder electrical wiring to rheostat.

(2) Position rheostat on lower pedestal panel and secure with attaching nut and washers.

(3) Position control knob on rheostat and secure with set screw.

(4) Secure pedestal panel with attaching screws.

E. Removal – Post Light Bulb

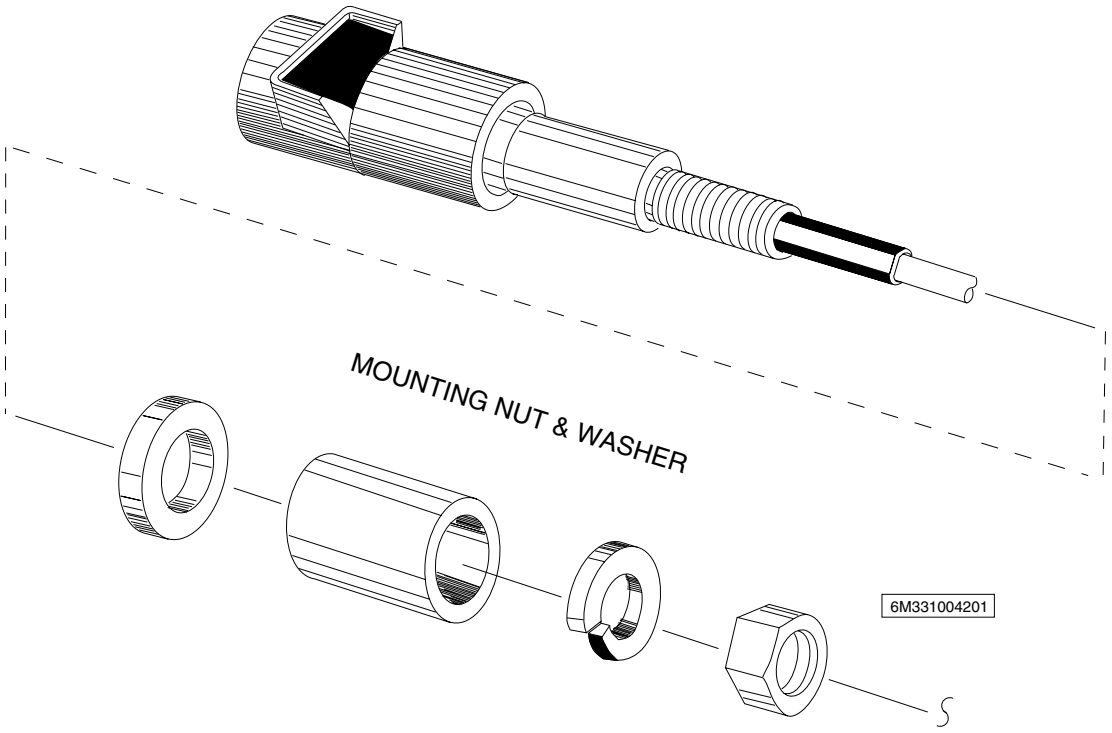
(1) Pull lamp hood straight out of it's retainer.

(2) Remove light bulb.

F. Installation – Post Light Bulb

(1) Install light bulb in post light receptacle.

(2) Snap hood of post light assembly in place.



Adjustable Glareshield Post Lights
Figure 201

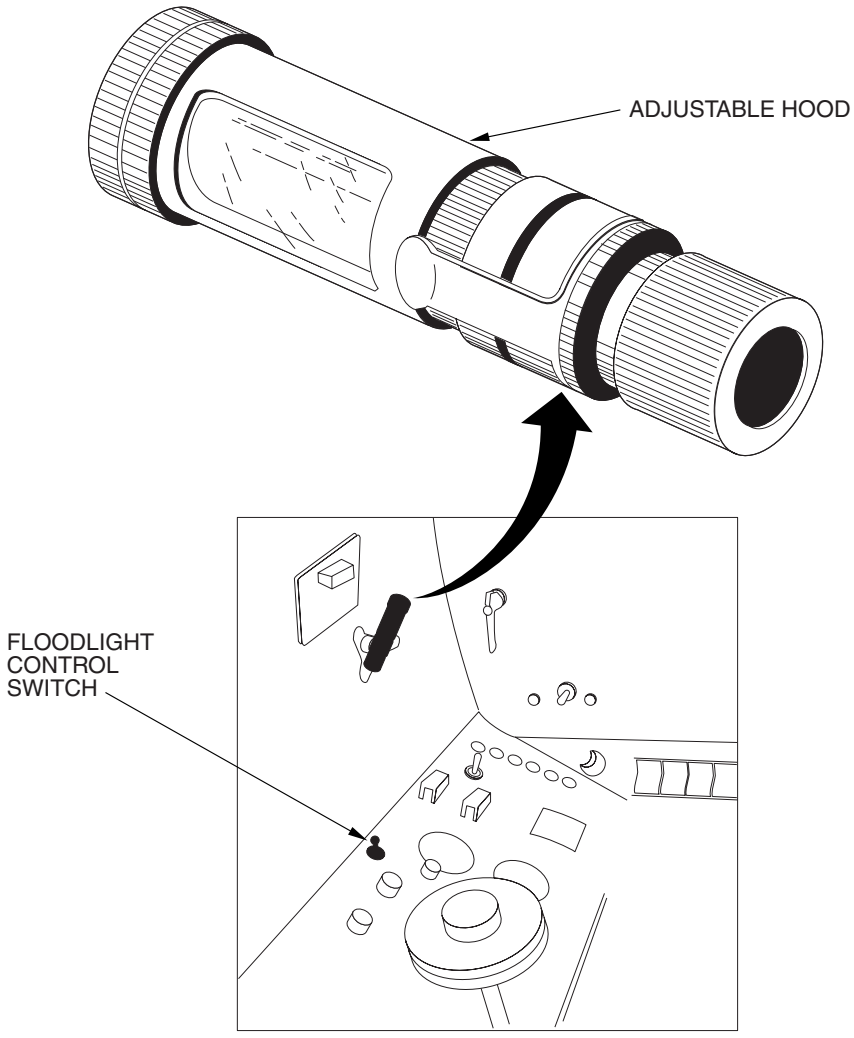
CONSOLE FLOOD LIGHT – DESCRIPTION AND OPERATION

1. General

A. Early SA226-T and AT aircraft are equipped with three adjustable console floodlights spaced evenly along the vertical side panel above the left and right consoles (see Figure 1).

NOTE: SA226-TC aircraft are equipped with only two floodlights on each side.

B. Lamp hoods on the console floodlights rotate completely around the socket assembly permitting directional control of the beam. An inner barrel contains a red colored filter and a white filter that can be rotated alternately in front of the beam opening. The console floodlight uses a miniature, single contact bayonet base lamp. A switch on the forward left console panel controls the floodlights and energizing current is supplied by a five amp circuit breaker on the 28 VDC nonessential bus.



Console Floodlight
Figure 1

CONSOLE FLOOD LIGHT – MAINTENANCE PRACTICES

1. Removal/Installation – Console Floodlight

A. Removal – Console Floodlight Light Bulb

(1) Pull hood straight out free of spring retainers.

(2) Withdraw light bulb from socket.

B. Installation – Console Floodlight Light Bulb

(1) Position light bulb in socket.

(2) Snap hood of light assembly in place.

C. Removal – Console Floodlight Assembly

(1) Remove hood.

(2) Unscrew back nut on body of light assembly.

(3) Remove clip and pull body of assembly of bracket.

(4) Remove two screws securing bracket to vertical side panel.

(5) Disconnect electrical wiring from light assembly.

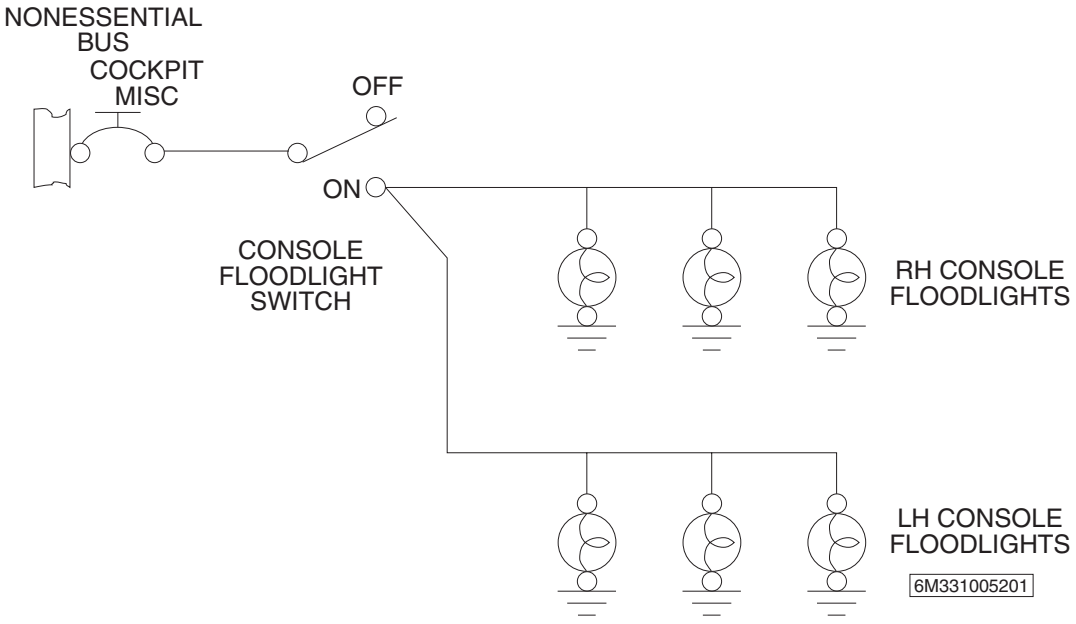
D. Installation – Console Floodlight Assembly

(1) Connect electrical wiring to light assembly.

(2) Secure bracket to vertical side panel with two screws.

(3) Assemble body of light assembly and clip.

(4) Screw back nut on body of light assembly.



Console Floodlights
Figure 201

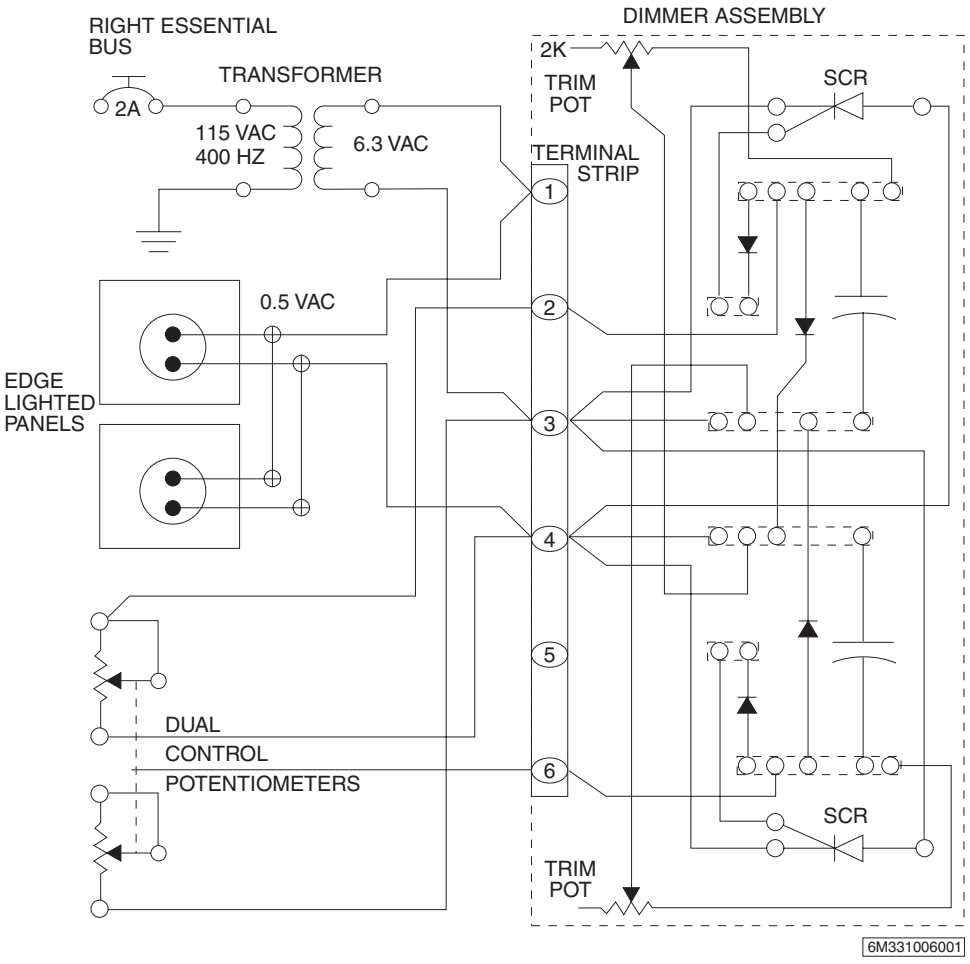
EDGE LIGHTED PANELS – DESCRIPTION AND OPERATION

1. General

A. Edge lighted panels provide lighting for console, switch panel, and pedestal instruments. Edge lighted panels are constructed with miniature light bulbs embedded in the panel plexiglass. Light rays from the embedded bulbs are directed through clear portions of the plexiglass causing instrument illumination.

B. Each lighted panel is controlled by a potentiometer/switch and dimmer assembly (see Figure 201). The dimmer assemblies supply 0 to five VAC to the edge lighted panel. Each dimmer assembly consists of two silicon controlled rectifiers (SCR'S), trimmer potentiometers, and electronic circuitry. A dual control potentiometer increases or decreases voltage to the gates of the silicon controlled rectifiers. Light dimming is accomplished by the triggering and conductivity characteristics of the silicon controlled rectifiers. When the control potentiometer increases voltage to the gates of the silicon controlled rectifier, conductivity of the SCRs decrease and light dimming occurs. When the control potentiometer decreases voltage to the gates of the SCRs, conductivity increases and panel lights become brighter.

C. Each dimmer assembly is supplied with 6.3 VAC from a 115/6V step-down transformer located within the aft right console. Each transformer is energized from a two amp circuit breaker on the right essential 115 VAC bus.



Edge Lighted Panels
Figure 1

EDGE LIGHTED PANELS – MAINTENANCE PRACTICES

1. Removal/Installation – Edge Lighted Panels

A. Removal – Edge Lighted Panel Control Potentiometer/Switch Forward Right Console

CAUTION: ENSURE ELECTRICAL POWER TO EDGE LIGHTED PANEL POTENTIOMETER/SWITCH IS OFF.

(1) Loosen set screw securing edge lighted panel potentiometer/switch control knob and remove knob (see Figure 201).

(2) Remove nut and washers securing potentiometer/switch to console.

(3) Gain access to the potentiometer/switch electrical wiring connections by removing screws securing forward right console access panel and removing panel.

(4) Mark and cut electrical wiring attached to potentiometer/switch.

(5) Remove potentiometer/switch.

B. Installation – Edge Lighted Panel Control Potentiometer/Switch

CAUTION: ENSURE ELECTRICAL POWER TO EDGE LIGHTED PANEL POTENTIOMETER/SWITCH IS OFF.

(1) Solder electrical wiring to potentiometer/switch.

(2) Position and secure potentiometer/switch to console with attaching nut and washers.

(3) Position and secure potentiometer/switch control knob with set screw.

(4) Position forward right console access panel and secure with attaching screws.

C. Removal – Edge Lighted Panel Control Potentiometer/Switch Forward Left Console

CAUTION: ENSURE ELECTRICAL POWER TO EDGE LIGHTED PANEL POTENTIOMETER/SWITCH IS OFF.

(1) Loosen set screw securing edge lighted panel potentiometer/switch control knob and remove knob.

(2) Remove nut and washers securing potentiometer/switch to console.

(3) Gain access to the potentiometer/switch electrical wiring connections by removing screws securing forward left console access panel and removing panel.

(4) Mark and cut electrical wiring attached to potentiometer/switch.

(5) Remove potentiometer/switch.

D. Installation – Edge Lighted Panel Control Potentiometer/Switch

CAUTION: ENSURE ELECTRICAL POWER TO EDGE LIGHTED PANEL POTENTIOMETER/SWITCH IS OFF.

(1) Solder electrical wiring to potentiometer/switch.

(2) Position and secure potentiometer/switch to console with attaching nut and washers.

(3) Position and secure potentiometer/switch control knob with set screw.

(4) Position forward left console access panel and secure with attaching screws.

E. Removal – Edge Lighted Panel Potentiometer/Switch Pedestal Panel

CAUTION: ENSURE ELECTRICAL POWER TO POTENTIOMETER/SWITCH IS OFF.

(1) Loosen set screw securing potentiometer/switch control knob and remove knob.

(2) Remove nut and washers securing potentiometer/switch.

(3) Gain access to the potentiometer/switch electrical wiring connections by removing screws securing pedestal panel and by withdrawing panel enough to cut and detach electrical wiring attached to potentiometer/switch.

(4) Mark and cut electrical wiring attached to potentiometer/switch.

(5) Remove potentiometer/switch.

F. Installation – Edge Lighted Panel Potentiometer/Switch

CAUTION: ENSURE ELECTRICAL POWER TO EDGE LIGHTED PANEL POTENTIOMETER/SWITCH IS OFF.

(1) Solder electrical wiring to potentiometer/switch.

(2) Position and potentiometer/switch on pedestal panel and secure with attaching nut and washers.

(3) Position control knob on potentiometer/switch and secure with set screw.

(4) Secure pedestal panel with attaching screws.

G. Removal – Edge Lighted Panel

CAUTION: ENSURE ELECTRICAL POWER TO PANEL IS OFF.

(1) Remove knobs and switches from overlay/panel.

(2) Remove mounting screws securing overlay/panel.

(3) Mark and detach electrical wiring from overlay/panel.

(4) Remove overlay/panel.

H. Installation – Edge Lighted Panel

(1) Connect electrical wiring to overlay/panel.

(2) Position and secure overlay/panel with mounting screws.

(3) Position and secure knobs and switches to overlay/panel.

I. Removal – Edge Lighted Panel Power Supply Transformer Assembly

(1) Ensure all power is off.

(2) Gain access to transformer assembly by removing right aft console access door (see Figure 201).

(3) Mark and detach electrical wiring from terminal strip on assembly.

(4) Remove attaching screws and remove transformer assembly.

J. Installation – Edge Lighted Panel Power Supply Transformer Assembly

(1) Position transformer assembly and secure with attaching screws.

(2) Position console access door and secure with attaching screws.

(3) Perform operational check.

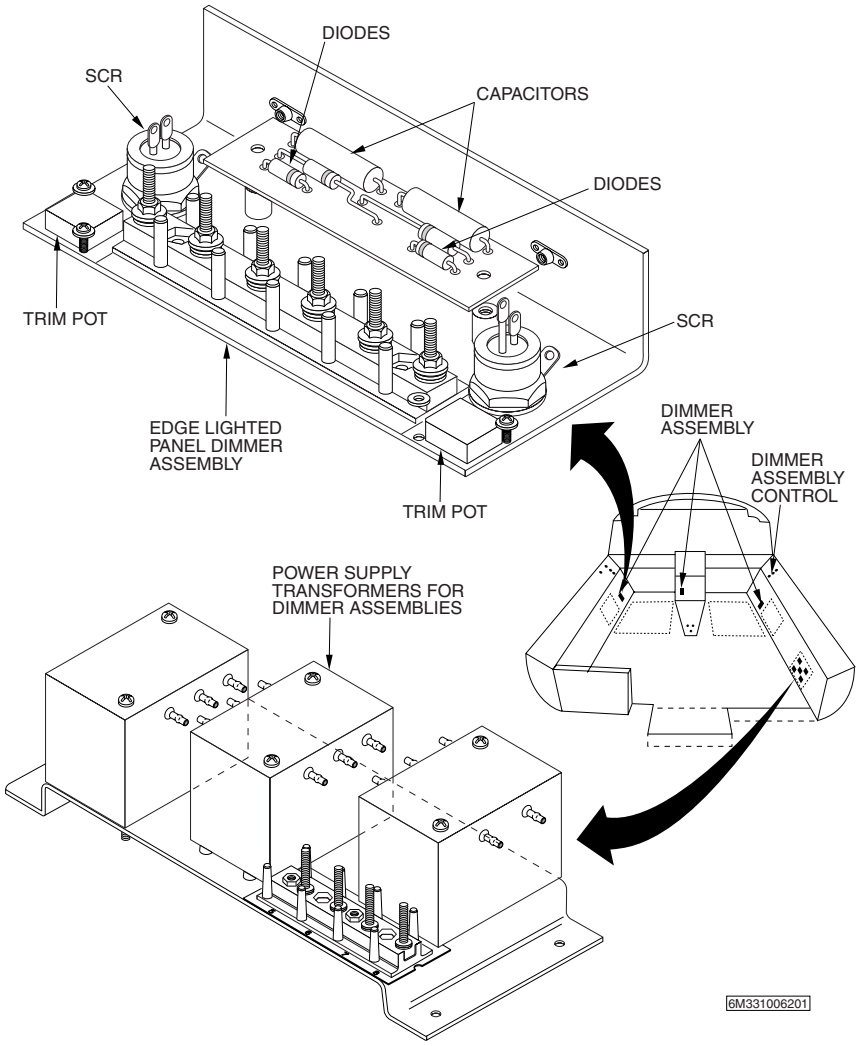
K. Removal – Dimmer Assembly Right and Left Consoles

(1) Gain access to dimmer assembly by removing forward left or right console access door (see Figure 201).

(2) Remove screws securing dimmer assembly.

(3) Mark and detach wiring from terminal strip on dimmer assembly.

(4) Remove dimmer assembly.



Edge Lighted Panel Dimmer Assembly
Figure 201

L. Installation – Dimmer Assembly Right and Left Consoles

(1) Secure electrical wiring to assembly terminal strip.

(2) Position dimmer assembly and secure to inner surface of console with attaching screws.

(3) Position console access door and secure with attaching screws.

(4) Perform operational check.

M. Removal – Dimmer Assembly Pedestal

(1) Gain access to dimmer assembly by removing attaching screws and removing center edge lighted panel from pedestal (see Figure 201).

(2) Remove attaching screws and remove center access panel from pedestal.

(3) Remove screws securing dimmer assembly to inside left surface of pedestal.

(4) Mark and detach electrical wiring from dimmer assembly terminal strip.

(5) Withdraw dimmer assembly from pedestal.

N. Installation – Dimmer Assembly Pedestal

(1) Secure electrical wiring to dimmer assembly terminal strip.

(2) Position dimmer assembly to inside left surface of pedestal and secure with attaching screws.

(3) Position pedestal access panel and secure with attaching screws.

(4) Position pedestal lighted panel and secure with attaching screws.

(5) Perform operational check.

2. Adjustment/Test – Edge Lighted Panels

A. Adjustment – (AC Dimmer Assembly)

(1) Remove appropriate access panel (see Figure 201).

(2) Remove dimmer assembly but do not remove electrical harness.

(3) Withdraw dimmer assembly sufficiently to gain access to trimmer pot.

(4) Connect APU; place battery switch ON; place inverter ON.

(5) Turn dual control pot approximately 1/8 turn clockwise from OFF position.

(6) Attach a voltmeter to terminals 1 and 4 of dimmer assembly terminal strip.

(7) Adjust each trimmer pot until the voltmeter indicates a voltage reading then back off trimmer pot until 0 VAC is observed on voltmeter.

(8) Turn dual control pot until a maximum reading is observed on voltmeter (approximately 5 VAC; lights full bright) and allow to operate five minutes. Then turn control pot back to 0 VAC on voltmeter (lights off).

(9) Repeat steps **2.**, A., (5) through (8). If voltage does not return to 0 VAC, repair or replace dimmer assembly.

(10) Position and secure dimmer assembly with attaching screws. Position and secure access panel with attaching screws.

B. Operational/Test – Edge Lighted Panels

(1) Connect APU to aircraft.

(2) Turn battery and inverter switches ON.

(3) Turn appropriate light control full counterclockwise (OFF). Observe edge lighted panels are not illuminated.

(4) Turn appropriate light control clockwise. Observe edge lighted panels increase in brilliance evenly (not in steps) to full bright.

CABIN LIGHTING – DESCRIPTION AND OPERATION

1. General

A. The passenger compartment lighting consists of reading lights, aisle lights, entrance lights, and lighted signs.

B. Cabin window fluorescent light assemblies are provided for SA226-AT aircraft.

C. SA226-AT and T aircraft are equipped with a lavatory fluorescent light fixture.

CABIN READING LIGHTS – DESCRIPTION AND OPERATION

1. General

A. SA226-AT and T aircraft are equipped with an adjustable reading light assembly located above each passenger seat to provide a source of diffused illumination for the seat occupant.

B. On early SA226-AT series aircraft, the reading light is individually controlled by an ON/OFF switch mounted on the light assembly. On later model SA226-AT series aircraft, all of the reading lights are controlled by the CABIN AISLE light switch located on the right switch panel.

C. Energizing current to the reading light is supplied by a ten amp circuit breaker on the 28 VDC nonessential bus.

D. SA226-TC series aircraft are equipped with reading light assemblies. In addition to the single ON/OFF switch on individual light assemblies, ON/OFF control of all the reading lights is provided by the CABIN AISLE switch located on the right switch panel.

E. Energizing current to the reading lights is supplied by a 15 amp circuit breaker on the 28 VDC nonessential bus.

CABIN READING LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Cabin Reading Lights

NOTE: Reading light assemblies are attached to the interior paneling sections of the aircraft. To remove a reading light assembly, the entire panel section attached to the reading light assembly must be removed.

A. Removal – Cabin Reading Light Assembly

(1) Remove applicable panel.

(2) Disconnect electrical wiring from light assembly.

(3) Remove light assembly from panel.

B. Installation – Cabin Reading Light Assembly

(1) Insert light assembly in its panel position.

(2) Connect electrical wiring to light assembly.

(3) Install panel.

C. Removal – Cabin Reading Light Bulb

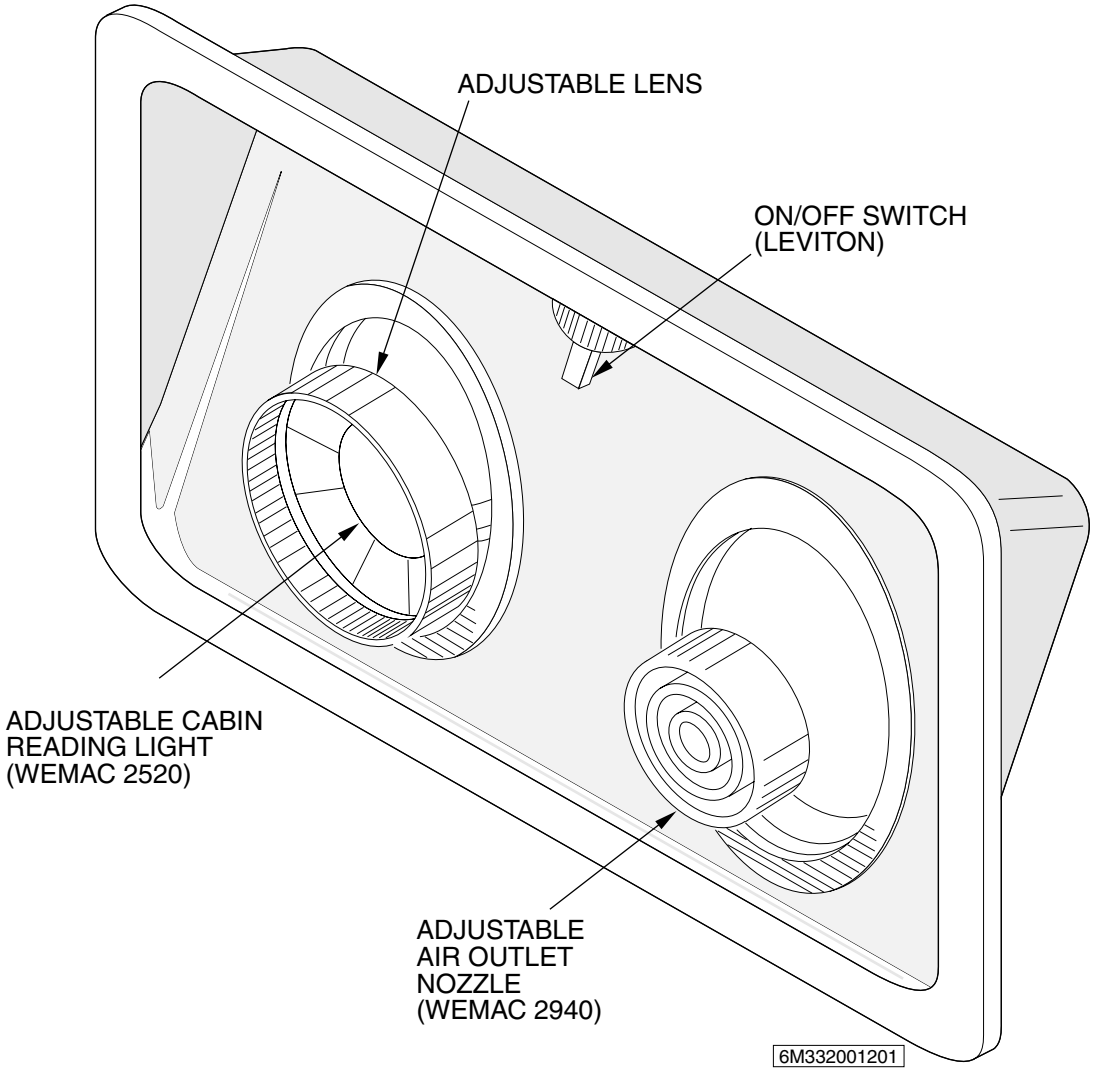
(1) Unsnap cover plate.

(2) Remove bulb.

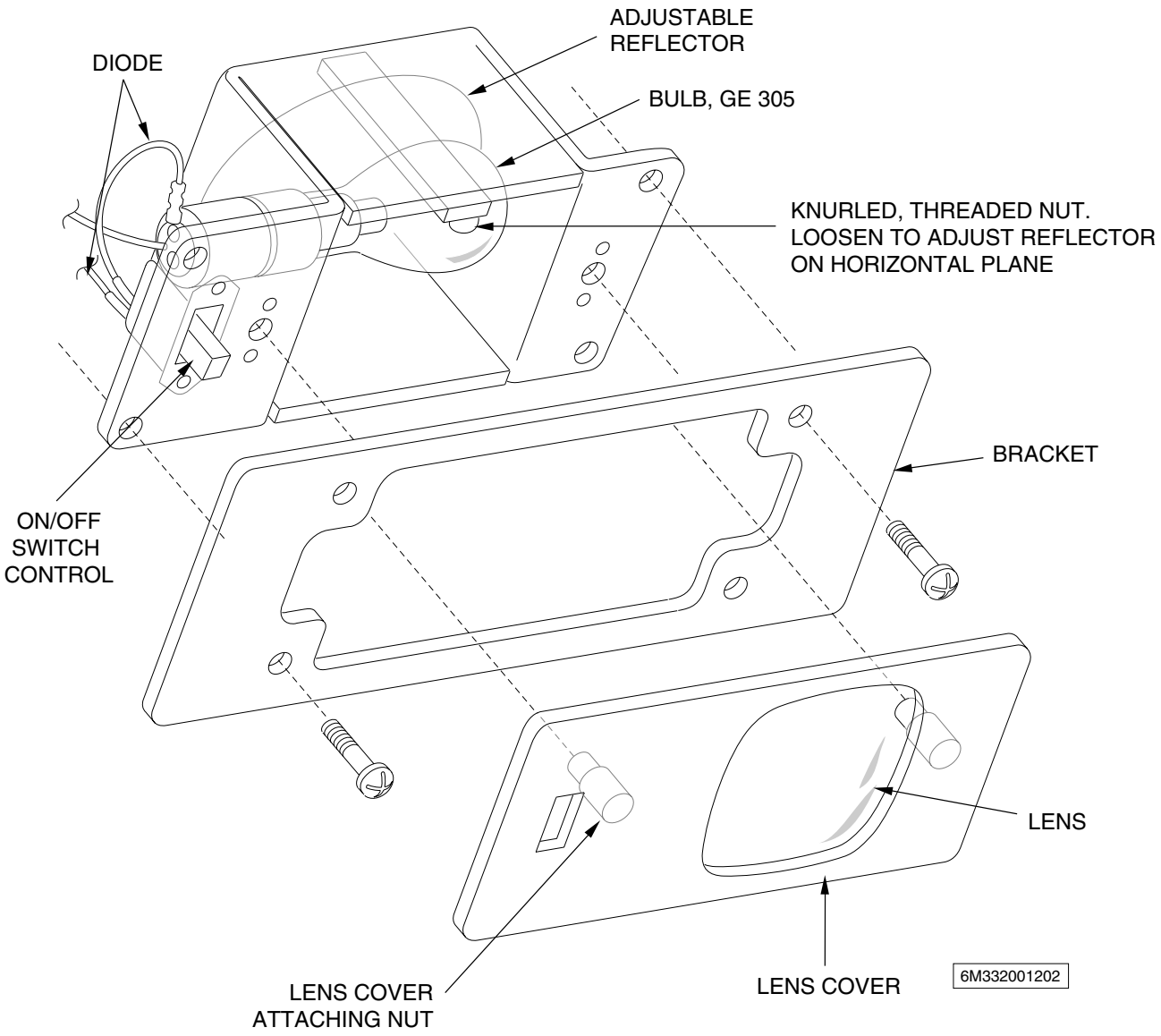
D. Installation – Cabin Reading Light Bulb

(1) Insert bulb in receptacle and turn until seated securely.

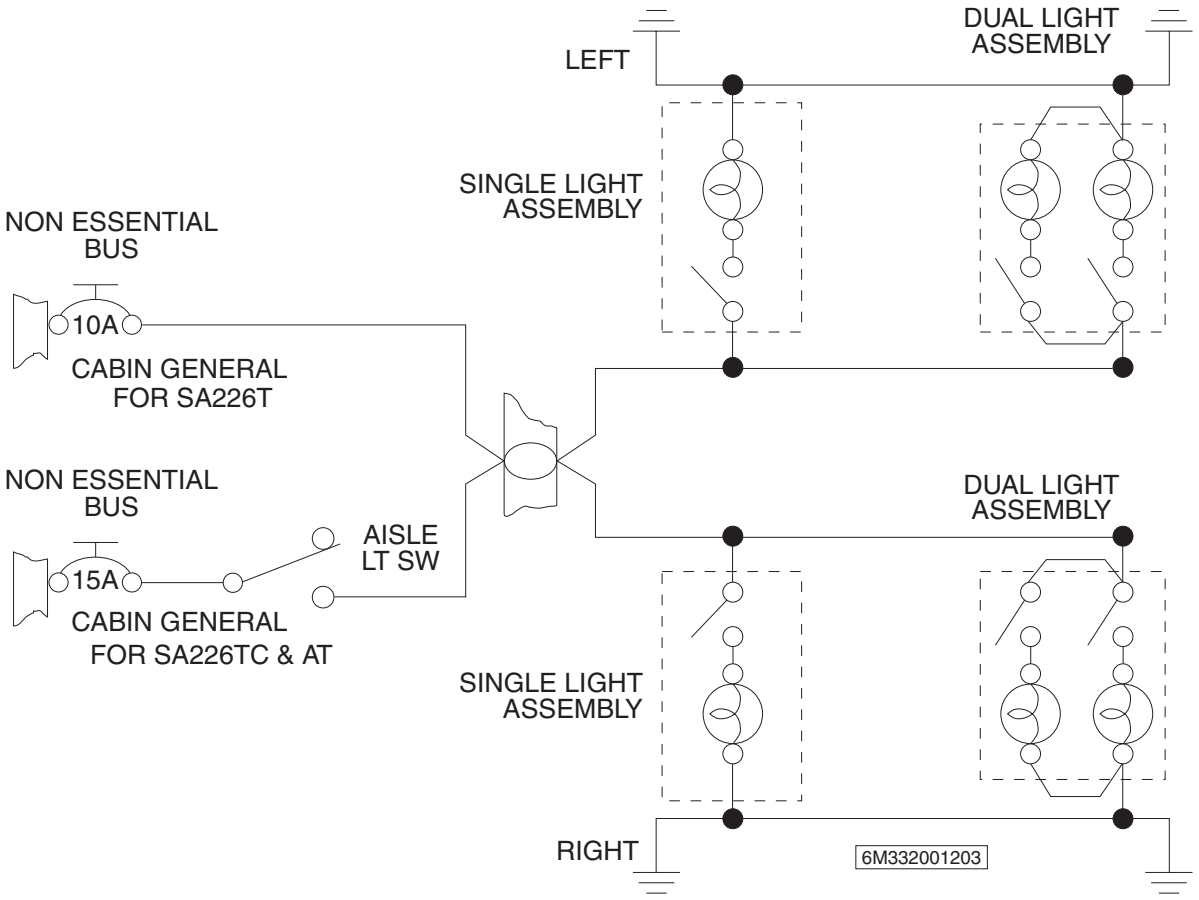
(2) Install cover plate.



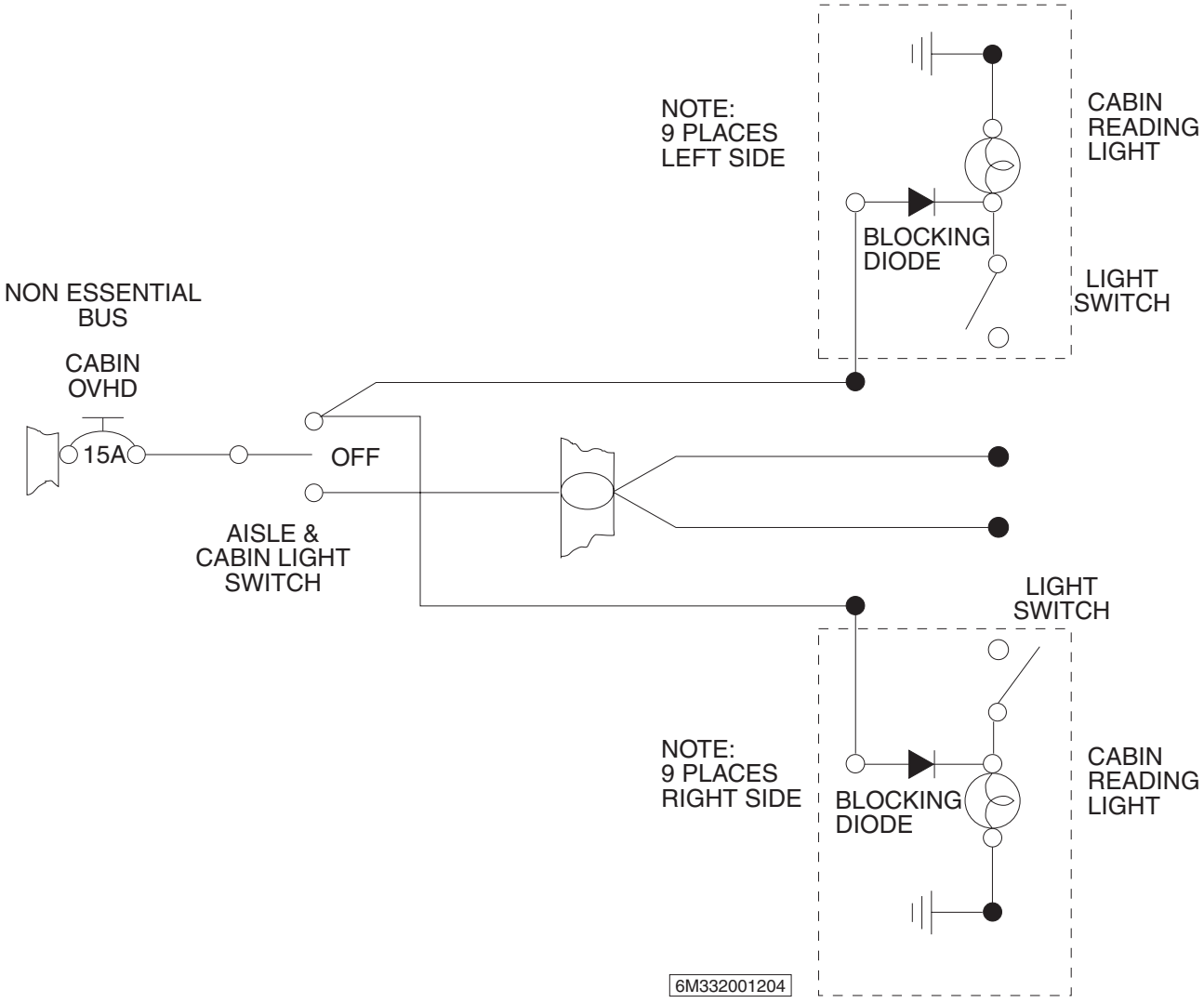
Reading Light Assembly
Figure 201



Reading Light
Figure 202



Reading Light
Figure 203

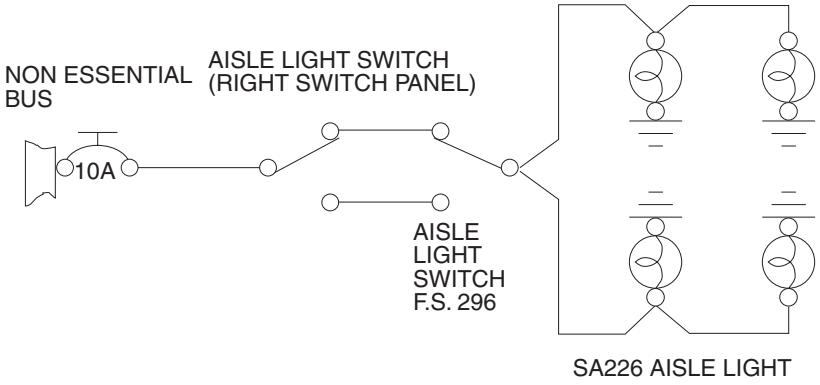


Reading Light
Figure 204

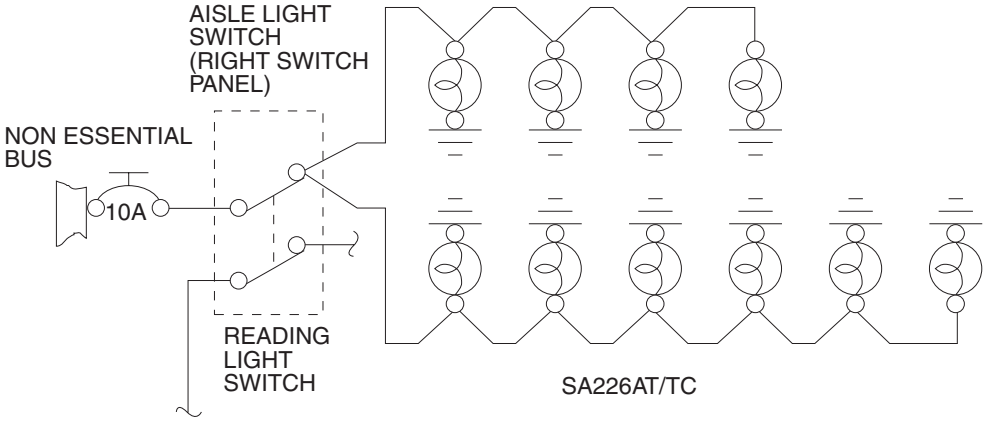
CABIN AISLE LIGHTS – DESCRIPTION AND OPERATION

1. General

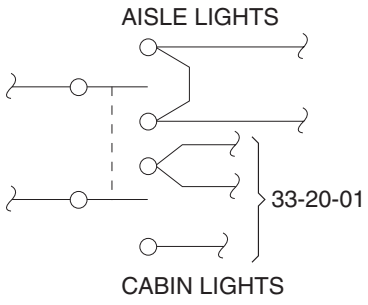
A. The aisle lights are small, rectangular, flush-mounted lights staggered every 30 inches on the vertical sides of the recessed aisles. Power for the lights is supplied from a ten amp circuit breaker on the 28 VDC nonessential bus. On SA226-T series a control switch is located on the right switch panel. An additional switch is located inside on the aft side of cabin door (F.S. 296). Aisle light control on SA226-AT and TC series is provided by the CABIN AISLE switch located on the right switch panel.



SA226 AISLE LIGHT



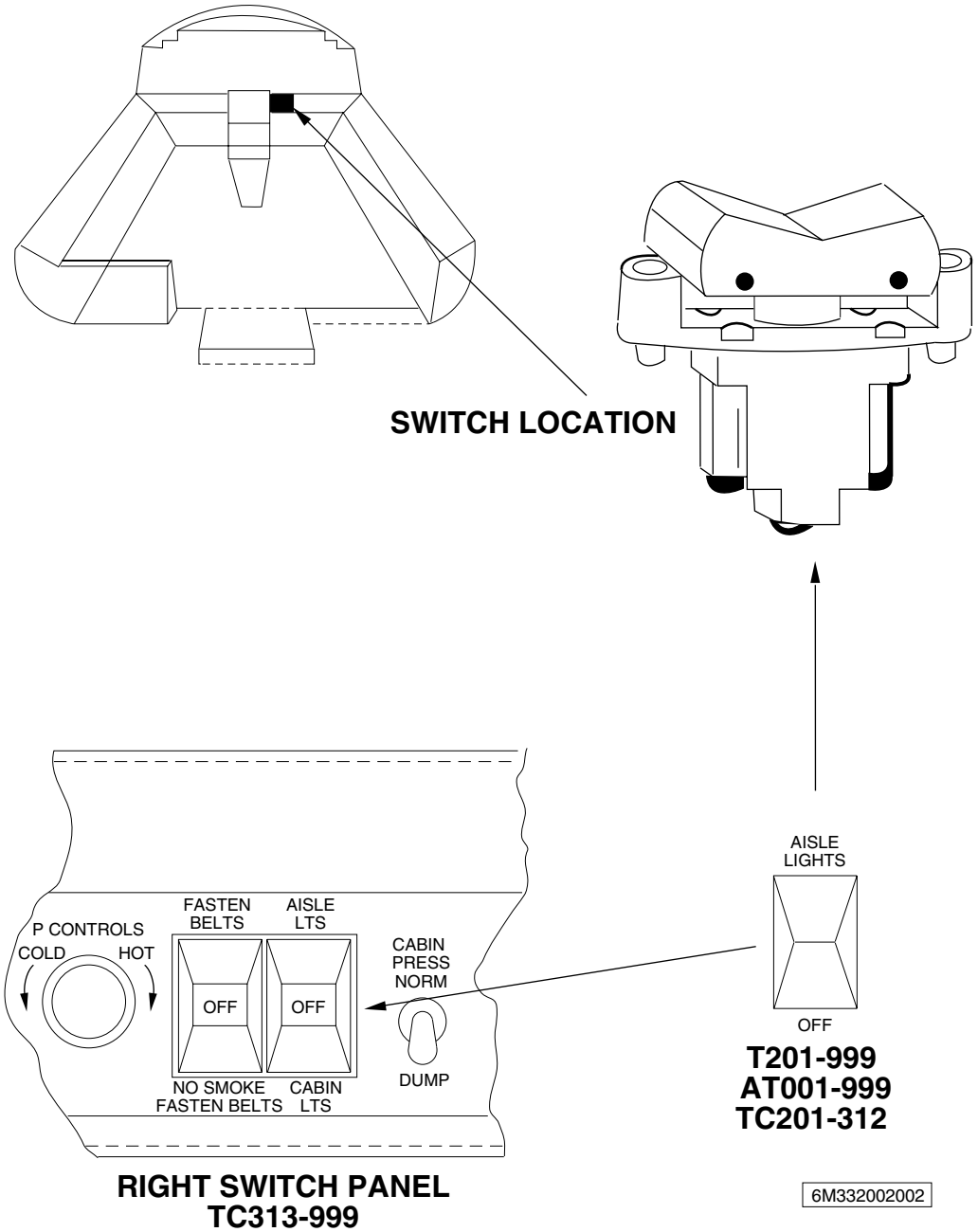
SA226AT/TC



CABIN/AISLE LIGHT SWITCH
FOR LATER MODEL SA226TC AIRCRAFT

6M332002001

Cabin Aisle Lights
Figure 1



Aisle Light Switch
Figure 2

1. Removal/Installation – Cabin Aisle Lights

A. Removal – Bulb

(1) Unsnap lens cover.

(2) Remove bulb.

B. Installation – Bulb

(1) Insert light bulb in socket and turn until seated properly.

(2) Install lens cover.

C. Removal – Aisle Light Assembly

(1) Unsnap lens cover.

(2) Remove screws securing light assembly to vertical side of recessed aisle.

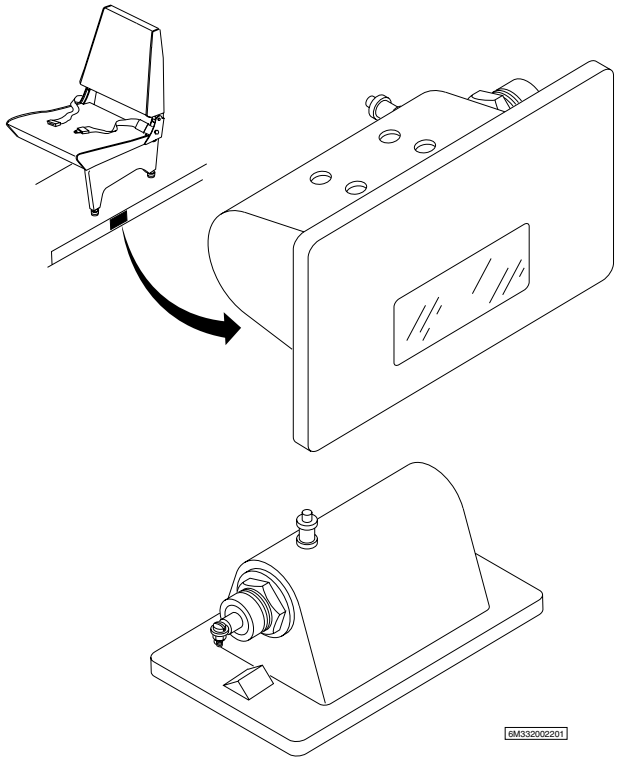
(3) Withdraw light assembly and disconnect electrical wiring.

D. Installation – Aisle Light Assembly

(1) Connect electrical wiring to light assembly.

(2) Secure light assembly to vertical side of recessed aisle with screws.

(3) Snap lens cover on light assembly.



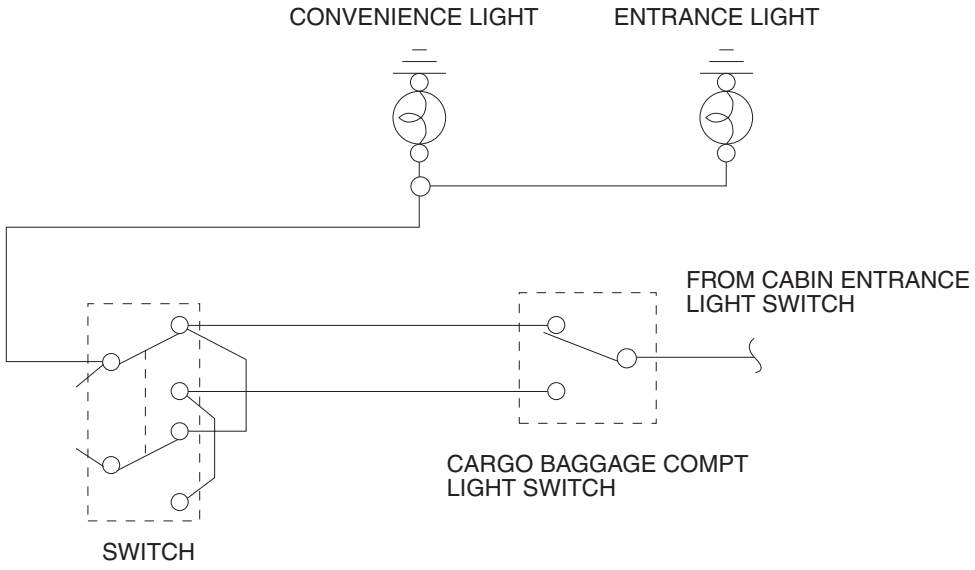
Aisle Light Assembly
Figure 201

BUFFET, GALLEY, AND LAVATORY LIGHTS – DESCRIPTION AND OPERATION

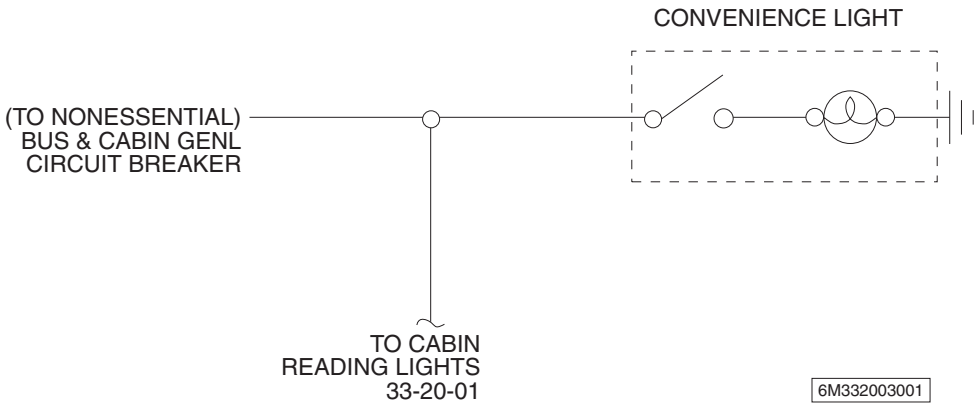
1. General

A. As a customer option SA226 TC aircraft are equipped with two light fixtures which illuminate the modular toilet. A light fixture is flush-mounted in the overhead paneling over the cargo entrance door and illuminates the modular toilet through an overhead aperture. A second light fixture is mounted on the side panel (FS450). The overhead modular toilet light is controlled by a single switch located on the light assembly and by the cargo door compartment light switch.

B. Energizing current to the modular toilet overhead light is supplied from the 28.5 VDC essential bus through the pilot's flight instrument circuit breaker. The side panel modular toilet light is energized from a 15 amp circuit breaker on the 28.5 VDC nonessential bus and is controlled by a single switch on the light assembly or by the aisle light switch.



MODULAR TOILET OVERHEAD LIGHT



MODULAR TOILET SIDE PANEL LIGHT

Modular Toilet Lighting
Figure 1

1. Removal/Installation – Buffet, Galley, and Lavatory Lights

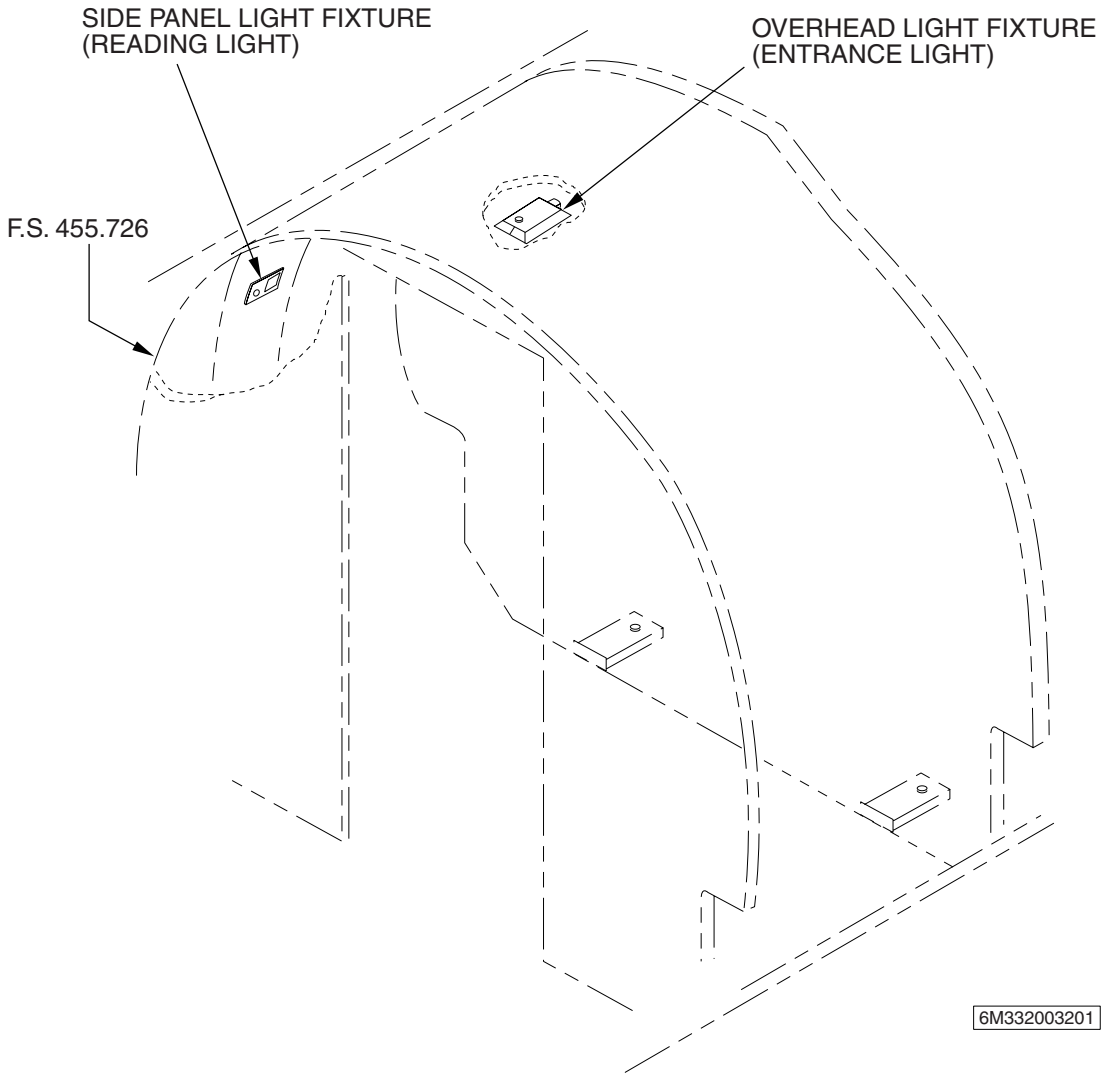
A. Removal – Overhead Light Assembly

NOTE: The modular toilet side panel light is attached to the interior paneling section of the aircraft. To remove the light assembly the entire panel section attached to the light assembly must be removed.

(1) Remove applicable panel.

(2) Disconnect electrical wiring from light assembly.

(3) Remove light assembly from panel.



Modular Toilet Lighting
Figure 201

B. Installation – Overhead Light Assembly

(1) Insert light assembly in its panel position.

(2) Connect electrical wiring to light assembly.

(3) Install panel.

C. Removal – Modular Toilet Panel Light Bulb

(1) Unsnap cover plate.

(2) Remove bulb.

D. Installation – Modular Toilet Panel Light Bulb

(1) Insert bulb in receptacle and turn until seated securely.

(2) Replace cover plate.

E. Removal – Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Remove screws and remove lens retainer.

(2) Remove screws securing light assembly to bracket and withdraw light assembly enough to detach electrical wiring.

(3) Disconnect electrical wiring.

(4) Remove light assembly.

F. Installation – Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Connect electrical wiring to light assembly.

(2) Position light assembly on bracket and secure with retaining screws.

(3) Position lens retainer on lens and secure with retaining screws.

(4) Perform operational check.

FLUORESCENT WINDOW AND LAVATORY LIGHTS – DESCRIPTION AND OPERATION

1. General

A. Cabin Window Lights

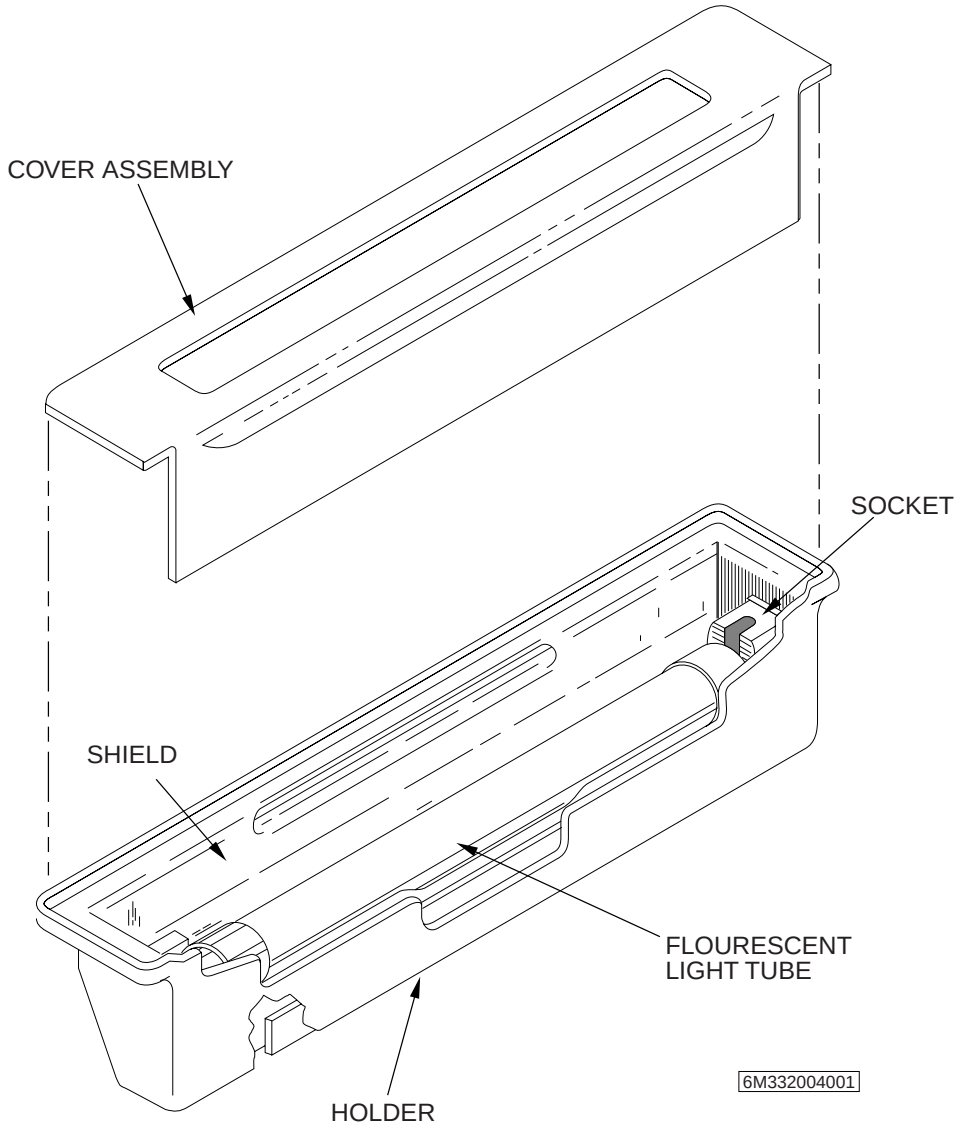
(1) SA226-AT series aircraft are equipped with a cabin window light. The cabin window light consists of a fluorescent light fixture recessed in the paneling above the escape hatch window. 115 VAC energizing current to the light is supplied by a small converter mounted behind each window panel. The converter receives power from the CABIN AISLE light ten amp circuit breaker on 28 VDC nonessential bus. The window light is controlled by a switch on the forward and aft cabin bar.

NOTE: SA226-TC series aircraft are provided with a cabin window light as a customer option.

B. Lavatory Light

(1) SA226-T and AT series aircraft are provided with a fluorescent lavatory light. The lavatory light is controlled by a single ON/OFF switch located on the base of the light fixture. 115 VAC energizing current is supplied by a small converter mounted on the light assembly. The converter is powered by 28 VDC from nonessential bus (GALLEY) early SA226 aircraft and from left essential bus (GALLEY) on later SA226 aircraft.

FLUORESCENT WINDOW AND LAVATORY LIGHTS – DESCRIPTION AND OPERATION



Window Light
Figure 1

1. Removal/Installation – Window and Escape Hatch Lights

A. Removal – Fluorescent Light Tube

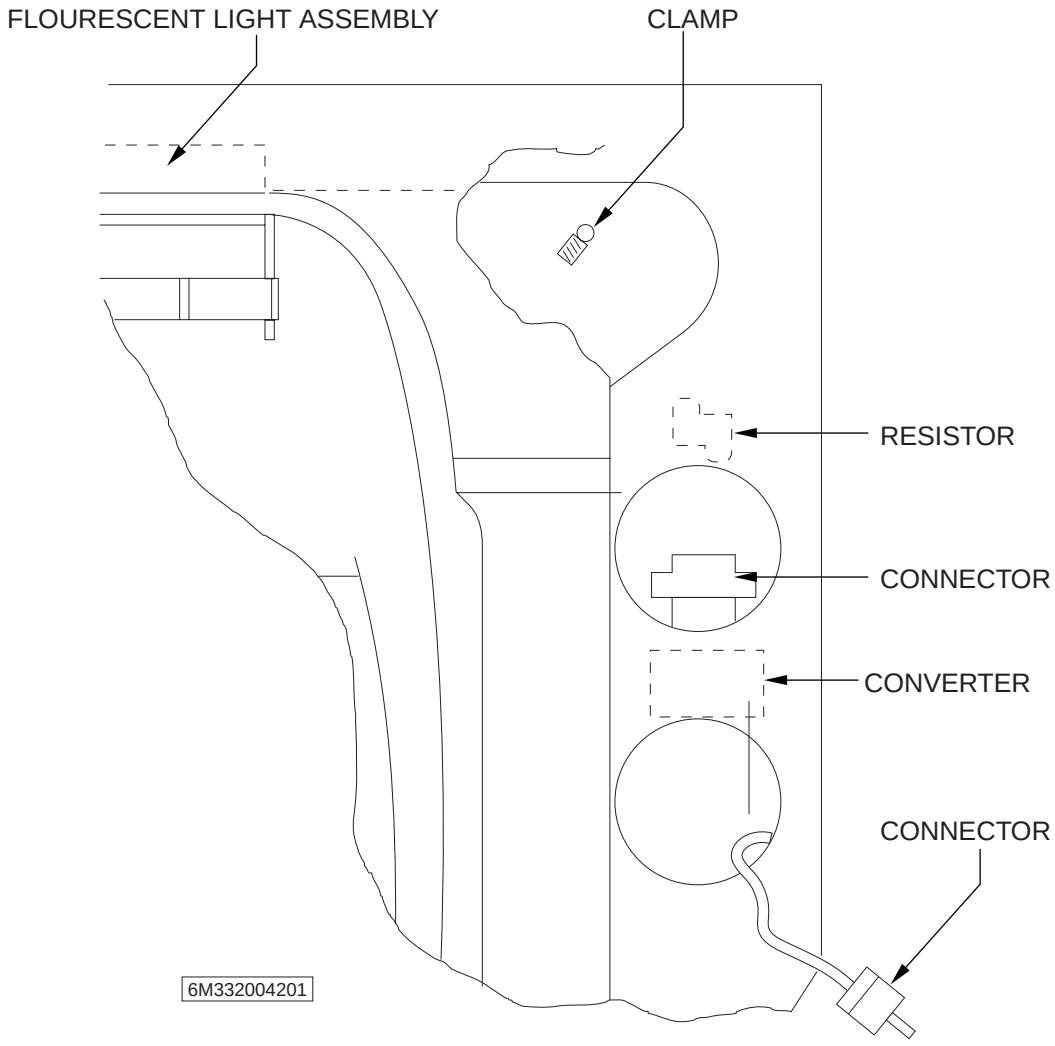
(1) Snap cover from window light assembly.

(2) Turn fluorescent tube in sockets until free and remove.

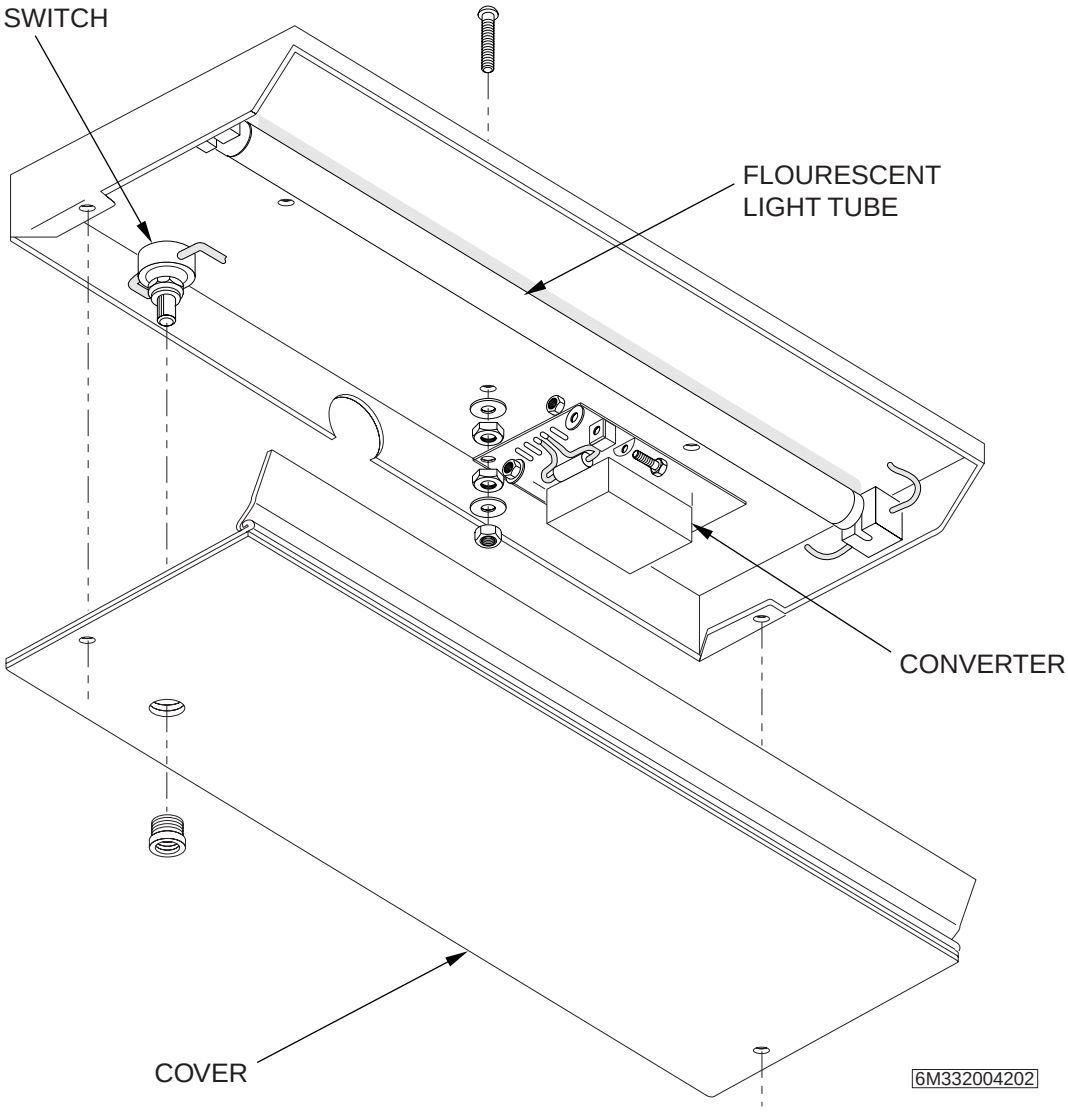
B. Installation – Fluorescent Light Tube

(1) Position fluorescent light tube in sockets and turn until secure.

(2) Snap cover on light assembly.



Escape Hatch Window lighting
Figure 201



Lavatory Light
Figure 202

FLUORESCENT WINDOW AND LAVATORY LIGHTS – MAINTENANCE PRACTICES

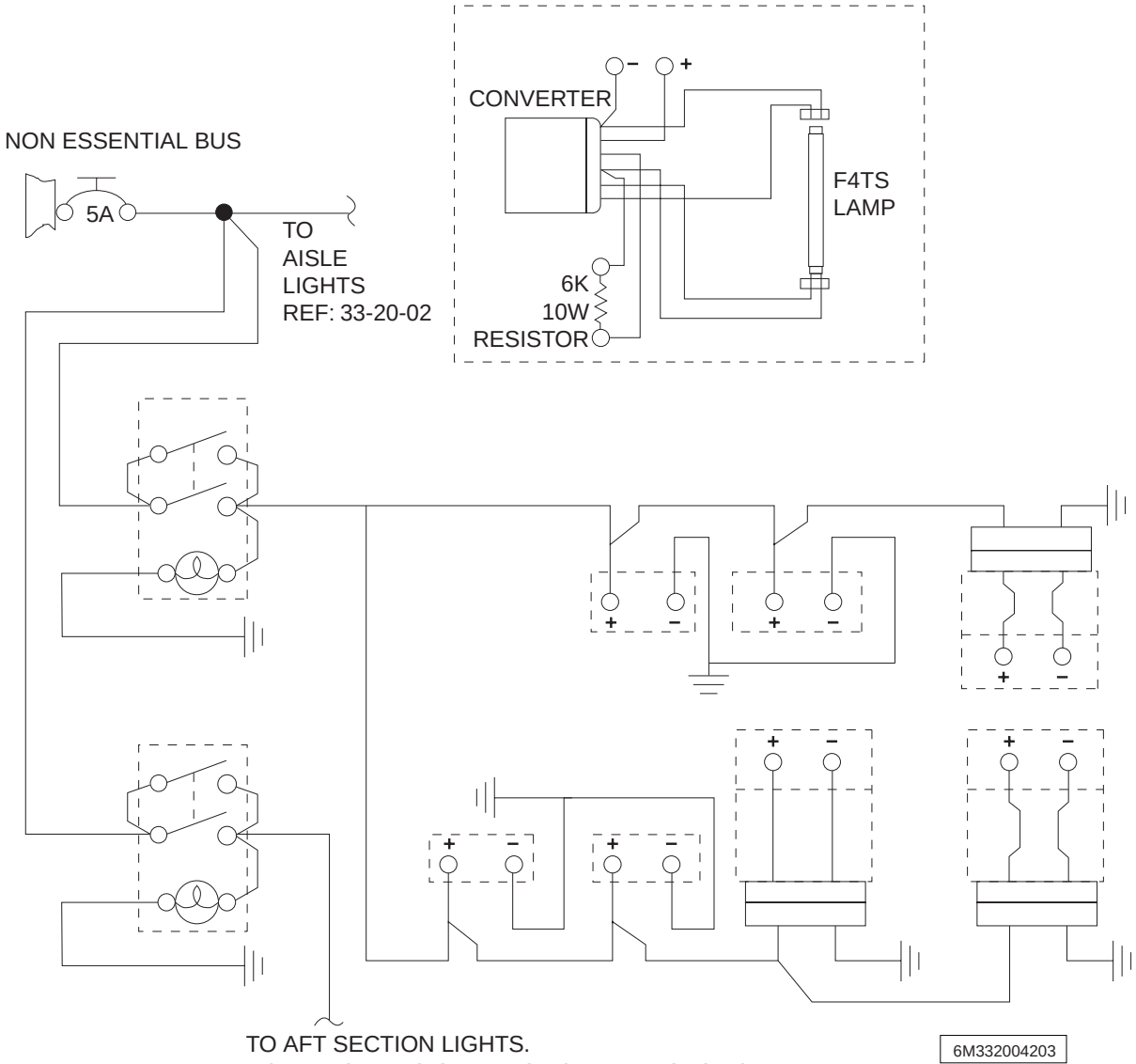
C. Removal – Window Light Converter

CAUTION: ENSURE ELECTRICAL POWER TO CONVERTER IS OFF.

(1) Gain access to converter by removing aircraft interior window paneling. Converter is attached to aircraft rib behind paneling.

(2) Disconnect electrical wiring from converter.

(3) Remove converter from rib.



Window Lighting
Figure 203

D. Installation – Window Light Converter

(1) Connect electrical wiring to converter.

(2) Position and secure converter to rib.

(3) Install panel assembly.

(4) Perform operational check.

E. Removal – Window Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO WINDOW LIGHT ASSEMBLY IS OFF.

(1) Withdraw light assembly from recessed position in paneling.

(2) Disconnect electrical wiring from light assembly of quick disconnect.

(3) Remove light assembly.

F. Installation – Window Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO WINDOW LIGHT ASSEMBLY IS OFF.

(1) Connect electrical wiring to light assembly.

(2) Position light assembly and snap into recessed position in paneling.

(3) Perform operational check.

G. Removal – Escape Hatch Light Assembly Converter

CAUTION: ENSURE ELECTRICAL POWER TO ESCAPE HATCH LIGHT ASSEMBLY IS OFF.

(1) Remove escape hatch to gain access to converter.

(2) Remove panel from escape hatch.

(3) Disconnect electrical wiring from converter at connector.

(4) Remove bracket and retaining screws securing converter and remove converter.

H. Installation – Escape Hatch Light Assembly Converter

CAUTION: ENSURE ELECTRICAL POWER TO ESCAPE HATCH LIGHT ASSEMBLY IS OFF.

(1) Secure converter to escape hatch panel with bracket and retaining screw.

(2) Connect electrical wiring to converter at connector.

(3) Perform operational check.

(4) Install escape hatch.

I. Removal – Lavatory Light Converter

CAUTION: ENSURE ELECTRICAL POWER TO LAVATORY LIGHT ASSEMBLY IS OFF.

(1) Snap cover from light assembly.

(2) Disconnect electrical wiring from converter.

(3) Remove screws securing converter to light assembly and remove converter.

J. Installation – Lavatory Light Converter

CAUTION: ENSURE ELECTRICAL POWER TO LAVATORY LIGHT ASSEMBLY IS OFF.

(1) Position converter on light assembly and secure with retaining screws.

(2) Connect electrical wiring to converter.

(3) Snap cover on light assembly.

(4) Perform operational check.

FLUORESCENT WINDOW AND LAVATORY LIGHTS – MAINTENANCE PRACTICES

K. Removal – Lavatory Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Snap cover from light assembly.

(2) Disconnect electrical wiring from lavatory light.

(3) Remove screws securing light assembly to lavatory and remove light assembly.

L. Installation – Lavatory Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Position lavatory light assembly and secure to lavatory with screws.

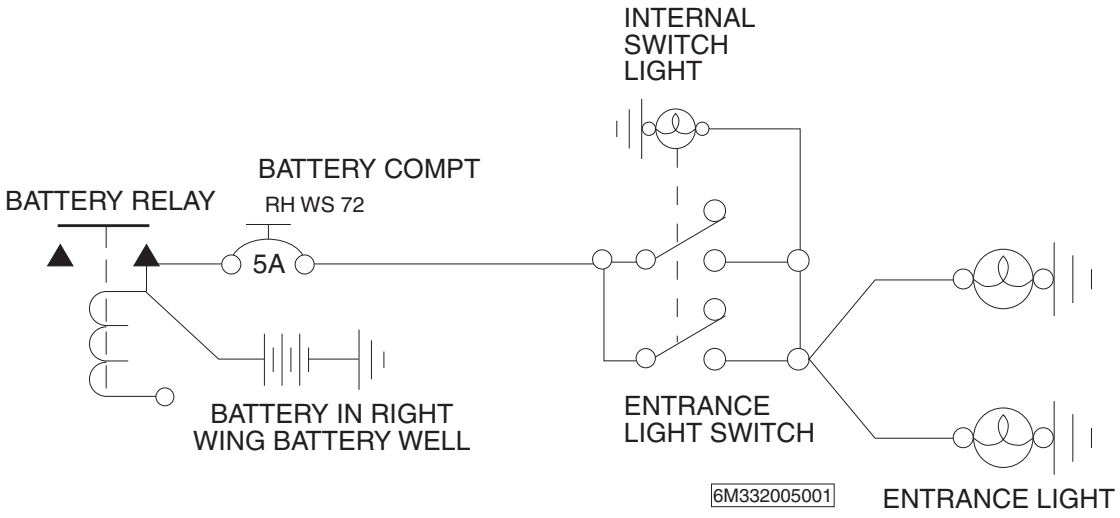
(2) Connect electrical wiring to lavatory light.

(3) Install cover on lavatory light assembly.

ENTRANCE LIGHTS – DESCRIPTION AND OPERATION

1. General

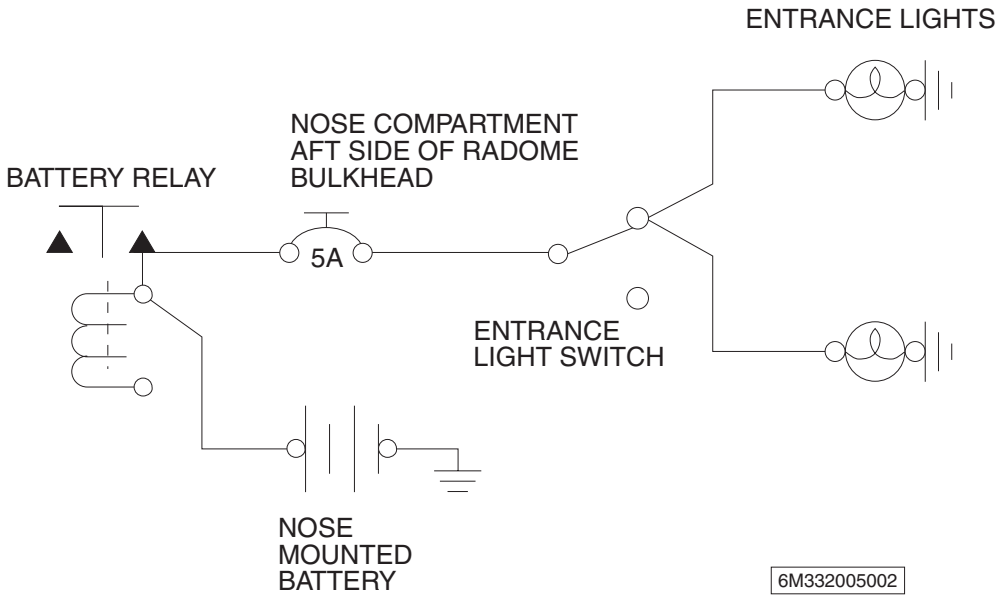
A. SA226 series aircraft T201 through T248 are equipped with two entrance lights over the cabin door. The entrance light consists of a shallow, compact light assembly flush-mounted in the overhead paneling Power for the light is obtained from the right battery through a five amp circuit breaker located in the battery compartment. The entrance lights control switch (CABIN LIGHTS) is located in the galley bar across from the cabin door.



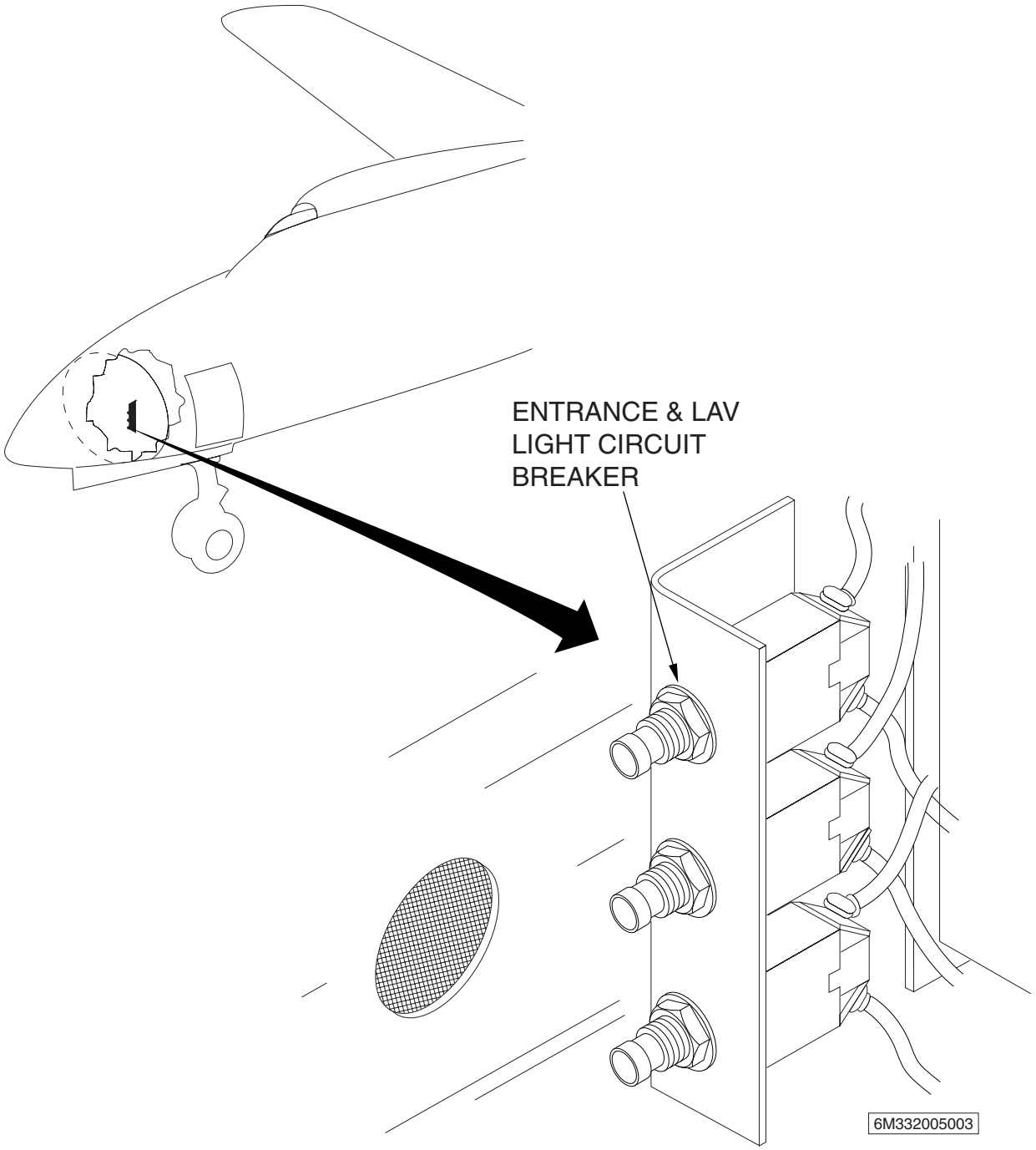
Entrance Lights
Figure 1

ENTRANCE LIGHTS – DESCRIPTION AND OPERATION

B. SA226 series aircraft T249 through T999 are equipped with two cabin entrance lights flush-mounted in paneling over the cabin entrance door. Energizing current for the entrance lights is supplied from the nose mounted battery through a five amp circuit breaker located on the aft side of the radome bulkhead. Access to the circuit breaker is gained through an aperture in the bulkhead. The control switch for the entrance lights (CABIN LIGHTS ENTRANCE) is located inside on the aft side of the cabin door.



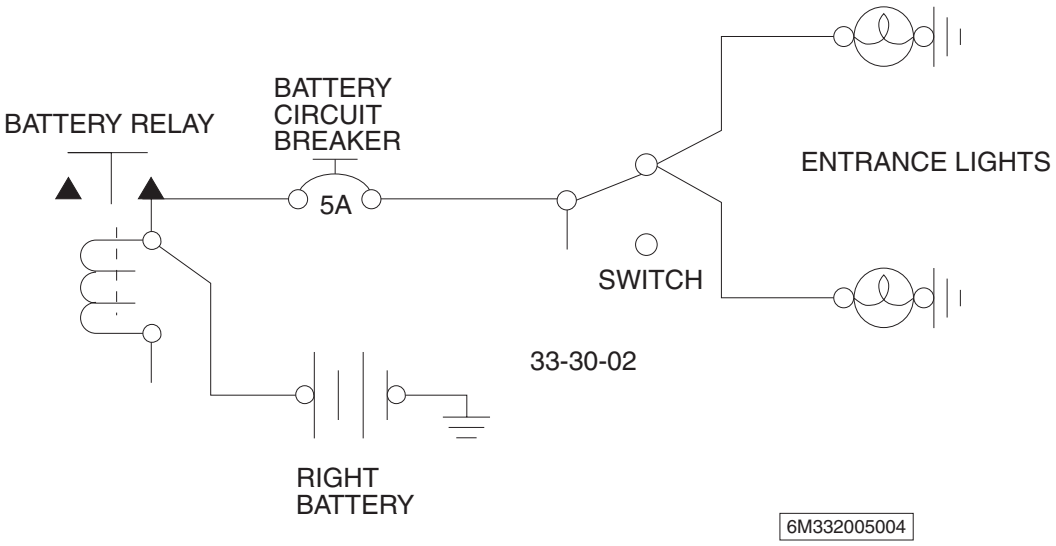
Entrance Lights
Figure 2



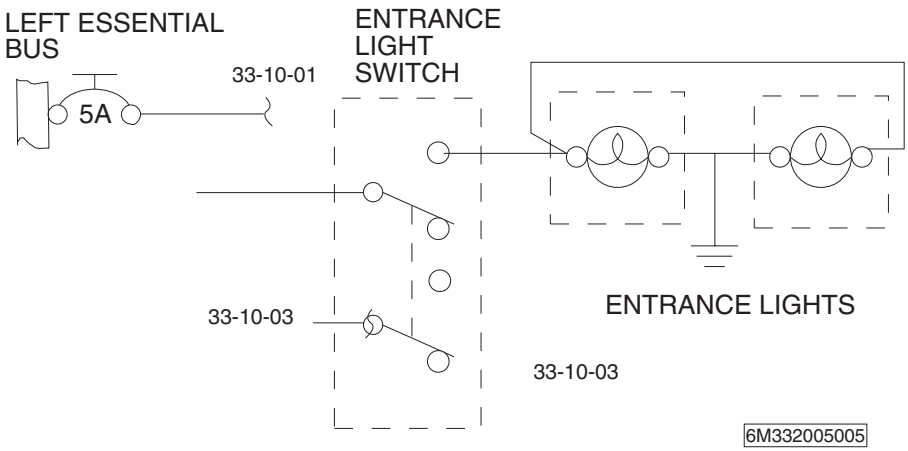
Entrance Light Circuit Breaker
Figure 3

ENTRANCE LIGHTS – DESCRIPTION AND OPERATION

C. SA226-AT001-074 series aircraft are equipped with cabin entrance lights flush-mounted in paneling over the cabin entrance door. Energizing current is supplied to the lights from the right battery through a five amp circuit breaker. A switch (ENTRANCE LIGHTS) located on the aft side of cockpit partition controls the entrance lights.



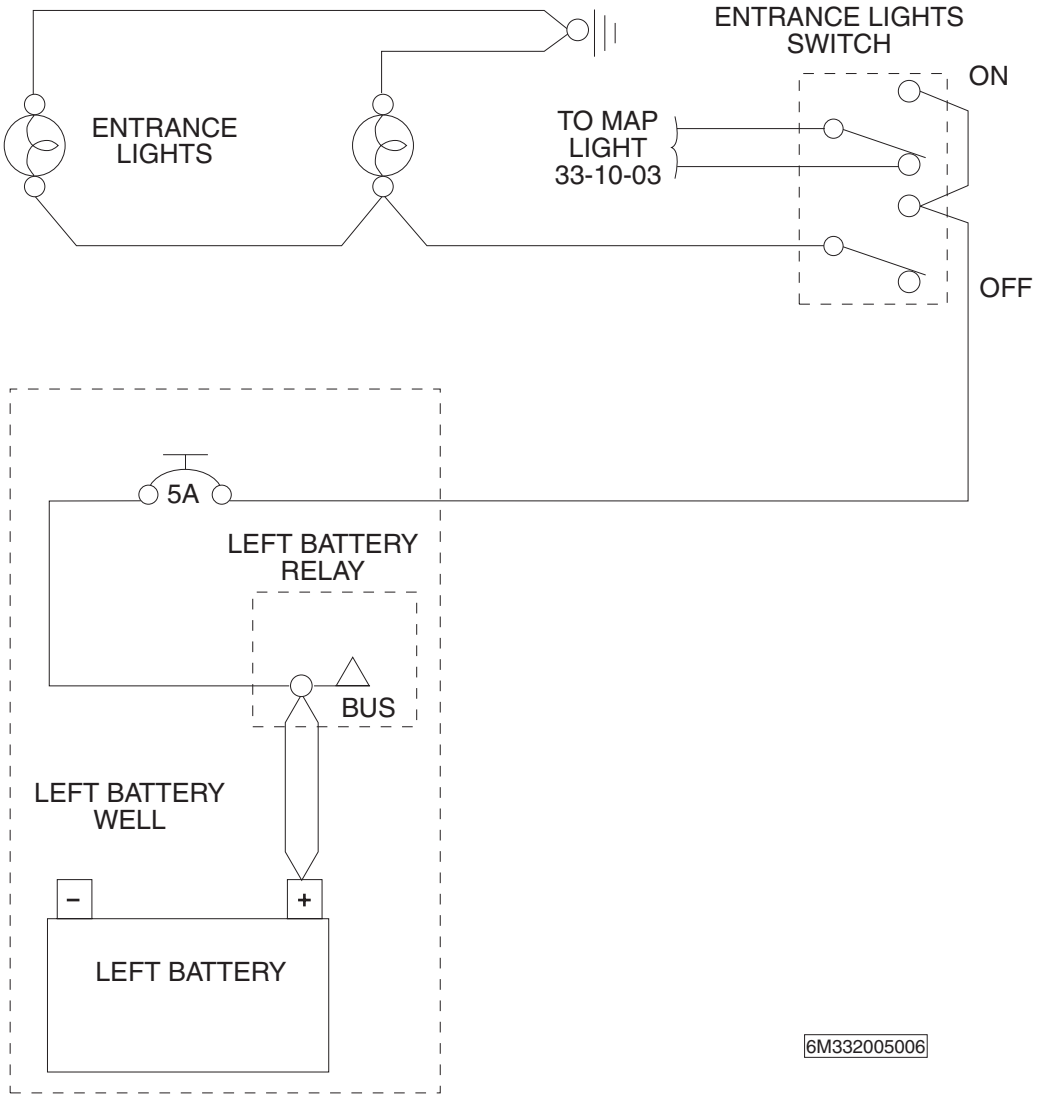
Entrance Lights
Figure 4



Entrance Lights
Figure 5

ENTRANCE LIGHTS – DESCRIPTION AND OPERATION

D. SA226-TC series aircraft are equipped with two entrance lights over the cabin door to provide lighting for enplaning and deplaning passengers. The entrance lights are shallow, compact light assemblies, flush-mounted in the overhead paneling. A switch (ENTRANCE LIGHTS), located on the aft side of the cockpit partition, controls the lights. Energizing current for the entrance lights is supplied from a five amp circuit breaker on the 28 VDC essential bus on SA226-TC201-354. Entrance lights on SA226-TC355-999 are powered from the left battery through a five amp circuit breaker (see Figure 6).



Entrance Lights
Figure 6

ENTRANCE LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Entrance Lights

A. Removal – Entrance Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Remove screws and remove lens retainer.

(2) Remove screws securing light assembly to bracket and withdraw light assembly enough to detach electrical wiring.

(3) Disconnect electrical wiring.

(4) Remove light assembly.

B. Installation – Entrance Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Connect electrical wiring to light assembly.

(2) Position light assembly on bracket and secure with retaining screws.

(3) Position lens retainer on lens and secure with retaining screws.

(4) Perform operational check.

C. Removal – Light Bulb, Entrance Light.

(1) Remove screws securing lens retainer and remove lens retainer.

(2) Remove light bulb.

D. Installation – Light Bulb, Entrance Light

(1) Insert light bulb in receptacle and turn until seated properly.

(2) Position lens retainer on lens and secure with screws.

2. Adjustment/Test – Entrance Lights

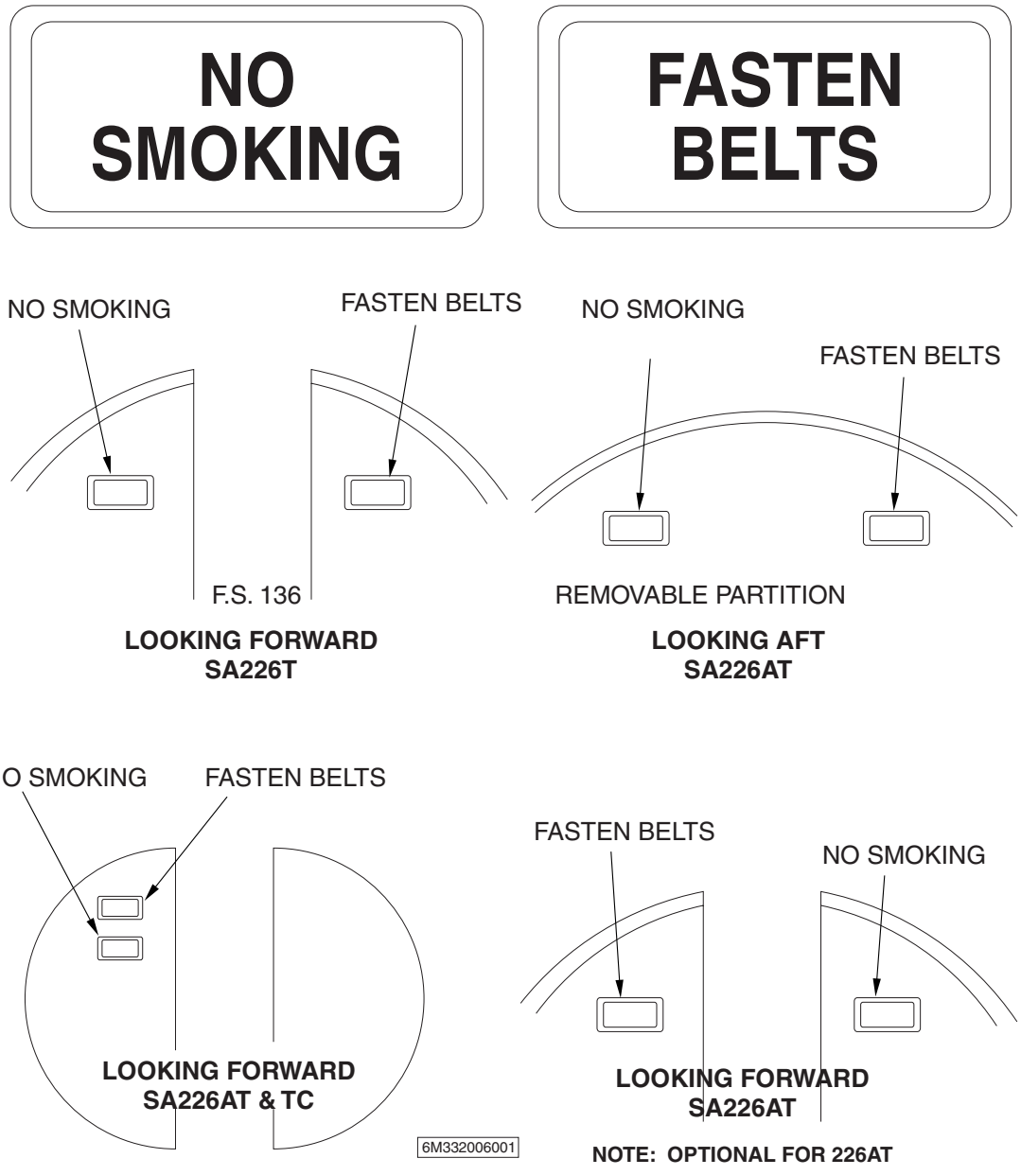
A. Adjustment – Circuit Breaker Resetting

CAUTION: WHEN RESETTING CIRCUIT BREAKERS IN BATTERY COMPARTMENT, DO NOT USE ANY METAL EXTENSION OBJECTS TO REACH CIRCUIT BREAKER. USE EITHER PLASTIC OR WOODEN NONCONDUCTIVE ROD.

NO SMOKING/FASTEN BELTS LIGHTED SIGNS – DESCRIPTION AND OPERATION

1. General

A. SA226 series aircraft are provided with lighted warning signs located on the aft side of the cockpit partition and on the forward and aft cabin bulkheads. The lighted warning sign assembly consists of a 3-1/2 by 2-1/2 inch clear plexiglass lens which is affixed to a phenolic backed holder. The sign is illuminated by four miniature light bulbs which are spaced around the perimeter of the lens holder. Energizing current for the lighted sign is supplied from a five amp circuit breaker on the 28 VDC nonessential bus. A switch (FASTEN BELTS & NO SMOKING), located on the right switch panel, controls the lighted signs.



Lighted Warning Signs
Figure 1

NO SMOKING/FASTEN BELTS LIGHTED SIGNS – MAINTENANCE PRACTICES

1. Removal/Installation – No Smoking/Fasten Belts Lighted Signs

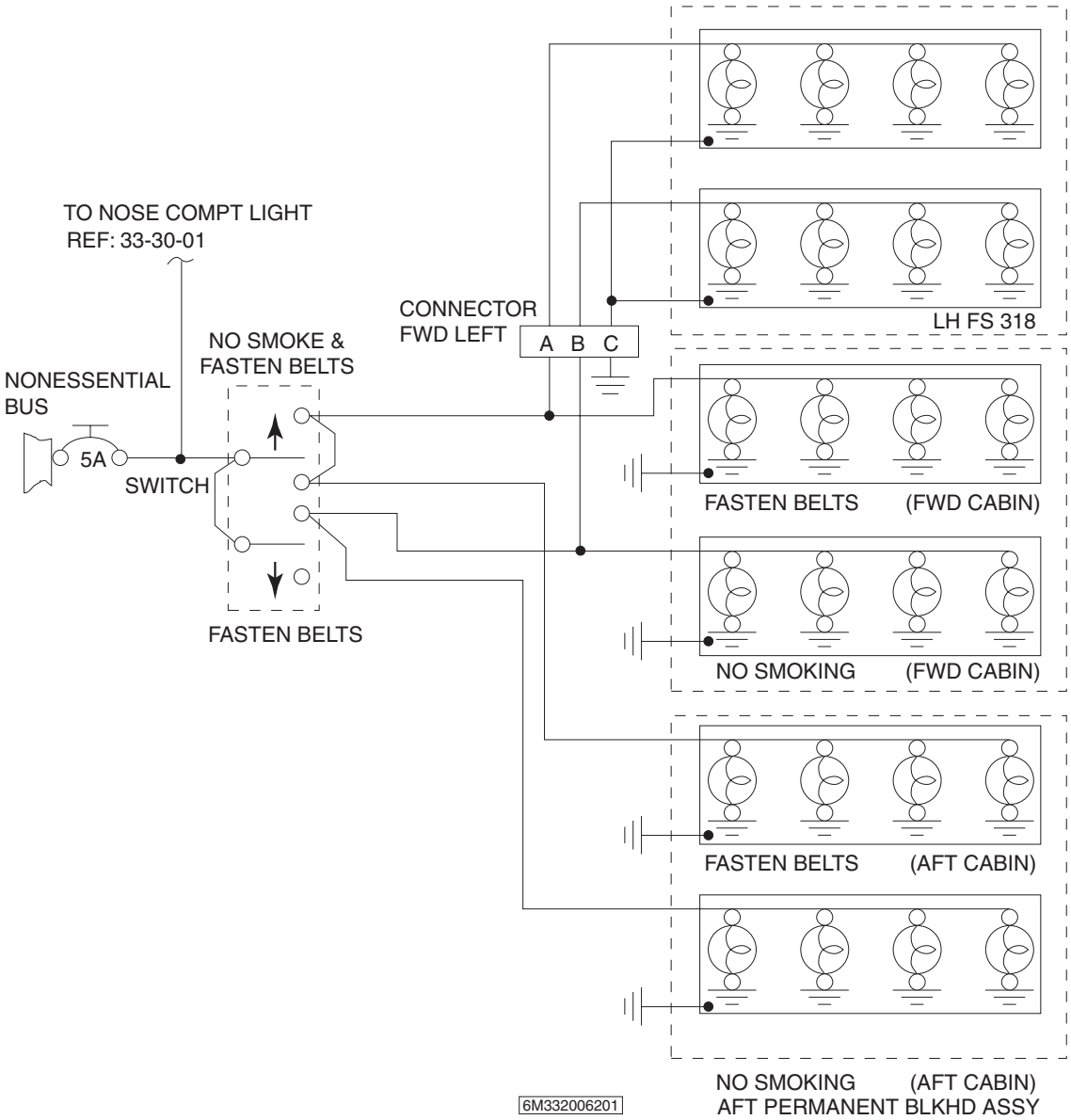
A. Removal – Light Assembly

(1) Unsnap cover plate.

(2) Remove screws holding light assembly to bulkhead.

(3) Disconnect electrical wiring.

(4) Remove light assembly.



No Smoking/Fasten Belt Signs
Figure 201

B. Installation – Light Assembly

(1) Connect electrical wiring to light assembly.

(2) Position light assembly on bulkhead.

(3) Secure light assembly to bulkhead with attaching screws.

(4) Replace cover plate.

C. Removal – Light Bulb, Light Assembly

(1) Unsnap cover plate.

(2) Remove light bulb.

D. Installation – Light Bulb, Light Assembly

(1) Install light bulb in receptacle.

(2) Push in receptacle.

(3) Replace cover plate.

CARGO COMPARTMENT LIGHTING – DESCRIPTION AND OPERATION

1. General

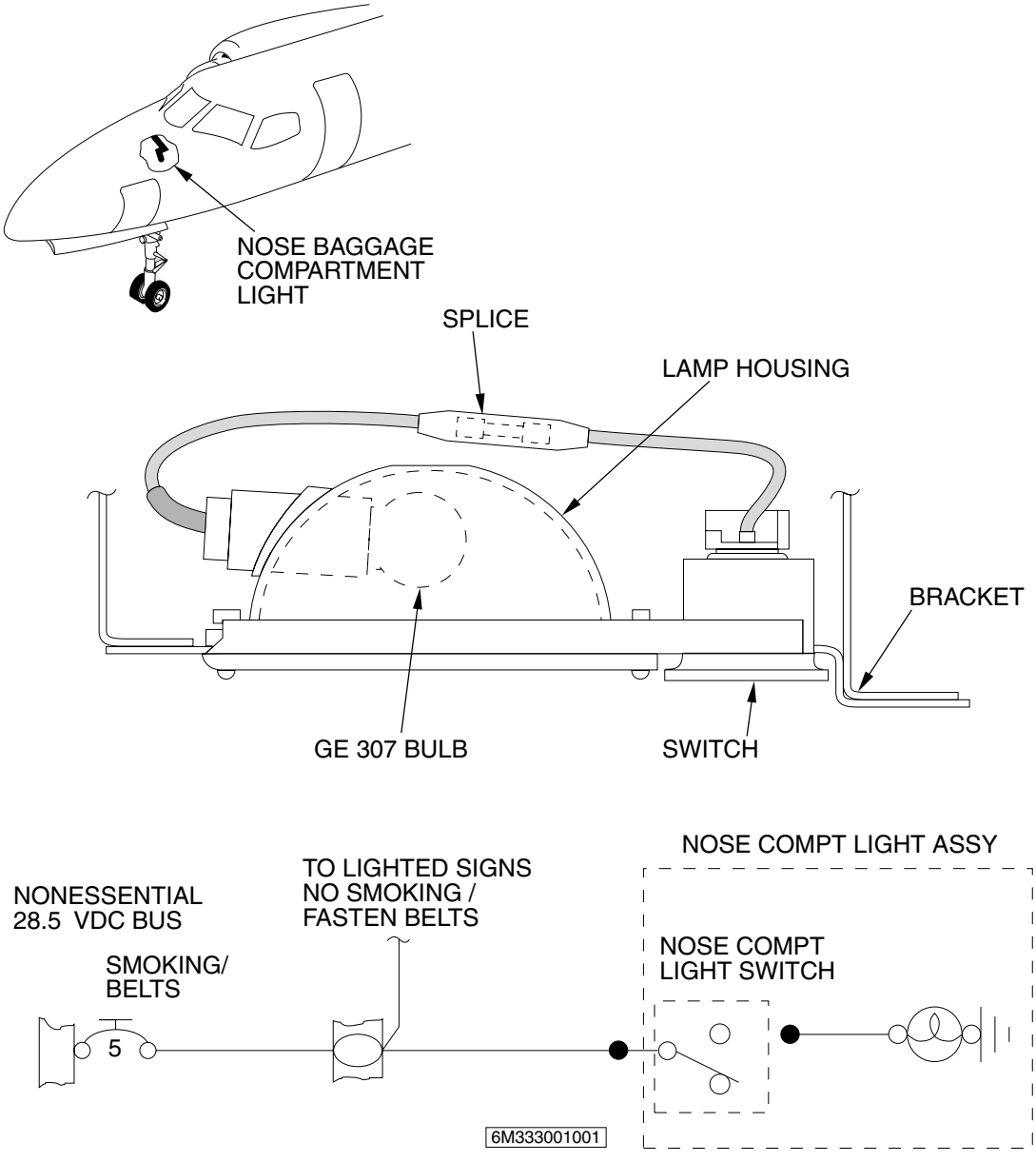
A. All SA226 series aircraft are provided with a nose baggage compartment light. SA226-AT and TC series aircraft are equipped with two lights mounted over the aft cargo entrance door. Cargo compartment lighting on SA226-TC aircraft are provided with auxiliary power provisions which are described in Sections 33-30-03 and 33-30-04.

NOSE BAGGAGE COMPARTMENT LIGHT – DESCRIPTION AND OPERATION

1. General

A. The nose baggage compartment light consists of a small, rectangular, dome light, bracket mounted overhead aft of nose baggage compartment left access door. A single ON/OFF switch located on the light assembly controls the light. Energizing current for the nose baggage compartment light on SA226-T and AT series aircraft is supplied from a five amp circuit breaker on the 28 VDC nonessential bus.

B. For a description of the SA226-TC nose baggage compartment light as a part of the auxiliary lighting system, refer to Sections 33-30-03 and 33-30-04.



Nose Compartment Light
Figure 1

NOSE BAGGAGE COMPARTMENT LIGHT – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Baggage Compartment Light

A. Removal – Light Assembly.

(1) Remove screws securing lens and remove lens.

(2) Remove screws securing light assembly to bracket.

(3) Remove attaching nut from switch.

(4) Disconnect electrical wiring.

(5) Remove light assembly and switch.

B. Installation – Light Assembly

(1) Connect electrical wiring to light assembly and switch.

(2) Install switch.

(3) Secure light assembly to bracket with retaining screws.

(4) Secure light lens.

C. Removal – Light Bulb

(1) Remove screws securing lens retainer and remove lens retainer.

(2) Remove light bulb.

D. Installation – Light Bulb

(1) Insert light bulb in receptacle and turn until seated properly.

(2) Position lens retainer on lens and secure with screws.

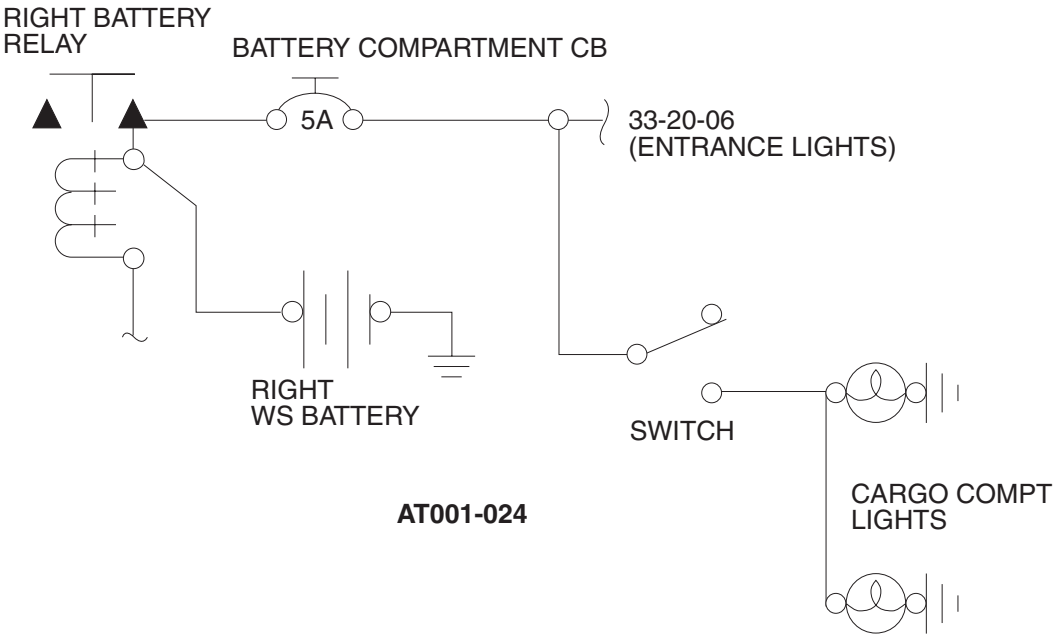
CARGO COMPARTMENT LIGHTS – DESCRIPTION AND OPERATION

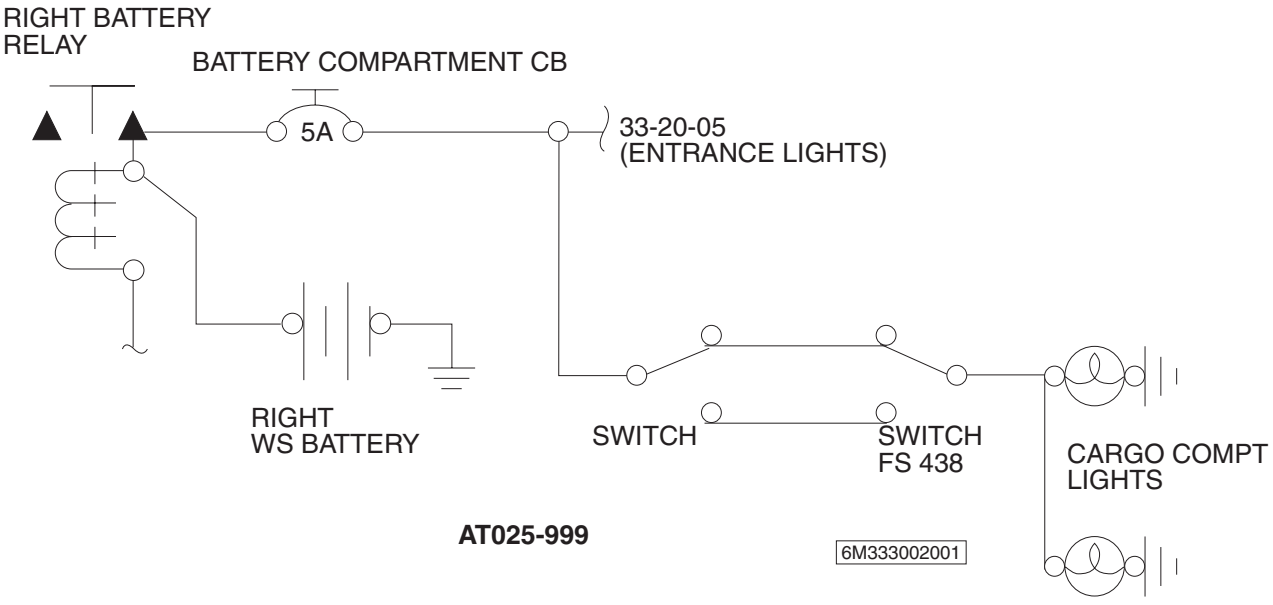
1. General

A. The cargo compartment of SA226-AT and TC series aircraft is lighted by two shallow, compact light assemblies, flush-mounted in the paneling, overhead of the cargo door. A switch, located inside on the aft side of the cargo door controls the lights.

B. SA226-AT025-999 series aircraft are provided with an additional control switch on the aft bulkhead at F.S. 438. SA226-TC series 201-354 are powered from a five amp circuit breaker on the left essential bus. SA226-TC series 355-999 are powered from a two amp circuit breaker on the right essential bus.

C. Reference Sections 33-30-03 and 33-30-04 for a description of the cargo compartment lights as a part of the auxiliary lighting system on SA226-TC series aircraft.





Cargo Compartment Lights
Figure 1

CARGO COMPARTMENT LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Cargo Compartment Lights

A. Removal – Cargo Compartment Light Assembly.

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Remove screws and remove lens retainer.

(2) Remove screws securing light assembly to bracket and withdraw light assembly enough to detach electrical wiring.

(3) Disconnect electrical wiring.

(4) Remove light assembly.

B. Installation – Cargo Compartment Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Connect electrical wiring to light assembly.

(2) Position light assembly on bracket and secure with retaining screws.

(3) Position lens retainer on lens and secure with retaining screws.

(4) Perform operational check.

C. Removal – Light Bulb, Cargo Compartment Light

(1) Remove screws securing lens retainer and remove lens retainer.

(2) Remove light bulb.

D. Installation – Light Bulb, Cargo Compartment Light

(1) Insert light bulb in receptacle and turn until seated properly.

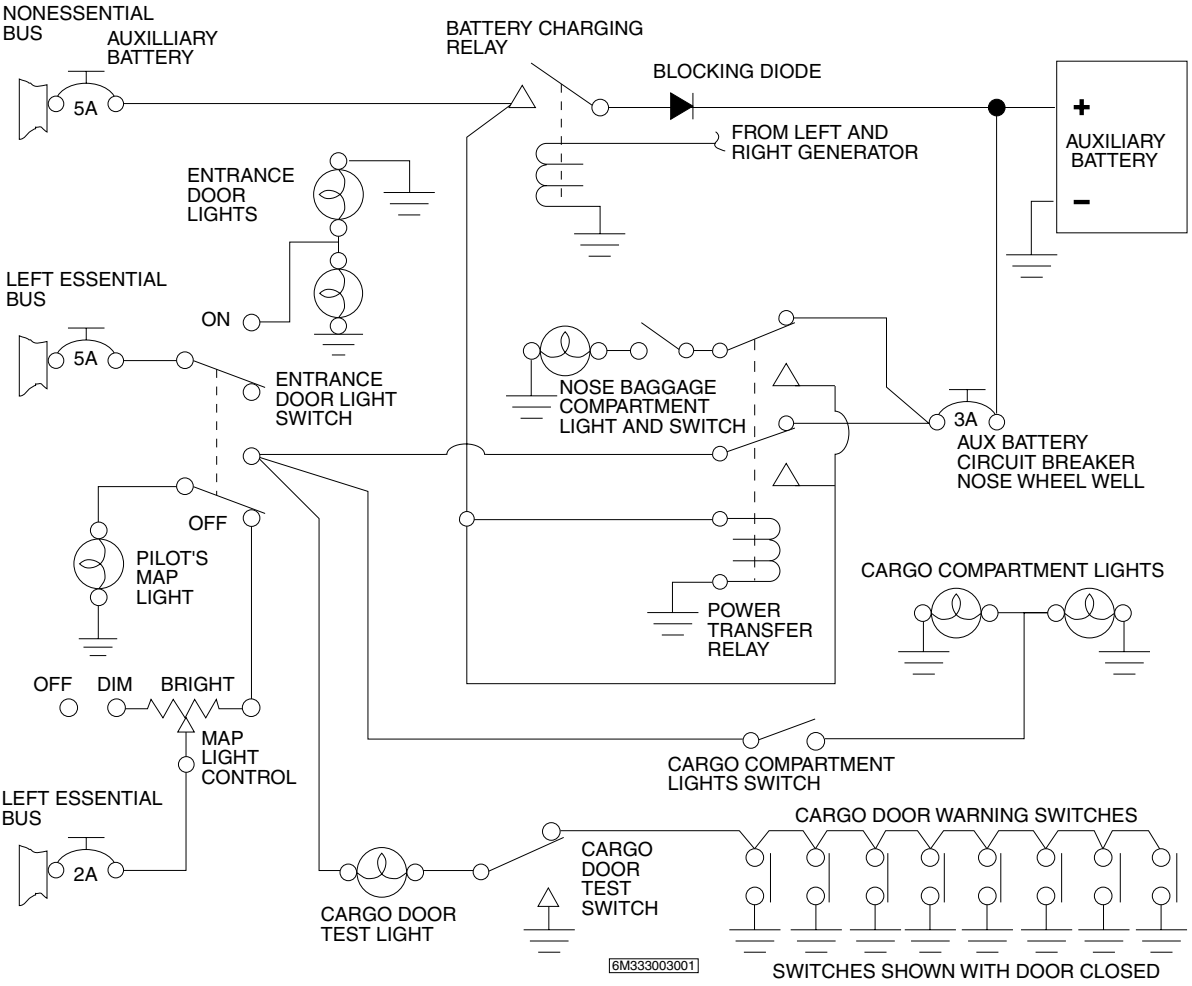
(2) Position lens retainer on lens and secure with screws.

AUXILIARY LIGHTING WITH AUXILIARY BATTERY – DESCRIPTION AND OPERATION

1. General

A. SA226-TC201-354 series aircraft are equipped with an auxiliary battery located under the left floorboard in the nose baggage compartment. The auxiliary battery supplies power to the cargo compartment lights, pilot's map light, and nose baggage compartment light. Upon loss of normal aircraft battery power, a power transfer relay located under left floorboard in nose baggage compartment de-energizes and is positioned to supply electrical current from the auxiliary battery to the cargo compartment lights, pilot's map light, and nose baggage compartment light. Circuit protection for the auxiliary battery lighting system is provided by a three amp circuit breaker located in the nose wheel well.

NOTE: The auxiliary lighting system also provides for illumination of the cargo door warning test light located on the aft side of cockpit partition forward of cabin entrance door.

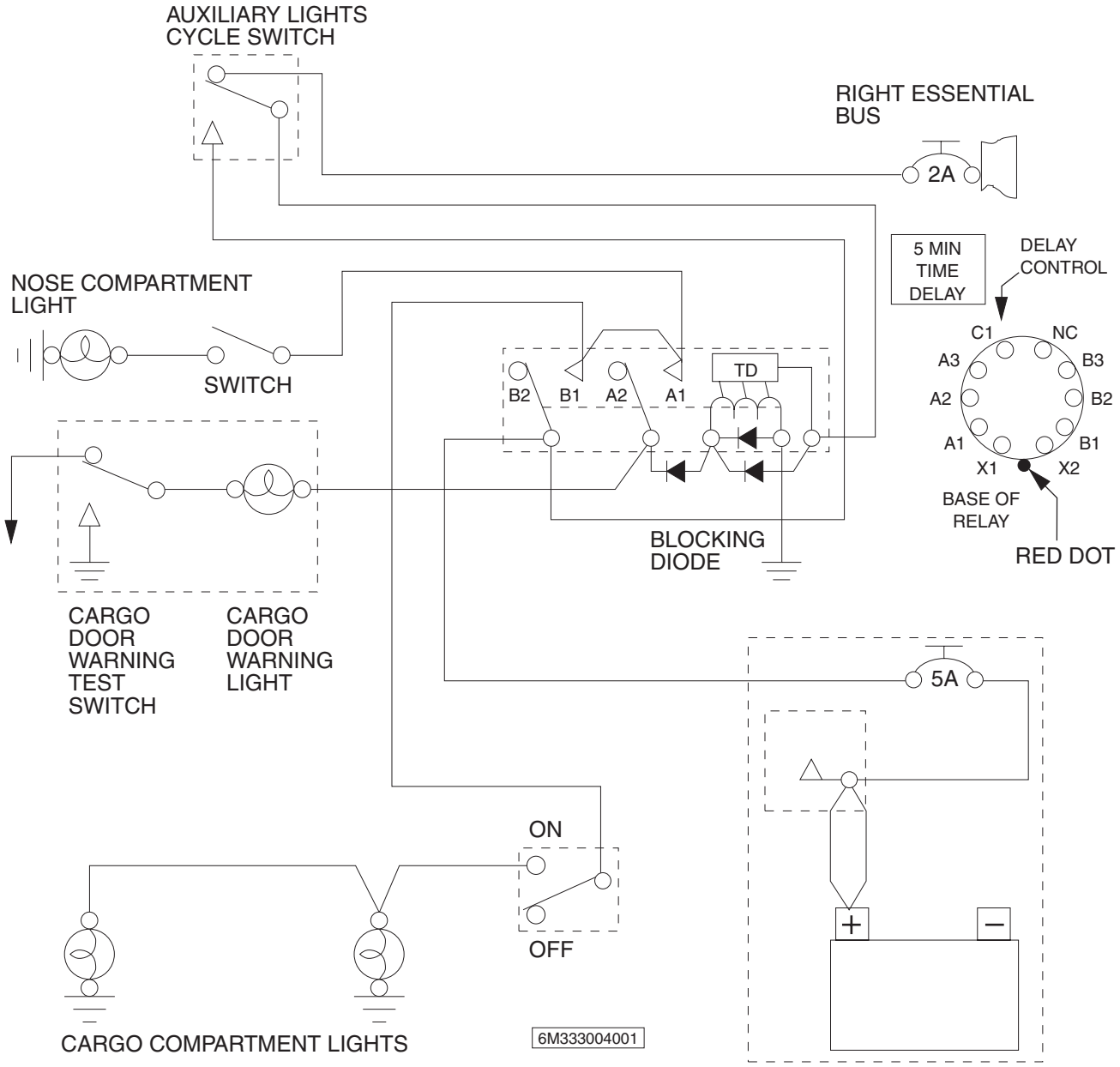


Auxiliary Battery Lighting
Figure 1

AUXILIARY LIGHTING WITHOUT AUXILIARY BATTERY – DESCRIPTION AND OPERATION

1. General

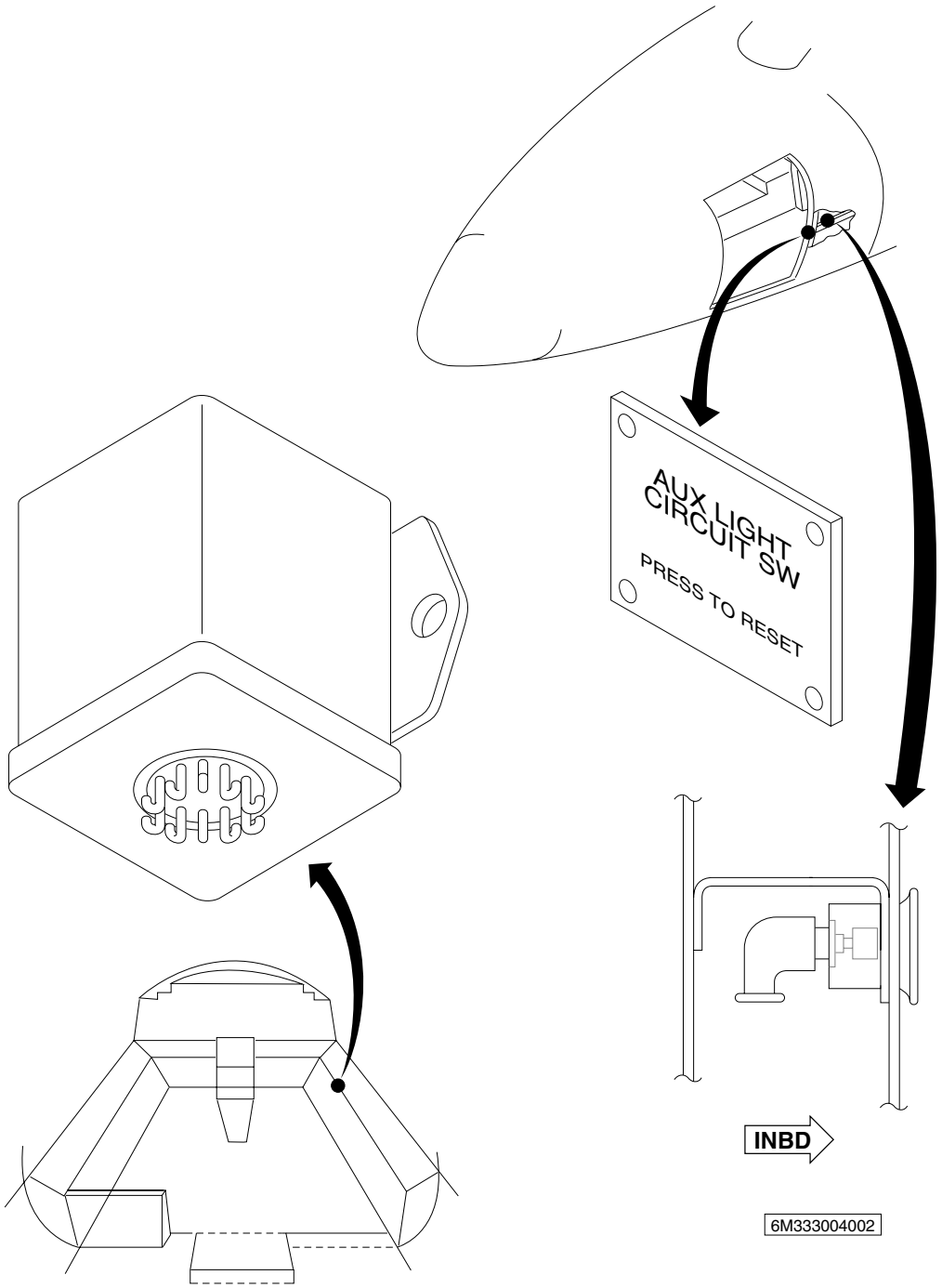
A. To allow operation of the auxiliary lights without operation of the battery master switch and to facilitate maintenance/installation, the auxiliary battery has been removed effective SA226 (TC355). Auxiliary lighting is provided on these later model SA226 series aircraft from the right aircraft battery through a time delay circuit.



Auxiliary Lighting Without Auxiliary Battery
Figure 1

AUXILIARY LIGHTING WITHOUT AUXILIARY BATTERY – DESCRIPTION AND OPERATION

B. The auxiliary lighting system consists of a light cycle switch, time cycle relay, circuit breakers, light assemblies, and control switches.



Auxiliary Lighting
Figure 2

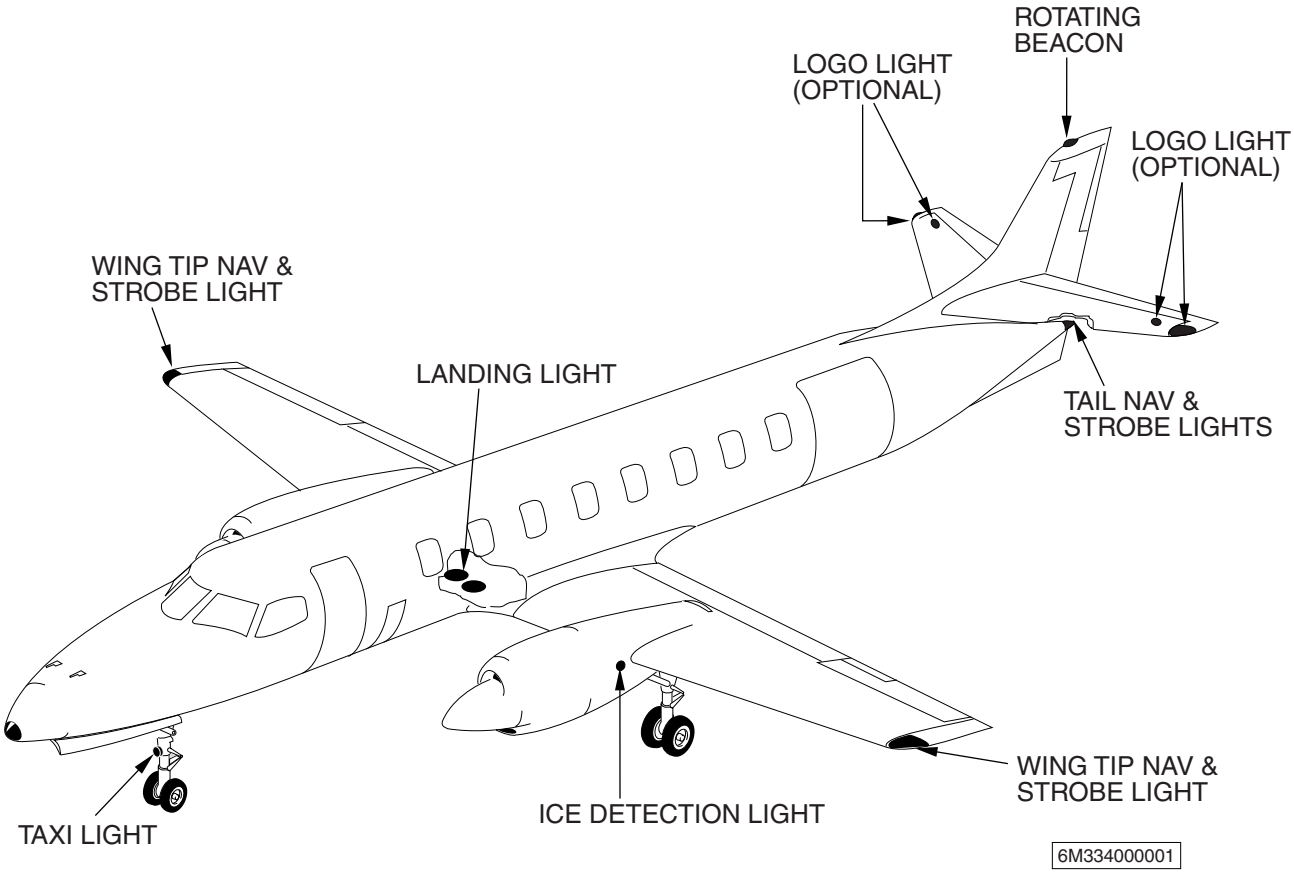
C. The auxiliary light cycle switch, located inside on the aft side of nose baggage compartment access door, when pressed, energizes a time cycle relay, located under the forward right console. When the time cycle relay is energized, electrical current is supplied from the right battery to the cargo compartment and nose baggage compartment lights. The time cycle relay allows illumination of the auxiliary lights for approximately five minutes.

NOTE: The cargo door warning light, located on the aft side of cockpit partition (forward of cabin entrance door), is included in the auxiliary light circuitry and illuminates during the auxiliary light time cycle.

EXTERIOR LIGHTS – DESCRIPTION AND OPERATION

1. General

A. Exterior lighting provisions for the SA226 series aircraft consists of navigation lights, strobe lights, a rotating beacon, two wing deice detection lights, a taxi light, and as a customer option, tel-tail (logo) lights are installed on the upper and lower horizontal stabilizers. 28 VDC electrical power for the exterior lighting is provided from the nonessential bus, except strobe lights which are powered from the left essential bus, Control of exterior lighting is provided by switches located on the left switch panel.



Exterior Lights
Figure 1

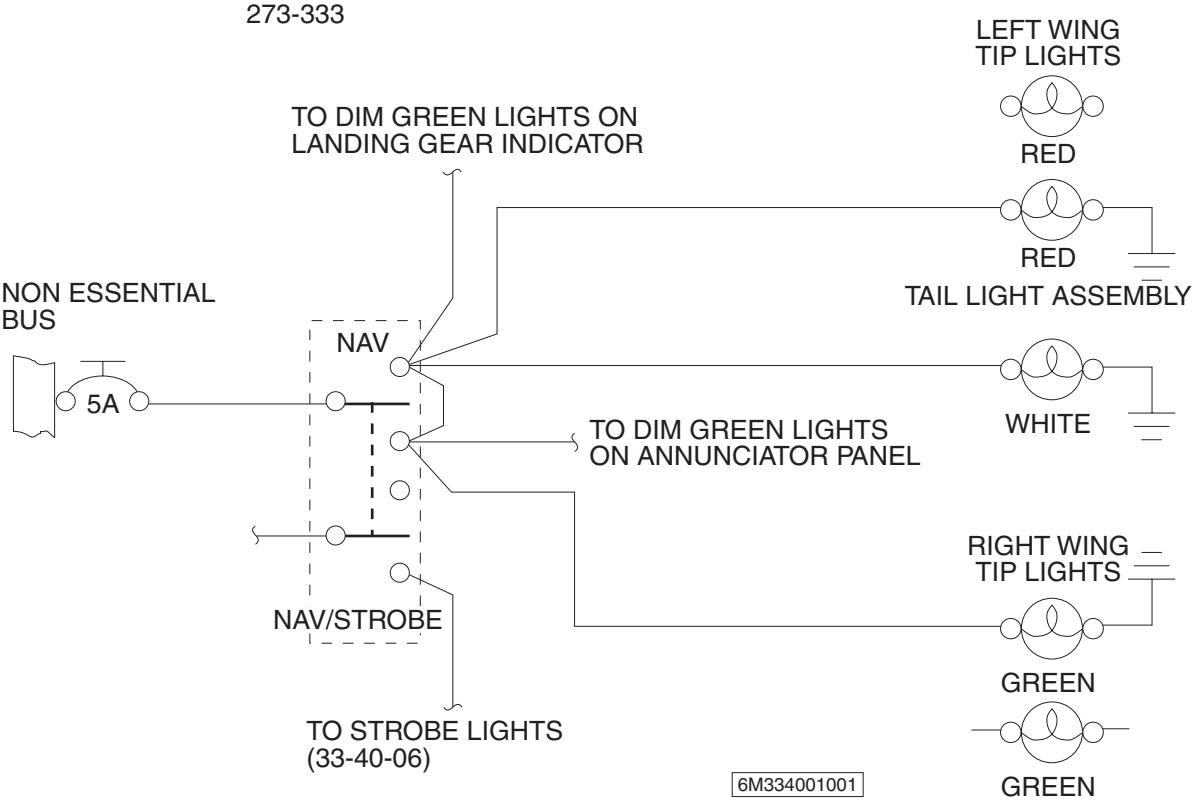
NAVIGATION LIGHTS – DESCRIPTION AND OPERATION

1. General

A. All SA226 series Aircraft are equipped with wing and tail navigation lights. The wingtip navigation lights are recessed into the right and left wingtips. The left wingtip navigation light assembly consists of two red reflectorized lamp units stacked one above the other. The right wingtip navigation light assembly consists of two green reflectorized lamp units stacked one above the other. A clear plexiglass shield, forming the leading tip of the wing, covers the navigation light assemblies. The tail navigation light shares a single lens and head assembly with the tail strobe light. A small white navigation light is mounted in a receptacle in the center of the strobe head assembly. The navigation light system is controlled by a switch (NAV & STROBE) located on the left switch panel and is energized from a five amp circuit breaker on the 28 VDC nonessential bus.

NOTE: When navigation lights are energized, an annunciator panel relay causes the green warning lights on annunciator panel to dim.

NOTE: LANDING GEAR INDICATOR LIMITED OUT EFFECTIVE T279-333
AT062-074
TC232, 233
273-333



Navigation Light System
Figure 1

NAVIGATION LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Navigation Lights

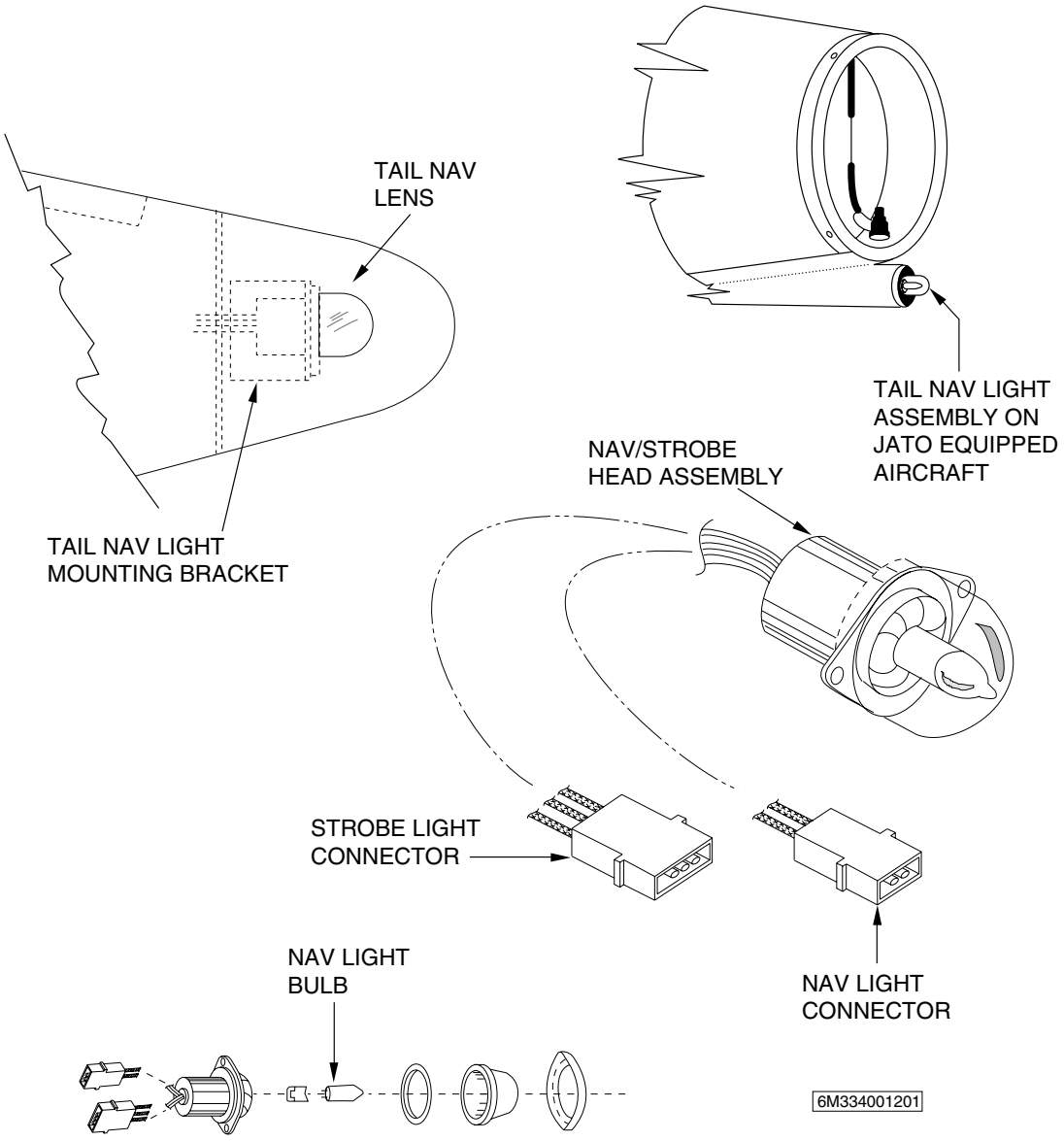
CAUTION: ENSURE ELECTRICAL POWER TO TAIL NAVIGATION LIGHT ASSEMBLY IS OFF.

A. Removal – Tail Navigation Light Assembly

NOTE: For Removal/Installation of Tail Navigation Light Assembly refer to Section 33-40-06, Removal – Tail Strobe Light Assembly.

B. Removal – Tail Navigation Light Bulb

(1) Gain access to tail navigation light assembly by removing retaining screws and removing clear plexiglass covering aft end of stinger assembly.



Tail Navigation Light
Figure 201

(2) Remove retaining screws and remove lens retainer.

(3) Remove lens.

(4) Remove navigation light bulb.

C. Installation – Tail Navigation Light Bulb

(1) Insert light bulb in tail nav/strobe head assembly.

(2) Position lens retainer on lens and secure with two screws.

(3) Position clear plexiglass covering and secure with retaining screws.

D. Removal – Tail Navigation Light Lens

(1) Gain access to tail navigation light assembly by removing retaining screws and removing clear plexiglass covering aft end of stinger assembly.

(2) Remove two screws and remove lens retainer.

(3) Remove lens.

E. Installation – Tail Navigation Light Lens

(1) Position lens retainer on lens and secure with two screws.

(2) Position clear plexiglass covering and secure with retaining screws.

F. Removal – Wingtip Navigation Light

CAUTION: ENSURE ELECTRICAL POWER TO WINGTIP NAVIGATION LIGHT IS OFF.

(1) Gain access to electrical wiring to navigation light by removing screws securing wingtip and removing wingtip.

(2) Detach electrical wiring to navigation light.

(3) Gain access to navigation light installation by removing screws and dimpled washers securing the clear plastic wingtip navigation light cover shield and by removing shield.

(4) Remove screw securing navigation light retainer.

(5) Withdraw navigation light.

G. Installation – Wingtip Navigation Light

(1) Position navigation light and secure with retainer.

(2) Connect electrical wiring to navigation light.

(3) Position clear plastic navigation light cover shield and secure with screws.

(4) Position wingtip and secure with screws.

(5) Perform operational check.

NOTE: Screws in navigation light cover and wingtip should be non magnetic.

H. Removal – Wingtip Navigation Light Lens and Bulb

(1) Gain access to navigation light lens and bulb by removing screws and dimpled washers securing the clear plastic wingtip navigation light cover shield.

(2) Remove screw securing lens retainer.

(3) Remove lens.

(4) Remove bulb.

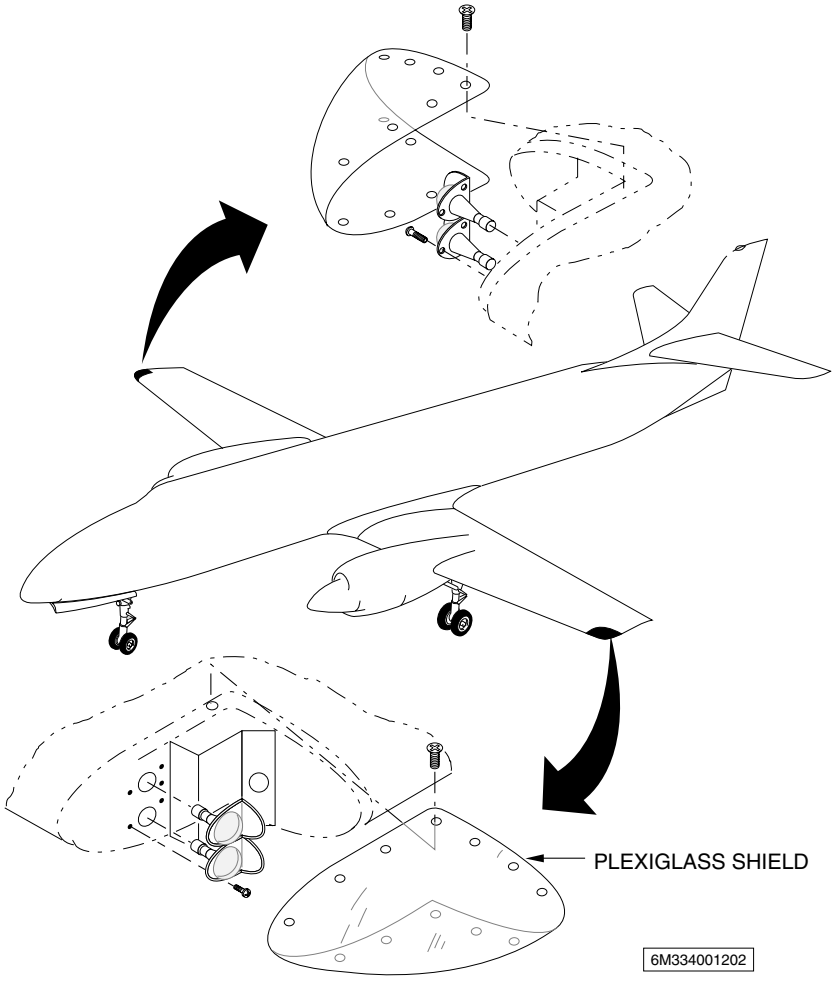
I. Installation – Wingtip Navigation Light Lens and Bulb

(1) Install bulb.

(2) Position lens retainer around lens and secure with screw.

(3) Position clear plastic wingtip cover shield and secure with screws and dimpled washers.

NOTE: Screws in navigation light cover and wingtip should be non magnetic.

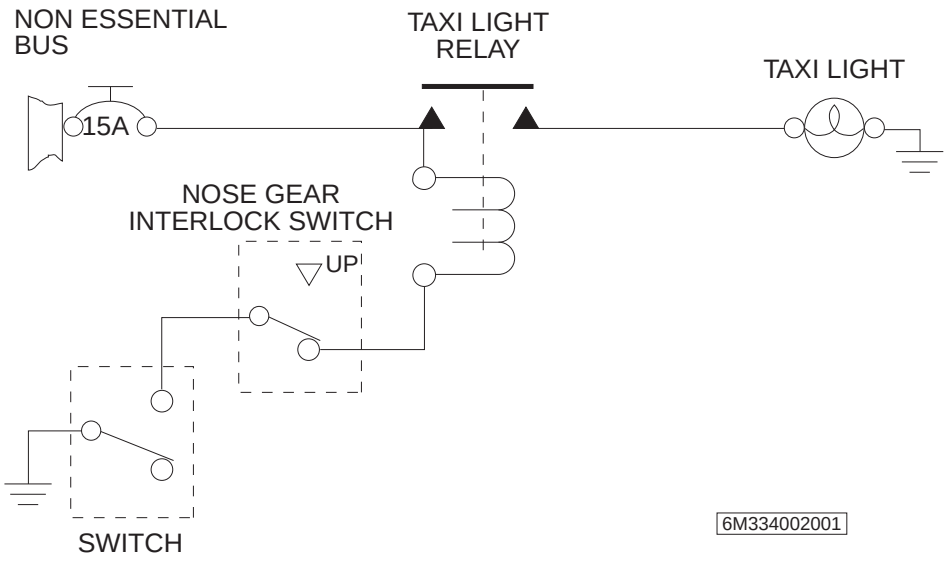


Wingtip Navigation Lights
Figure 202

TAXI LIGHT – DESCRIPTION AND OPERATION

1. General

A. A taxi light is mounted on the nose gear strut to provide illumination for the pilot during night ground operations. The taxi light system consists of a sealed beam and housing assembly, control switch, circuit breaker, power relay (located on the forward side of the forward pressure bulkhead), and nose gear switch. The sealed beam lamp is enclosed in a housing assembly which is attached to the nose gear strut by a reinforced bracket assembly. A control switch (TAXI LIGHT) located on the left switch panel energizes the taxi light power relay which feeds current to the light from the 28 VDC nonessential bus through a 15 amp circuit breaker. The nose gear interlock switch functions to ensure that taxi light does not remain illuminated when retracted into the wheel well. When the nose gear retracts, the door actuator assembly presses the interlock switch which breaks the taxi light electrical circuit extinguishing the light.



Taxi Light System
Figure 1

TAXI LIGHT – MAINTENANCE PRACTICES

1. Removal/Installation – Taxi Light

A. Removal – Taxi Light Housing

(1) Remove sealed beam.

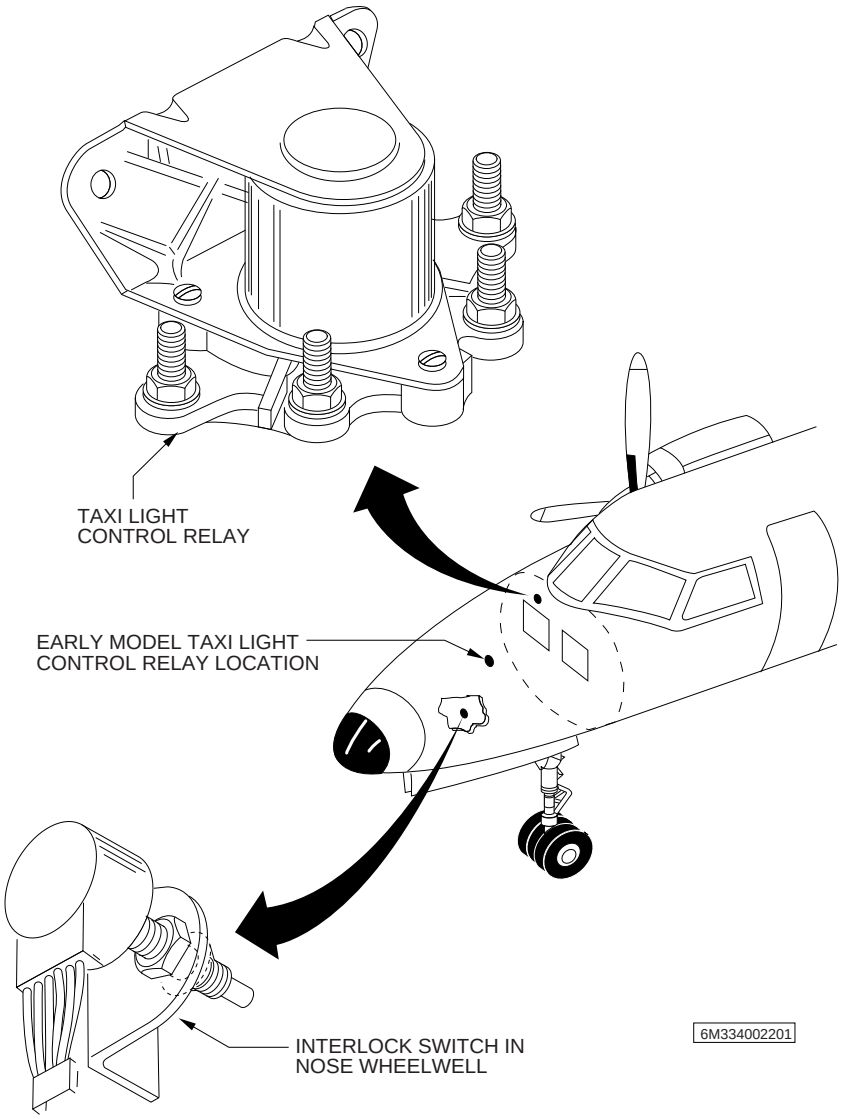
(2) Remove locknut and washer attaching housing to bracket.

(3) Remove housing.

B. Installation – Taxi Light Housing

(1) Position taxi light housing and secure with locknut and washer.

(2) Install sealed beam.



Taxi Light Interlock Switch and Control relay
Figure 201

TAXI LIGHT – MAINTENANCE PRACTICES

C. Removal – Sealed Beam

(1) Remove screw from sealed beam retaining ring and remove retaining ring.

(2) Remove sealed beam from lamp housing.

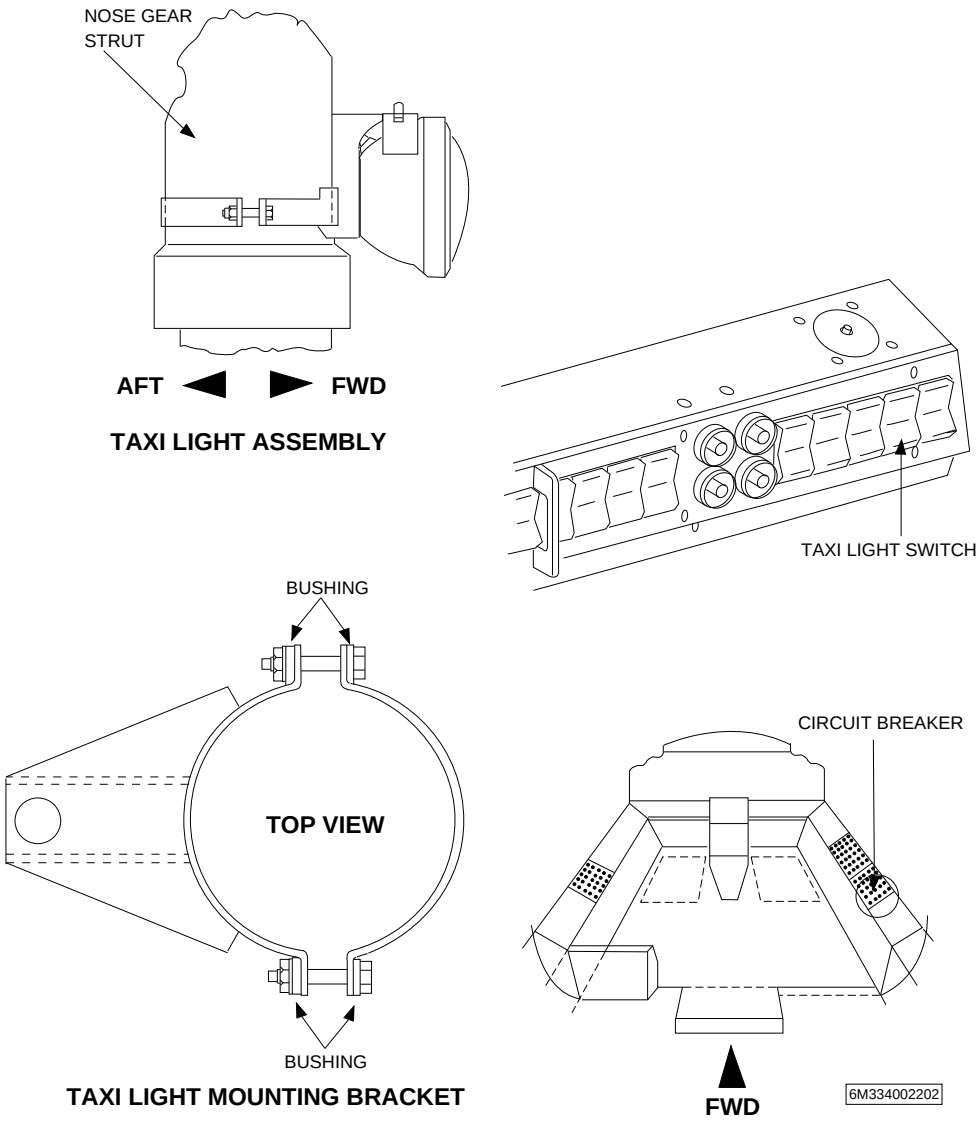
(3) Remove screws securing electrical wiring to sealed beam.

(4) Remove sealed beam.

D. Installation – Sealed Beam

(1) Secure electrical wiring to sealed beam with attaching screws.

(2) Position sealed beam in housing.



Taxi Light Assembly and Controls
Figure 202

(3) Position retaining ring and gasket on sealed beam.

(4) Secure retaining ring with screw.

(5) Perform functional check.

E. Removal – Taxi Light Bracket

(1) Remove sealed beam.

(2) Remove housing.

(3) Remove bolts, nuts, washers, and bushings holding bracket together.

(4) Separate and remove brackets.

F. Installation – Taxi Light Bracket

(1) Position brackets and install bolts, nuts, washers, and bushings. Refer to Figure 201 for bushing locations.

(2) Install housing.

(3) Install sealed beam.

2. Adjustment/Test – Functional Test (Taxi Light)

A. Adjustment – Taxi Light

(1) Connect APU to aircraft.

(2) Turn battery switches ON.

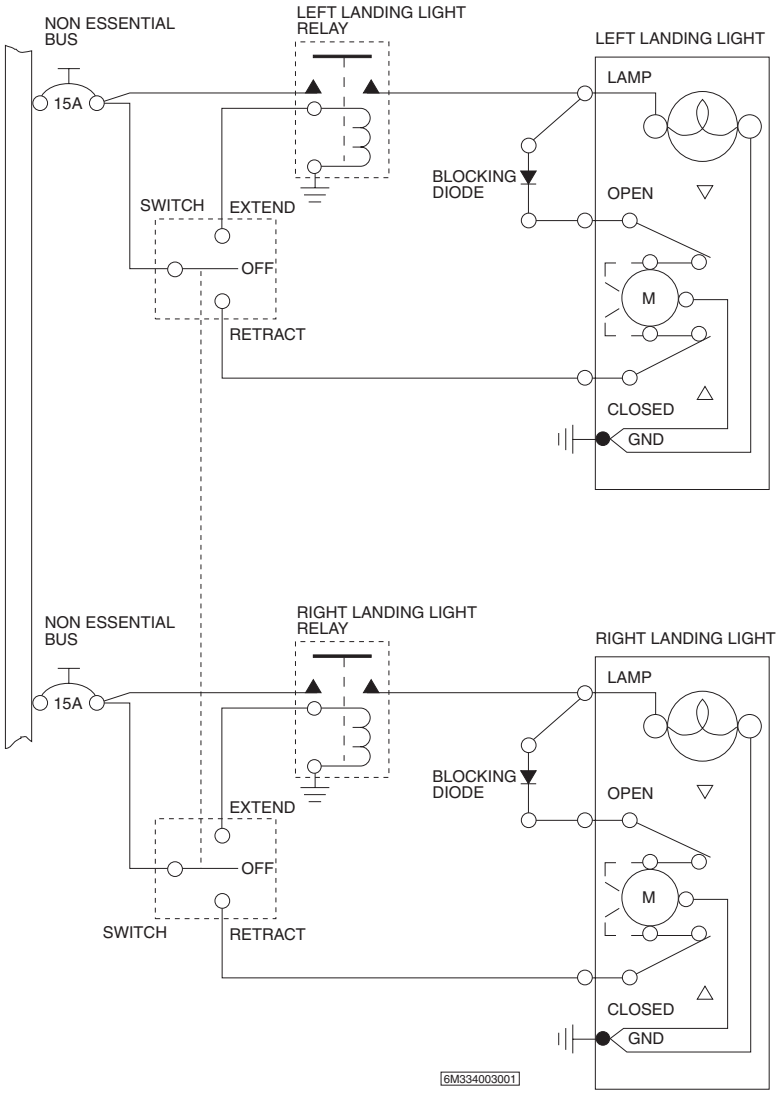
(3) Turn taxi light switch ON.

(4) Depress nose gear interlock switch. Notice that taxi light extinguishes.

LANDING LIGHTS – DESCRIPTION AND OPERATION

1. General

A. Two landing lights mounted to the underside of the center wing section form an arc of light forward of the aircraft allowing the pilot to view critical areas during night landing and taxiing operations. The aircraft's landing light system consists of a sealed beam assembly, housing and motor assembly, power relays, control switch, and circuit breakers. The sealed beam is secured within a lamp housing unit. The housing unit is attached to a gear arm which extends the light to A 90° position, or retracts it to normal flush-mounted position. (Note: SA226-T aircraft landing lights retract only to approximately 3/8 inch of flush position). The sealed beam is made directional by a reflector shield which directs the light forward and outward. The power unit, consisting of an armature, a three stage reduction gear, and microswitches, powers the light assembly to an extended or retracted position. The power relays, when energized, provide an electrical path to their respective landing light. A single switch on the left switch panel (EXTEND, OFF, RETRACT) controls light operation, and energizing current is provided from separate 15 amp circuit breakers on the 28 VDC nonessential bus.



Landing Light System
Figure 1

LANDING LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Landing and Recognition Lights

A. Removal – Sealed Beam

(1) Remove retaining ring securing sealed beam.

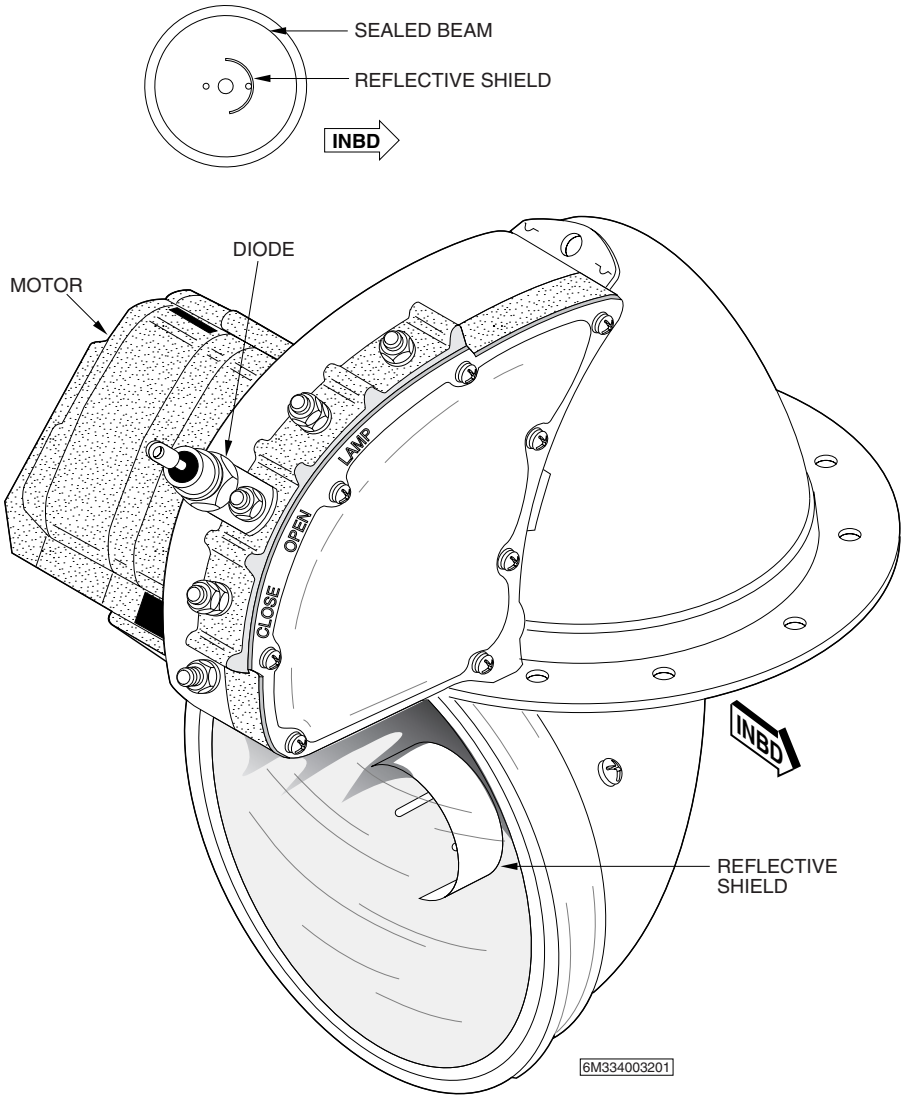
(2) Withdraw sealed beam and remove screws securing electrical wiring to sealed beam.

B. Installation – Sealed Beam

(1) Attach electrical wiring contacts to sealed beam.

(2) Position sealed beam in light housing with reflecting shield of sealed beam positioned inboard, (see Figure 201).

(3) Secure sealed beam in light housing with retaining ring.



Landing Light
Figure 201

C. Removal – Landing Light Assembly

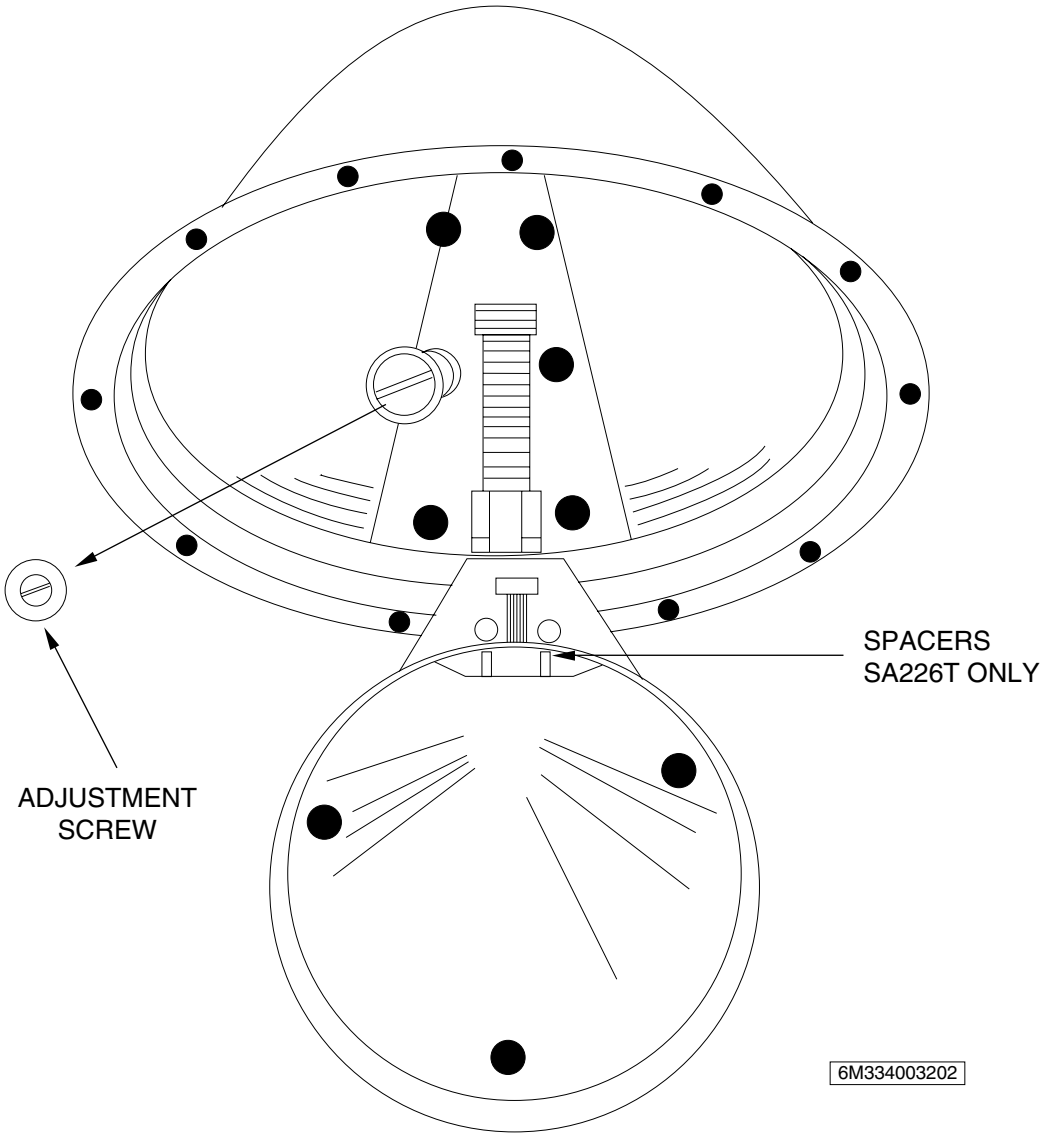
CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Remove screws securing landing light assembly to underside of center wing section.

(2) Withdraw light assembly enough to detach electrical wiring from light assembly.

(3) Detach electrical wiring.

(4) Remove light assembly.



Landing Light
Showing Adjustment Screw
Figure 202

D. Installation – Landing Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO LIGHT ASSEMBLY IS OFF.

(1) Attach electrical wiring to light assembly.

CAUTION: ENSURE ELECTRICAL TERMINALS CLEAR ALL FUEL LINES BY 1/2 INCH (SEE SERVICE NOTE 009).

(3) Position light assembly and secure with attaching screws.

NOTE: Ensure the filament shield is toward the pilot.

2. Adjustment/Test – Landing Lights

A. Adjustment – Landing Lights

(1) Adjustment (increasing/decreasing degree of landing light opening).

CAUTION: NEVER TURN ADJUSTMENT SCREW COUNTERCLOCKWISE MORE THAN THREE TURNS WITHOUT CLOSING AND OPENING LIGHT FOR FURTHER ADJUSTMENT.

(a) Turn adjustment screw clockwise with a common screwdriver to increase degree of opening, see Figure 202.

NOTE: Each full turn changes opening about one degree.

(b) Turn adjustment screw counterclockwise to decrease degree of opening, see Figure 202.

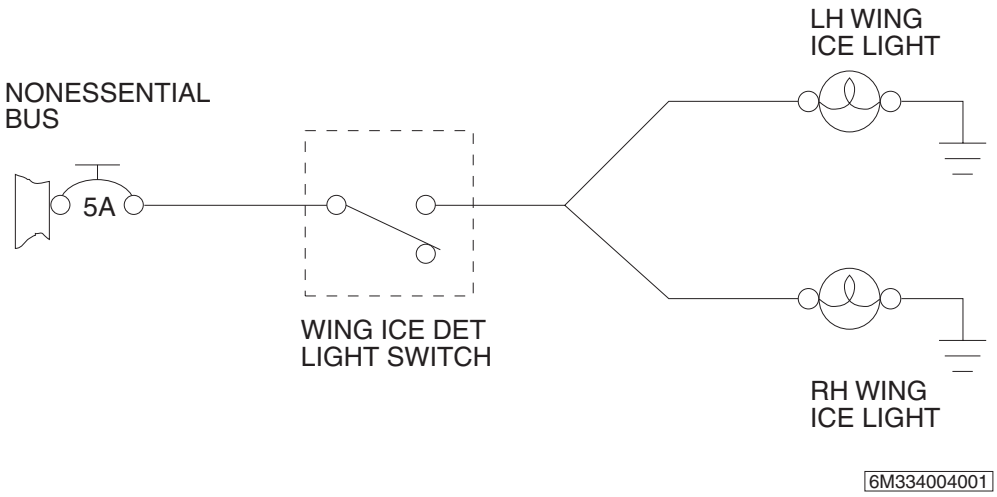
NOTE: Lamp housing will only adjust to 90° from base housing. The SA226-T aircraft landing lights should be extended to more than 90°, see Figure 202.

B. Lubrication – lubricate armature and gears of power unit with mil spec grease P/NO-6085.

WING ICE DETECTION LIGHTS – DESCRIPTION AND OPERATION

1. General

A. Wing ice detection lights are installed on the outboard side of each engine nacelle to detect the formation of ice on critical parts of the wing. The wing ice detection light system consists of a sealed beam, a bracket assembly, clear plexiglass lens panel, two turnbuckles, a control switch, and a circuit breaker. The sealed beam is bracket mounted behind a separate clear lens panel, and the two turnbuckles provide accurate alignment of the light. Energizing current is supplied from a five amp circuit breaker on the 28 VDC nonessential bus. A switch (WING ICE) on the left switch panel controls the lights.



Ice Detection Light
Figure 1

WING ICE DETECTION LIGHTS – MAINTENANCE PRACTICES

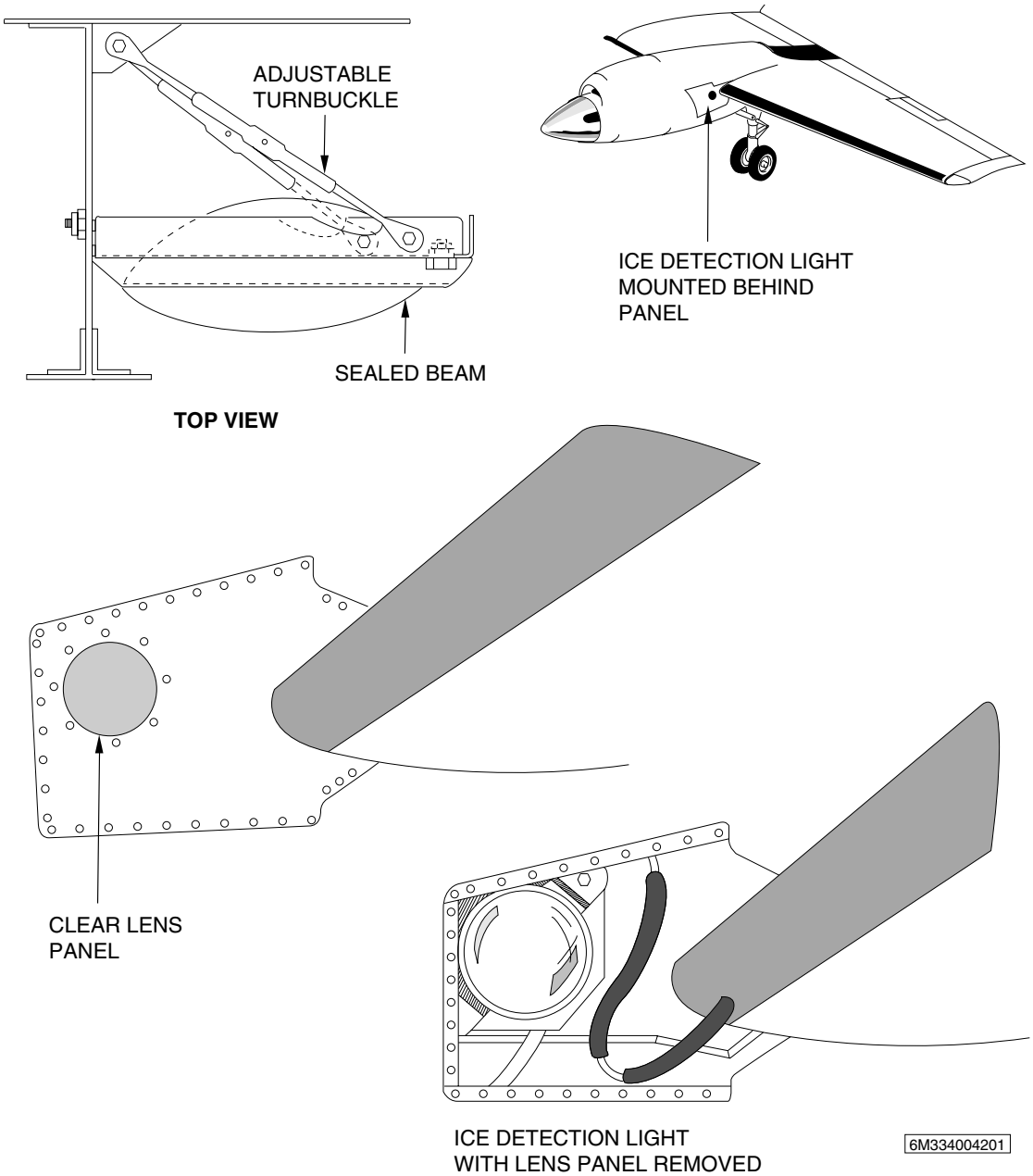
1. Removal/Installation – Wing Ice Detection Lights

A. Removal – Ice Detection Light Assembly

CAUTION: ENSURE THAT ELECTRICAL POWER TO ICE DETECTION LIGHT IS OFF.

(1) Remove screws securing ice detection light access panel and remove panel.

(2) Detach electrical connections from ice detection light assembly.



Wing Ice Detection Light
Figure 201

(3) Remove screws securing turnbuckles to wing ice detection light.

(4) Remove screws securing frame of ice detection light to aircraft structure and remove ice detection light.

B. Installation – Ice Detection Light Assembly

CAUTION: ENSURE ELECTRICAL POWER TO ICE DETECTION LIGHT IS OFF.

(1) Secure frame of ice detection light to aircraft structure with attaching screws.

(2) Secure turnbuckles to ice detection light with attaching screws.

(3) Connect electrical wiring to ice detection light.

(4) Check ice detection light alignment.

(5) Position ice detection light access panel and secure with retaining screws.

(6) Perform operational check.

C. Removal – Sealed Beam

(1) Remove screws securing ice detection light access panel and remove panel.

(2) Remove bolts and remove clips holding sealed beam to bracket.

(3) Withdraw sealed beam and remove screws securing electrical wiring to sealed beam.

D. Installation – Sealed Beam

(1) Attach electrical wiring contacts to sealed beam.

(2) Position sealed beam in bracket assembly and secure with clips and bolts.

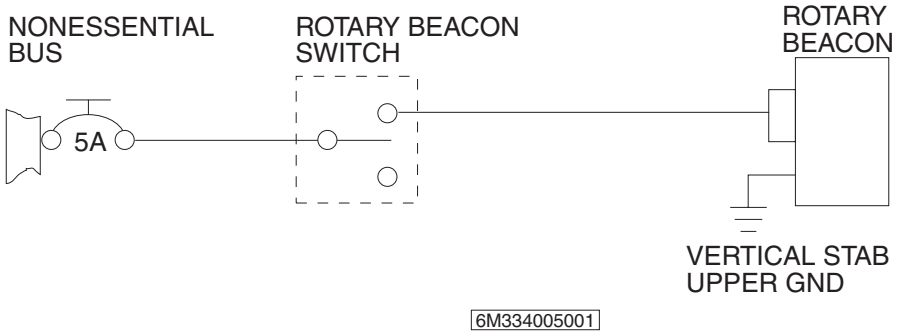
(3) Check Alignment.

(4) Position ice detection light access panel and secure with retaining screws.

ROTARY BEACON LIGHT – DESCRIPTION AND OPERATION

1. General

A. The rotary beacon is a rotating anti-collision light mounted on the top leading edge of the vertical stabilizer. The rotary beacon assembly consists of dual, tandem-mounted, motor driven lamps oscillating 180° out of phase providing a flashing characteristic of 60-100 flashes per minute. Electrical power for the rotary beacon is supplied from a five amp circuit breaker on the 28 VDC nonessential bus, and control is provided by a switch (ROT BEACON) on the left switch panel.



Rotary Beacon Light System
Figure 1

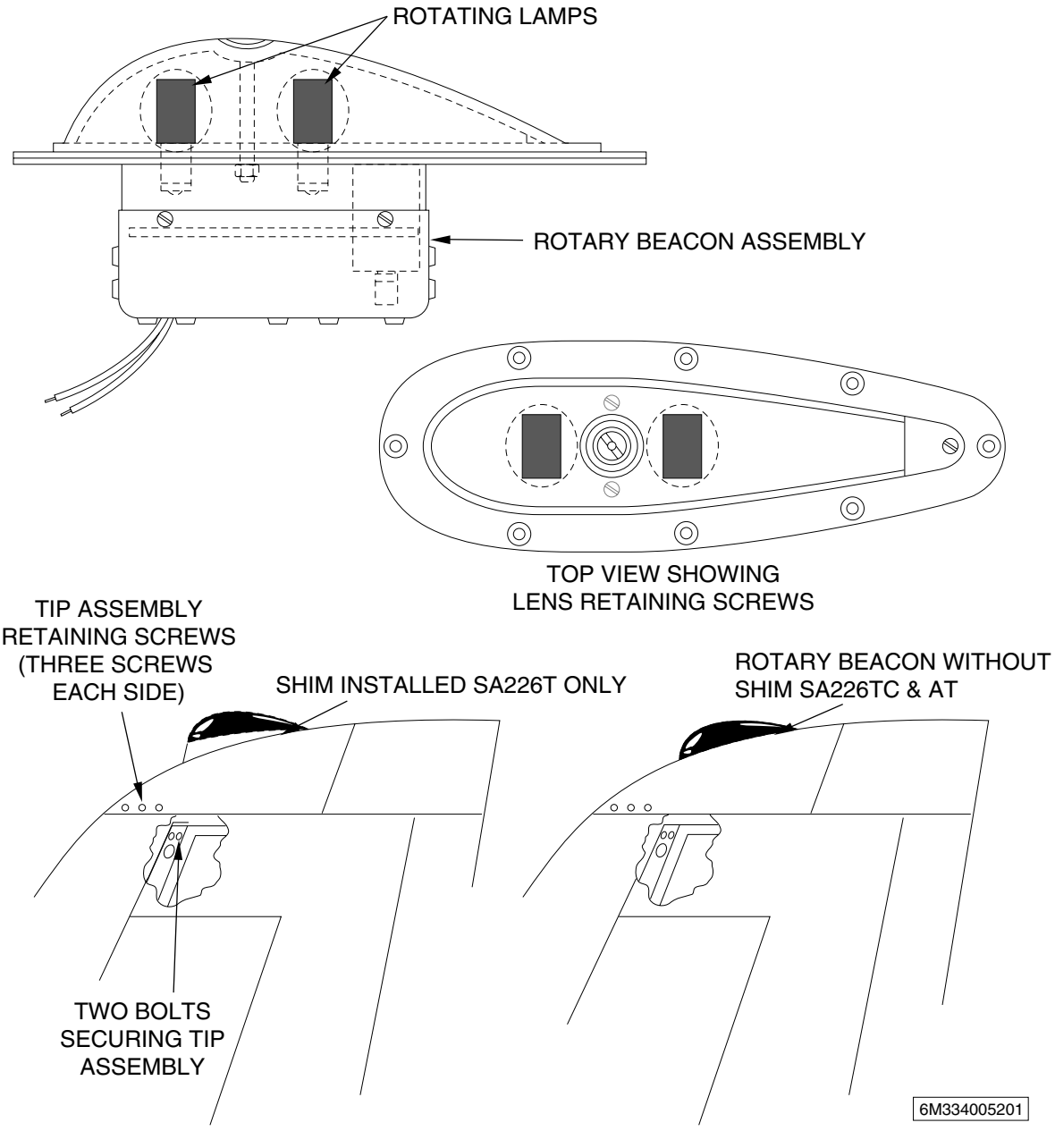
ROTARY BEACON LIGHT – MAINTENANCE PRACTICES

1. Removal/Installation – Rotary Beacon Light

A. Removal – Rotary Beacon Assembly

CAUTION: ENSURE ELECTRICAL POWER TO ROTARY BEACON ASSEMBLY IS OFF.

(1) Remove screws and bolts securing tip assembly to vertical stabilizer, see Figure 201.



Rotary Beacon Assembly
Figure 201

(2) Break sealer and remove tip assembly.

(3) Detach rotary beacon electrical wiring at QUICK DISCONNECT. Disconnect ground wire.

(4) Remove lens retaining screws and remove lens.

(5) Remove screws securing rotary beacon to tip assembly and remove rotary beacon.

B. Installation – Rotary Beacon Assembly

CAUTION: ENSURE ELECTRICAL POWER TO ROTARY BEACON ASSEMBLY IS OFF.

(1) Connect rotary beacon electrical wiring at QUICK DISCONNECT in tip assembly. Attach ground wire.

(2) Secure rotary beacon assembly to tip assembly with mounting screws and secure lens with its retaining screws.

(3) Apply 801 sealer to base of tip assembly. Position tip assembly and secure with six screws and two bolts.

C. Removal – Rotary Beacon Lamp Bulb

(1) Remove screws securing lens.

(2) Remove lens.

(3) Remove lamp bulb from its receptacle.

D. Installation – Rotary Beacon Lamp Bulb

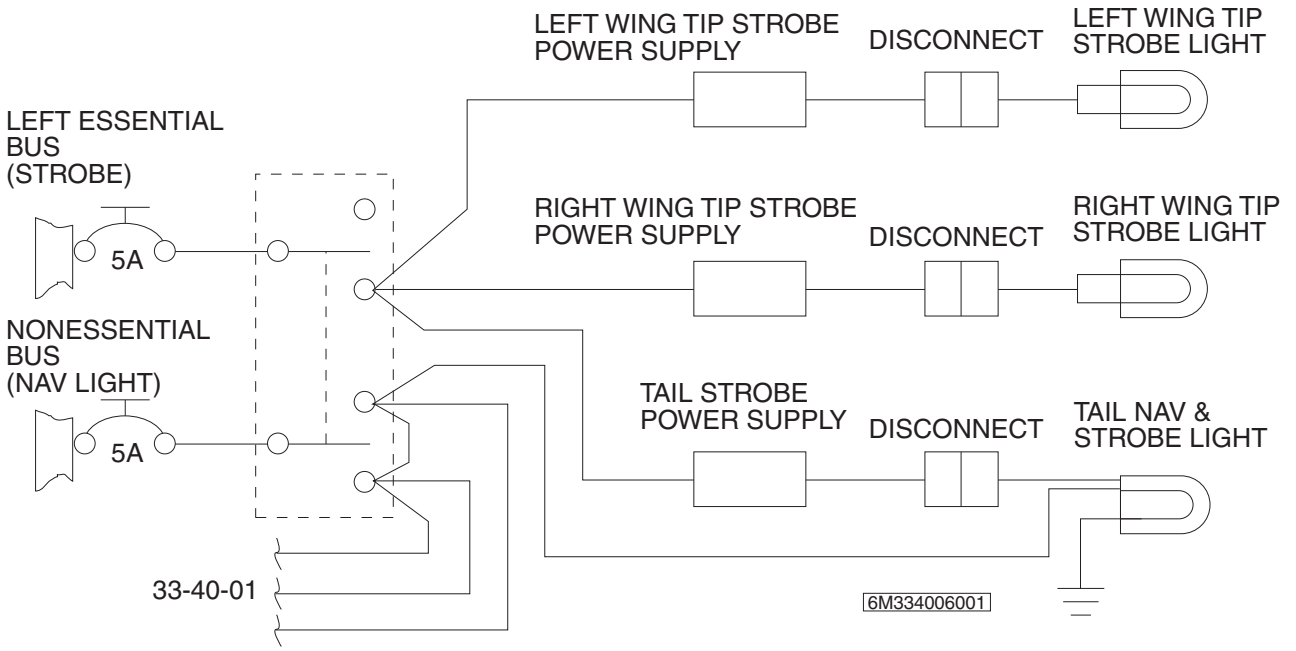
(1) Position lamp bulb in receptacle and turn until seated securely.

(2) Position lens and secure with screws.

STROBE LIGHTS – DESCRIPTION AND OPERATION

1. General

A. SA226 series aircraft are equipped with wingtip and tail strobe lights which serve as the aircraft's anti-collision light system. The strobe light system consists of a strobe head, power supply, circuit breaker, and control switch. The strobe head consists of a xenon flash tube, condenser, lens, and retainer. The power supply contains a solid state timing circuit and provides the high voltage required to flash the xenon tube. The xenon tube is flashed by a condenser discharge system. A condenser is charged to approximately 450 VDC, and then is discharged across the xenon tube at controlled intervals. The condenser discharge is triggered by an external pulse received from the strobe power supply timing circuit. Energizing current us supplied to the strobe power supply from a five amp circuit breaker on the 28 VDC left essential bus. Strobe lights are controlled by a switch (NAV & STROBE) located on the left switch panel.



Strobe Lights
Figure 1

1. Removal/Installation – Strobe Lights

A. Removal – Wingtip Strobe Light Power Supply Unit

(1) Gain access to power supply by removing wingtip. Remove screws securing wingtip. Remove wingtip and secure wingtip to upper wing with two screws.

(2) Disconnect electrical wiring from strobe light power supply to strobe head and cut power wire at splice.

(3) Remove four bolts securing power supply to wing rib and remove power supply unit.

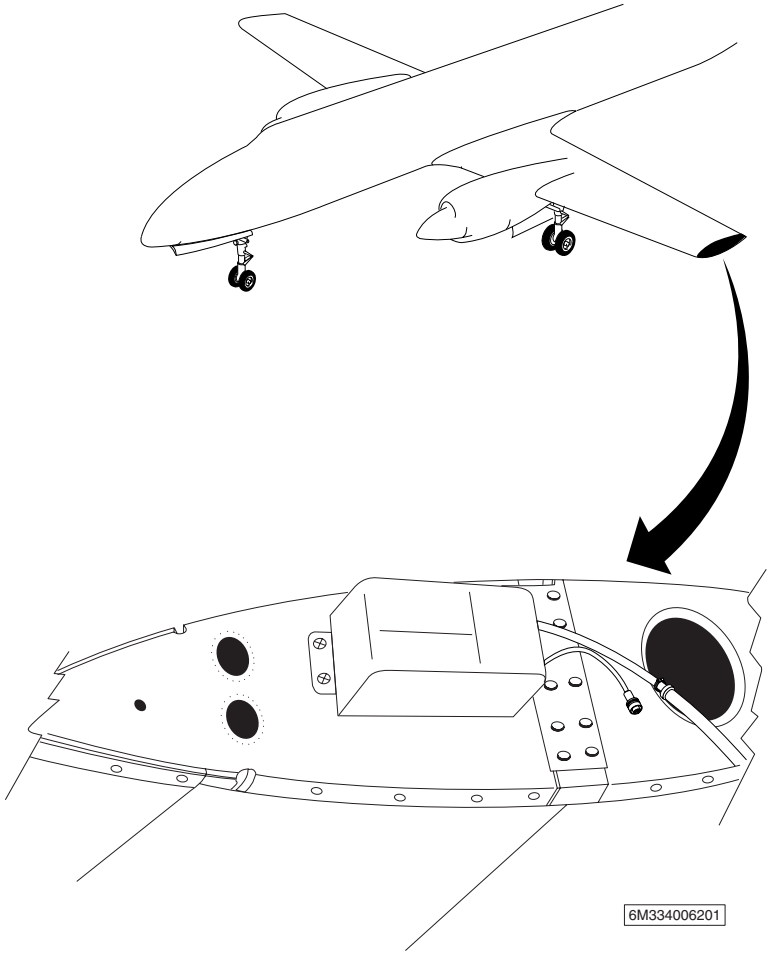
B. Installation – Wingtip Strobe Light Power Supply Unit

(1) Position strobe light power supply unit and secure with four bolts.

(2) Connect electrical wiring to power supply unit. Splice power wire.

(3) Position wingtip and secure with attaching screws.

(4) Perform operational check.



Wingtip Strobe Light Power Supply
Figure 201

C. Removal – Wingtip Strobe Light Lens

(1) Remove screws securing lens retainer.

(2) Remove lens retainer.

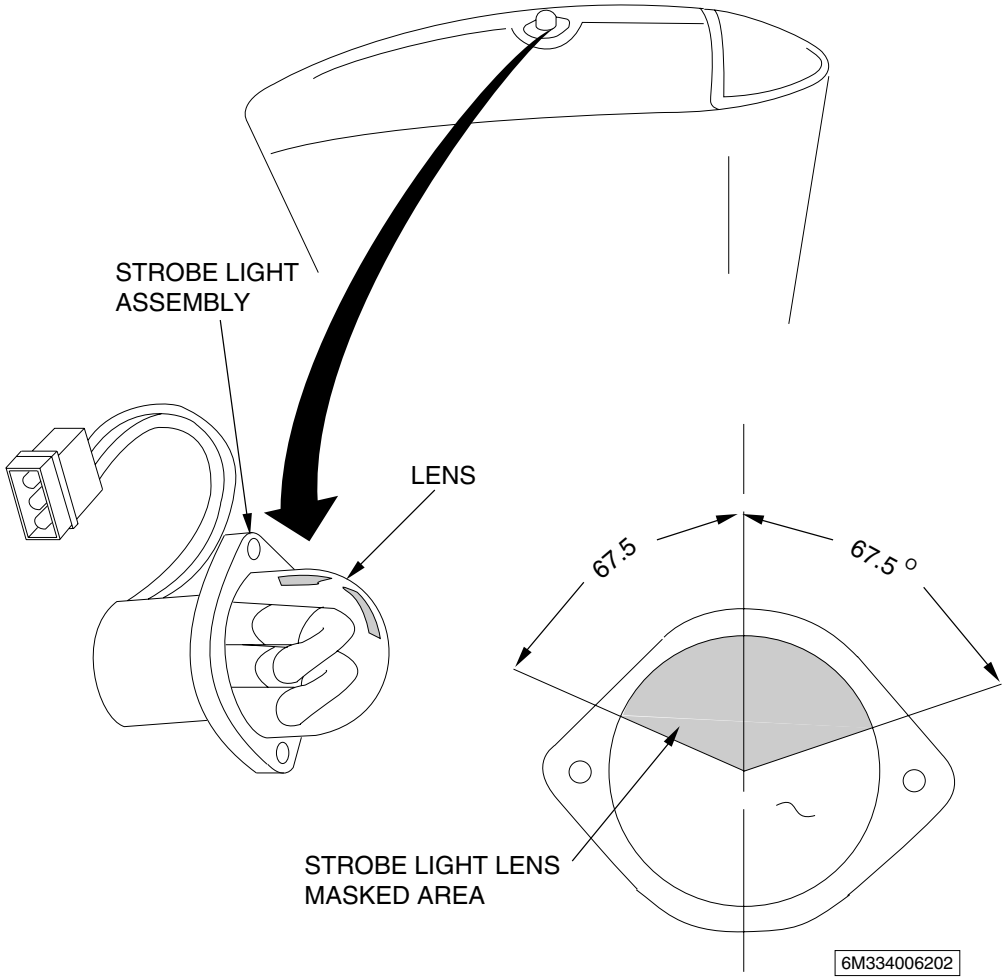
(3) Remove lens from strobe light head assembly.

D. Installation – Wingtip Strobe Lights Lens

NOTE: To reduce pilot interference caused by reflection of the wingtip strobe light, mask upper 135° of lens by painting with black enamel paint, see Figure 202 (Effectivity T247-999).

(1) Position lens on strobe head assembly.

(2) Position lens retainer and secure with attaching screws.



NOTE: TO PREVENT STROBE LIGHT INTRUSION INTO THE COCKPIT ON ALL SA226-T AIRCRAFT WITH WEYLEN STROBE LIGHTS, MASK THE LENS WITH BLACK PAINT.

Wingtip Strobe Light
Figure 202

E. Removal Wingtip Strobe Light Head Assembly

(1) Remove screws attaching retainer and lens to strobe light head assembly.

(2) Pull out assembly and disconnect electrical wiring attached to strobe head assembly.

(3) Remove strobe light head assembly.

F. Installation – Wingtip Strobe Light Head Assembly

(1) Attach electrical wiring to strobe light head assembly.

(2) Attach strobe light head assembly to wingtip with retainer and screws.

(3) Perform operational check.

G. Removal – Tail Strobe Light Lens

(1) Gain access to strobe light lens by removing four screws and removing clear plexiglass covering aft end of stinger assembly.

(2) Remove two screws and remove lens retainer of tail strobe light.

(3) Remove lens.

H. Installation – Tail Strobe Light Lens.

(1) Position lens and secure with retainer and two screws.

(2) Position clear plexiglass covering aft end of stinger assembly and secure with four screws.

I. Removal – Tail Strobe Light Power Supply

(1) Gain access to strobe light power supply by removing tail cone assembly.

(2) Disconnect electrical wiring connected to strobe light power supply.

(3) Remove four screws attaching power supply to upper surface of tail cone.

(4) Remove power supply.

J. Installation – Tail Strobe Light Power Supply.

(1) Position strobe light power supply and attach to upper surface of tail cone with four screws.

(2) Connect electrical wiring to power supply.

(3) Position tail cone assembly and secure with attaching screws.

K. Removal – Tail Strobe Light Assembly.

(1) Remove tail cone assembly to gain access to tail strobe electrical wiring disconnect.

(2) Remove tape covering and disconnect electrical wiring to tail strobe at quick disconnect.

(3) Remove screws and remove plexiglass lens cover.

(4) Remove screws and remove tail strobe lens retainer/lens and back out strobe head from bracket.

L. Installation – Tail Strobe Light Assembly

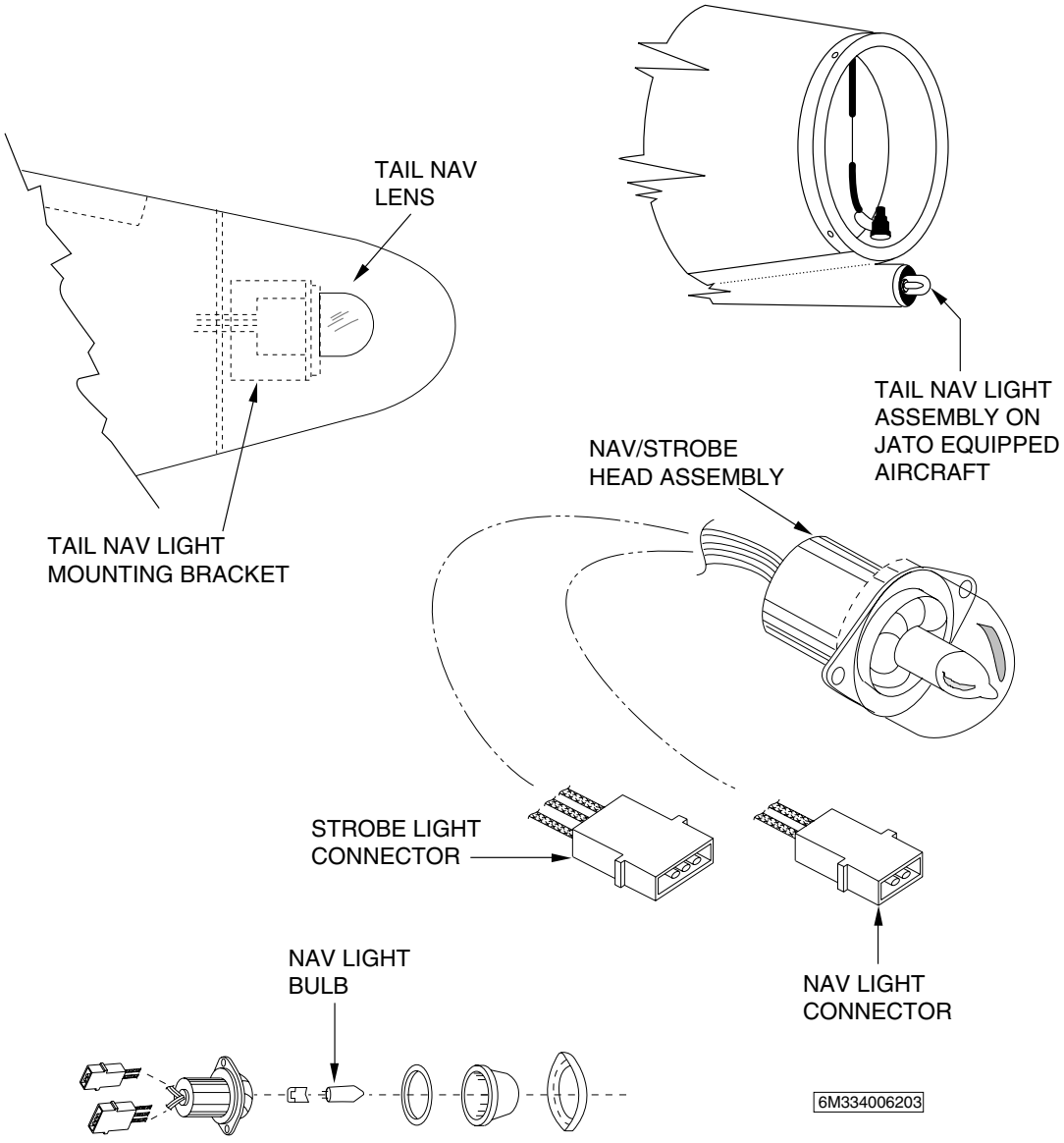
(1) Connect electrical wiring to tail strobe at tail cone disconnect and cover with tape covering.

(2) Insert tail strobe assembly in mounting bracket and attach with retainer and attaching screws.

(3) Position clear plexiglass lens cover and secure with attaching screws.

(4) Position tail cone assembly and secure with attaching screws.

NOTE: To prevent strobe light intrusion into cockpit on all SA226-T aircraft with weylen strobe lights, mask the lens with black paint.



Tail Strobe Light
Figure 203

2. Adjustment/Test – Strobe Lights

A. Adjustment – Functional Test and Troubleshooting

(1) Determine that there is the proper 28 VDC input voltage at the power supply input.

CAUTION: REVERSED POLARITY OF THE INPUT POWER FOR JUST AN INSTANT WILL PERMANENTLY DAMAGE THE POWER SUPPLY. THIS DAMAGE IS SOMETIMES NOT IMMEDIATELY APPARENT, BUT WILL CAUSE A FAILURE IN TIME. THE REVERSED POLARITY DESTROYS A PROTECTIVE CIRCUIT IN THE POWER SUPPLY, CAUSING SELF-DESTRUCTION FROM OVERHEATING OF THE POWER SUPPLY.

LET THE POWER SUPPLY BLEED DOWN FOR FIVE MINUTES BEFORE HANDLING.

(2) Clear all possible shorts at power supply, and check the power supply operationally. Disconnect the output cables from the power supply outlets, and connect an operating strobe head assembly directly to power supply outlet or a Whelen Strobe check unit.

(3) Apply the required voltage to the power supply input. If this application functions normally, look for the following problems in interconnecting cable.

(a) Check for malfunctions in interconnecting cables. If pins 1 and 3 are reversed, or if there is a short between pins 1 and 2 of the interconnecting cable, the power supply will be nonoperable until the short is cleared. A short of this type will not cause any permanent damage to the power supply. A discharge of the condenser across pin 1 and pin 3 will destroy the trigger circuit.

(b) Continuity checks: Check for continuity between the connectors of each interconnecting cable:

1 Pin 1 to pin 1 (red wire-anode)

2 Pin 2 to pin 2 (Black wire-ground) -

3 Pin 3 to pin 3 (White wire-trigger)

(c) Check for shorts from pins 1, 2, 3 to aircraft ground (pin 2 is aircraft ground in the power supply).

(d) Check for continuity in the wiring of the A432 tube socket assembly:

AMP CONNECTOR		TUBE SOCKET	COLOR WIRE
1	to	Pin 1	Red (+ 450 VDC Anode)
2	to	Pin 6 and 7	Black (Aircraft Ground)
3	to	Pin 4	White (Trigger)

Continuity Check
Table 201

(e) When pins 1 and 2 and 2 and 3 are reversed, the system will appear to operate normally, but these conditions will cause early flash tube failure.

(4) Replace the Xenon Flash Tube for the Following Reasons:

(a) Leaks around the seal of the wire to the glass. Leaks through eggshelling of the glass.

(b) Continuous glow (self-ionization).

(c) Hard firing tube (with age may become hard firing and will not fire on low battery).

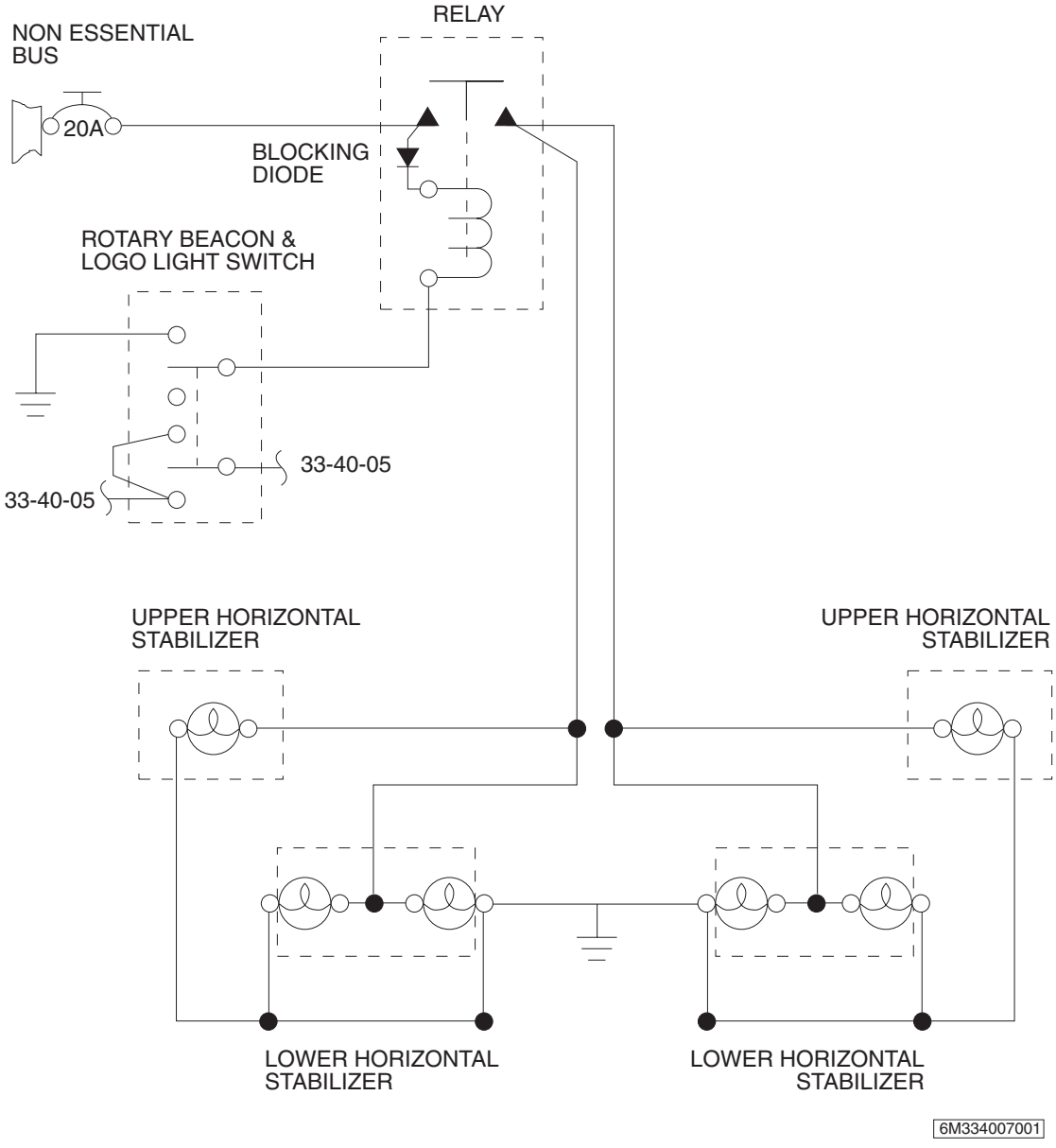
(d) Photo sensitive. Flashes when exposed to external light source but does not flash in darkness.

NOTE: To check a questionable flash tube, install it in a system and apply 22 volts. If the flash tube will operate at the reduced level, it has a great deal of service life in it.

LOGO LIGHTS – DESCRIPTION AND OPERATION

1. General

A. As a customer option, SA226 series aircraft are equipped with vertical tail flood lights mounted on the upper and lower horizontal stabilizer surfaces. The logo light system consists of upper and lower light assemblies, a control switch, control relay, and a circuit breaker. Each lower horizontal stabilizer light consists of double lamps, base assembly, reflector, yoke assembly, and class cover. Each upper stabilizer logo light assembly consists of one lamp, a base assembly, reflector, yoke assembly, and glass cover. A switch (ROTARY BEACON & TEL TAIL) located on the left switch panel, controls a relay, located inside the forward right console. When energized the relay powers the logo lights from a 20 amp circuit breaker on the 28 VDC nonessential bus.



Logo (Tel-Tail) Lighting System
Figure 1

LOGO LIGHTS – MAINTENANCE PRACTICES

1. Removal/Installation – Logo Lights

A. Removal – Logo Lights Upper or Lower Horizontal Stabilizer

CAUTION: ENSURE ELECTRICAL POWER TO LOGO LIGHTS IS OFF.

(1) Remove screws securing lens retainer and remove lens.

(2) Remove screws and washers securing lamp housing to stabilizer.

(3) Remove lamp housing.

(4) Withdraw lamp housing enough to detach electrical wiring.

(5) Remove logo light assembly.

(6) Scrape away sealant from around the gasket and doubler with wood or plastic scraper.

B. Installation – Logo Lights Upper or Lower Horizontal Stabilizer

CAUTION: ENSURE ELECTRICAL POWER TO LOGO LIGHTS IS OFF.

(1) Apply a bead of sealant around the gasket and doubler, then carefully wipe away excess sealant to provide smooth fillet installation.

NOTE: Installation must be waterproof.

(2) Connect electrical wiring to logo lamp assembly.

(3) Position lamp housing and secure with screws and washers.

(4) Align lamps for best illumination of vertical stabilizer, see Figure 201 for alignment procedures.

(5) Position and secure lens with lens retainer and screws. To properly secure cover glass, the screws should be tightened to the point that the cover glass is properly seated on the gasket and the "O" ring under the hold down screw washer is compressed without undue strain on the glass. Proceed as follows:

(a) Single lamp fixture should be torqued 5 to 6 inch-pounds maximum.

(b) Dual lamp fixtures should be torqued 8 to 9 inch-pounds maximum.

(6) Perform operational check.

2. Adjustment/Test – Logo Lights

A. Adjustment – Logo lights

NOTE: Adjust reflectors with lamps turned off.

Make adjustments at night or in darkened hangar.

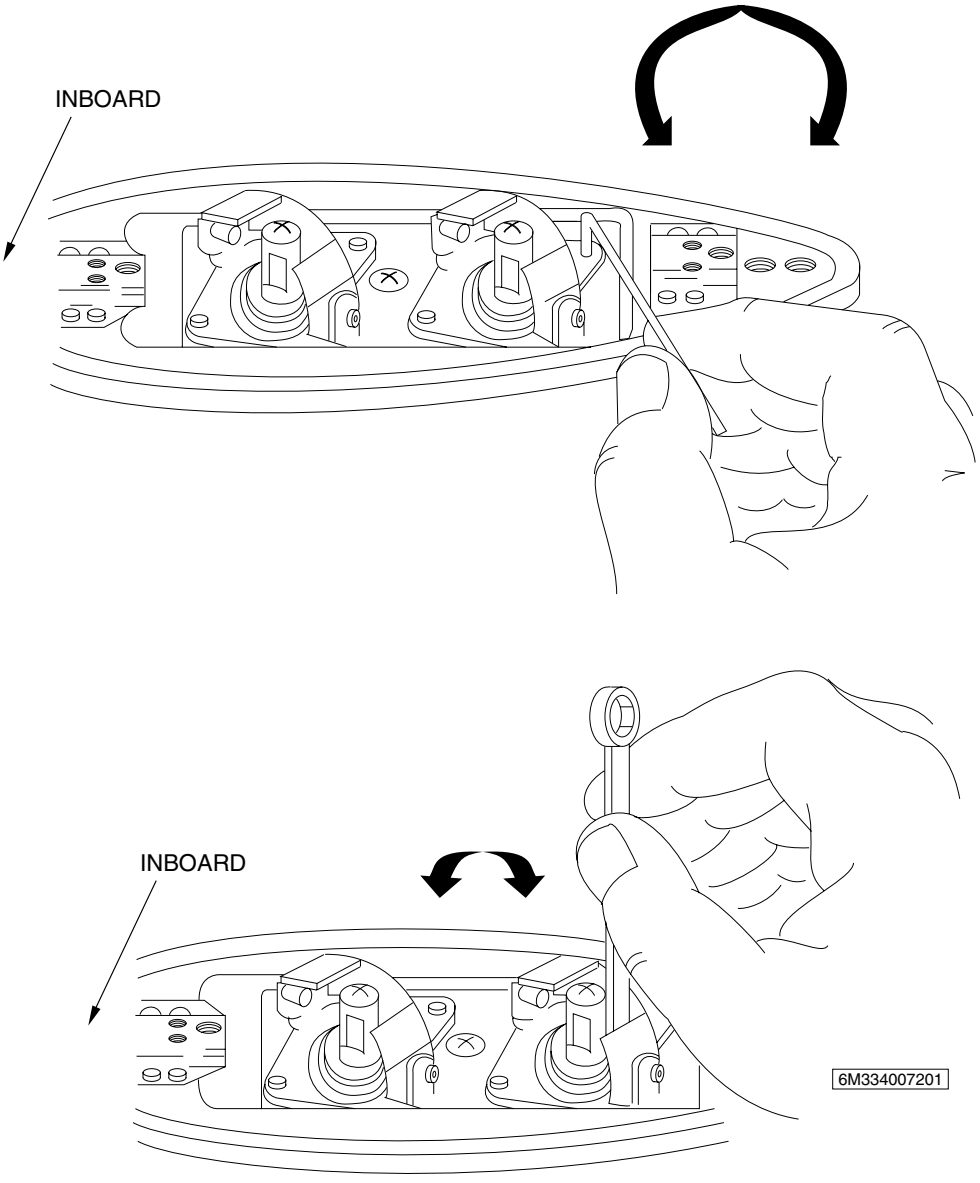
Do not apply power until steps **2.**, A., (1) through (4) are completed.

LOGO LIGHTS – MAINTENANCE PRACTICES

CAUTION: WHEN OPERATING LAMPS WITH GLASS COVER REMOVED, WEAR SAFETY GLASSES OR EQUIVALENT EYE PROTECTION. DO NOT LOOK DIRECTLY AT LAMP FILAMENT. LAMPS AND REFLECTORS BECOME EXTREMELY HOT. USE EXTREME CARE WHILE MAKING ADJUSTMENTS. DO NOT DEPOSIT FINGERPRINTS AND SKIN OIL ON LAMP. REMOVE BEFORE ILLUMINATION BY CLEANING WITH A GREASE FREE SOLVENT.LOGO LIGHTS – MAINTENANCE PRACTICES

(1) Remove cover glass.

(2) Loosen the two capscrews at the yoke clamp of each lamp (bottom of assembly base, see Figure 201) to allow the reflector to be rotated around the lamp.



Logo Light Adjustment
Figure 201

(3) Loosen the two l/4" hexhead screws on either side of lamp (see Figure 201) in the reflector yoke assembly to allow the reflector to be tilted.

(4) Aim reflectors at the tail surface. When adjusting the double light assembly, the reflector for the aft lamp should be aimed toward a point near the aft portion of the fuselage/vertical tail, and the reflector for the forward lamp should be aimed toward the forward portion.

(5) Turn on aircraft DC power and actuate light switch.

(6) Observe the light pattern and reposition reflectors if necessary.

(7) Set the glass cover on the fixture. Do not install attaching screws at this time.

(8) Observe that the cover glass changes the light pattern to eliminate bright and dark spots.

(9) Reposition reflectors, if required, to provide even coverage.

(10) Turn off electrical power and tighten adjustment screws. Recheck to be sure adjustment was not changed when screws were tightened.

(11) Install glass cover. To properly secure the cover glass, the screws should be tightened to the point that the cover glass is properly seated on the gasket and the "O" ring under the hold down screw washer is compressed without undue strain on the glass. Do not overtighten screws. Proceed as follows:

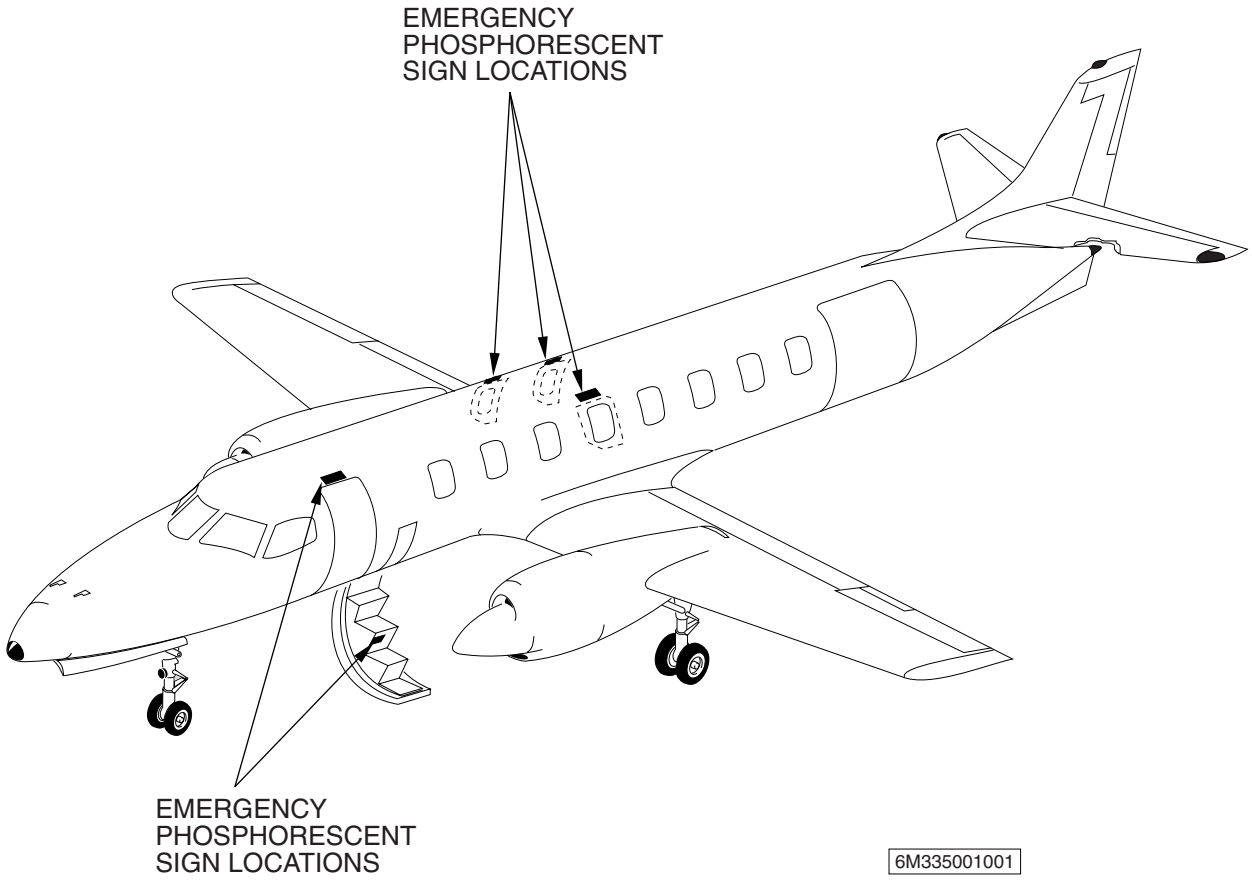
(a) Single lamp fixture should be torqued 5 to 6 inch-pounds (0.56 to 0.68 N·m) maximum.

(b) Dual lamp fixtures should be torqued 8 to 9 inch-pounds (0.90 to 1.02 N·m) maximum.

EMERGENCY PHOSPHORESCENT SIGNS – DESCRIPTION AND OPERATION

1. General

A. SA226-TC series aircraft are provided with phosphorescent signs which indicate emergency exit locations. The signs are flush-mounted in the interior paneling. See Figure 1 which shows locations displaying emergency exit signs.



Emergency Phosphorescent Signs
Figure 1

EMERGENCY PHOSPHORESCENT SIGNS – MAINTENANCE PRACTICES

1. Removal/Installation – Emergency Phosphorescent Signs

NOTE: Signs are attached to the interior paneling sections of the aircraft. To remove a sign assembly, the entire panel section attached to the sign assembly must be removed.

A. Removal – Emergency Sign

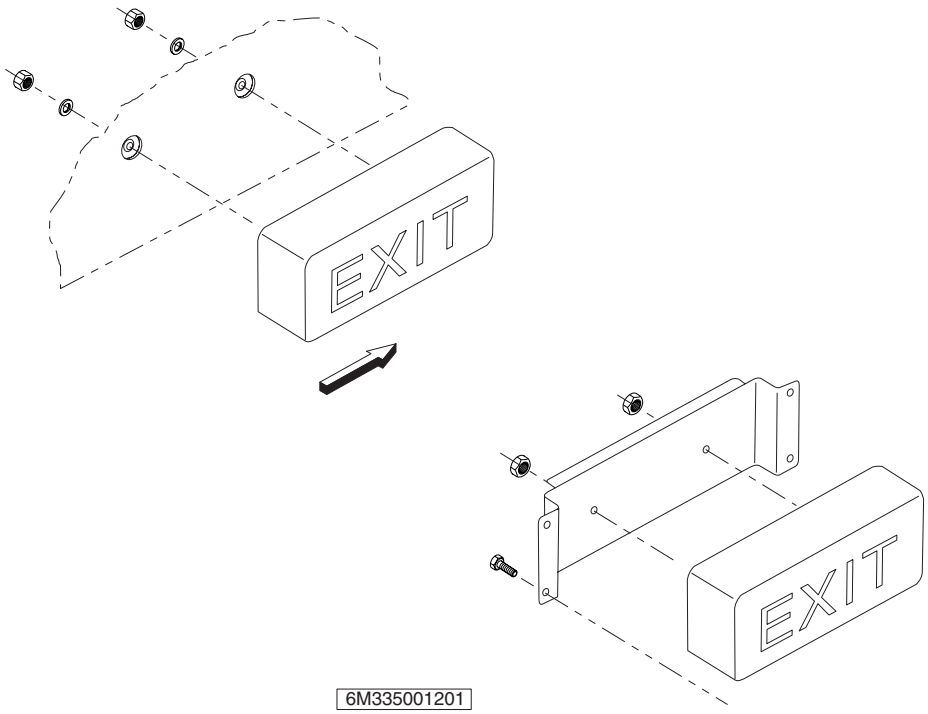
(1) Remove applicable panel.

(2) Remove screws attaching sign to bracket and remove sign.

B. Installation – Emergency Sign

(1) Secure sign to bracket with attaching screws.

(2) install panel.



Emergency Phosphorescent Signs
Figure 201

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MAINTENANCE MANUAL

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NAVIGATION – SERVICE INFORMATION

1. General

A. The following information is supplied as a supplement to this chapter. Subject matter is included which is not applicable to an ATA 100 format, but is still of interest to Merlin and Metro owner/operators. The information in this section is not part of the basic manual, and may be removed or retained at the discretion of the owner/operator.

B. Revisions and additions to this section will be made as additional information is available. Input from individual owner/operators is encouraged, and may be submitted to:

Fairchild Aircraft
Post Office Box 790490
San Antonio, Texas 78279-0490
Attention: Technical Publications Department

C. 34.01 – Pitot Static Check

(1) Checking the pitot static system can result in damage to a number of expensive instruments if not performed properly. The most common mistakes are:

(a) Increasing or decreasing the evacuation of the system at too rapid of rate, thereby damaging the rate of climb indicators, and in some cases the altimeters as well.

(b) Causing the airspeed indicator(s) to move backwards past zero and thereby damaging the internal gears of the instrument.

(c) Applying pressure to instruments designed for a lower than atmospheric pressures (altimeter, rate-of-climb, static side of the airspeeds).

(2) Always try to locate the pitot static tester inside of the aircraft so that the instrument panel is visible to the person operating the tester. The pitot static instruments in the instrument panel should always be monitored more closely than the instruments in the pitot static tester itself. Make sure that personnel are aware that a pitot static test is in progress (damage has occurred to various instruments by someone inadvertently pulling off a pitot tube or standing on a static line while a pitot static test was in progress).

(3) The airspeed indicators should be pulled off the peg during the static portion of a pitot static test. As vacuum is applied to the static system, the altimeter will show increasing altitude, the rate of climb will show an increasing rate of climb, and the airspeed will begin to indicate increasing airspeed.

(4) The airspeed will increase because the indicator senses the ratio between impact pitot pressure and static. As a vacuum is applied to the static system, the ratio between the static side of the airspeed and the pitot side (which is ambient) begins to increase. Before the necessary evacuation of the static system is complete (19,000 feet indicated on the altimeter) the ratio between pitot and static will be large enough that thee airspeed indicator will try to exceed the maximum indication and damage the internal mechanism.

(5) To prevent this occurrence, some personnel "T" the pitot and static systems together and connect them both to the vacuum source pitot static tester. This ensures that the ratio between the pitot side of the airspeed indicator and the static side of the airspeed indicator will remain constant and; therefore, the indicator will not move off zero. This method works satisfactorily unless there is a large leak in the aircraft static system.

(6) In this case, there will be a lower pressure on the pitot side than on the static side. This condition will cause the airspeed needle to attempt to move backwards, which may also result in damage to the indicator.

(7) A solution to this possibility is to use the crossbleed valve on the pitot static tester which allows control over the amount of vacuum being applied to the pitot side of the airspeed indicators. Begin applying vacuum to the static system with the crossbleed valve closed. As the static instruments begin to move (altimeter, rate-of-climb) the airspeed will also begin to increase. Allow the airspeed to increase to some easy point of reference (approximately half scale will result in the needle pointing to the six o'clock position). Slowly open the crossbleed valve and the airspeed needle will stop. Modulate the crossbleed as necessary to cause the airspeed needle to remain stationary. When the test altitude is reached, close off the vacuum and crossbleed connections from the tester. As the altimeter bleed rate is being timed, monitor the airspeed indicators.

(8) Use the crossbleed valve and the static valve as necessary to ensure that the airspeed indicator needle does not contact the peg in either the increasing or decreasing direction. The airspeed indicator needle will move slightly during the test, because there is always some small amount of leakage in the system. No action is necessary unless the airspeed needle begins moving towards the peg at a rapid rate. If it is necessary to take action to prevent the airspeed indicator from contacting peg during the test, there is a large leak in the system that will have to be corrected.

(9) If the airspeed indicator needle moves toward increasing airspeed, the leak is in the pitot side of the system. If the needle moves toward zero, there is a leak in the static side of the system.

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NAVIGATION SYSTEM – DESCRIPTION AND OPERATION

1. General

A. Standard navigational equipment provides flight environmental information such as airspeed, altitude, and vertical speed as well as outside air temperature, attitude (turn/slip), and direction (magnetic compass) information.

NOTE: For installations of special equipment selected by the owner/operator, refer to applicable vendor avionics document packages. For additional information refer to the applicable vendor's avionics package.

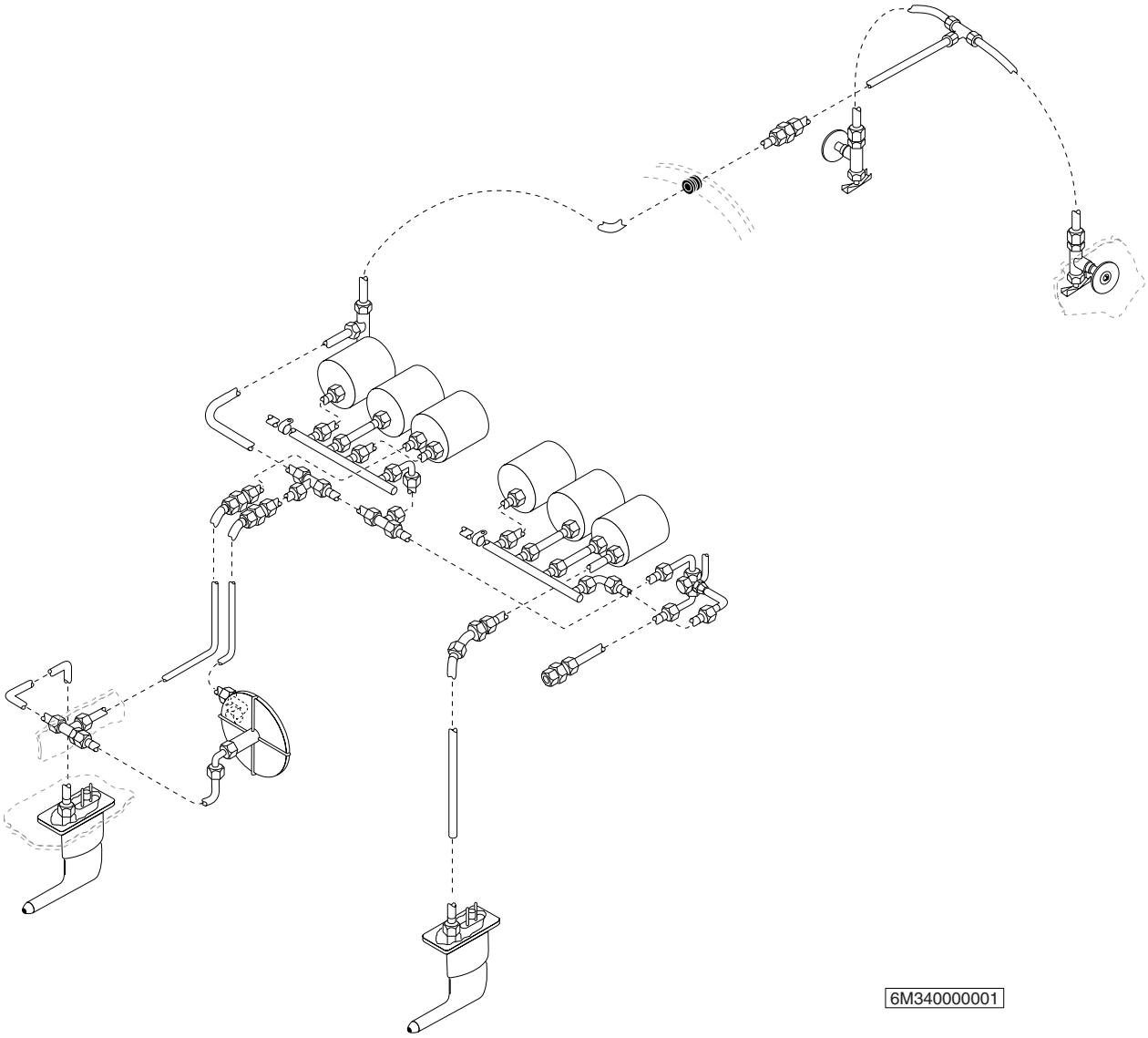
B. The flight environment system measures airspeed, altitude, vertical speed, and external temperature.

(1) Pitot/static plumbing routes outside air to airspeed, altitude, and vertical speed indicators and impact air to airspeed indicators.

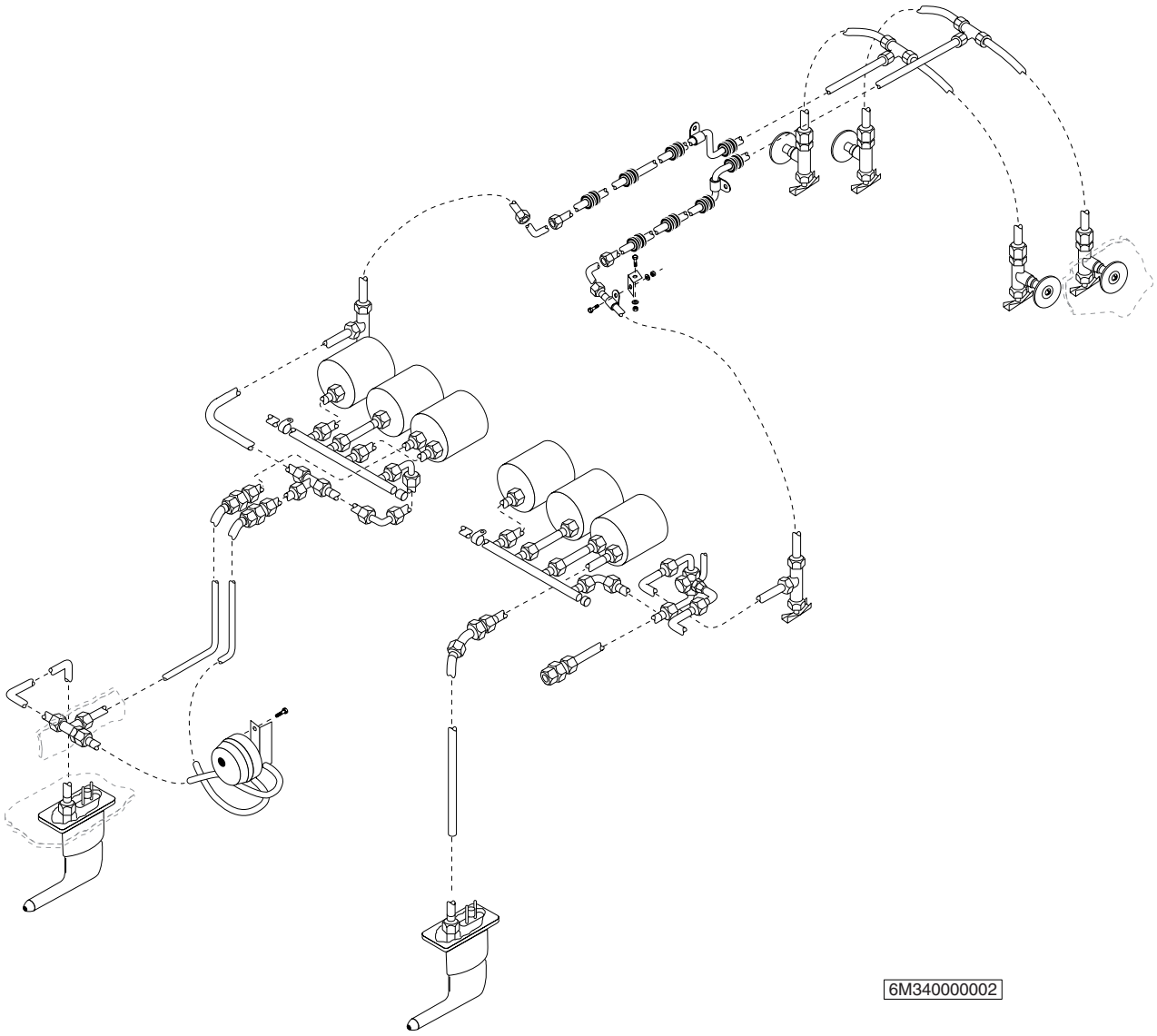
(2) Temperature is sensed as electrical resistance by a probe mounted on the fuselage skin beneath the cockpit. Temperature is indicated in degrees Celsius and displayed on the copilot outside air temperature indicator.

C Attitude and rate of slip/skid of a turn is indicated by the turn and slip indicator.

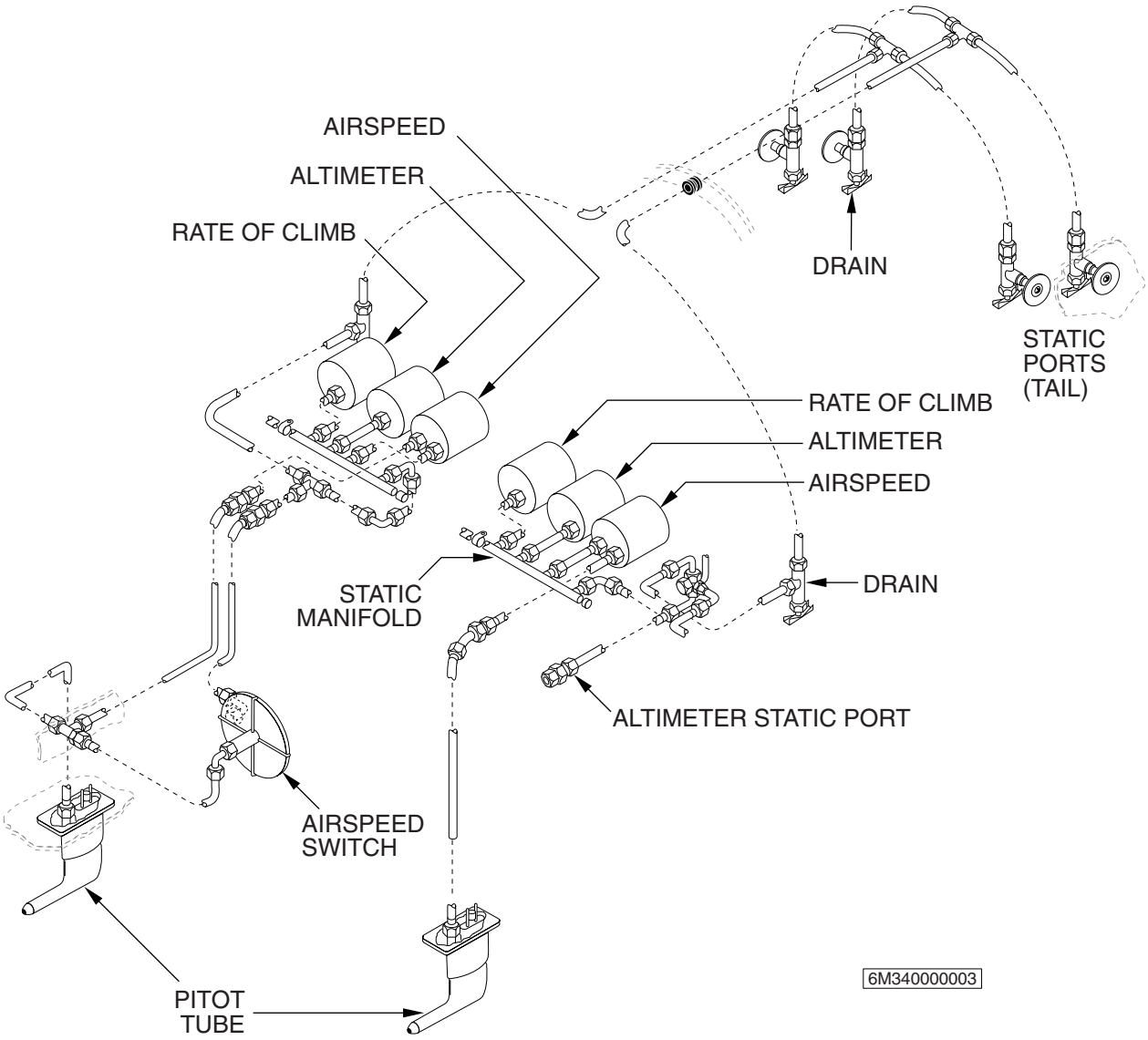
D. Aircraft directional heading in reference to the earth's magnetic field is indicated by the magnetic compass.



Navigation System
SA226 – T
Figure 1



Navigation System
SA226 Early
Figure 2



Navigation System
SA226 – AT, TC
Figure 3

PITOT-STATIC – MAINTENANCE PRACTICES

1. Removal/Installation – Pitot-Static

A. Refer to Plumbing, Chapter 34-10-60 for removal and installation of pitot-static plumbing.

2. Adjustment/Test – Pitot-Static

NOTE: The following procedures include instructions for Pitot-Static leak check, Airspeed Calibration check, SAS Arm Light ON/OFF speeds, and Altimeter Calibration check.

If any pitot-static component is removed and/or replaced, a leak check and applicable calibration checks must be performed.

A. Test Equipment

(1) Pitot/Static Tester or Calibrator.

(2) Water Manometer, calibrated in knots.

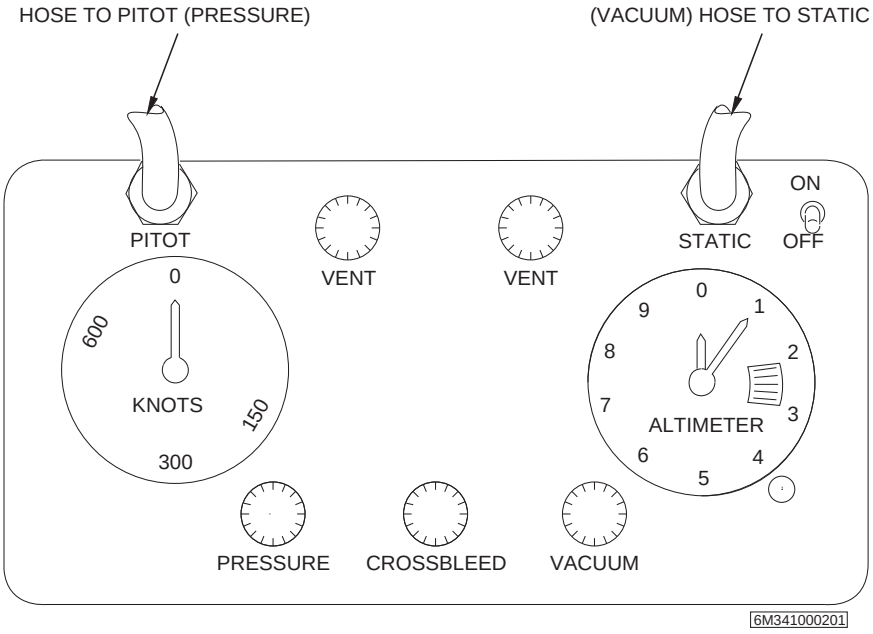
NOTE: Use of a Barfield DPS-400 pitot static tester (or tester with equivalent specifications) provides sufficient precision to qualify as an alternative for a water manometer when conducting tests in this section.

(3) Hoses and fittings to connect tester to pitot-static system.

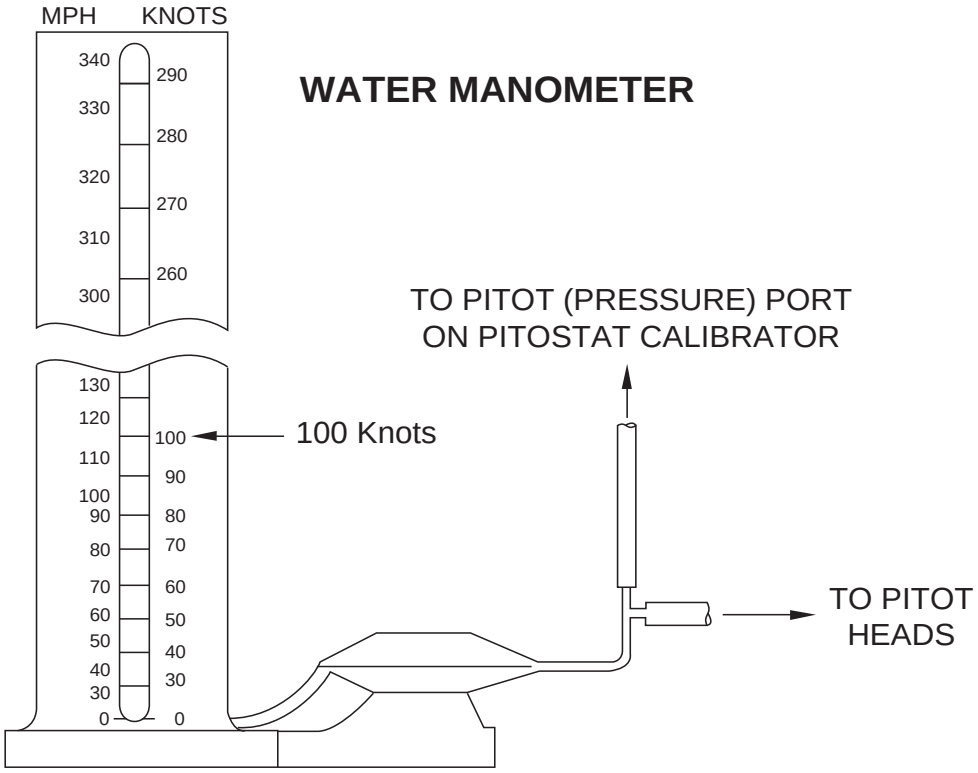
(4) Static Port Adapter NAV-AIDS P/N 33410-70-4.

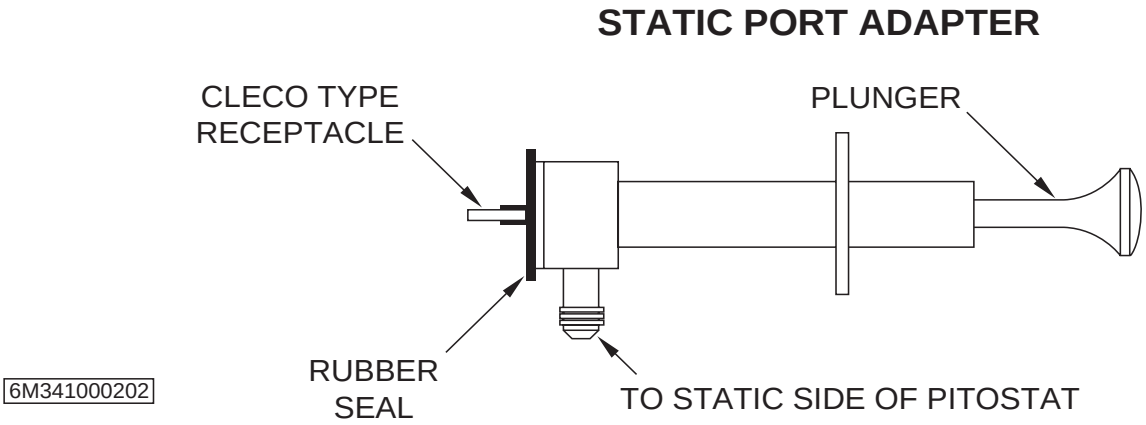
(5) LW-401 black leakproof tape.

NOTE: Ensure test equipment is leak free, before starting test. Equipment should be stored carefully when not in use, to prevent damage. Store tester with pressure and static vents open to avoid build up of internal pressure.



Pitot/Static Instrument Calibrator/Check
Figure 201





Water Manometer and Static Port Adapter
Figure 202

B. Pitot Leak Check

CAUTION: A THOROUGH WORKING KNOWLEDGE OF PITOT/STATIC AND TEST EQUIPMENT IS ESSENTIAL FOR THIS PROCEDURE. CAUTIONS SHOULD BE POSTED, VISIBLE TO ALL PERSONNEL, INDICATING THAT TEST IS IN PROGRESS.

NOTE: Use of a calibrated Barfield DPS-400 pitot/static tester (or tester with equivalent specifications) provides sufficient precision to qualify as an alternative for a water manometer. When conducting calibration tests in this section, setup and operate the tester IAW tester manufacturer's instruction manual and refer to the pitot static tester indicators instead of the water manometer.

(1) Connect pressure hose from manometer to pressure side of pitot/static test set. Connect hose from both pitot tubes to side port of manometer.

(2) Temporarily apply LW-401 black leakproof tape over drain holes in pitot tubes.

(3) Verify manometer water level is zero (0) with pressure vent open.

CAUTION: DAMAGE TO STATIC SYSTEM WILL RESULT IF THE CROSSBLEED VALVE IS NOT FULLY CLOSED DURING THE PITOT TEST. DO NOT APPLY SUCTION TO THE PITOT LINES.

(4) Close crossbleed valve on pitot/static test set.

(5) With crossbleed, pressure valve and pressure vent closed, pump approximately 10 to 15 psi (0.703 to 1.055 kg/cm^2) into pressure reservoir (replenish as required to complete test).

CAUTION: DO NOT APPLY PRESSURE TO INSTRUMENTS DESIGNED TO OPERATE AT LOWER THAN ATMOSPHERIC PRESSURE. (ALTIMETERS, VERTICAL SPEED, AND STATIC SIDE OF THE AIRSPEED INDICATORS OR STATIC SIDE OF AIRSPEED SWITCH).

(6) Slowly open pressure valve and apply pressure until manometer indicates 150 KIAS.

(7) Close pressure valve and observe manometer for any drop in water level. Maximum allowable drop of 10 knots in airspeed in one (1) minute is acceptable.

(8) Open vent slowly to gradually decrease airspeed to zero (0).

CAUTION: DO NOT DISCONNECT ANY PART OF THE PITOT-STATIC PLUMBING, INSTRUMENTS, OR TEST EQUIPMENT (INCLUDING LEAKPROOF TAPE) UNLESS ALL AIR PRESSURE OR VACUUM PRESSURE HAS BEEN RELEASED.

(9) If leak is in pitot system, isolate leak by disconnecting segments of the pitot plumbing, starting with the most accessible points, and applying pressure to each side of junction separately until leak is located. Correct leaks as necessary.

NOTE: Conducting the pitot leak check first has the advantage of isolating potential leaks to a smaller plumbing system before checking the static system, a larger plumbing system.

(10) If no other test is required at this time, return aircraft to original configuration.

SAS ARM LIGHT CHECK

OFF	_____ KIAS
ON	_____ KIAS

INSPECTOR: _____

DATE: _____

Manometer * True KIAS	PILOT (S/N)		COPILOT (S/N)	
	Up	Down	Up	Down
60				
70				
80				
85				
90				
95				
100				
105				
110				
115				
120				
130				
140				
160				
180				
200				
220				
240				
260				
280				

6M341000203

* Use of a calibrated Barfield DPS-400 pitot static tester (or tester with equivalent specifications) provides sufficient precision to qualify as an alternative for a water manometer. When conducting calibration tests in this section, setup and operate the tester IAW tester manufacturer's instruction manual and refer to the pitot static tester indicators instead of the water manometer.

Airspeed Indicator Calibration Chart
Figure 203

C. SAS Arm Light ON/OFF Check

NOTE: This check may be accomplished immediately after the pitot leak check and while in the same configuration.

(1) With crossbleed, pressure vent, and pressure valve closed, pump 10 to 15 psi (0.703 to 1.055 kg/cm^2) into pressure reservoir (replenish as required to complete test).

(2) Carefully open pressure valve and allow indicated airspeed to slowly increase until SAS Arm light located on annunciator panel extinguishes.

(3) Observe copilot's indicated airspeed when the SAS Arm light extinguishes. Record in Figure 203.

(4) Carefully open pressure vent and allow airspeeds to decrease slowly.

(5) Observe copilot's indicated airspeed when the SAS Arm light illuminates. Record in Figure 203.

NOTE: The SAS Arm light should be extinguished by 140 KIAS (± 5 knots) for the SA226-AT and -TC Models, 185 knots for the SA226-T (± 5 knots).

(6) Allow airspeed to gradually decrease to zero (0).

(7) If no other test is required at this time, return aircraft to original configuration.

D. Airspeed Calibration Check

NOTE: This calibration check may be accomplished immediately after the SAS Arm light check from the same configuration.

NOTE: Use of a calibrated Barfield DPS-400 pitot static tester (or tester with equivalent specifications) provides sufficient precision to qualify as an alternative for a water manometer. When conducting calibration tests in this section, setup and operate the tester IAW tester manufacturer's instruction manual and refer to the pitot static tester indicators instead of the water manometer.

(1) Verify that manometer water level is at zero (0) with vent open.

(2) With crossbleed, pressure valve, and pressure vent closed, pump 10 to 15 psi (0.703 to 1.055 kg/cm^2) into pressure reservoir (replenish as required to complete test).

(3) Carefully open pressure valve and allow both airspeeds to increase slowly.

(4) Close pressure valve when manometer indicates 60 KIAS. Record pilot and copilot indicated airspeed on the Airspeed Calibration Chart (Figure 203) in the UP column.

NOTE: Close the pressure valve as close as possible to the water manometer reference values when increasing pressure. Overshooting the reference and trying to reduce pressure to regain the reference introduces friction errors in the instruments.

(5) Using pressure valve to control manometer, repeat steps **2.**, D., (3) and (4) for each reference value listed in Figure 203, ending at 280 KIAS.

(6) With manometer indicating 280 KIAS, carefully open pressure vent so that airspeed gradually decreases.

(7) When manometer indicates 260 KIAS close pressure valve.

(8) Observe pilot and copilot airspeeds and record value in the DOWN columns of Figure 203.

(9) Continue decreasing airspeed in the above manner, stopping at each reference value listed in Figure 203 and record each value.

(10) Open pressure vent and allow airspeed indicators to return to zero (0).

(11) Replace or repair indicators that fail to meet accuracy requirements listed in Table 201 (maximum allowable split is 6 knots):

INDICATED AIRSPEED	MAX AIRSPEED	ALLOWED TOLERANCE
0 to 79.0	0 to 79.0	± 4 Knots
80.0 to 230.0	80.0 to 230.0	± 3 Knots
231.0 to 268.0	231.0 to 268.0	+ 3 Knots - 0 Knots
269.0 to 300.0	269.0 to 300.0	± 4 Knots

Airspeed Indicator Accuracy Requirements
Table 201

(12) Repeat Pitot System Leak Check and Airspeed Calibration Check if either the pilot or copilot airspeed indicator was replaced due to inaccuracy.

(13) If no other test is required at this time, return aircraft to original configuration.

E. Static System Check

CAUTION: DO NOT APPLY PRESSURE TO INSTRUMENTS DESIGNED TO OPERATE AT LOWER THAN ATMOSPHERIC PRESSURE (ALTIMETERS, VERTICAL SPEED INDICATORS, AND STATIC SIDE OF AIRSPEED INDICATORS OR STATIC SIDE OF AIRSPEED SWITCH).

(1) Disconnect manometer hose from pitot (pressure) side of pitot/static tester.

(2) Disconnect pitot line from side port of manometer.

(3) Position pitot/static tester inside aircraft so that all navigational instruments are visible.

NOTE: For easier isolation of leaks, pilot and copilot static systems should be checked separately.

(4) Connect static (vacuum) hose from pitot/static tester to pilot static ports using static port adapter.

NOTE: To ensure a positive leak free seal, apply a coat of petroleum jelly to face of rubber seal and support the connecting hose by attaching to fuselage with duct (or similar) tape.

(5) Connect pitot hose from pitot/static tester to pilot's pitot system. Do not connect any hose from pitot/static tester to copilot's pitot system at this time.

(6) Temporarily cover copilot static ports with leakproof tape. Verify that drain holes in pitot tubes are also temporarily taped closed.

(7) Close pressure valve, pressure vent, static valve, and static vent on pitot/static tester.

(8) Open cross bleed valve approximately three (3) turns.

CAUTION: WHEN APPLYING VACUUM PRESSURE TO STATIC SYSTEM, OBSERVE INSTRUMENTS CLOSELY. AIRSPEED INDICATORS SHOULD REMAIN AT ZERO OR INCREASE TO NO MORE THAN 50 KNOTS. VERTICAL SPEED SHOULD INDICATE A CLIMB. ALTIMETER SHOULD INCREASE.

IF AIRSPEED INDICATORS ACCELERATE OR OTHER INSTRUMENTS MOVE IN REVERSE DIRECTION CLOSE VACUUM VALVE IMMEDIATELY. CHECK CONNECTIONS AND TEST EQUIPMENT SINCE THESE CONDITIONS COULD INDICATE A REVERSE CONNECTION.

(9) Altimeters on the aircraft and pitot/static tester should be set at 29.92 Hg.

(10) Slowly open vacuum valve and apply continuous vacuum pressure to static system until altimeter indicates 18,000 ft.

CAUTION: VERTICAL SPEED INDICATOR SHOULD NOT EXCEED 5000 FPM. USE STATIC (VACUUM) VALVE OR VENT TO CONTROL RATE OF CLIMB OR DESCENT.

(11) When altimeter indicates 18,000 feet, close vacuum valve to test for leaks. Maximum allowable leak is 337 feet per minute.

(12) Slowly open static vent and release vacuum pressure. Vertical speed should not exceed 5000 feet per minute descent.

CAUTION: DO NOT DISCONNECT ANY PART OF THE PITOT/STATIC PLUMBING, INSTRUMENTS OR TEST EQUIPMENT (INCLUDING LEAKPROOF TAPE) UNLESS ALL AIR PRESSURE OR VACUUM PRESSURE HAS BEEN RELEASED.

(13) If pilot's static system is found to leak, isolate leak by subdividing plumbing into testable segments beginning at most accessible points.

(14) Correct leaks as necessary.

NOTE: If any component of the static system is removed and/or replaced, a leak check must be performed.

(15) If copilot's static system is to be tested, remove leakproof tape from copilot static ports. Remove static port adapters from pilot static ports and connect to copilot static ports. Place leakproof tape over the pilot static ports.

(16) Repeat steps **2.**, E., (1) through (14) for copilot static system.

(17) If no other test is required at this time, return aircraft to original configuration.

AIRSPEED INDICATOR – DESCRIPTION AND OPERATION

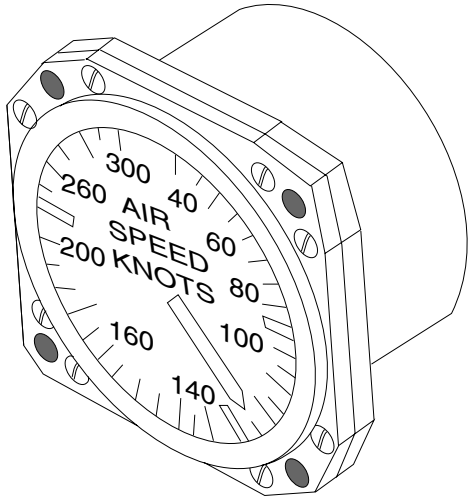
1. General

A. The airspeed indicators located on the pilot and copilot's instrument panel are connected to its respective pitot tube and static pressure manifold.

(1) The indicator receives impact air pressure from the pitot tube and converts it to a scale reading in knots.

(2) Static is connected to each indicator so the indicator is accurate at all altitudes.

(3) The indicator, scaled from 40 to 300 knots, gives a continuous reading of the aircraft airspeed.



6M341010001

Airspeed Indicator
Figure 1

AIRSPEED INDICATOR – FAULT ISOLATION

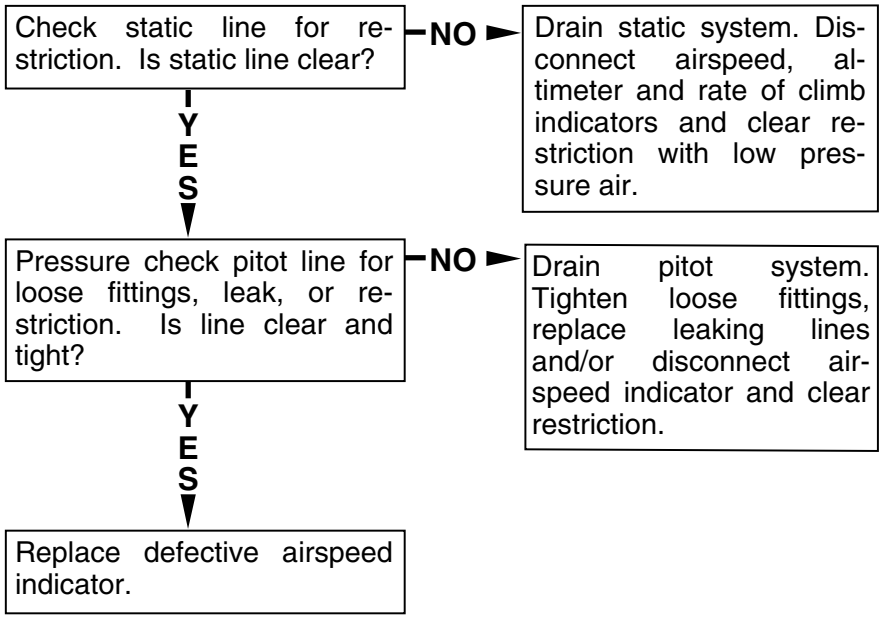
PROBLEM:

AIRSPEED INDICATOR INOPERATIVE OR INCORRECT.

PREREQUISITES:

Verify airspeed indicators properly installed, connected to pitot-static lines, and circuit breakers IN.

START:



END

6M341010101

AIRSPEED INDICATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Airspeed Indicator

A. Removal – Airspeed Indicator

(1) Lock the elevator control column in neutral position.

(2) Loosen Airloc fasteners holding secondary instrument panel to main instrument panel.

(3) Rotate secondary instrument panel aft to gain access to indicator plumbing.

(4) Disconnect pitot and static lines from the indicator.

(5) Cap the lines to prevent contamination.

(6) Remove indicator mount screws and withdraw indicator from rear side of secondary instrument panel.

C. Installation – Airspeed Indicator

(1) Position indicator in secondary instrument panel and secure with screws.

(2) Connect pitot and static lines to indicator. Tighten fittings IAW instructions in Plumbing, this chapter.

(3) Secure secondary instrument panel to main instrument panel with Airloc fasteners.

(4) Unlock elevator control column as required.

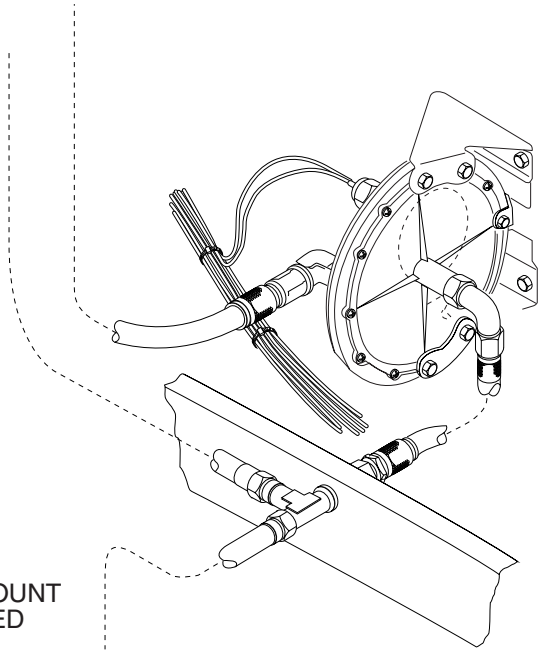
2. Inspection/Check – Airspeed Indicator

A. Inspect the airspeed indicator for condition and security.

AIRSPEED SWITCH – DESCRIPTION AND OPERATION

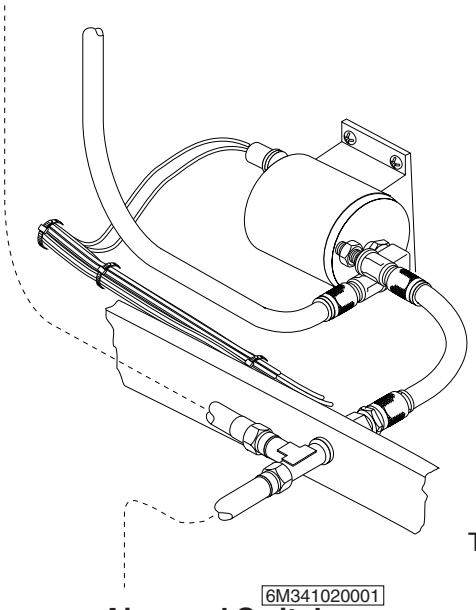
1. General

A. The airspeed switch, located under the nose compartment right aft floorboard, is mounted to the nose gear keelson. The airspeed switch senses pitot and static differential pressure for the stall avoidance system. Pitot pressure is supplied by the copilot's pitot system. A line connected to the static system behind the copilot's instrument panel supplies static pressure to the airspeed switch.



ROSEMOUNT AIRSPEED SWITCH

T201-275, 277-291
AT001-999
TC201-999



CONRAC AIRSPEED SWITCH

T276, 292-999

6M341020001

Airspeed Switch
Figure 1

AIRSPEED SWITCH – MAINTENANCE PRACTICES

1. Removal/Installation – Airspeed Switch

A. Refer to Maintenance Manual, Chapter 27, Section 27-30-12 for Removal/Installation and Adjustment/Test procedures.

2. Inspection/Check – Airspeed Switch

A. Inspect the stall avoidance system airspeed switch for condition and security.

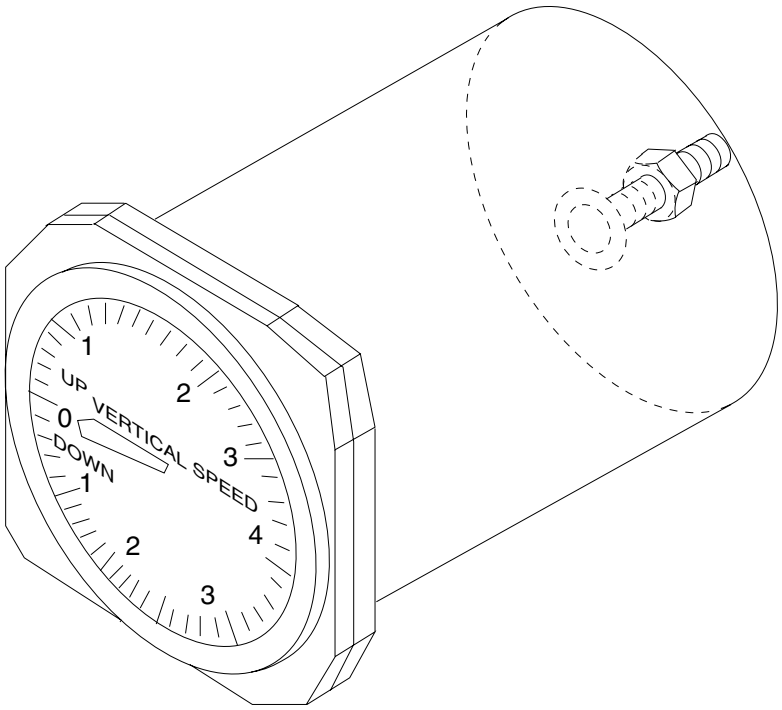
RATE OF CLIMB (VERTICAL VELOCITY) INDICATOR – DESCRIPTION AND OPERATION

1. General

A. The rate of climb indicator installed in the pilot and copilot's instrument panel is connected to its respective static pressure manifold.

(1) The indicator provides the crew with an indication of the rate of climb or descent, calibrated in 1,000's of feet per minute. The indicator has an airtight diaphragm vented directly to the static air. The instrument case is airtight and vented to static through a slow bleed device inside the indicator.

(2) As the aircraft gains altitude, the pressure in the diaphragm decreases, and the higher pressure in the case compresses the diaphragm, causing the pointer to move through a mechanical linkage. The diaphragm expands in the same manner to give an accurate indication of the rate of descent to flight level, the pressure in the instrument case will bleed off through the slow bleed device, thus returning the pointer to zero.



6M341030001

Rate of Climb Indicator
Figure 1

RATE OF CLIMB (VERTICAL VELOCITY) INDICATOR – FAULT ISOLATION

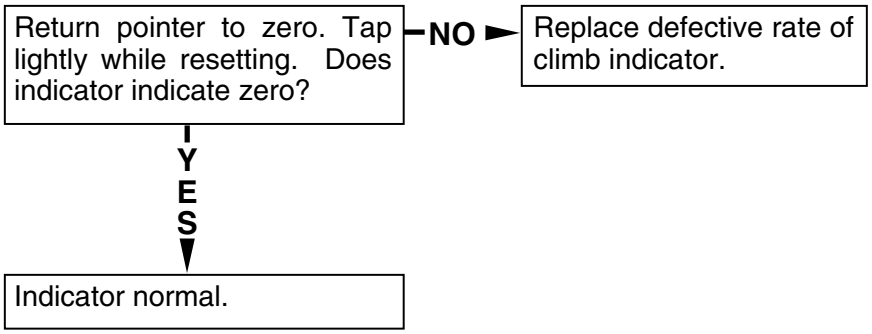
PROBLEM:

RATE OF CLIMB INDICATOR DOES NOT INDICATE ZERO AT CONSTANT ALTITUDE

PREREQUISITES:

Verify rate of climb indicator properly installed, static connection tight, and circuit breaker IN.

START:



END

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RATE OF CLIMB (VERTICAL VELOCITY) INDICATOR – FAULT ISOLATION

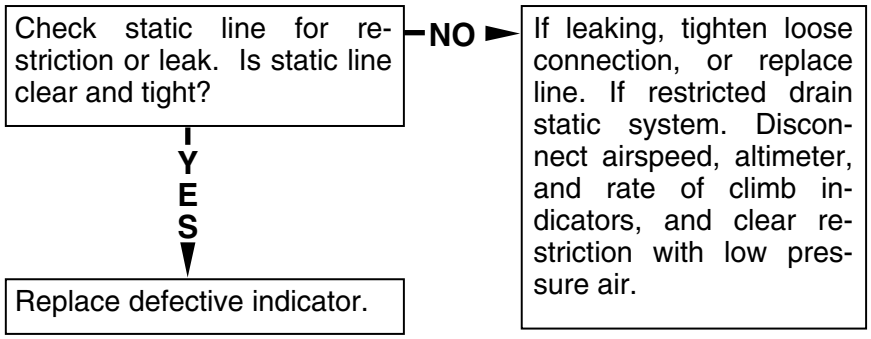
PROBLEM:

RATE OF CLIMB INDICATOR ERRATIC OR INOPERATIVE.

PREREQUISITES:

Verify rate of climb indicator properly installed, static connection tight, and circuit breaker IN.

START:



END

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RATE OF CLIMB (VERTICAL VELOCITY) INDICATOR – MAINTENANCE PRACTICES

1. Removal/Installation Rate of Climb (Vertical Velocity) Indicator

A. Removal – Rate of Climb Indicator

(1) Lock control column in neutral position.

(2) Loosen fasteners securing secondary instrument panel to main instrument panel.

(3) Rotate secondary panel aft to gain access to indicator plumbing.

(4) Disconnect Static line from indicator.

(5) Cap line to prevent contamination.

(6) Remove screws and withdraw indicator from secondary instrument panel.

B. Installation – Rate of Climb Indicator

(1) Position indicator in secondary instrument panel and secure with screws.

(2) Remove cap from static line.

(3) Connect static line to indicator. Tighten fitting IAW instructions in Plumbing, this chapter.

(4) Secure secondary instrument panel to main instrument panel with fasteners.

(5) Unlock elevator control column as required.

2. Inspection/Check Rate of Climb (Vertical Velocity) Indicator

A. Inspect rate of climb indicator for condition and security.

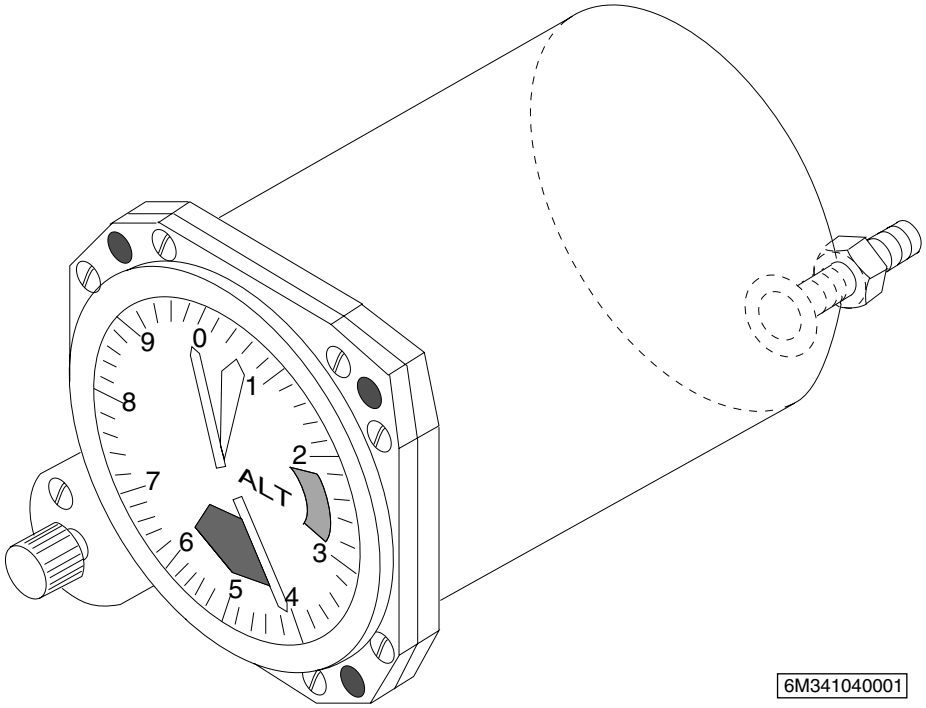
ALTIMETER – DESCRIPTION AND OPERATION

1. General

A. A pressure altimeter located on the pilot and copilot's instrument panel is connected to its respective static pressure manifold.

(1) The altimeter operates by sensing static pressure changes and converting it to a scale reading in feet.

(2) The altimeter incorporates a barometric scale on the instrument face that can be set with a rotatable knob on the lower left of the indicator case. The scale is used to recalibrate the indicator for known pressure variations.



ALTIMETER
Figure 1

ALTIMETER– FAULT ISOLATION

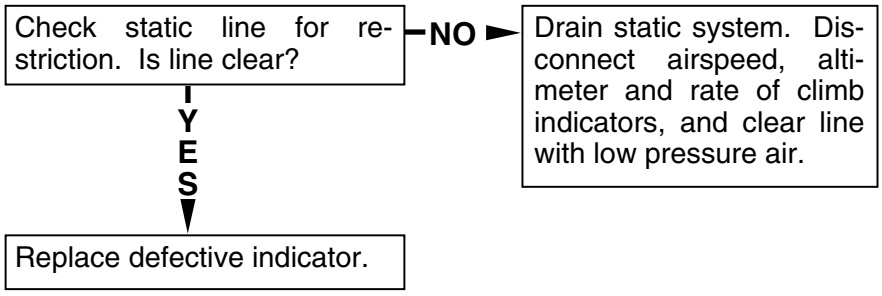
PROBLEM:

ALTIMETER ERRATIC OR INOPERATIVE.

PREREQUISITES:

Verify altimeter properly installed, static connection tight and circuit breaker IN.

START:



END

4M341040101

ALTIMETER– FAULT ISOLATION

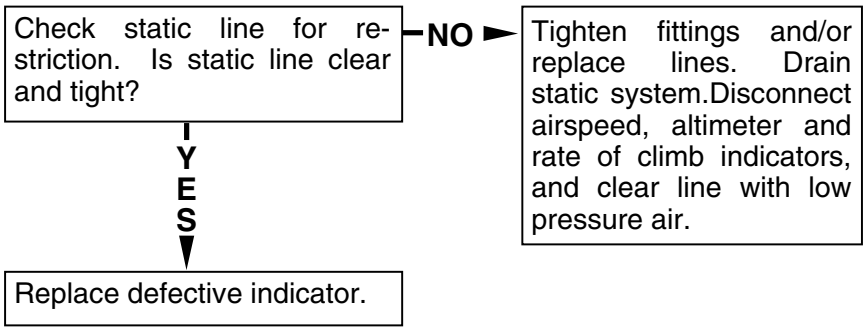
PROBLEM:

ALTIMETER POINTER OSCILLATES.

PREREQUISITES:

Verify altimeter properly installed, and static connection tight.

START:



END

4M341040102

ALTIMETER – MAINTENANCE PRACTICES

1. Removal/Installation – Altimeter

A. Removal – Altimeter

(1) Lock the elevator control column in neutral position.

(2) Loosen fasteners securing secondary instrument panel to main instrument panel.

(3) Rotate secondary panel aft to gain access to indicator plumbing.

(4) Disconnect Static line from indicator.

(5) Cap line to prevent contamination.

(6) Remove screws and withdraw indicator from secondary instrument panel.

B. Installation – Altimeter

(1) Position indicator in secondary instrument panel and secure with screws.

(2) Remove cap from static line.

(3) Connect static line to indicator. Tighten fitting IAW instructions in Plumbing, this chapter.

(4) Secure secondary instrument panel to main instrument panel with fasteners.

(5) Unlock elevator control column as required.

2. Inspection/Check – Altimeter

A. Inspect altimeter for condition and security.

PITOT TUBE – DESCRIPTION AND OPERATION

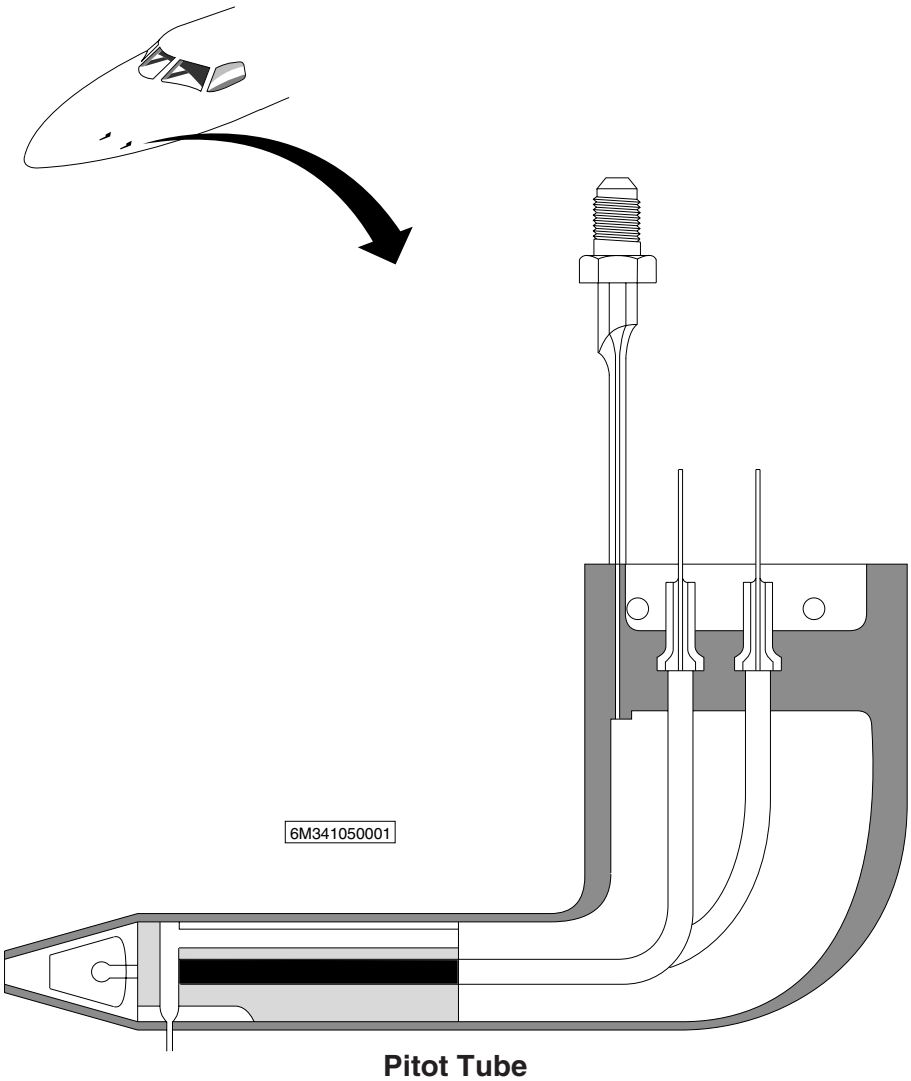
1. General

A. A pitot tube is installed in a base mount on the lower left and right nose skin, with the axis parallel to the longitudial axis of the aircraft.

(1) The left pitot tube supplies impact air pressure to the pilot's airspeed indicator. The right pitot tube routes impact air pressure to the copilot's airspeed indicator and to the stall avoidance system airspeed switch.

(2) The pitot tube incorporates an internal heater to prevent ice from clogging the tube. The heater is powered by 28 VDC through a ten amp circuit breaker on the left and right essential bus circuit breaker panel.

(3) Pitot heat is controlled by two switches located on the pilot's forward switch panel. Each switch has three positions, up supplies power to the respective pitot tube and to the SAS deice system. An ammeter on the forward left side console provides pitot heat monitoring. A switch next to the meter allows selection of left or right pitot heat indication.



Pitot Tube
Figure 1

PITOT TUBE – MAINTENANCE PRACTICES

1. Removal/Installation – Pitot Tube

A. Removal – Pitot Tube

CAUTION: IF PITOT HEAT HAS BEEN OPERATED, ENSURE THAT PITOT TUBES HAVE COOLED DOWN BEFORE REMOVAL.

(1) Remove screws securing pitot tube to base mount.

(2) Withdraw pitot tube from base mount far enough to allow access to electrical leads and plumbing connection.

NOTE: On aircraft with water injection installed, the right pitot tube is sealed inside with Pro-Seal 890B2 to prevent the heater element from shorting out should a water leak develop in the area.

(3) Disconnect electrical and plumbing connectors.

(4) Cap plumbing line to prevent contamination.

B. Installation – Pitot Tube

(1) Remove cap from plumbing line.

(2) Connect electrical and plumbing connections.

(3) Position pitot tube into base mount.

(4) Secure pitot tube to mount with screws and washers.

NOTE: On aircraft with water injection installed, seal the right pitot tube around the electrical and plumbing connectors with Pro-Seal 890B2.

2. Inspection/Check – Pitot Tube

A. Inspect the pitot tube and base for damage, obstructions, and security.

MAINTENANCE MANUAL

PITOT-STATIC PLUMBING – DESCRIPTION AND OPERATION

1. General

A. Pitot-static plumbing consists of rubber hoses and metal fittings in the forward nose compartment pitot system.

B. Plastic lines and fittings make up the plumbing in the pressurized section of the aircraft from the forward pressure bulkhead to the static ports in the aft fuselage.

C. A moisture drain valve connects the left and right static ports to plastic tubing, which is routed upward to a "T" fitting, that connects each side to a single line routed overhead to the cockpit, where it comes down the right side and connects to a moisture drain valve behind the copilot's side console. The tubing continues forward to connect to the airspeed switch, copilot's manifold, alternate static selector valve, and the pilot's static pressure manifold.

D. Aircraft SA226 – AT and TC use a dual static system. One routed to the pilot's side and connects to the alternate static selector valve and pilot's static pressure manifold; the other is routed to the copilot's side and connects the airspeed switch and copilot's static pressure manifold.

PITOT-STATIC PLUMBING – MAINTENANCE PRACTICES

1. Removal/Installation – Pitot-Static Plumbing

A. Installation – Plastic Plumbing

NOTE: Nylo-Seal or Swageloc fittings may be used on various aircraft. The fittings come completely assembled. Do not disassemble.

(1) Insert tubing into the fitting so that it rests firmly on the inner shoulder of the fitting.

CAUTION: TO PREVENT DAMAGE AND MINIMIZE THE POSSIBILITY OF LEAKS, THE PLASTIC FITTINGS MUST BE TIGHTENED BY A GEOMETRIC METHOD AND NOT BY TORQUE.

(2) Tighten Swageloc fittings one and one quarter (1 1/4) turns from finger tight, Tighten Nylo-Seal fittings two to two and one half (2 to 2 1/2) turns from finger tight.

NOTE: Both Nylo-Seal and Swageloc fittings may be disconnected and retightened. Tighten Swageloc fittings an additional one quarter (1/4) turn when retightening.

CAUTION: WHEN CLAMPING PLASTIC TUBING, TAKE CARE TO PREVENT DEFORMATION OF TUBING.

PROTECT TUBING FROM TEMPERATURES IN EXCESS OF 180°F (82.2°C).

(3) Plastic tubing is supported by plastic tie-straps and protected by grommets.

2. Inspection/Check – Pitot-Static Plumbing

A. Inspect accessible plumbing for kinks, breaks, security, and general condition.

STATIC PRESSURE MANIFOLD – DESCRIPTION AND OPERATION

1. General

A. A static pressure manifold, mounted below the pilot and copilot's windshield, contains ports for connecting components to the static system.

STATIC PRESSURE MANIFOLD – MAINTENANCE PRACTICES

1. Removal/Installation – Static Pressure Manifold

A. Removal – Static Pressure Manifold

(1) Remove glareshield mounting screws on left and right side.

(2) Lift glareshield up until mounting clips clear moulding.

(3) Move glareshield aft by turning it sideways while withdrawing it.

(4) Disconnect glareshield electrical connector, and remove glareshield.

(5) Disconnect static lines from manifold.

(6) Cap static lines to prevent contamination.

(7) Remove Adel clamps screws.

(8) Remove manifold.

B. Installation – Static Pressure Manifold

(1) Position manifold in Adel clamps and secure to frame with screws, washers, and nuts.

NOTE: Each manifold clamp screw also secures a hydraulic gauge line clamp.

(2) Remove caps from static lines.

(3) Connect static lines to manifold ports. Tighten fittings IAW instructions in Plumbing, this chapter.

(4) Connect glareshield electrical connector.

(5) Position glareshield and install clip screws.

2. Inspection/Check – Static Pressure Manifold

A. Inspect static pressure manifolds for condition and security.

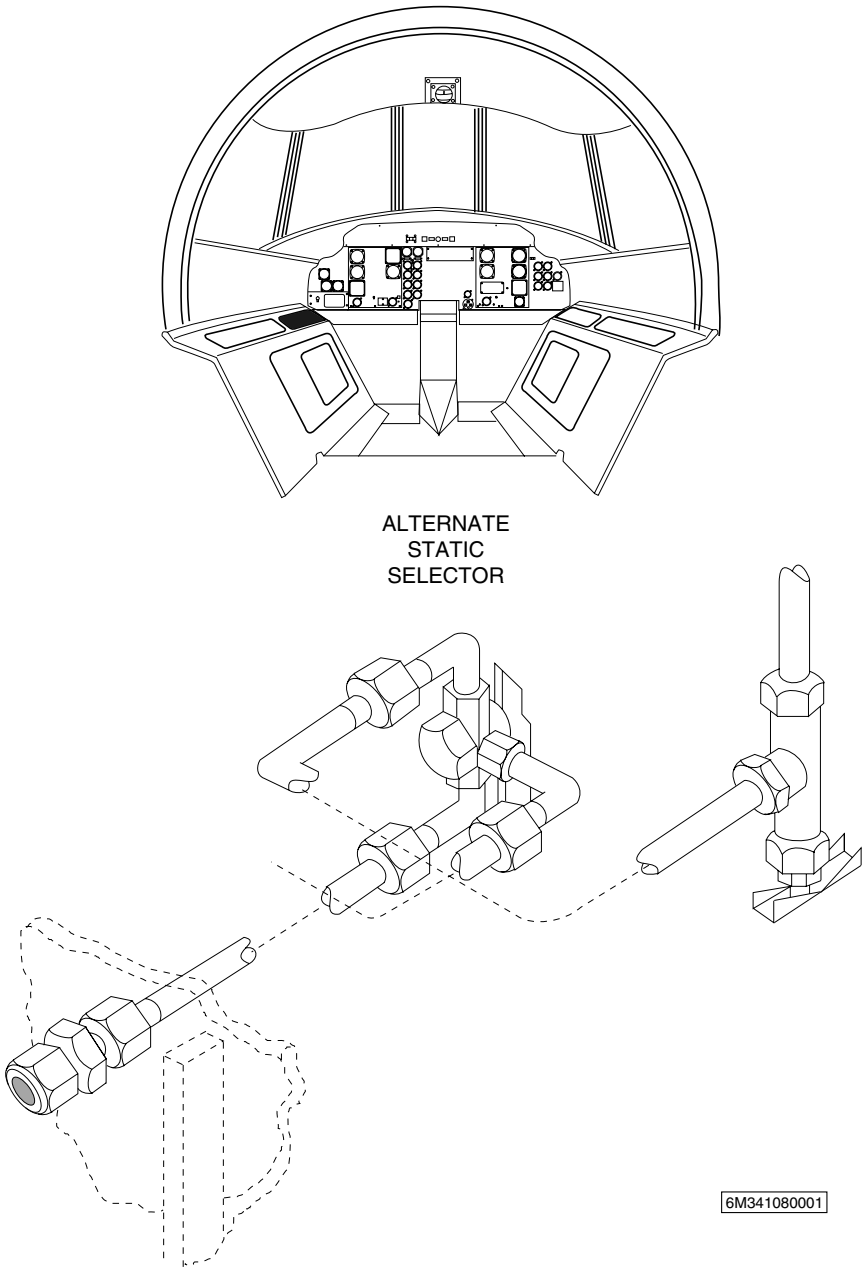
ALTERNATE STATIC SELECTOR VALVE – DESCRIPTION AND OPERATION

1. General

A. The static selector valve located on the pilot's instrument panel lower left side is installed in the normal static system and connected to an alternate static source.

B. The alternate static line is routed from the valve to a fitting in the forward pressure bulkhead. The bulkhead fitting has a screen cap installed to prevent foreign objects from entering the system.

C. The selector valve provides an alternate static source to the pilot's instruments only.



Static Pressure Selector Valve
Figure 1

ALTERNATE STATIC SELECTOR VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – Alternate Static Selector Valve

A. Removal – Static Selector Valve

(1) Lock elevator control column in neutral position.

(2) Loosen fasteners securing secondary instrument panel to main instrument panel.

(3) Rotate secondary panel aft to gain access to static valve plumbing.

(4) Disconnect static lines. Be sure to identify them for reinstallation.

(5) Cap lines to prevent contamination.

(6) Remove screw securing handle to valve shaft. Remove handle.

(7) Remove mounting screws and withdraw valve from rear of instrument panel.

B. Installation – Static Selector Valve

(1) Position static selector valve in instrument panel and secure with screws.

(2) Install valve handle.

NOTE: Index handle in relation to the shaft and body.

(3) Remove caps from static lines.

CAUTION: IT IS POSSIBLE TO CONNECT LINES TO WRONG PORTS ON THE VALVE.

(4) Connect static lines to valve ports. Tighten fittings IAW instructions in Plumbing, this chapter.

(5) Secure secondary instrument panel to main instrument panel with fasteners.

(6) Unlock elevator control column as required.

2. Inspection/Check – Alternate Static Selector Valve

A. Inspect the alternate static selector valve for operation, condition, and security.

STATIC PORT – DESCRIPTION AND OPERATION

1. General

A. Static ports are installed in the left and right aft fuselage skin. The pressure port has seven vent holes that supply outside air static pressure for the system's various components.

B. The SA226 – T aircraft uses one static port on each side of the fuselage to operate both the pilot and copilot's instruments, while the SA226 – AT and TC has two separate static systems.

C. The pilot's instruments use the lower left and right static ports. The copilot's instruments use the upper left and right static ports.

NOTE: There are four static ports installed on the nose of the aircraft below the pilot's and copilot's side window. One port on each side is connected to the pressurization system. The remaining port on each side is spare at the time of manufacture, but is usually connected to the autopilot if the aircraft is so equipped. The four ports in the nose are not connected to the pitot static system.

STATIC PORT – MAINTENANCE PRACTICES

1. Removal/Installation – Static Port

NOTE: The static ports are fay sealed and riveted to the aft fuselage skin and should not need to be replaced.

2. Inspection/Check – Static Port

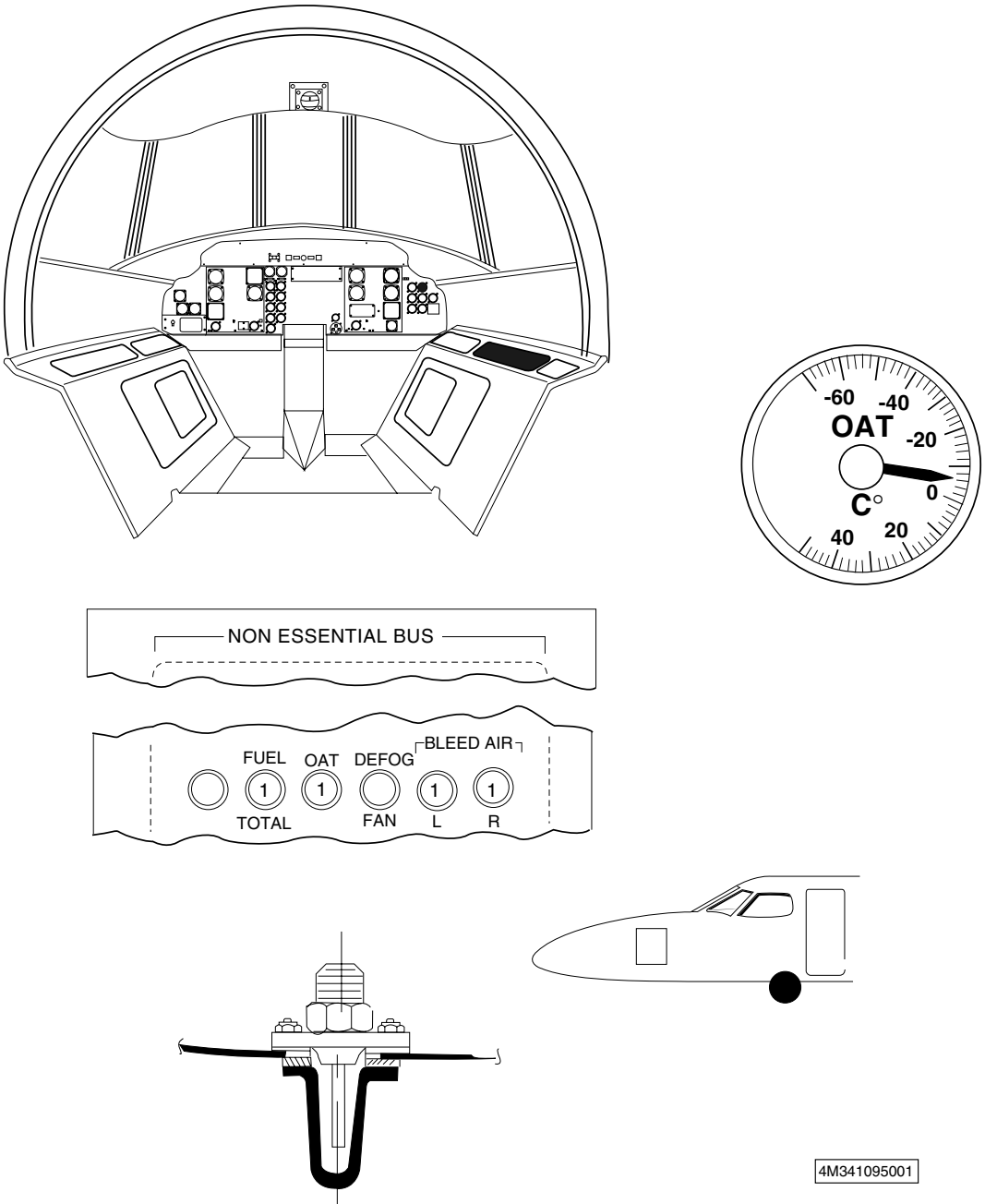
A. Inspect static ports for obstructions.

OUTSIDE AIR TEMPERATURE – DESCRIPTION AND OPERATION

1. General

A. Outside air temperature is operated by the aircraft 28 VDC and protected by a 1 amp circuit breaker located on the copilot's nonessential bus circuit breaker panel.

B. A resistance type air temperature bulb is mounted to the lower fuselage skin between the pilot and copilot's seats. The temperature is electrically transmitted to the outside air temperature indicator on the lower center instrument panel and indicated in degrees celsius.



Outside Air Temperature Indicator
Figure 1

OUTSIDE AIR TEMPERATURE – FAULT ISOLATION

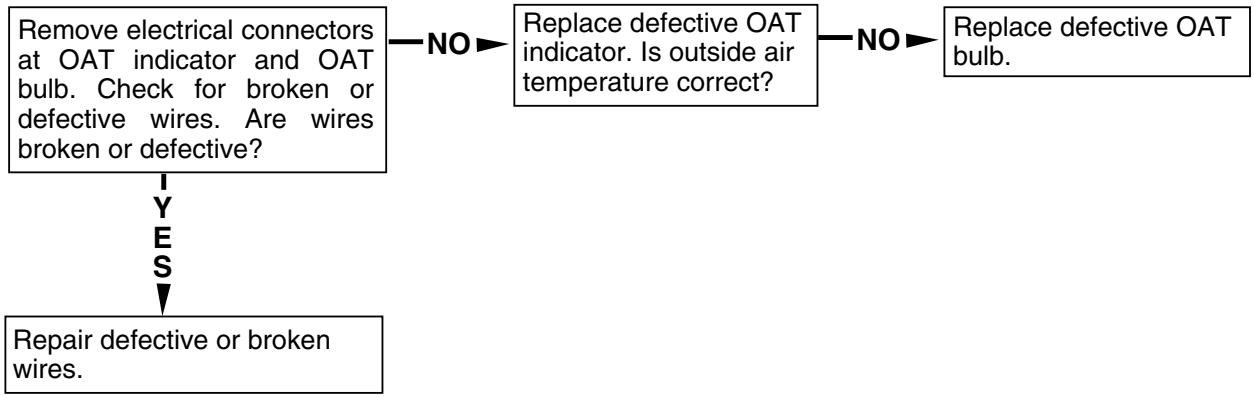
PROBLEM:

INTERMITTENT INDICATION ON OUTSIDE AIR TEMPERATURE INDICATOR.

PREREQUISITES:

Verify outside air temperature indicator properly installed and connected. Outside air temperature bulb properly installed and connected. OAT circuit breaker IN.

START:



END

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OUTSIDE AIR TEMPERATURE – FAULT ISOLATION

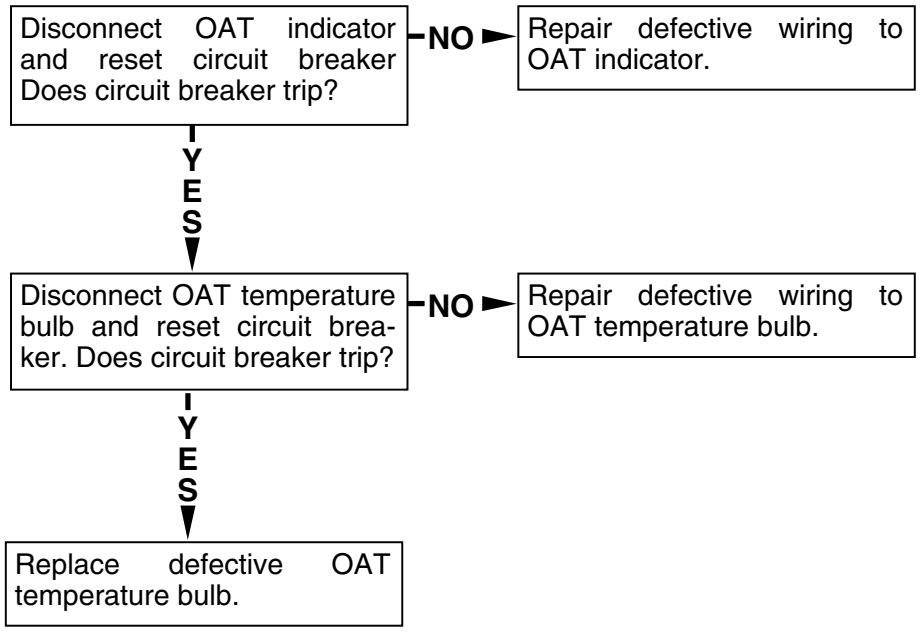
PROBLEM:

CIRCUIT BREAKER TRIPS.

PREREQUISITES:

Verify outside air temperaure indicator properly installed and connected. Outside air temperature bulb properly installed and connected. OAT circuit breaker IN.

START:



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OUTSIDE AIR TEMPERATURE – FAULT ISOLATION

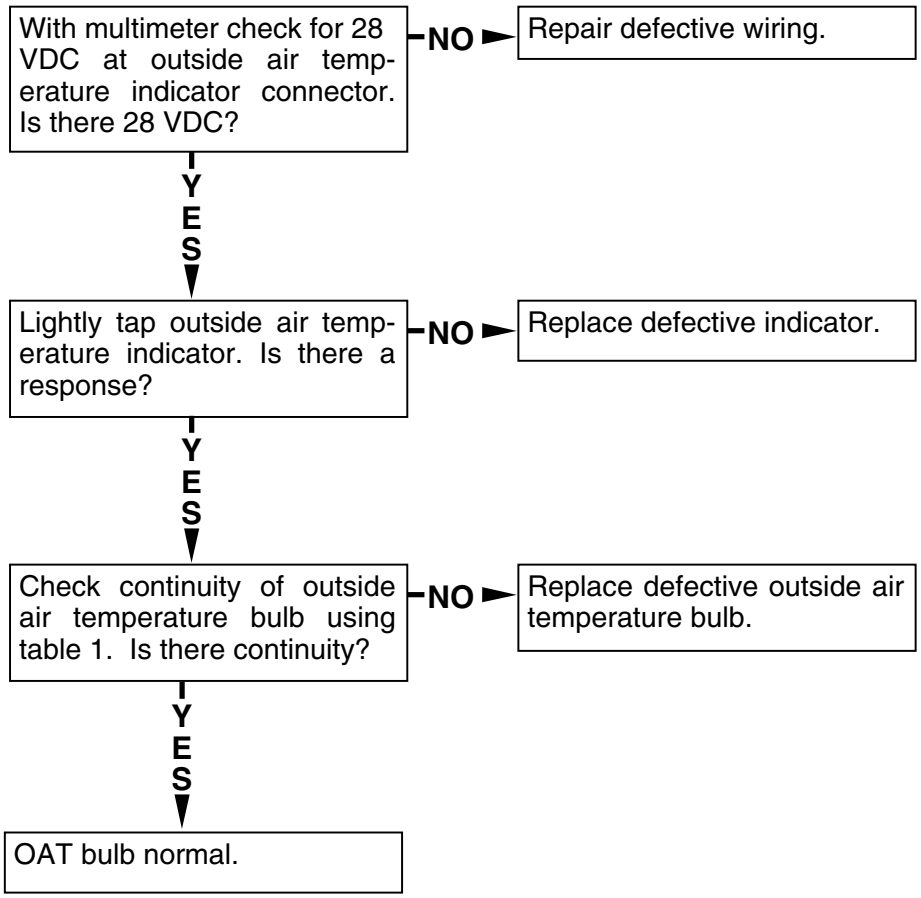
PROBLEM:

NO RESPONSE ON THE OUTSIDE AIR TEMPERATURE INDICATOR

START:

PREREQUISITES:

Verify outside air temperaure indicator properly installed and connected. Outside air temperature bulb properly installed and connected. OAT circuit breaker IN.



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OUTSIDE AIR TEMPERATURE – MAINTENANCE PRACTICE

1. Removal/Installation – Outside Air Temperature

A. Removal – Outside Air Temperature Bulb

(1) Remove the floorboard between the pilot and copilot's seat.

(2) Disconnect electrical connector from temperature bulb.

(3) Remove screws from outside aircraft.

(4) Remove outside air temperature bulb and gaskets.

B. Installation – Outside Air Temperature Bulb

(1) Position temperature bulb in mounting bracket.

(2) Install rubber gasket to prevent pressurization leaks.

NOTE: Install enough rubber gaskets so that the bulb is flush with aircraft skin.

(3) Secure bulb with screws.

(4) Connect electrical connector and safety.

(5) Install floorboard between pilot and copilot's seat.

A. Removal – Outside Air Temperature Indicator

(1) Remove post light cap.

(2) Loosen indicator retaining clamp screw.

(3) Withdraw indicator from instrument panel.

(4) Disconnect electrical connector.

B. Installation – Outside Air Temperature Indicator

(1) Position indicator, and connect electrical connector.

(2) Install indicator in instrument panel.

(3) Secure indicator by tightening retaining clamp screw.

(4) Install post light caps.

2. Inspection/Check – Outside Air Temperature

A. Inspection – outside Air Temperature Bulb

(1) Visually inspect outside air temperature bulb for cracks, damage, corrosion, or contamination that would affect temperature sensing.

(2) Inspect air temperature bulb for bent, broken, or burned pins.

(3) Check resistance across temperature bulb electrical pins, see Table 201 for resistance values.

B. Inspection – outside Air Temperature Indicator

(1) Inspect outside air temperature indicator for general condition and security of mounting.

RESISTANCE IN OHMS	F°	C°
71.984	-94	-70
76.028	-76	-60
80.056	-58	-50
84.070	-40	-40
88.070	-22	-30
92.060	-4	-20
96.040	+14	-10
100.000	32	0
103.950	50	10
107.890	68	20
111.820	86	30
115.740	104	40
119.650	122	50

OAT Bulb Resistance Values
Table 201

ATTITUDE AND DIRECTIONAL – DESCRIPTION AND OPERATION

1. General

A. A turn and slip indicator installed on the pilot's instrument panel, provides the pilot with visual indication of the rate and coordination of a controlled turn. The gyro operated rate of turn is powered by 28 VDC. The bank portion of the indicator is composed of a ball sealed in a curved inclinometer tube.

B. A magnetic compass mounted above the cockpit center windshield is a self contained instrument. The compass provides directional heading indication in reference to the earth's magnetic field.

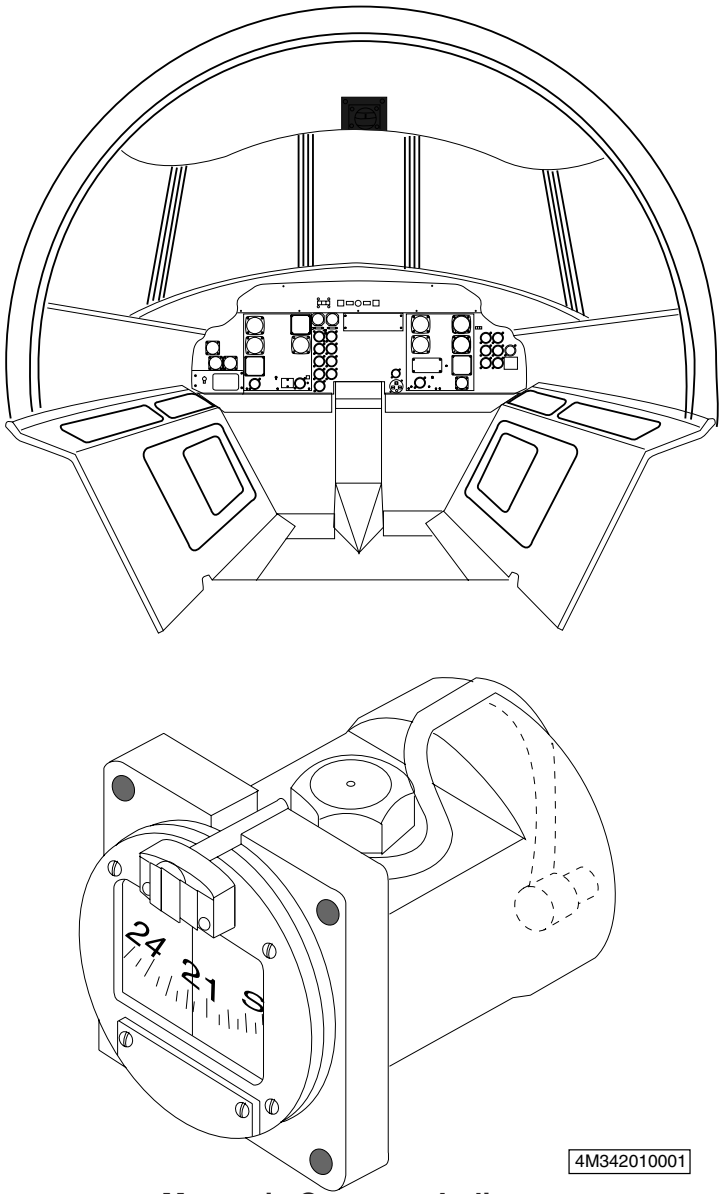
MAGNETIC COMPASS INDICATOR – DESCRIPTION AND OPERATION

1. General

A. The magnetic compass is mounted in the headliner above the center windshield.

B. The compass is a non-magnetic bowl containing a compass indicating card and filled with liquid. The card is mounted on a floating disk with numbers painted on its edge. A set of small magnetized bars or needles is fastened to the card. The card-magnet assembly is suspended on a jeweled pivot that allows the magnets to align with north-south direction of the earth's magnetic field.

C. The compass card and a fixed position reference marker (lubber line) are visible through a glass window on the front of the indicator.



Magnetic Compass Indicator
Figure 1

MAGNETIC COMPASS INDICATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Magnetic Compass

A. Removal – Magnetic Compass

(1) Remove cockpit headliner.

(2) Disconnect electrical connector.

(3) Remove mounting screws and nuts.

(4) Remove indicator from headliner.

B. Installation – Magnetic Compass

(1) Position indicator in headliner.

(2) Secure with compass to headliner with screws and nuts.

(3) Connect electrical connector.

(4) Install cockpit headliner.

2. Adjustment/Test – Magnetic Compass

A. When the magnetic compass does not provide satisfactory directional indications, it can be adjusted by the ground swinging technique.

(1) Tools and equipment required:

(a) Brass or other non-ferrus screwdriver.

(b) Aircraft tow vehicle.

(c) Aircraft tow bar.

(d) Compass rose, or master compass.

(2) Preparation:

(a) Remove all nonessential magnetic materials and tools from the area adjacent to the aircraft and compass.

(b) Place any other magnetic materials in the location they will occupy during normal flight.

(3) Operate the following equipment during compass swing:

(a) Engines.

(b) Fuel boost pumps.

(c) Electrical power supplies.

(d) All avionics.

(e) Rotating beacon.

(f) Interior Lights.

NOTE: Neutralize compensating magnets by matching dots on the compensating screws with the dots on the case.

(4) Compass Swing:

(a) If available, use a compass rose to align the aircraft. As an alternate method, use a master compass on the cardinal headings.

1 Align aircraft with north magnetic heading, and compensate for heading deviation with north-south compensating screw.

2 Align aircraft with east magnetic heading, and compensate for heading deviation with east-west compensating screw.

3 Align aircraft with south magnetic heading, and compensate for one-half of the heading deviation with north-south compensating screw.

4 Align aircraft with west magnetic heading, and compensate for one-half (1/2) of the heading deviation with east-west compensating screw.

5 Turn aircraft on successive 30° headings through 360°.

6 Prepare a compass correction card to show correction to be applied at each of these headings.

7 Allow heading to stabilize before recording the readings.

NOTE: Tolerance is 10° maximum deviation on commercial aircraft.

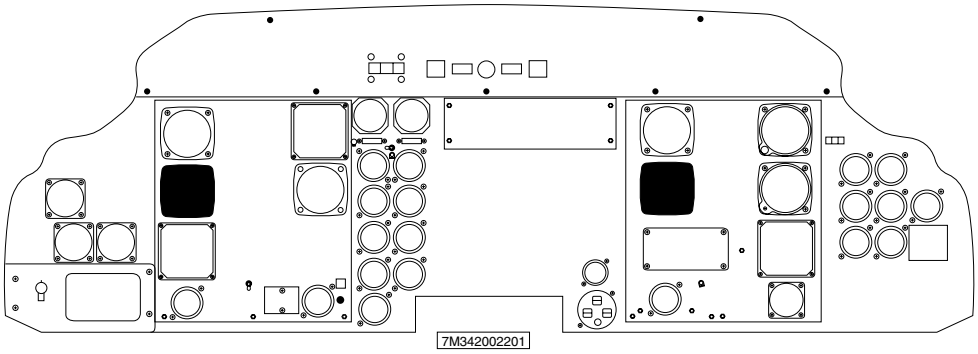
TURN AND SLIP INDICATOR – DESCRIPTION AND OPERATION

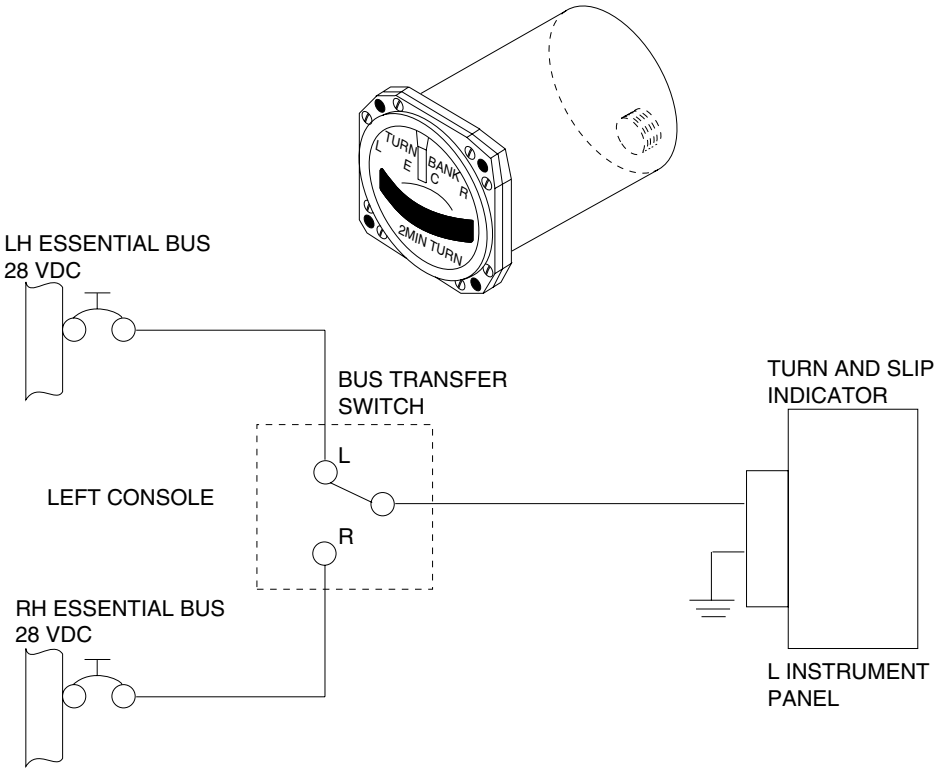
1. General

A. A turn and slip indicator is installed on the pilot's instrument panel. The indicator is used to provide the pilot with a visual indication of the rate and coordination of a controlled turn.

B. The turn portion of the indicator is controlled by an electrical gyroscope. The slip portion of the indicator is composed of a ball sealed in a curved inclinometer tube which indicates the degree of bank.

C. The gyroscope is driven by 28VDC from either the left or right essential bus and protected by a 3 amp circuit breaker; on T model aircraft, a 5 amp circuit breaker is used. A bus transfer switch, located on the left aft side console, allows selection of the bus.





Turn and Slip Indicator and Schematic
Figure 201

TURN AND SLIP INDICATOR – FAULT ISOLATION

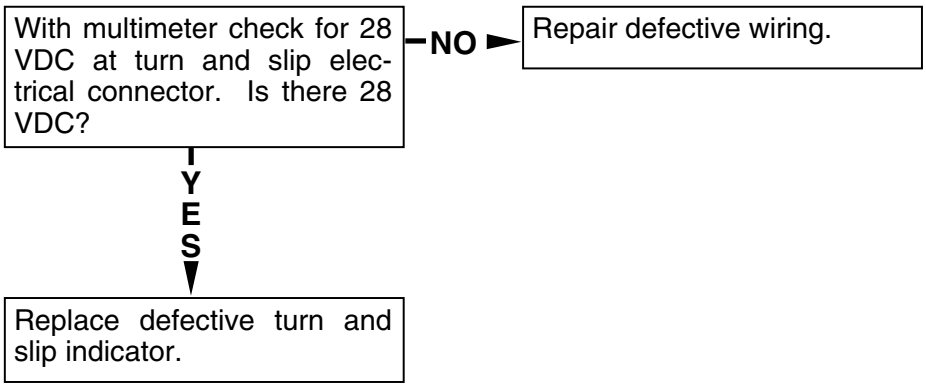
PROBLEM:

TURN AND SLIP INDICATOR ERRATIC OR INOPERATIVE.

PREREQUISITES:

Verify turn and slip indicators installed and connected, and circuit breaker IN.

START:



END

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TURN AND SLIP INDICATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Turn and Slip Indicator

A. Removal – Turn and Slip Indicator

(1) Lock elevator control column in neutral position.

(2) Loosen fasteners securing pilot's secondary panel to main instrument panel.

(3) Rotate secondary instrument panel aft to gain access to indicator electrical connector.

(4) Disconnect electrical connector from turn and slip indicator.

(5) Remove screws and withdraw indicator from secondary instrument panel.

B. Installation – Turn and Slip Indicator

(1) Position indicator in secondary instrument panel, and secure with mounting screws.

(2) Connect electrical connector.

(3) Secure secondary panel to main instrument panel with fasteners.

(4) Unlock elevator control column.

2. Inspection/Check – Turn and Slip Indicator

A. Inspect turn and slip indicator for general condition and security of mounting.

GROUND PROXIMITY WARNING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The Mark VI (MK VI) ground proximity warning system (GPWS) is designed to provide alerts and warnings of potential controlled flight into terrain. The primary component of this warning system is the ground proximity warning computer (GPWC). The MK VI warning computer is an analog controlled computer intended to provide ground proximity warnings to the flight crew. Data is collected from sensors on the aircraft and processed to determine if the aircraft is in danger. Visual and aural outputs advise the flight crew of dangerous situations and indicate if the system is operational.

B. The GPWS is comprised of the following components:

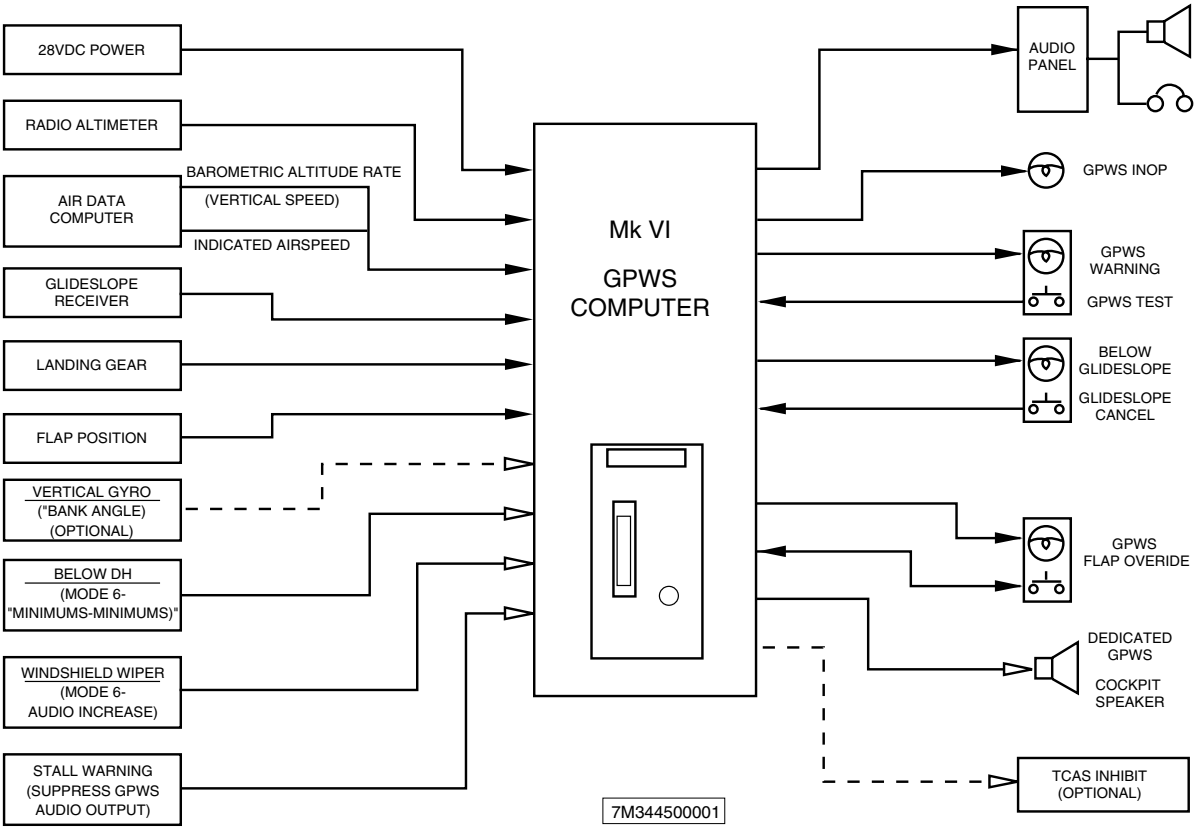
(1) Bendix-King KRA 405 Radio Altimeter R/T
or
Collins ALT 55B Radio Altimeter R/T.

(2) Bendix-King KNI 415 Radio Altimeter Indicator
or
Collins ALI 55 Radio Altimeter Indicator.

(3) Sundstrand Mark VI GPWS Computer, P/N 965-0686-001.

(4) CIC Air Data Computer, P/N 04077.

(5) Korry or equivalent annunciator warning cockpit lights/switches.



MK VI GPWS Block Diagram
Figure 1

GROUND PROXIMITY WARNING SYSTEM – MAINTENANCE PRACTICES

1. Adjustment/Test – Ground Proximity Warning System

A. Radio Altimeter Test

(1) Ensure the Radio Altimeter Indicator and the R/T unit are properly installed and electrically connected. Also, connect the forward antenna coax to the XMIT fitting and the aft antenna coax to the RCVR fitting. Push in the RAD ALT circuit breaker. Verify the indicator flag pulls shortly and the needle shows “0” feet. Set the DH bug below “0” feet. Verify the DH light extinguishes. Proceed with the following tests.

B. Normal Self Test

(1) Install the GPWS computer. Ensure the unit is secure in its tray and the electrical connector is fastened properly.

(2) Push in the GPWS CMPTR circuit breaker. Verify the following CB’s are in: GPWS ADC, RAD ALT, PILOT AUDIO, COPILOT AUDIO.

(3) Verify all GPWS annunciator lights are OFF.

C. Decision Height

(1) Set the DH to 200 feet and push the Press-to-Test for the Radio Altimeter. Verify the following occurs:

(a) DH Light - ON.

(b) Radio Altitude - 50 feet.

D. GPWS Self-Test – Pilot

(1) Depress and hold the pilot GPWS P/TEST switch.

(2) Verify GPWS INOP light and GPWS FLAP OVRD light illuminate immediately.

(3) Verify that approximately one half (1/2) second later:

(a) BELOW G/S light illuminates.

(b) Aural message “Glideslope” is heard once at reduced test volume.

(4) Verify that approximately one (1) to two (2) seconds later:

(a) RED GPWS warning lamp illuminates.

(b) Aural message “Pull-up” is heard two (2) to six (6) times at a reduced test volume.

(5) After the “PULL-UP” message is heard at least twice, release GPWS P/TEST switch.

(6) Verify all messages terminated and all lights extinguish.

NOTE: This is a successful SELF-TEST. If you hear any other sequence, message or omission of lights or messages, there is a fault in the GPWS or one of its inputs.

Important: Wait thirty (30) seconds for the GPWS CMPTR to re-initialize before attempting another SELF-TEST procedure.

If you are ready to perform a SELF-TEST, and you observe that the GPWS INOP lamp is on, the GPWS has a short menu of fault messages to tell you which input may be the problem.

(7) With the GPWS INOP light on, depress and hold the GPWS P/TEST switch. There are three possible aural fault messages:

(a) RADAR ALTIMETER FAULT.

(b) GLIDESLOPE FAULT.

(c) BARO RATE FAULT.

(8) The fault messages will repeat once, then release the GPWS P/TEST switch. These messages indicate that the problem is a failure in a system providing input into the GPWS computer, or a broken wire between the system and the GPWS computer.

E. GPWS Self-Test – Copilot

(1) Depress and hold the copilot GPWS P/TEST switch.

(2) Verify that all annunciations presented for pilot test are repeated.

F. Abnormal (Fault) Self-Test

(1) To repeat the GPWS FAULT sequence, allow a one (1) second delay between tests.

G. Radio Altimeter Fail

(1) Pull the RAD ALT circuit breaker and verify the following events:

(a) GPWS INOP light - ON within five (5) seconds

(b) Aural message during GPWS P/TEST - “RADAR ALTIMETER FAULT”.

(c) Reset the circuit breaker. NO GPWS warnings are accecptable during this circuit breaker OFF/ON cycle.

H. Air Data Fail

(1) Pull the GPWS ADC circuit breaker and verify the following events:

(a) GPWS INOP light - ON.

(b) Aural message during GPWS P/TEST - “BARO RATE FAULT”.

(c) Reset circuit breaker. NO GPWS warnings are accecptable during this circuit breaker OFF/ON cycle.

I. GPWS Fail

(1) Pull GPWS CMPTR circuit breaker. Verify the following:

(a) GPWS INOP light - ON.

(b) Reset the GPWS CMPTR circuit breaker. After a delay, verify the light extinguishes. No nuisance GPWS warnings shall result from this power OFF/ON cycle.

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CHAPTER 35 – OXYGEN SYSTEM

LIST OF EFFECTIVE PAGES

SUBJECT	SECTION	PAGE	DATE	SUBJECT	SECTION	PAGE	DATE
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35-EFF-PGS	-	2	APR 01/04	35-00-10	MP	202	APR 01/98
35-CONTENTS	-	1	APR 01/98	35-00-20	DO	1	APR 01/98
35-CONTENTS	-	2	APR 01/05	35-00-20	MP	201	APR 01/98
35-00-00	DO	1	APR 01/98	35-00-20	MP	202	APR 01/98
35-00-00	DO	2	APR 01/98	35-00-30	DO	1	APR 01/98
35-00-00	DO	3	APR 01/98	35-00-30	MP	201	APR 01/98
35-00-00	DO	4	APR 01/04	35-00-30	MP	202	APR 01/98
35-00-00	FI	101	APR 01/98	35-00-40	DO	1	APR 01/98
35-00-00	FI	102	APR 01/98	35-00-40	DO	2	APR 01/98
35-00-00	FI	103	APR 01/98	35-00-40	MP	201	APR 01/98
35-00-00	FI	104	APR 01/98	35-00-50	DO	1	APR 01/98
35-00-00	FI	105	APR 01/98	35-00-50	MP	201	APR 01/98
35-00-00	FI	106	APR 01/98	35-00-60	DO	1	APR 01/98
35-00-00	MP	201	APR 01/98	35-00-60	MP	201	APR 01/98
35-00-00	MP	202	APR 01/98	35-00-70	DO	1	APR 01/98
35-00-00	MP	203	APR 01/04	35-00-70	MP	201	APR 01/98
35-00-00	MP	204	APR 01/04	35-00-80	DO	1	APR 01/98
35-00-00	MP	205	APR 01/04	35-00-80	MP	201	APR 01/98
35-00-00	MP	206	APR 01/04	35-00-90	DO	1	APR 01/98
35-00-00	MP	207	APR 01/04	35-00-90	MP	201	APR 01/98
35-00-10	DO	1	APR 01/98	35-SERVC	SI	1	APR 01/98
35-00-10	DO	2	APR 01/98	35-SERVC	SI	1	APR 01/98

MAINTENANCE MANUAL

CHAPTER 35 – OXYGEN SYSTEM

LIST OF EFFECTIVE PAGES

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CHAPTER 35 – OXYGEN SYSTEM

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CHAPTER 35 – OXYGEN SYSTEM

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ALT. COMPENSATOR REGULATOR DESCRIPTION AND OPERATION	35-00-70	1	T AT TC	– – – 001-018 201-211EE
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OXYGEN – SERVICE DESCRIPTION AND OPERATION	35-SERVC	1	T AT TC	201-999 001-999 201-999

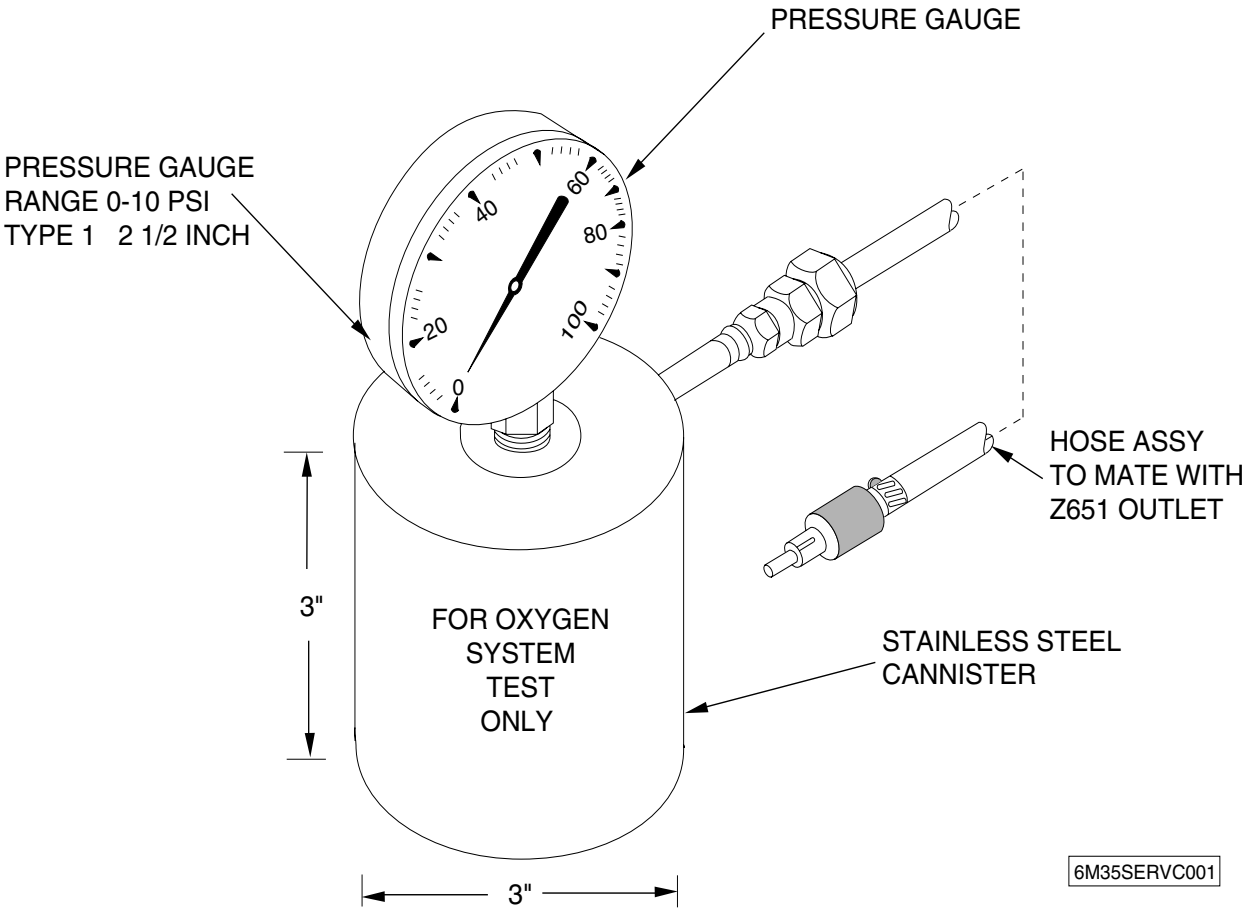
OXYGEN – SERVICE INFORMATION

1. General

A. The following information is supplied as a supplement to this chapter. Subject matter is included which is not applicable to an ATA 100 format, but is still of interest to Merlin and Metro owner/operators. The information in this section is not part of the basic manual, and may be removed or retained at the discretion of the owner/operator.

B. 35.01 – Type-1 Oxygen System Tester

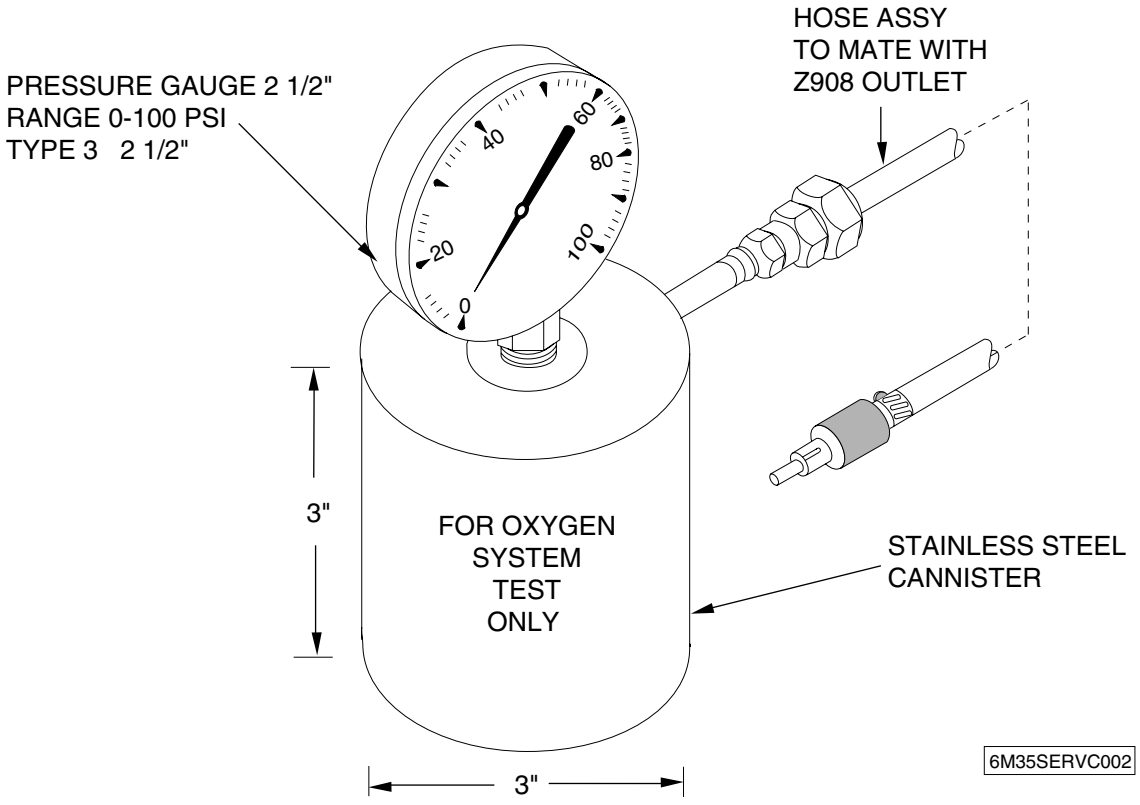
(1) The type-1 test pressure gauge fixture shown in Figure 1 provides a means of checking the oxygen system pressure and oxygen leakage rate on models AT001-018 and TC201-211EE. The test fixture may be manufactured by welding 1/8 inch stainless steel into the canister size shown. Weld a nut into the top of the fixture to mate the threads of the oxygen pressure gauge used. Gauge range should be 0-10 psi. On the side weld a 1/4 inch stainless steel tube and install a hose assembly with a connector to mate with the oxygen outlet.



Oxygen System Tester
Figure 1

C. 35.02 – Type-3 Oxygen System Tester (Figure 2)

(1) The type-3 test pressure gauge fixture shown in Figure 2 provides a means of checking the oxygen system pressure and oxygen leakage rate on models T201-999, AT019-999, TC211EEE-999. The test fixtures may be manufactured by welding 1/8 inch stainless steel into the canister size shown. Weld a nut into the top of the fixture to mate the threads of the oxygen pressure gauge used. Gauge range should be 0-100 psi. On the side, weld a 1/4 inch stainless steel tube and install a hose assembly with a connector to mate with the oxygen outlet.



Oxygen System Tester
Figure 2

OXYGEN SYSTEM – DESCRIPTION AND OPERATION

1. General

A. An emergency oxygen system is installed as standard equipment on all SA226 aircraft. It is intended for use in the event of pressurization loss at altitude, smoke in the cockpit or passenger cabin, and may be used for medical purposes.

B. Components of the system are: pressure regulator, oxygen cylinder, lines and fittings, outlets, pressure gauge, service connection, Toggle Valve, and control systems.

C. The oxygen system installed on models T201-248 is shown in Figure 1 (Push/Pull cable regulator control). Figure 2 shows the oxygen system installed in models T249-999, AT001-999, and TC201-999 (manual regulator control and passenger toggle valve control switch).

2. Oxygen Pressure Regulator

A. The pressure regulator is mounted directly to the oxygen cylinder. It reduces cylinder pressure to a system operating pressure of 70 psi, ± 10 psi. The regulator incorporates a shutoff valve for system control, a fill port for cylinder servicing, a pressure gauge port for cylinder pressure monitoring, and (on models T249-999, AT001-999, and TC201-999) an overboard dump port to relieve excess pressures.

3. Oxygen Cylinder

A. The oxygen storage cylinder is mounted horizontally in the nose baggage compartment forward of the nose wheel well. The cylinder stores high pressure gaseous oxygen regulated to system pressure for flight crew and passenger use.

4. Oxygen Lines and Fittings

A. Oxygen is distributed through aluminum, copper, and plastic lines, and fittings to the various system components during normal system operation.

5. Oxygen Outlets

A. Plug in type oxygen outlets are provided at every seat location, including the lavatory. Passengers are provided disposable constant-flow masks that are stored in the seat back pockets.

B. With passenger oxygen ON, oxygen will flow any time one of the passenger mask bayonet fittings is plugged into an outlet. For this reason, all masks should be unplugged when not in use.

C. The flight crew is furnished permanent demand type oxygen masks and smoke goggles that are stored under the crew chair or in the seat back pockets.

6. Oxygen Pressure Gauge

A. The oxygen pressure gauge mounted on the copilot's instrument panel provides a continuous indication of system pressure.

7. Oxygen Service Connection

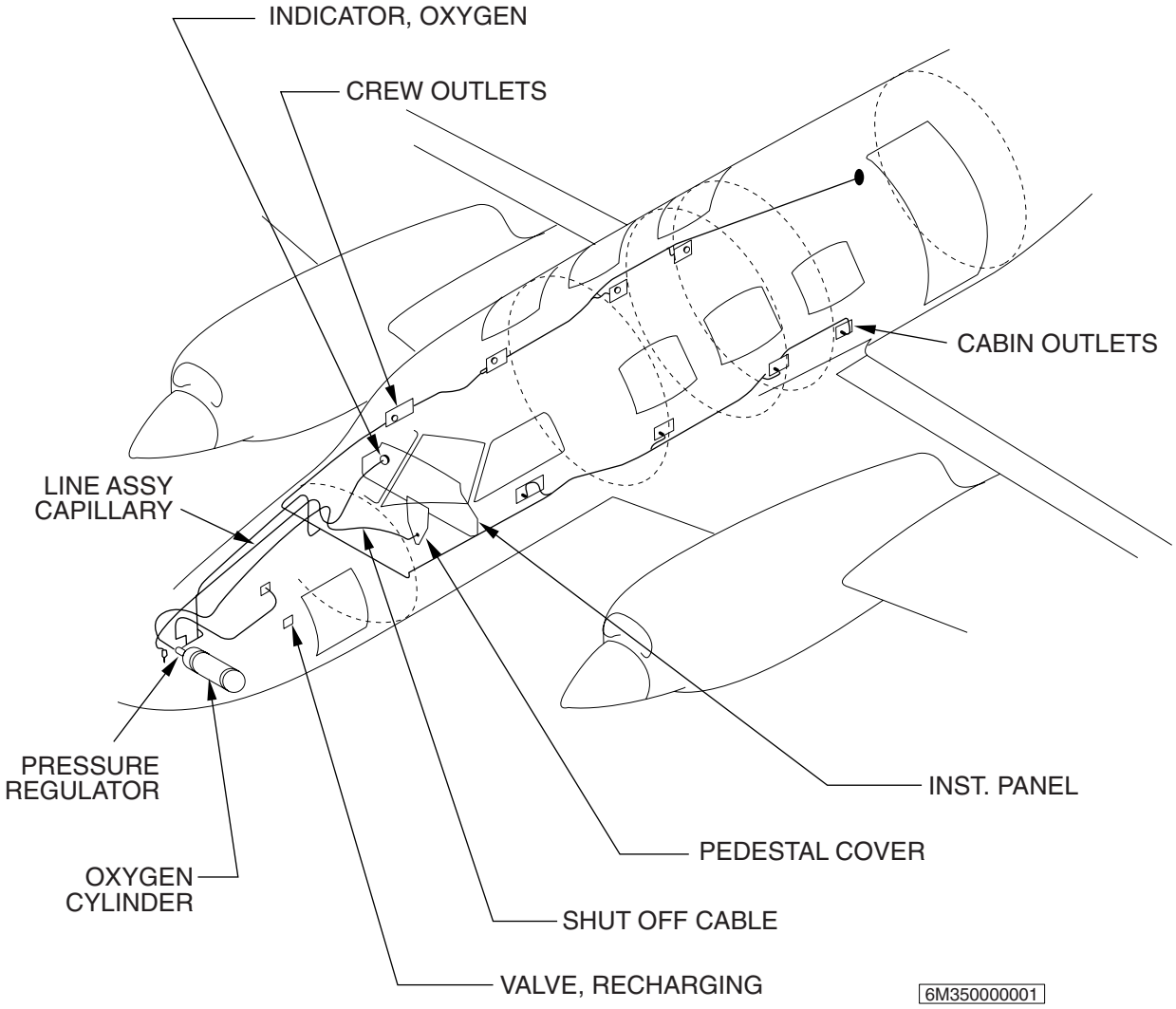
A. The oxygen filler valve located at the upper forward corner of the right hand baggage door frame is used for remote filling of the high pressure oxygen cylinder.

8. Altitude Compensating Regulator

A. On models AT001-018 and TC201-211EE oxygen pressure is routed from the toggle valve through an altitude compensating regulator to conserve oxygen supplied to the passenger compartment. An aneroid sensing device in the regulator responds to cabin altitude variations by increasing or decreasing the regulated output of oxygen. The altitude compensating regulator has been deleted from later models.

9. Push-Pull Cable Control System (Figure 1)

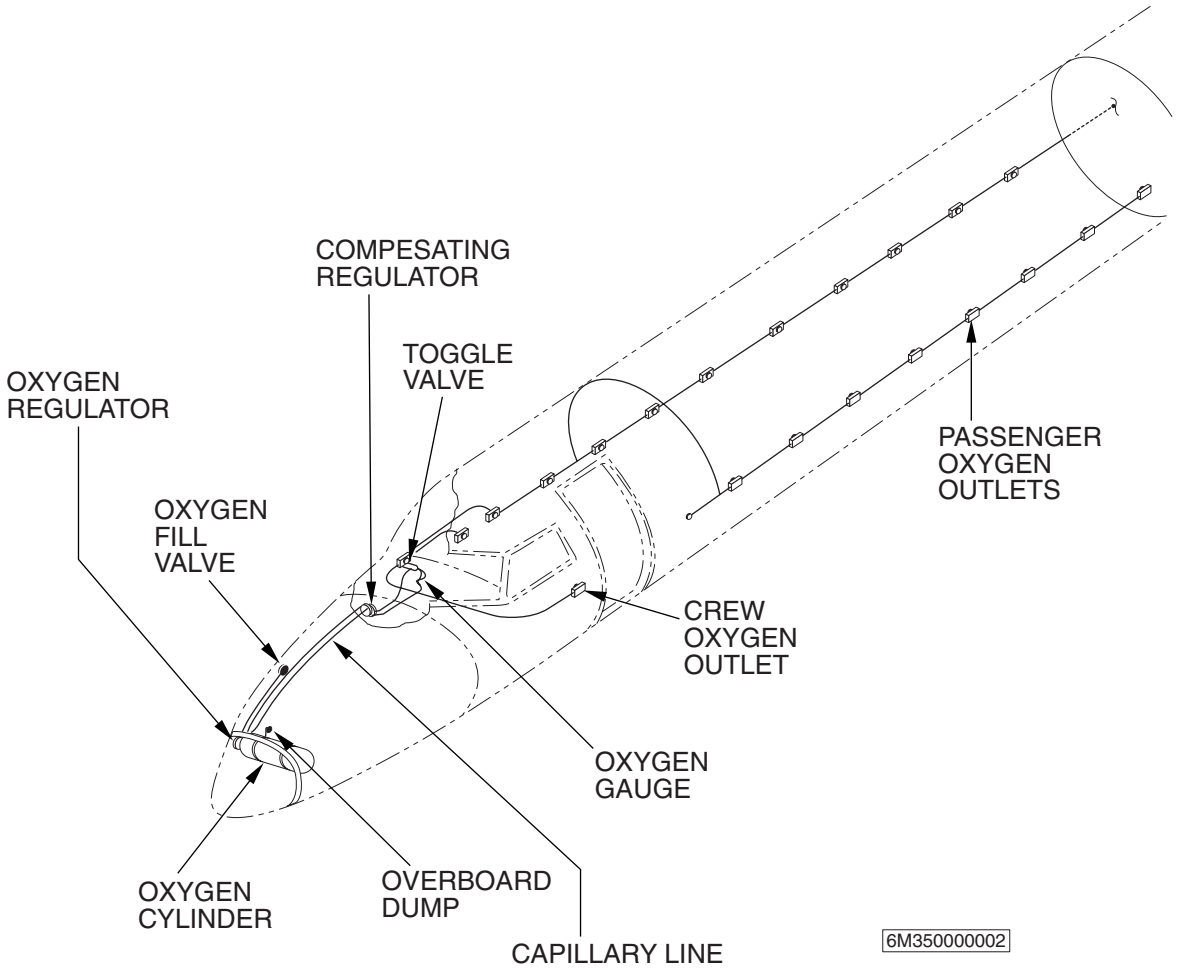
A. On models T201-248, a cockpit pedestal mounted push-pull knob and cable control system opens or closes the cylinder mounted regulator shutoff valve. Pulling the control cable knob opens the regulator, allowing 70 psi, ± 10 psi oxygen pressure to flow through the supply tubing from the pressure regulator to the crew and passenger outlets.



Oxygen System
Models T201-248
Figure 1

10. Manual Control & Toggle Valve System (Figure 2)

A. On models T249-999, AT001-999, and TC201-999, a pressure regulator mounted directly to the cylinder incorporates a manually operated shutoff valve. With the shutoff valve open, pressure regulated to 70 psi, ± 10 psi is allowed to flow through the supply tubing to the pilot and copilot's outlet. A manually operated toggle switch mounted on the copilot's instrument panel gives the flight crew on and off control of the passenger oxygen system.



Oxygen System
Models T249-999, AT001-999, TC201-999
Figure 2

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OXYGEN SYSTEM – FAULT ISOLATION

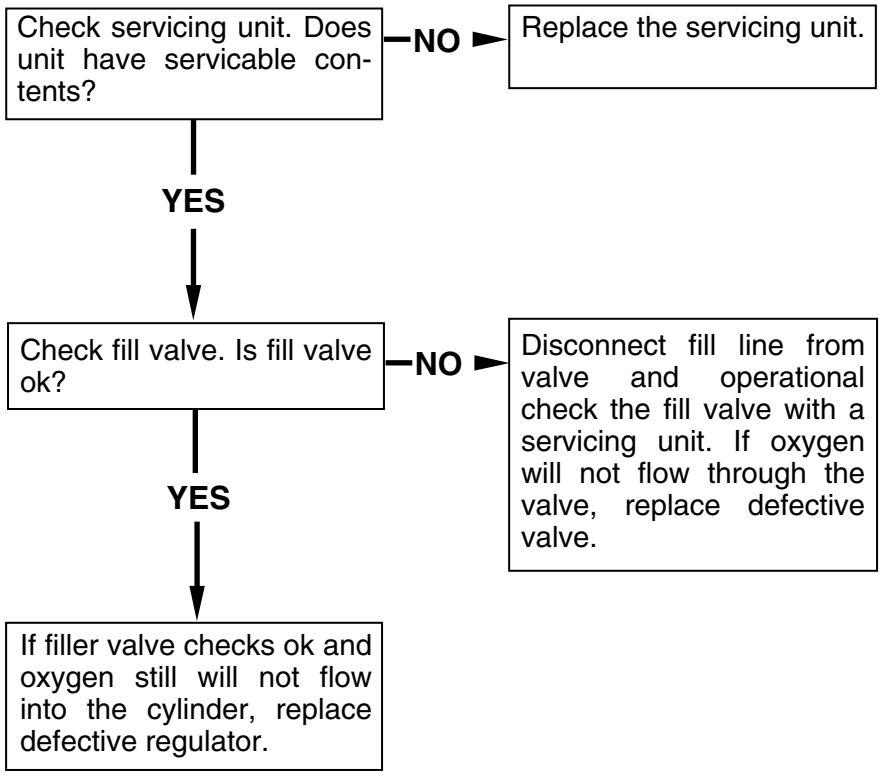
PROBLEM:

OXYGEN WILL NOT FLOW INTO THE CYLINDER FROM A SERVICING UNIT.

PREREQUISITES:

Fully charged servicing unit.

START:



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END

OXYGEN SYSTEM – FAULT ISOLATION

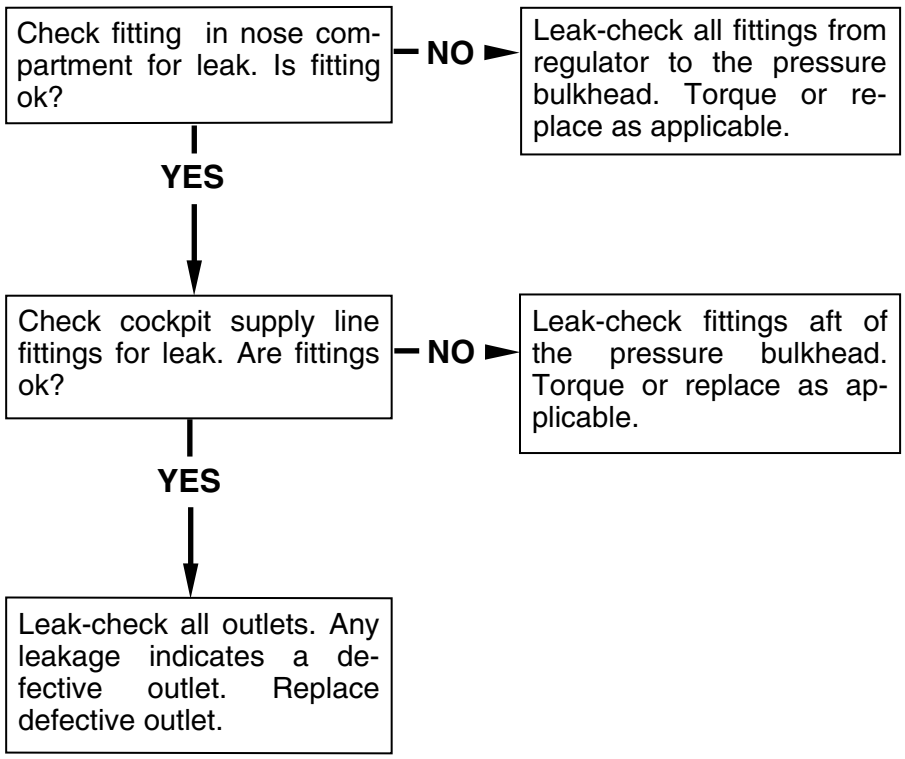
PROBLEM:

OXYGEN CYLINDER LOSES PRESSURE OVER NIGHT WITH THE PRESSURE REGULATOR TURNED ON.

PREREQUISITES:

Oxygen cylinder full.

START:



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END

OXYGEN SYSTEM – FAULT ISOLATION

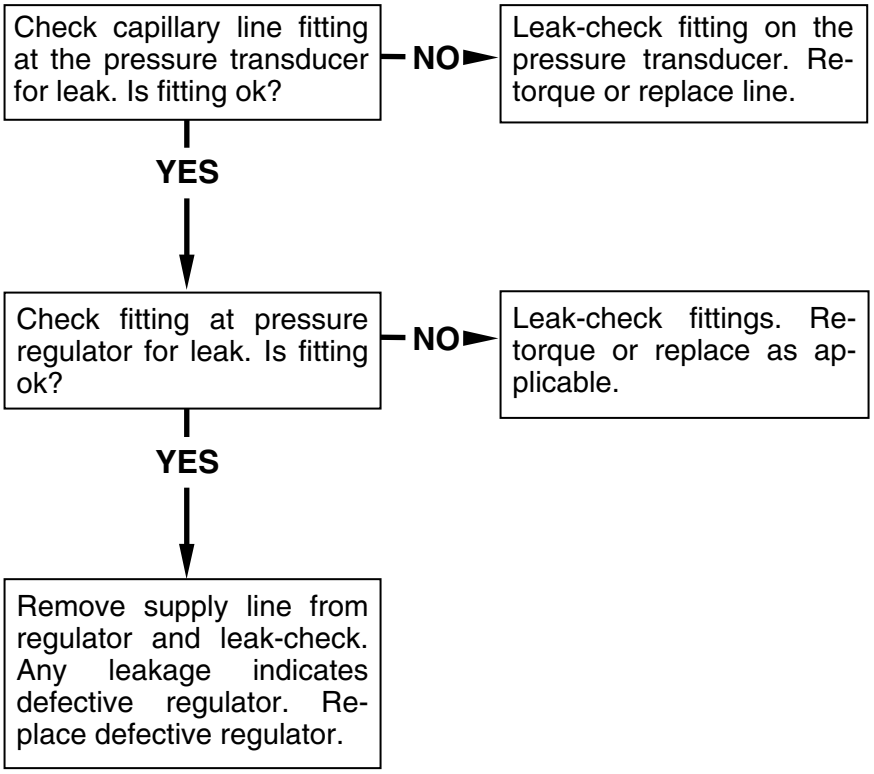
PROBLEM:

OXYGEN CYLINDER LOSES PRESSURE OVER NIGHT WITH THE PRESSURE REGULATOR TURNED OFF.

PREREQUISITES:

Oxygen cylinder full.

START:



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END

OXYGEN SYSTEM – FAULT ISOLATION

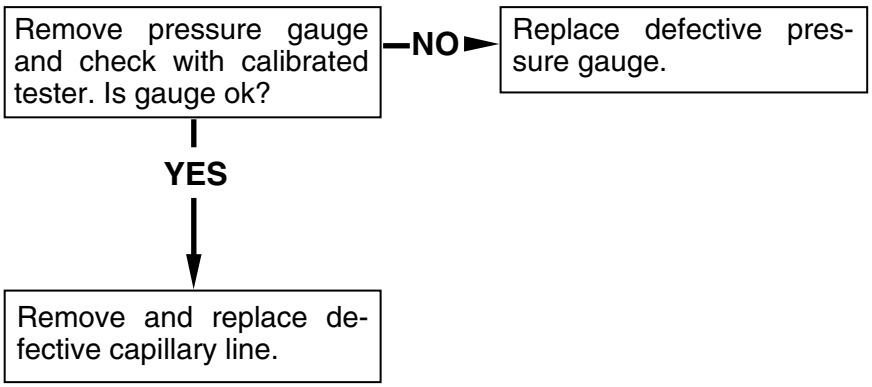
PROBLEM:

PRESSURE GAUGE INDICATES INCORRECT PRESSURE.

PREREQUISITES:

Power on, pressure gauge circut breaker IN, and oxygen system charged.

START:



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END

OXYGEN SYSTEM – FAULT ISOLATION

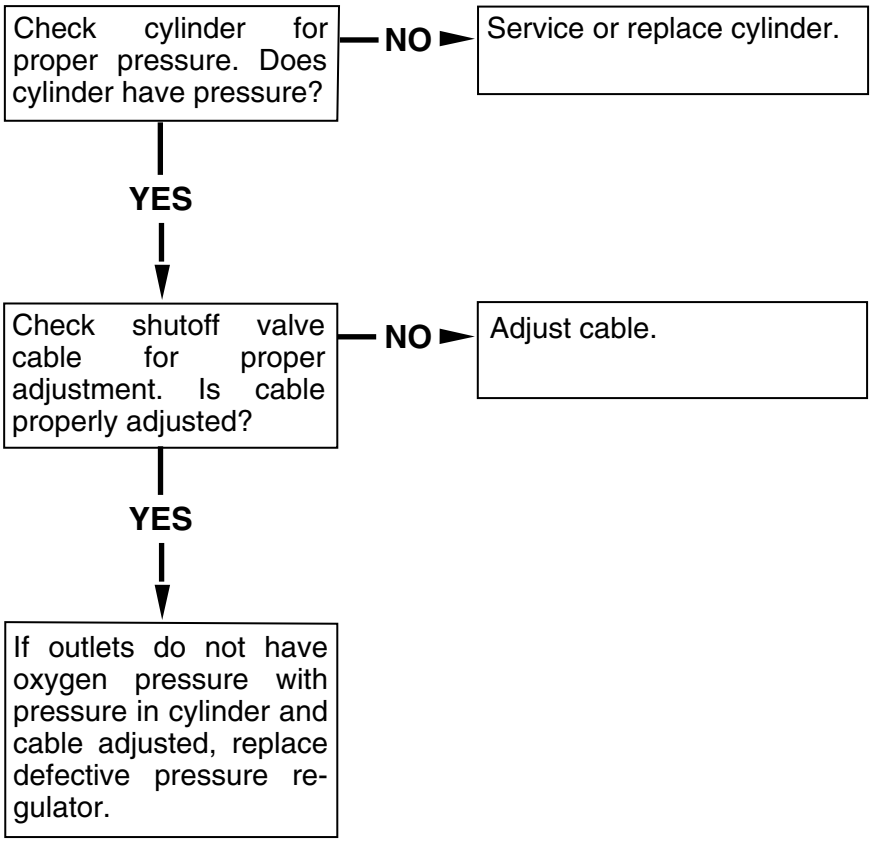
PROBLEM:

NO OXYGEN PRESSURE TO THE PASSENGER OUTLETS, MODELS T210-248 ONLY, EQUIPPED WITH PUSH/PULL ON/OFF CONTROL.

PREREQUISITES:

Oxygen system charged, pressure regulator and toggle valve ON.

START:



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END

OXYGEN SYSTEM – FAULT ISOLATION

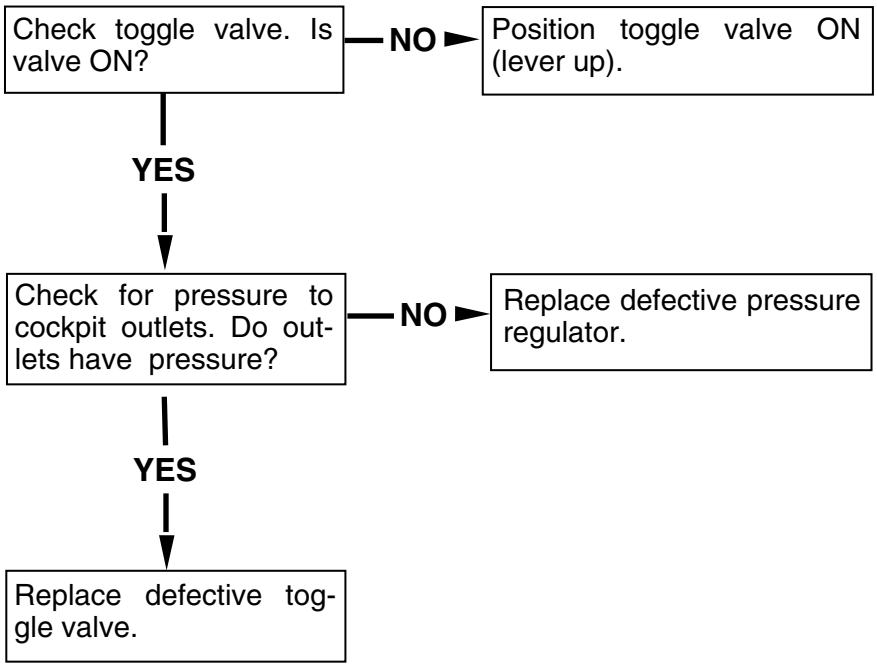
PROBLEM:

NO OXYGEN PRESSURE TO THE PASSENGER OUTLETS, MODELS T249-999, AT001-999, AND TC201-999 ONLY, EQUIPPED WITH TOGGLE VALVE.

PREREQUISITES:

Oxygen system charged, pressure regulator and toggle valve ON.

START:



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END

OXYGEN SYSTEM – MAINTENANCE PRACTICES

1. Servicing – Oxygen System

A. Oxygen System Safety Precautions

NOTE: The utmost care must be exercised in servicing, handling, and inspecting the oxygen system. Comply with the following precautions:

(1) Use only special lubricants and leak detecting material specifically made to be used with oxygen.

(2) Exercise caution to ensure the oxygen system will not become contaminated by leaking fuel, oil, and hydraulic fluids.

(3) Cap open ends of cleaned and dried tubing at all times. Ensure there are no uncapped openings in the installation at any time except during attachment or detachment of parts. Do not use tape to seal the open tube ends.

(4) Allow at least two (2) inches clearance between oxygen tubing and equipment, flexible control cables, and other rigid or moving parts, to prevent chaffing against each other.

(5) Route oxygen system tubing, fittings, and equipment away from hot ducts, conduits, or equipment. Insulate the hot ducts, conduits, or equipment, or spaces between these items from oxygen system, tubing, fittings, and equipment to prevent heating of the oxygen system.

(6) Mount all tubing, fittings, and connections to prevent vibration and chafing by the use of clamps, grommets, and other suitable materials.

(7) The maximum length of unsupported tubing shall be 24 inches (60.96 cm) for 3/16 O.D. copper, and 36 inches (91.44 cm) for 5/16 O.D. aluminum alloy. Clamps should be loop type line support conforming to AN742 or equivalent.

(8) Never tighten oxygen system fittings with pressure applied.

(9) The entire system should be free from the following:

(a) Oil and grease

(b) Water

(c) Other foreign matter

(10) Anti-seize and thread sealing compound should be IAW specification MIL-T-5542B.

(a) Do not apply thread sealing compound on aluminum alloy flared tube fitting straight threads. Proper flaring and tightening should be sufficient to make a flared tube connection leakproof.

(b) Treat all male tapered pipe threads with approved thread sealing compound.

(c) Exercise caution to ensure compounds are carefully and sparingly put only on male threads, coating the first three (3) male threads from the end of the fitting. Use no compound on the coupling sleeves or on the outside of the tube flares.

(11) Ensure tools, equipment, clothing, and hands are clean and free of oil or grease at all times while performing maintenance on the oxygen system. Take precautions to wash thoroughly immediately after eating so that there are no animal fats or other contaminants on the hands.

B. Oxygen System – Filling

NOTE: A hydrostatic (pressure) test is required every five years for cylinders bearing a ICC (or D.O.T.) 3AA designation, and every three years for those bearing an ICC (or D.O.T.) 3HT designation. Do not fill cylinders that have not been inspected within the specified time interval. Do not fill cylinders without an ICC (D.O.T.) 3AA or 3HT designation.

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

CAUTION: IF OXYGEN IS ALLOWED TO FLOW INTO THE CYLINDER TOO FAST, EXCESSIVE HEATING OF THE CYLINDER WILL OCCUR, RESULTING IN EXPANSION OF THE OXYGEN IN THE CYLINDER. THIS WILL INDICATE A HIGHER PRESSURE THAN ACTUALLY EXISTS.

(1) Open the right nose baggage compartment door. The filler valve is located next to the forward upper corner of the door frame.

(2) Remove the dust cover from the oxygen filler fitting (see Placards at Fitting), and place the servicing cylinder fitting on the filler fitting (pure breathing oxygen only), and tighten enough to prevent leakage.

(3) Open the servicing cylinder valve slowly until oxygen is heard going into the aircraft cylinder.

(4) When the aircraft cylinder reaches full pressure, (1,800 psi for T models and 1,850 psi for AT and TC models) as indicated on the cockpit pressure gauge, close the servicing cylinder valve.

(5) Remove the servicing cylinder fitting from the filler fitting, replace the filler fitting dust cover, and secure the area.

NOTE: If time and circumstances permit, allow the cylinder to cool down for 30 minutes to one hour and recheck the cylinder pressure. If the pressure has dropped appreciably, repeat steps **1.**, B., (1) through (5).

C. Bleeding Oxygen Cylinder – Cylinder Installed In Aircraft

NOTE: Complete bleeding of the oxygen cylinder is not necessary except to remove/install the oxygen cylinder regulator.

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(1) Turn shutoff valve OFF.

(2) Bleed trapped pressure from supply line by inserting an oxygen mask fitting into an outlet and breathing through the mask.

(3) Disconnect supply line from regulator.

(4) Connect a clean length of hose to supply line port on the regulator. Hose should be long enough to reach four (4) or five (5) feet outside the aircraft nose baggage compartment.

(5) Securely grasp the free end of the hose and have an assistant slowly open the regulator control valve.

WARNING: WHEN OXYGEN IS BEING REMOVED FROM THE CYLINDER DO NOT DIRECT THE OPEN END OF THE BLEED HOSE TOWARD ANY PERSON OR ANY AREA CONTAINING OIL OR GREASE.

(6) When cylinder is empty, remove bleed hose from regulator control valve.

(7) Securely close regulator control valve, and reconnect supply line to the regulator.

NOTE: When an oxygen cylinder has been completely depressurized, condensation may form in the cylinder if it is not charged within two (2) hours. If the cylinder is not charged within two (2) hours, it must be purged of any condensation prior to servicing, IAW Advisory Circular 43.13, para. 363.

D. Bleeding Oxygen Cylinder – Cylinder Removed From Aircraft

NOTE: Complete bleeding of the oxygen cylinder is not necessary except to remove/install the oxygen cylinder regulator.

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(1) Securely clamp the cylinder in a cradle.

(2) Ensure the area is free from oil, grease, combustible fumes, fires, and is well ventilated. Position the regulator outlet port so that it is directed away from personnel, or any grease or oil contaminated area.

(3) Slowly open the regulator control valve, and allow the oxygen to bleed out of the cylinder until it is empty.

(4) When the cylinder is empty, securely close the regulator control valve.

NOTE: When an oxygen cylinder has been completely depressurized, condensation may form in the cylinder if it is not charged within two (2) hours. If the cylinder is not charged within two (2) hours, it must be purged of any condensation prior to servicing, IAW Advisory Circular 43.13, para. 363.

2. Adjustment/Test – Oxygen System

A. Operational Test – Oxygen System

(1) Check pressure gauge for full system pressure (1,800 psi for T models and 1,850 psi for AT and TC).

(2) Pull out control cable, placing regulator in ON position (models T201-248 only). For all other aircraft (T249-999, AT001-999, TC201-999), turn the pressure regulator ON in the nose baggage compartment.

(3) Plug an oxygen mask hose fitting into the pilot and copilot outlets and check for flow by breathing through the mask.

(4) Turn toggle valve to ON (all aircraft except T201-248 which do not have a toggle valve).

(5) Plug an oxygen mask hose fitting into a passenger outlet and check for oxygen flow. Check all outlets accordingly.

(6) Turn toggle valve OFF (all aircraft except T201-248). Verify that flow to passenger masks has stopped.

(7) Push in control cable (models T201-248). Verify that flow to the pilot and copilot's masks has stopped.

3. Inspection/Check – Oxygen System

A. Inspection – Oxygen System

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(1) Inspect the oxygen cylinder for charge, maintaining 1,800 psi models T201-999 and 1,850 psi for all other models.

(2) Inspect cylinder and mounting bracket for condition and security. Remove from service, any cylinders that show signs of abuse, dents, bulges, cracks, distortion, damaged threads, or defects that might render the cylinder unsafe. Replace or repair any cylinder mount brackets that show signs of wear.

(3) Inspect the cylinder mounted regulator/shutoff valve for operation and condition.

(4) Inspect lines, fittings, and filler valve for leaks, security, chafing, dents, cracks, and clearance from other lines and adjacent structure.

(5) Check for the presence of the green overboard discharge indicator disk on the lower right side of the fuselage (models T249-999, AT001-999, and TC201-999). Inspect the rupture disk on the pressure regulator.

(6) Inspect outlets for condition and security. Inspect dust cover for proper installation.

(7) Perform an oxygen system leak check IAW instructions this chapter.

(8) Inspect crew and passenger masks for cleanliness, deterioration, operation, and proper storage.

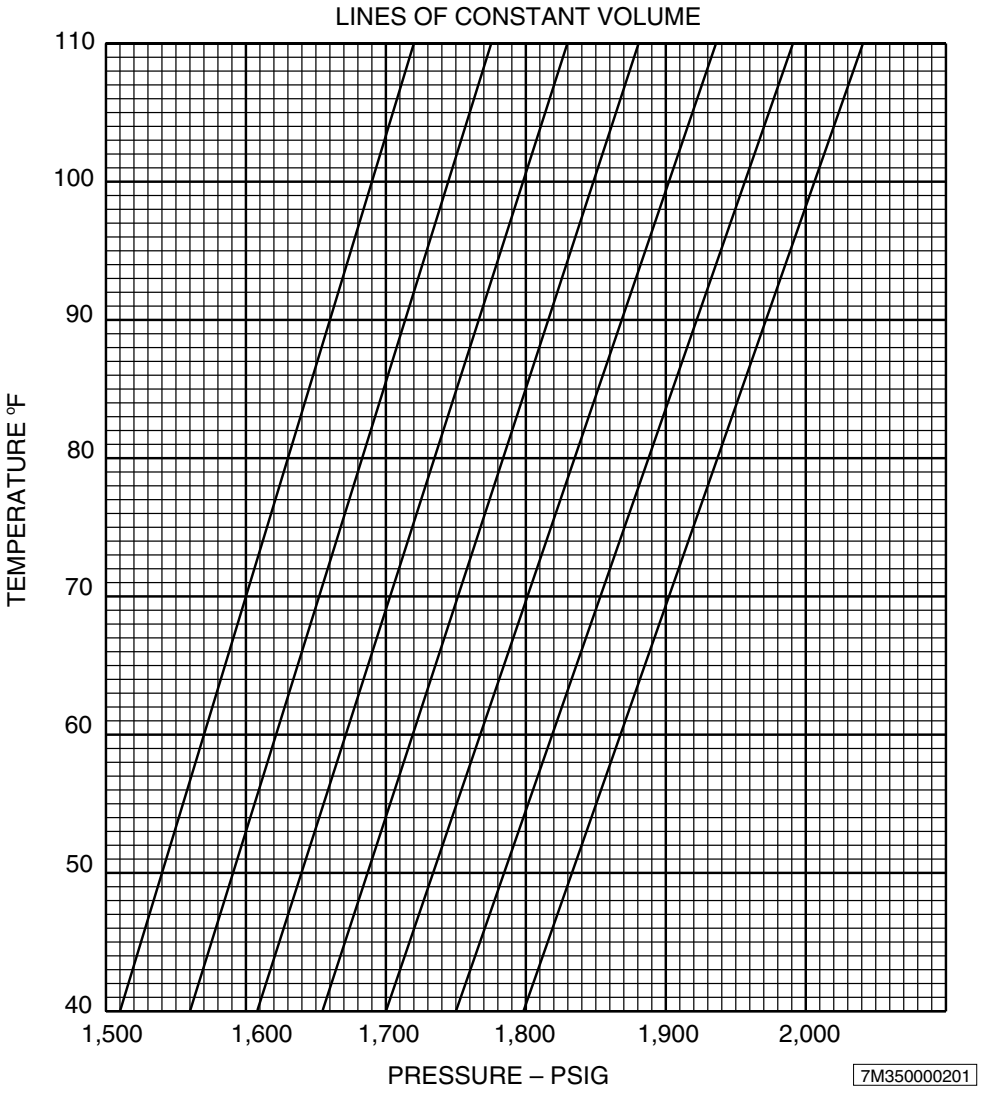
(9) Inspect hoses for condition and serviceability.

(10) Inspect hydrostatic test date. Hydrostatic test standard weight cylinders every five (5) years.

NOTE: The last date of testing is indicated on top of cylinder along with the metal stamp designation ICC or (D.O.T.) 3AA 1800, which classifies the cylinder as a standard weight cylinder. Refer to Chapter 5 for cylinder part number and testing specifications.

(11) Inspect hydrostatic test date. Hydrostatic test lightweight cylinders every three (3) years.

NOTE: The last date of testing is indicated on top of cylinder along with the metal stamp designation ICC or (D.O.T.) 3AA 1800, which classifies the cylinder as a standard weight cylinder. The ICC or (D.O.T.) 3HT 1850 designated cylinders must be retired from service after 24 years or 4,380 pressurizations, whichever occurs first. Refer to Chapter 5 for cylinder part number and testing specifications.



Cylinder Pressure vs. Temperature
Figure 201

B. Leak Check – Oxygen System

(1) Materials and Equipment:

(a) Oxygen System, Leak Detector Fluid (MIL-L-25567, Type 1 Zep Aero Sherlock Leak Detector or equivalent).

(b) Test Pressure Gauge Fixture, Type 1, Range 0-10 psi (see 35-SERVC).

(c) Test Pressure Gauge Fixture, Type 3, Range 0-100 psi (see 35-SERVC).

(2) Procedure:

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(a) Ensure that oxygen cylinder is serviced to at least 1,700 psi.

WARNING: BLEED OFF OXYGEN AS NECESSARY TO PREVENT PRESSURE GAUGE IN COCKPIT FROM INDICATING FULL SCALE (2,000 PSI) OR ABOVE. WHEN PRESSURE IS SO HIGH THAT IT CANNOT REASONABLY BE DETERMINED, DANGEROUS CONDITIONS COULD EXIST.

(b) Plug test gauge hose fitting into any convenient passenger outlet. Use Type 1 Gauge Fixture for models AT001-018, and TC201-211EEE (aircraft equipped with Altitude Compensating Regulators). Use Type 3 Gauge Fixture for all other aircraft.

(c) Turn oxygen system shutoff valve ON. Turn toggle valve ON for aircraft so equipped (models T249-999, AT001-999, TC201-999).

(d) Verify that gauge reads 70 psi, ± 10 psi for all aircraft except those equipped with Altitude Compensating Regulators. For these aircraft (models AT001-018, TC201-211EE) verify gauge reads 4.5 psi, ± 1.0 psi.

(e) For aircraft equipped with toggle valves:

1 For aircraft equipped with Altitude Compensating Regulators (AT001-018, TC201-211EE), turn toggle valve OFF. Verify that trapped pressure does not decay more than 1.0 psi in five (5) minutes.

2 For aircraft not equipped with Altitude Compensating Regulators, turn toggle valve OFF. Verify pressure does not decay more than 5.0 psi in five (5) minutes.

(f) For aircraft without toggle valves:

1 Turn oxygen regulator shutoff valve OFF. Verify that trapped pressure does not decay more than 5.0 psi in five (5) minutes.

(g) Turn regulator shutoff valve ON.

(h) Turn toggle valve OFF (if applicable).

(i) Check all accessible fittings in the supply line to the pilot and copilot outlets, cylinder filler line, oxygen pressure gauge, toggle valve, and cylinder regulator with leak detector fluid.

(j) Turn toggle valve ON (if present) and regulator shutoff valve OFF.

(k) Bleed off system pressure until test gauge reads zero (0) by inserting a passenger mask fitting in an outlet.

(l) Remove the mask fitting and observe the test gauge for five (5) minutes and watch for pressure buildup. Any pressure buildup indicates a leaking shutoff valve and is unacceptable.

(3) Leak Correction:

(a) If test results indicate leakage in excess of the given tolerance check the unions and connections, using Sherlock Leak Detector. If leaks are found bleed off the oxygen pressure for that part of the system to zero (0) and repair the leaks. Repeat the leak test and repair procedures until all leaks are corrected.

C. Leak Check – Oxygen Cylinder and Related High Pressure Components

NOTE: Equipment needed is one thermometer, calibrated in degrees F, type 21543 - Curtin Matheson Scientific, P.O.Box 1546, Houston, TX 77001 or equivalent.

(1) Observe safety precautions at start of this section.

WARNING: BLEED OFF OXYGEN AS NECESSARY TO PREVENT PRESSURE GAUGE IN COCKPIT FROM INDICATING FULL SCALE (2,000 PSI) OR ABOVE. WHEN PRESSURE IS SO HIGH THAT IT CANNOT REASONABLY BE DETERMINED, DANGEROUS CONDITIONS COULD EXIST.

(2) Ensure the oxygen cylinder is serviced to at least 1,700 psi.

(3) Tape thermometer to oxygen cylinder with masking tape.

(4) Turn pressure regulator ON.

(5) Turn toggle valve OFF (not applicable T201-248).

(6) Allow temperature to stabilize. Tap cockpit pressure gauge, and read as accurately as possible.

(7) Log temperature, pressure, and time.

(8) Tag regulator, pilot's and copilot's outlets and toggle valve with "DO NOT DISTURB, TEST IN PROGRESS" placard.

(9) Repeat temperature and pressure readings in approximately 48 hours.

(10) If the two temperatures are the same, the second pressure reading should not be less than the first by more than one percent. If the temperatures are different they should be corrected to a common reference temperature using Figure 201. Plot the first temperature and pressure on the chart. From the point draw a line parallel to the nearest line of constant volume to intersect the 70°F line.

(11) Repeat this procedure for the second temperature and pressure reading. The pressure at the 70°F line for the second reading should not be less than the first by more than one percent.

(12) If test results indicate leakage in excess of the given tolerance, check the unions and connections, using Sherlock Leak Detector. If leaks are found, bleed off the oxygen pressure for that part of the system to zero (0) and repair the leaks. Repeat the leak test and repair procedure until no leaks remain.

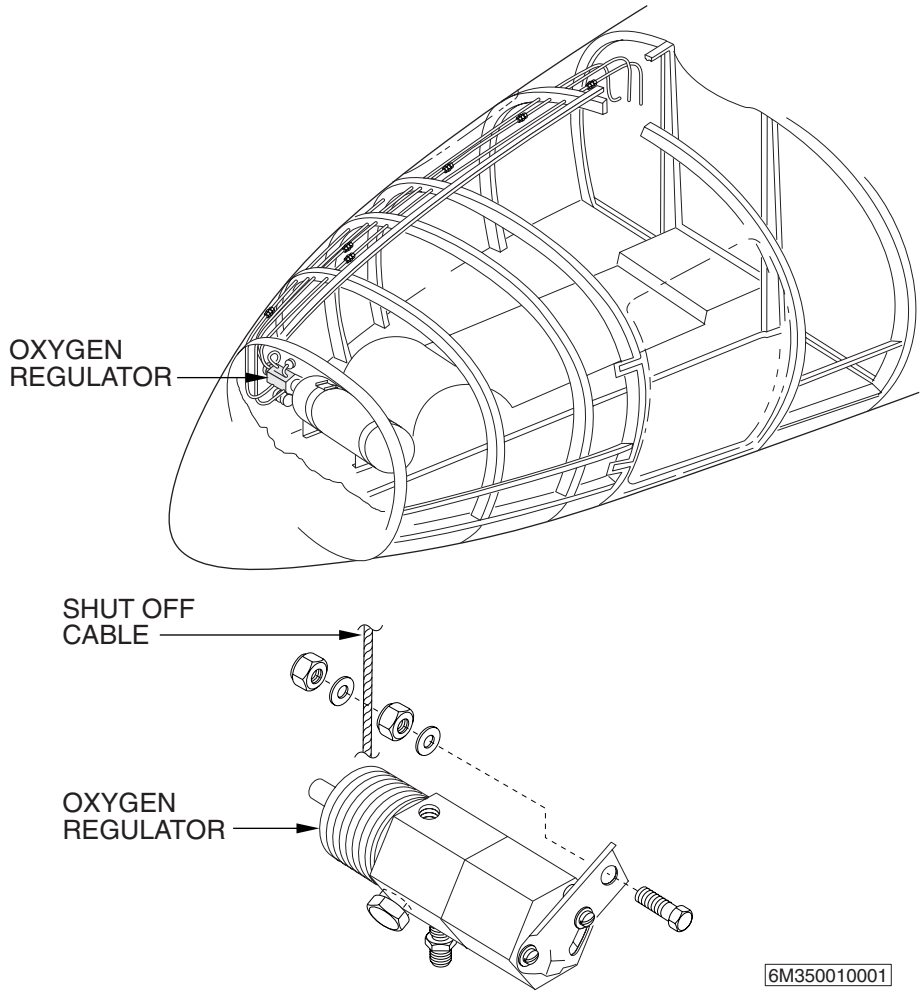
OXYGEN PRESSURE REGULATOR – DESCRIPTION AND OPERATION

1. General

A. The pressure regulator is mounted directly to the oxygen storage cylinder in the nose compartment. The regulator is factory set to maintain 70 psi, ± 10 psi of oxygen at a minimum flow rate of 210 liters per minute normal tank pressure discharge (LPM NTPD). The regulator incorporates a mechanical shutoff valve, servicing line (fill) port, supply line (out) port, and pressure gauge port.

2. Oxygen Pressure Regulator – Models T201-248 (Figure 1)

A. On models T201-248 only, the regulator shutoff valve is equipped with a push-pull cable control mounted on the lower cockpit control pedestal.

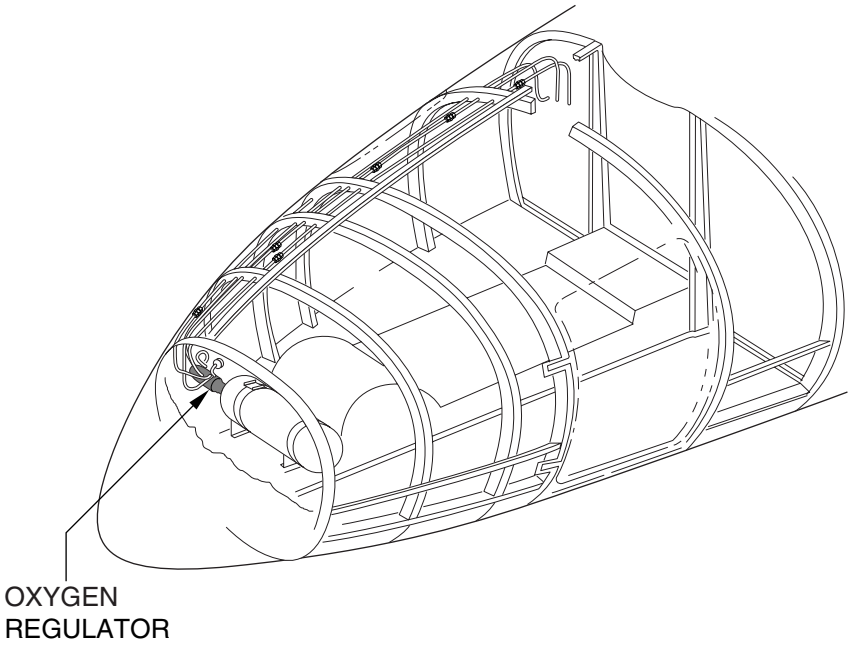


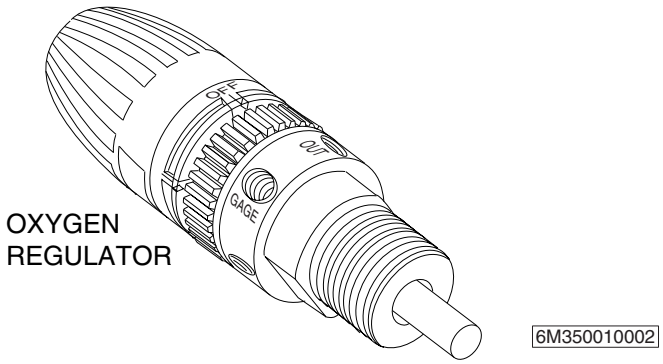
Oxygen Pressure Regulator
Models T201-248
Figure 1

3. Oxygen Pressure Regulator – All Other Models (Figure 2)

A. An overboard dump port to relieve excess pressure from over service or thermal expansion is installed on models T249-999, AT001-999 and TC201-999. The dump valve activates at 2,700 to 3,000 psi (2,500 to 2,700 psi (CC 3HT cylinder only) rupturing the green safety disc and discharging the oxygen overboard. The oxygen exits on the right side of the fuselage; just aft of the radome, forcing the green indicator disc and snap ring overboard.

B. If the green indicator disc is missing, the system has been overpressurized. If pressure gauge in cockpit indicates empty; replace ruptured metal disc in regulator, service system and replace green indicator disc and snap ring.





Oxygen Pressure Regulator
Models T249-999, AT001-999, TC210-999
Figure 2

OXYGEN PRESSURE REGULATOR – MAINTENANCE PRACTICES

1. Servicing – Oxygen Pressure Regulator

A. No regularly scheduled maintenance is required on the oxygen regulator. If bottle has been completely depressurized, be sure it is purged of moisture before recharge.

NOTE: The regulator is threaded into the cylinder; therefore, the regulator and cylinder should be sent to overhaul as one unit.

2. Removal/Installation – Oxygen Pressure Regulator

A. Removal – Oxygen Regulator

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(1) Turn shutoff valve OFF.

(2) Bleed pressure from the system supply line by inserting an oxygen mask fitting into an outlet and inhaling through the mask.

(2) Disconnect supply line at regulator port marked OUT.

(3) Connect bleed hose to regulator supply line port (marked OUT).

WARNING: WHEN OXYGEN IS BEING REMOVED FROM THE CYLINDER DO NOT ALLOW OPEN END OF BLEED HOSE TO BE DIRECTED TOWARD ANY PERSON OR ANY AREA CONTAINING OIL OR GREASE.

(4) Turn regulator shutoff valve ON, and Bleed off cylinder pressure.

(5) Disconnect all line fittings from regulator. Loosen cockpit pressure gauge and servicing lines slowly as there may be some trapped pressure in these lines.

NOTE: Before removing servicing lines It is advisable to insert a bleed fitting into the filler valve to bleed off trapped pressure.

(6) Disconnect cable from shutoff valve lever (only on aircraft models T201-248).

(7) Cap open lines to prevent contamination.

(8) Unscrew regulator from cylinder.

NOTE: If it becomes necessary to remove the shutoff valve lever (only on models T201-248), care must be exercised when installing the lever on the valve to prevent valve leakage:

B. Installation – Oxygen Regulator

(1) Apply MIL-T-5542 thread sealing compound to all male-tapered pipe threads.

(2) Screw regulator fitting into cylinder.

(3) Connect all line fittings to regulator ports.

NOTE: When an oxygen cylinder has been completely depressurized, condensation may form in the cylinder. The cylinder should be purged of any condensation prior to servicing.

(4) Install shutoff valve lever (only on models T201-248).

NOTE: Care must be exercised when installing the lever on the valve to prevent valve leakage:

CAUTION: DO NOT APPLY LOCKTITE ON SURFACE OR BEARING AREA OF REGULATOR OR ON THE CONTROL ARM.

(a) Apply Locktite Screwlock (No. 59-21) to screw threads.

(b) Install a spacer and washer on each screw.

(c) Torque screws 12 to 15 inch-pounds (1.36 to 1.69 N·m).

(d) Secure with safety wire.

NOTE: If other maintenance is required on the valve, it should be sent to a repair agency while still installed on the cylinder.

(5) Connect cable to shutoff valve lever (only on aircraft models T201-248, see 35-00-35 for adjustment procedure).

(6) Turn regulator shutoff valve ON and service the oxygen cylinder (pure breathing oxygen only).

(7) Perform leak check.

3. Inspection/Check – Oxygen Pressure Regulator

A. Leak Check – Oxygen Pressure Regulator

(1) Check pressure gauge for full cylinder pressure.

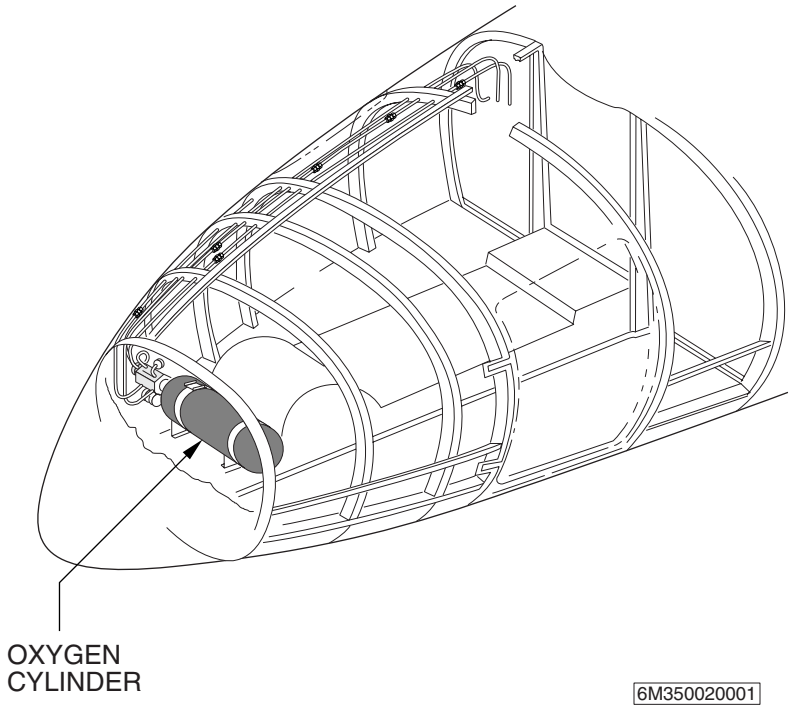
(2) Bubble check all connections for leaks, using oxygen test soap solution. Wipe all connections dry after check.

(3) Plug in pilot or copilot's mask fitting; check mask for 100% setting, breathe through mask, and check oxygen flow.

OXYGEN CYLINDER – DESCRIPTION AND OPERATION

1. General

A. The oxygen storage cylinder is standard equipment on all SA226 aircraft; the size varies with aircraft and customer choice. A 22, 49, or 115 cubic foot cylinder is located in the nose compartment mounted horizontally forward of the nose wheel well. The cylinder stores high pressure gaseous oxygen for use in the event of pressurization loss at altitude, smoke in cockpit or passenger cabin, or for medical purposes.



Oxygen Cylinder
Figure 1

OXYGEN CYLINDER – MAINTENANCE PRACTICES

1. Removal/Installation – Oxygen Cylinder

A. Removal – Oxygen Cylinder

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(1) Turn shutoff valve OFF.

NOTE: Oxygen cylinders with Puritan ZEP Regulators have check valves installed in the regulator ports, and may be removed without bleeding off cylinder pressure.

(2) Disconnect supply line from regulator port.

(3) Connect bleed hose to regulator supply line port.

(4) Bleed off cylinder pressure.

(5) Disconnect all line fittings from regulator and cap open lines to prevent contamination.

(6) On models T201-248 disconnect cable from shutoff valve lever.

WARNING: THE OXYGEN STORAGE CYLINDER IS A HIGH PRESSURE CONTAINER. EXERCISE CARE NOT TO DROP THE CYLINDER OR STRIKE THE VALVE ON ANYTHING. IF THE VALVE SHOULD BREAK OFF WITH THE CYLINDER PRESSURIZED THE CYLINDER MAY BECOME AN UNGUIDED MISSILE.

(7) Disconnect the two cylinder clamps and remove cylinder from aircraft.

B. Installation – Oxygen Cylinder

WARNING: THE OXYGEN STORAGE CYLINDER IS A HIGH PRESSURE CONTAINER. EXERCISE CARE NOT TO DROP THE CYLINDER OR STRIKE THE VALVE ON ANYTHING. IF THE VALVE SHOULD BREAK OFF WITH THE CYLINDER PRESSURIZED THE CYLINDER MAY BECOME AN UNGUIDED MISSILE.

(1) Position oxygen cylinder in mount.

(2) Apply MIL-T-5542B Thread compound sparingly to all male-tapered pipe threads.

(3) Connect all line fittings to the regulator ports.

(4) Torque clamp nuts 15 to 20 inch-pounds (1.69 to 2.26 N·m).

(5) Open shutoff valve and service cylinder.

2. Adjustment/Test – Oxygen Cylinder

A. Hydrostatic Test –- Oxygen Cylinder

NOTE: A hydrostatic (pressure) test is required every five years for cylinders bearing an ICC 3AA designation, and every three years for those bearing an ICC 3HT designation. Cylinders not having these designations or those not having been inspected within the time interval should not be filled.

(1) Remove the cylinder IAW the removal instructions.

(2) Send the cylinder to an authorized agency.

(3) When cylinder is returned from service, install it IAW installation instructions.

(4) If the cylinder is returned unserviced, service IAW servicing instructions.

(5) Turn the shutoff valve ON and check for normal operation.

(6) Bubble check the line fittings that were installed into the regulator using oxygen test soap solution. Wipe all connections dry after check.

3. Inspection/Check – Oxygen Cylinder

A. Leak Check – Oxygen Cylinder

(1) Check pressure gauge for full cylinder pressure.

(2) Bubble check all connections for leaks using an oxygen test soap solution. Wipe all connections dry after check.

(3) Plug in pilot or copilot's oxygen mask fitting; check mask for 100% setting, breathe through mask, and check oxygen flow.

OXYGEN LINES AND FITTINGS – DESCRIPTION AND OPERATION

1. General

A A network of aluminum, copper, and plastic tubing make up the ducting of the oxygen system carrying oxygen from the cylinder to various system components.

OXYGEN LINES AND FITTINGS – MAINTENANCE PRACTICES

1. Removal/Installation – Oxygen Line and Fittings

A. Removal – Oxygen Lines and Fittings

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(1) Turn shutoff valve OFF or deplete cylinder pressure.

(2) Bleed residue pressure from lines.

(3) To remove individual tubing, disconnect each end, loosen any clamps and tie straps supporting tubing, and remove tubing.

(4) Cap open lines to prevent contamination.

B. Installation – Oxygen Lines and Fittings

CAUTION: ENSURE ALL LINES AND FITTINGS ARE CLEAN AND FREE OF CONTAMINATION PRIOR TO INSTALLATION. LINES MAY BE FLUSHED WITH TRICHLOROETHYLENE, DRAINED AND THOROUGHLY DRIED.

INSTALL OXYGEN LINES ABOVE FUEL, OIL, AND HYDRAULIC LINES WHENEVER POSSIBLE. EXERCISE CAUTION TO ENSURE THE OXYGEN SYSTEM WILL NOT BECOME CONTAMINATED BY LEAKING FUEL, OIL, AND HYDRAULIC FLUID. USE DEFLECTOR PLATES IF NECESSARY.

PROTECT ALL TUBING FROM CHAFING AT EACH CUT OUT BY GROMMETS OR CLAMPS, AND PROVIDE SUFFICIENT CLEARANCE BETWEEN LINE AND PROJECTIONS SUCH AS BOLTS OR NUTS TO PREVENT CONTACT DUE TO VIBRATION.

MAINTAIN AT LEAST TWO INCHES OF CLEARANCE BETWEEN THE OXYGEN SYSTEM TUBING AND CONTROL CABLES AND OTHER MOVING PARTS. IF TWO INCHES CANNOT BE PROVIDED USE SUITABLE INSULATION TO PREVENT POSSIBLE DAMAGE.

(1) Install oxygen tubing nylo-seal and poly-flo tubing so that each separate piece of tubing will have at least 0.50 inch (1.27 cm) of slack per foot to allow for expansion and contraction.

CAUTION: ENSURE NO THREAD COMPOUND GETS IN THE BORE OF THE FITTING, OR ON THE FITTING END OR CONE SURFACE. NO COMPOUND SHOULD BE ALLOWED ON THE TUBING INSIDE FLARE SURFACE, INTERIOR OF THE TUBE, OR INSIDE ANY SYSTEM COMPONENTS. ALWAYS USE THREAD COMPOUND SPARINGLY.

(2) To prevent thread galling and seizure of straight threaded fittings, apply MIL-T-5542B thread sealing compound to the male threads of fittings as follows:

(a) Align tubing and press flare firmly against fitting end.

(b) Tighten nut one-half (1/2) to one (1) turn to ensure compound will not get on first one (1) to one and one-half (1 1/2) threads, then sparingly apply MIL-T-5542B thread sealing compound to male threads.

(3) Procedure for installing and tightening fittings is as follows:

NOTE: Nylo-Seal and Swagelock tube fittings are received completely assembled with the nut finger tight. Do not disassemble before use. Disassembly may result in contamination of the fitting.

(a) Insert tubing into fitting. Make sure tubing rests firmly on shoulder of fitting and nut is finger-tight.

(b) Scribe nut at 6 o'clock position. This mark is used for reference when tightening nut.

(c) On Nylo-Seal fittings, tighten nut two (2) to two and one-half (2 1/2) turns.

(d) On Swagelock fittings, tighten nut one and one-quarter (1 1/4) turns.

(e) Nylo-Seal and Swagelock fittings may be disconnected and re-tightened. To re-tighten, tighten nuts original number of turns, then tighten an additional one-quarter (1/2) turn.

(f) Torque Aluminum alloy and copper fittings 100 to 125 inch-pounds (11.30 to 14.12 N·m).

2. Cleaning/Painting – Oxygen Line and Fittings

A. Cleaning – Oxygen Lines and Fittings

NOTE: When it is suspected that tubing contains foreign matter clean IAW the following procedures.

(1) Remove all tubing to be cleaned.

WARNING: UNDER NO CIRCUMSTANCES USE ANY CLEANING SOLUTION OR ANY OTHER SOLUTION INSIDE ANY PART OF THE OXYGEN SYSTEM WHILE IT IS INSTALLED, EXPLOSIONS AND/OR FIRES CAN RESULT.

(2) Flush the tubing with trichloroethylene.

(3) Drain the tubing and dry thoroughly with clean dry air.

NOTE: As an alternate method flush tubing with isopropyl alcohol, rinse with fresh clean water, and dry thoroughly with clean dry air.

CAUTION: PROTECT TUBING, FITTING, AND COMPONENTS NOT IMMEDIATELY INSTALLED FROM CONTAMINATION BY CAPS, OR PLACE IN A CLEAN POLYETHYLENE PACKAGE AND SEAL.

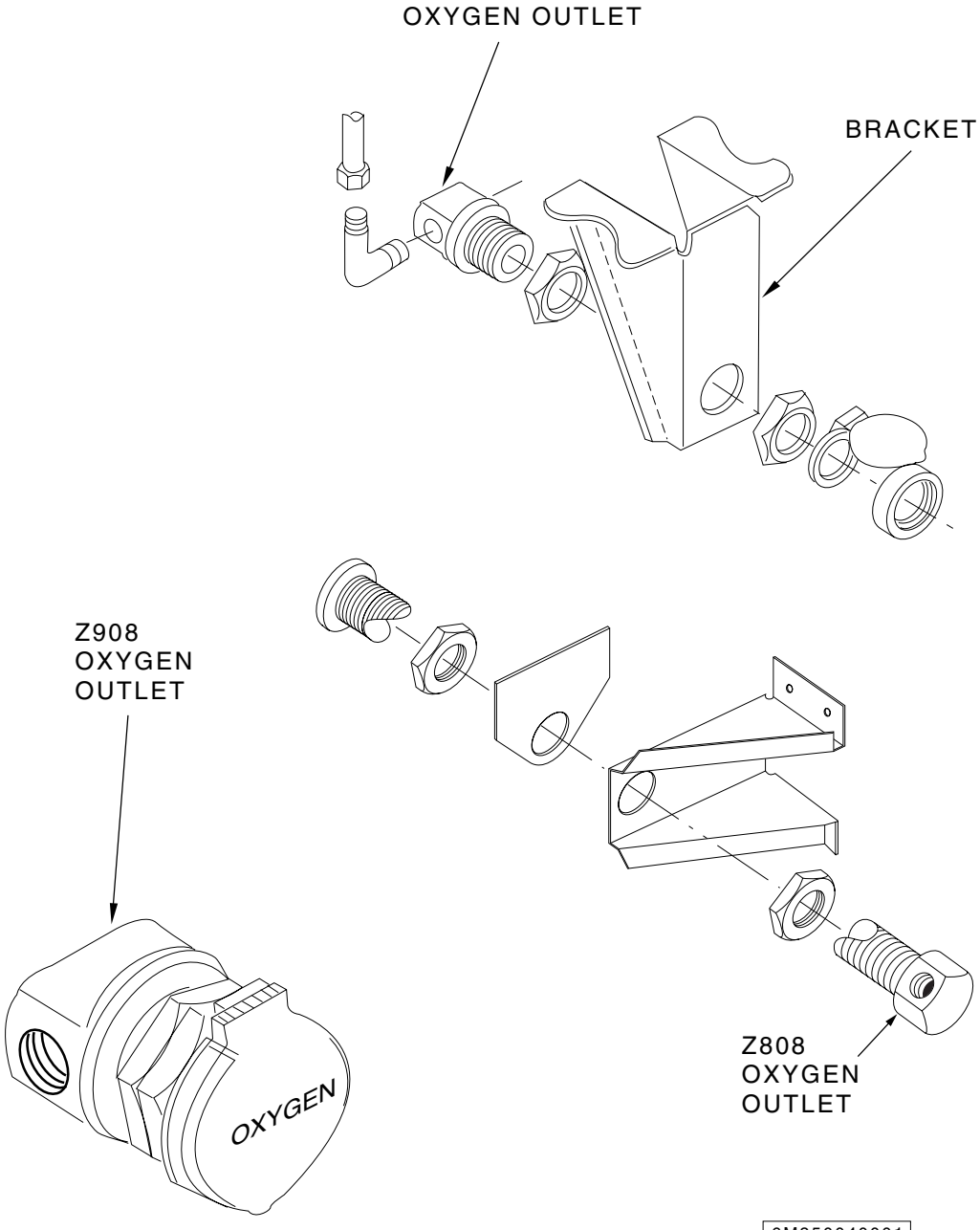
(4) Install tubing and bubble check line connections. After check wipe all connections dry.

OXYGEN OUTLETS – DESCRIPTION AND OPERATION

1. General

A. Standard plug-in type oxygen outlet receptacles are installed on SA226 aircraft located on the side wall adjacent to each seat location. They use a hinged dust cover that will lock when positioned open.

(1) A normally closed internal check valve is opened by plugging an oxygen mask bayonet fitting into the outlet.



Oxygen Outlets
Figure 1

2. Crew Oxygen Outlets

A. The flight crew is furnished permanent demand type oxygen masks and smoke goggles that are stored under the crew chair or in the seat back pockets.

(1) With the cylinder mounted shutoff valve on and the crew mask bayonet fitting inserted into an outlet the line to the mask is pressurized, but until the crew member breathes from the mask no oxygen flows.

(2) The crew mask is unplugged when not needed to preclude any flow should the demand mechanism develop a leak.

(3) Flow control is provided by a restricting orifice in the bayonet rather than in the outlet. The orifice in the crew mask provides twice the oxygen flow provided to the passenger masks. By having the orifice in the bayonet, the crew member receives additional flow regardless of the outlet to which the mask is connected.

3. Passenger Oxygen Outlets

A. Plug in type oxygen outlets are provided at every passenger seat location, including the lavatory. Passengers are provided disposable constant-flow masks that are stored in the seat back pockets.

(1) With the passenger oxygen ON, oxygen will flow any time one of the passenger mask bayonet fittings is inserted into an outlet. For this reason all masks should be unplugged when not in use

OXYGEN OUTLETS – MAINTENANCE PRACTICES

1. Removal/Installation – Oxygen Outlets

A. Removal – Oxygen Outlets

(1) Close the manual shutoff valve on the oxygen cylinder regulator.

(2) Bleed off any residual pressure in lines by plugging an oxygen mask fitting into an outlet.

(3) Gain access to outlets from behind the interior covering by first removing the outlet cover and placard, then by removing the necessary molding, etc.

(4) Loosen the jam nut on the outlet, remove oxygen lines, and withdraw the outlet from the mount bracket.

B. Installation – Oxygen Outlets

(1) Position the outlet in the mounting bracket, and position the back jam nut to allow just enough of the outlet to protrude through the bracket and interior covering, placard, and cover bracket so the cover will fit as flush as possible. Secure the outlet with the jam nuts and lockwasher.

(2) Connect and secure the oxygen lines.

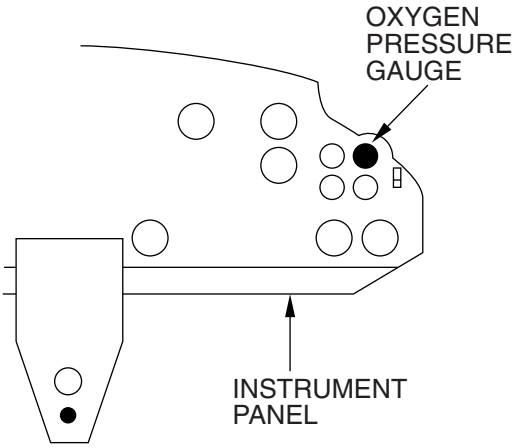
(3) Turn shutoff valve ON and bubble check connections for leaks using oxygen test soap solutions. Wipe dry after check.

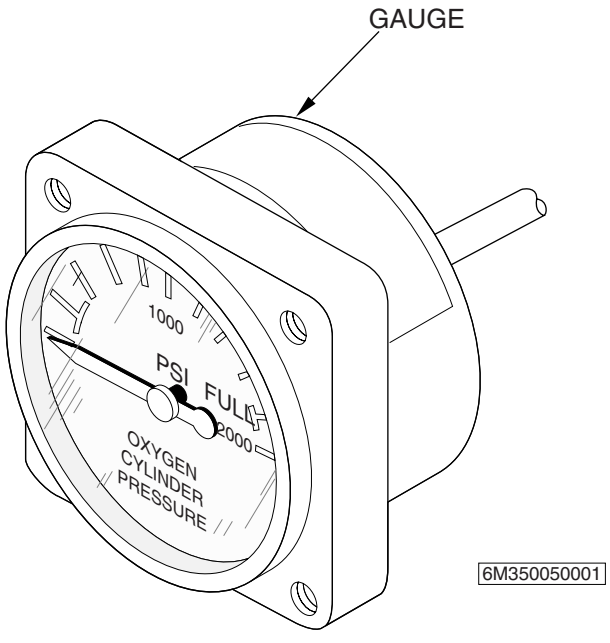
(4) Replace interior, placard, and outlet dust cover and secure.

OXYGEN PRESSURE GAUGE – DESCRIPTION AND OPERATION

1. General

A. An oxygen pressure gauge is mounted on the copilot's instrument panel. The gauge is calibrated from 0 to 2,000 psi and gives a continuous reading of cylinder pressure regardless of the shutoff valve position. The system is normally serviced 1,800 psi for T models and 1,850 psi for AT and TC for normal system operation.





Oxygen Pressure Gauge
Figure 1

OXYGEN PRESSURE GAUGE – MAINTENANCE PRACTICES

1. Servicing – Oxygen Pressure Gauge

A. No regularly scheduled maintenance is required on the oxygen gauge.

2. Removal/Installation – Oxygen Pressure Gauge

A. Removal – Oxygen Pressure Gauge

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN WHEN PERFORMING MAINTENANCE ON THE OXYGEN SYSTEM.

CAUTION: MODELS T201-248 WITH SCOTT PRESSURE REGULATORS INSTALLED MUST HAVE CYLINDER PRESSURE DEPLETED PRIOR TO REMOVING THE GAUGE. BLEED CYLINDER IAW 35-00-00.

(1) Turn shutoff valve OFF.

(2) Slowly loosen capillary line on back of gauge to allow any trapped pressure to evacuate.

(3) Remove and cap the capillary line.

(4) Remove screws securing gauge to copilot's instrument panel. Withdraw gauge from panel.

B. Installation – Oxygen Pressure Gauge

(1) Position the oxygen pressure gauge in the copilot's instrument panel, and secure with screws.

(2) Install the capillary line to the back of the gauge.

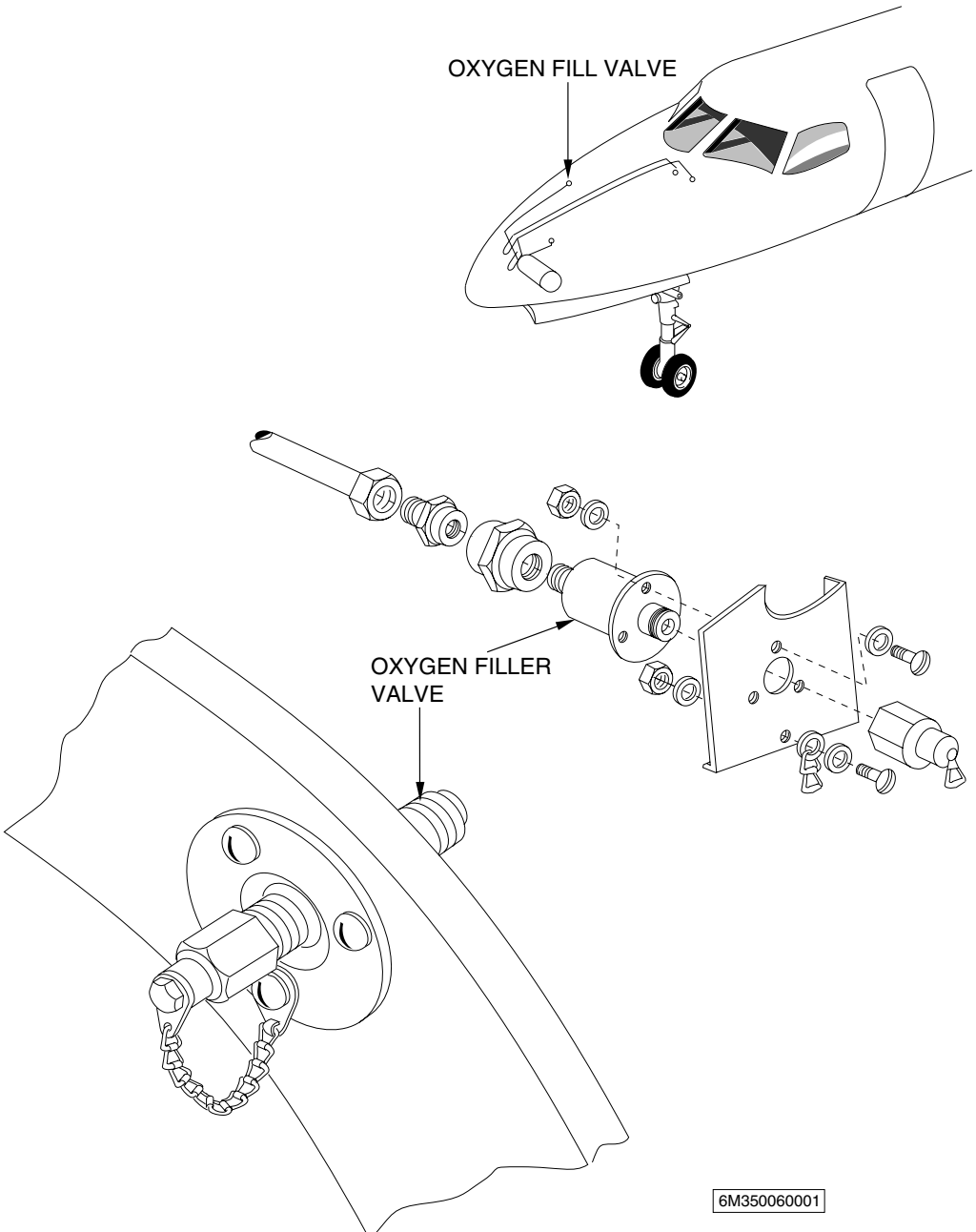
(3) Turn the shutoff valve ON. If required fill the cylinder IAW the servicing instructions.

(4) Bubble check capillary line fitting using oxygen test soap solution. Wipe dry after check

OXYGEN SERVICE CONNECTION – DESCRIPTION AND OPERATION

1. General

A. The oxygen service connection is located inside the nose baggage compartment upper right hand forward corner and is accessible through the right hand baggage door. The fill valve is designed for remote servicing of high pressure oxygen cylinders, and provided internally is a check valve, flow control restrictor, and integral filter. A sealing cap and chain is provided for protection against dust and contamination. The cap also serves as a secondary seal in the event of check valve leakage.



Oxygen Service Connection
Figure 1

OXYGEN SERVICE CONNECTION – MAINTENANCE PRACTICES

1. Servicing – Oxygen Service Connection

A. No regularly scheduled maintenance is required on the fill valve.

2. Removal/Installation – Oxygen Service Connection

A. Removal – Fill Valve

WARNING: OIL OR GREASE CONTAMINANTS WILL IGNITE UPON CONTACT WITH PURE OXYGEN. BE SURE EQUIPMENT, CLOTHING, AND HANDS ARE CLEAN PRIOR TO PERFORMING ANY MAINTENANCE ON THE OXYGEN SYSTEM.

(1) Turn shutoff valve OFF.

CAUTION: ON MODELS T201-248 WITH SCOTT PRESSURE REGULATORS INSTALLED THE SERVICE LINE WILL ALWAYS HAVE CYLINDER PRESSURE REGARDLESS OF SHUTOFF VALVE POSITION. BLEED CYLINDER PRESSURE (SEE SECTION 35-00-00, BLEEDING).

(2) Slowly remove service line from fill valve to allow trapped pressure to escape.

(3) Cap line to prevent contamination.

(4) Remove mounting screws.

(5) Remove fill valve from mount.

B. Installation – Fill Valve

(1) Position fill valve in mount.

(2) Secure with screws.

(3) Remove cap from service line and connect to the fill valve.

(4) Turn regulator shutoff valve on and service the oxygen cylinder.

3. Inspection/Check – Oxygen Service Connection

A. Leak Check

(1) Pressurize the service line with a servicing cylinder.

(2) Bubble check fitting for leaks using an oxygen test soap solution. Wipe dry after check.

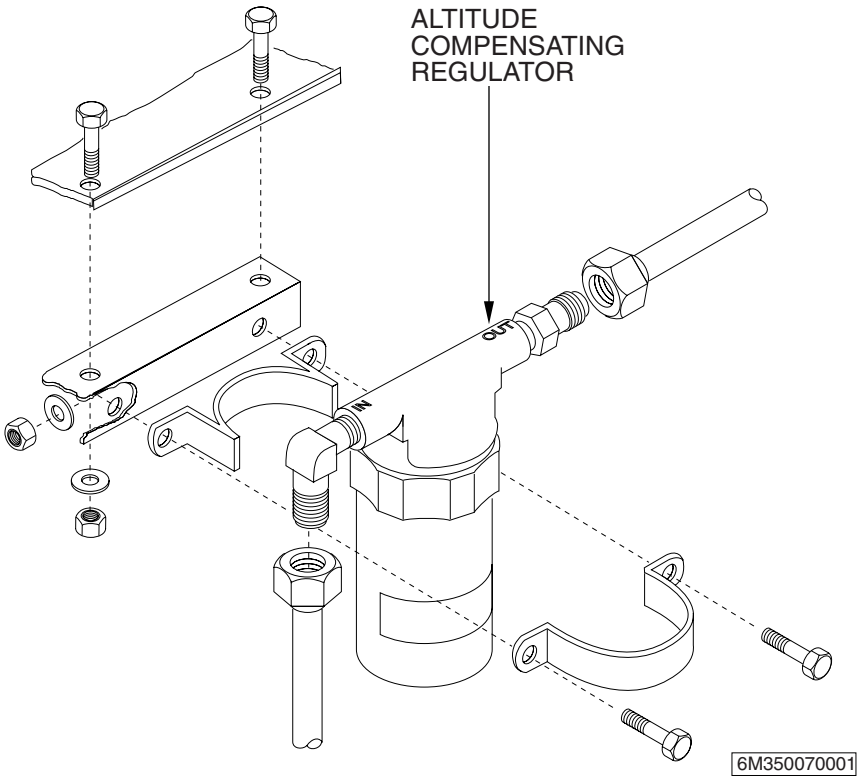
(3) Disconnect service line.

(4) Install cap.

ALTITUDE COMPENSATING REGULATOR – DESCRIPTION AND OPERATION

1. General

A. An altitude compensating regulator is installed in the passenger oxygen system to conserve and control the volume flow of oxygen, IAW altitude, to the passenger outlets. Oxygen is supplied to each passenger at a rate of 1.8 liters per minute (LPM) at 10,000 feet to 2.6 LPM at 31,000 feet. The regulator is mounted on the right hand side forward of the copilot's instrument panel, and has been deleted on later model aircraft.



Altitude Compensating Regulator
Figure 1

ALTITUDE COMPENSATING REGULATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Altitude Compensating Regulator

A. Removal – Regulator

(1) Turn the oxygen cylinder regulator shutoff valve OFF.

(2) Bleed off any residual line pressure by turning toggle valve ON and inserting a passenger mask fitting into one of the passenger outlets.

NOTE: Access to the altitude compensating regulator is under the copilot's instrument panel.

(3) Disconnect both oxygen lines (cap loose lines).

(4) Loosen the mounting clamp and remove the regulator.

B. Installation – Regulator

(1) Position the regulator in the mounting clamp and secure it.

(2) Connect both lines to the regulator.

(3) Turn the oxygen cylinder regulator shutoff valve ON.

(4) Plug in a passenger oxygen mask, and listen for a slight oxygen flow.

2. Inspection/Check – Altitude Compensating Regulator

A. Leak Check

(1) Turn shutoff valve ON.

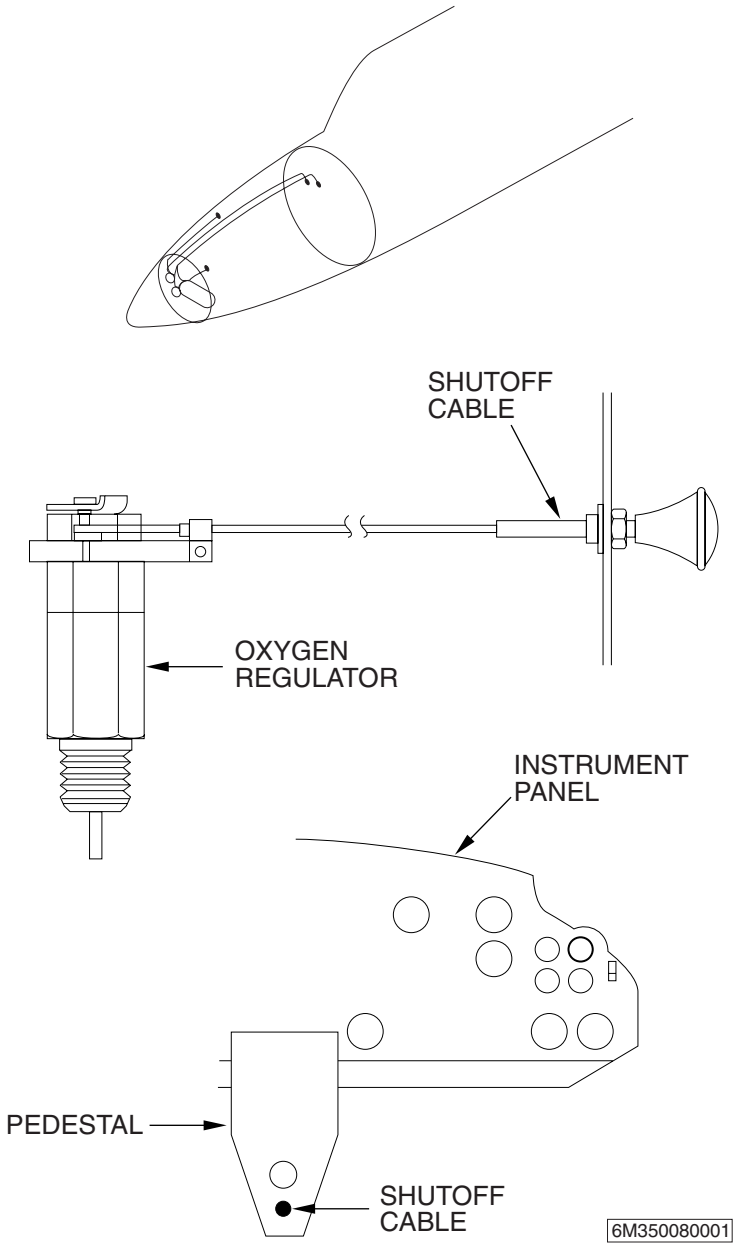
(2) Turn toggle valve ON.

(3) Bubble check connection for leaks using an oxygen test soap solution. Wipe dry after check.

MECHANICAL SHUTOFF CABLE – DESCRIPTION AND OPERATION

1. General

A. On T201-248 a mechanical pull (ON) push (OFF) cable is installed to operate the oxygen system. A knob located on the lower cockpit control pedestal is connected through a push-pull type cable to a lever operated shutoff valve on the oxygen cylinder in the nose compartment. The cable housing is constructed of wrapped steel encased in plastic and should not wear out or need replacement. The cable itself likewise should not need to be replaced; however, the following instructions describe how it can be removed and installed.



Mechanical Shutoff Cable
Figure 1

MECHANICAL SHUTOFF CABLE – MAINTENANCE PRACTICES

1. Removal/Installation – Mechanical Shutoff Cable

A. Removal – Mechanical Shutoff Cable

(1) Disconnect the cable from the shutoff lever on the oxygen cylinder in the nose.

(2) Pull the cable out from the cockpit end.

B. Installation – Mechanical Shutoff Cable

(1) From the cockpit end insert the cable and push it through the housing until the end emerges in the nose adjacent to the shutoff lever.

(2) Insert the cable through the hole in the cable mount bolt.

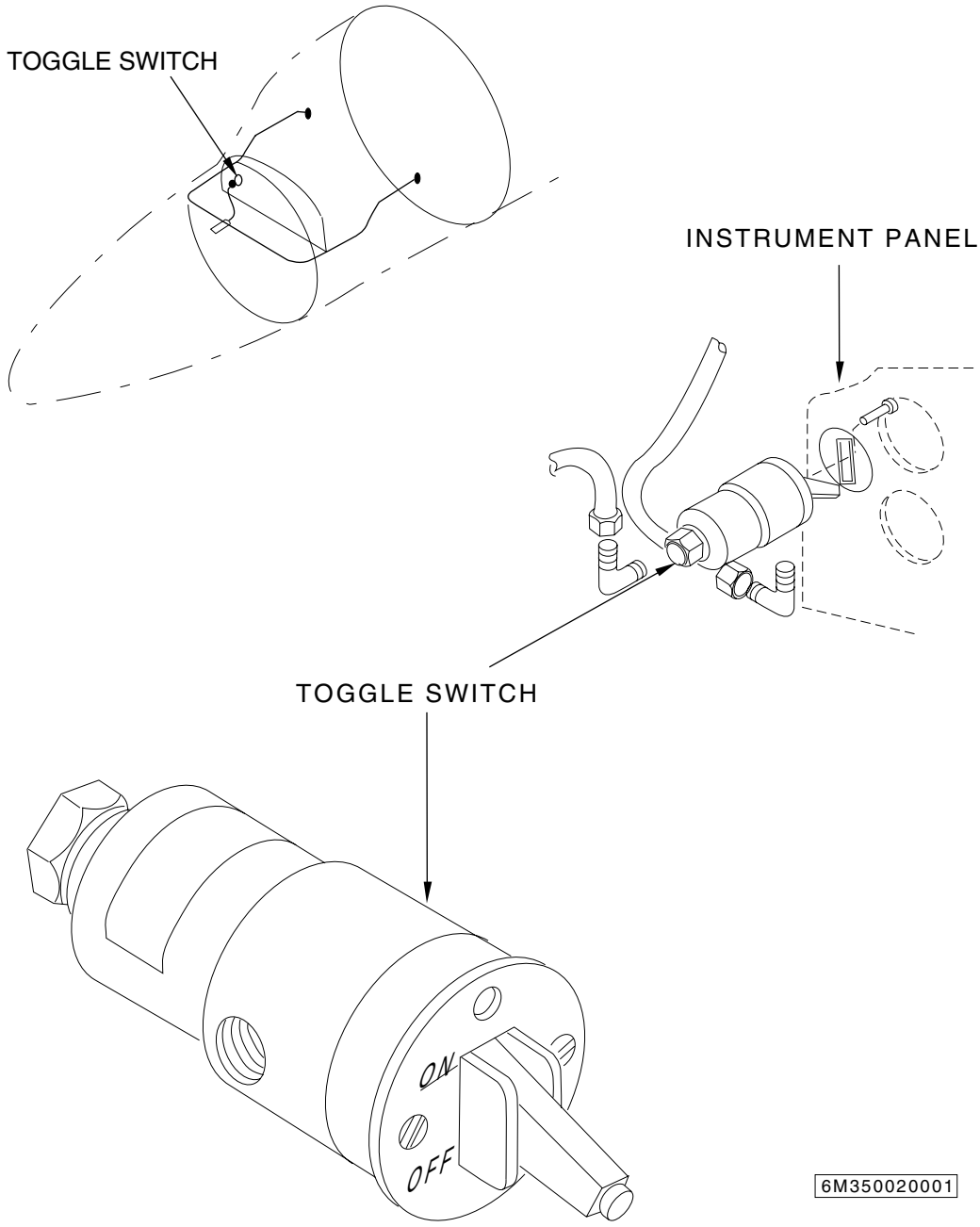
(3) With the shutoff valve lever in the off position push the cable all the way in and tighten the cable mount bolt.

(4) Check for correct travel (full ON to full OFF).

TOGGLE VALVE – DESCRIPTION AND OPERATION

1. General

A. A manually operated toggle switch type valve, mounted on the copilot's instrument panel adjacent to the oxygen pressure gauge, is used to turn ON or OFF the oxygen supply to the passenger outlets. It is placed between the crew passenger "T" off lines and the passenger outlets. The valve uses a soft elastomer seal for bubble free leakage.



Toggle Valve
Figure 1

TOGGLE VALVE – MAINTENANCE PRACTICES

1. Servicing – Toggle Valve

A. No regularly scheduled maintenance is required on the toggle valve.

2. Removal/Installation – Toggle Valve

A. Removal – Toggle Valve

(1) Turn regulator shutoff valve OFF.

(2) Bleed off any residual line pressure by selecting toggle valve to ON, and inserting a passenger mask fitting into one of the passenger outlets, and letting the pressure evacuate.

(3) Disconnect and cap both lines to the toggle valve.

(4) Remove the screws and valve.

B. Installation – Toggle Valve

(1) Position toggle valve in instrument panel so that up is ON and secure it with screws.

(2) Connect and secure both lines to the valve.

(3) Turn regulator shutoff valve ON.

3. Inspection/Check – Toggle Valve

A. Leak Check

(1) Bubble check all connections for leaks using an oxygen test soap solution. Wipe all connections dry after check.

(2) Turn the toggle valve ON, insert a mask fitting into a passenger outlet, and check for a slight oxygen flow.

MAINTENANCE MANUAL

CHAPTER 37 – VACUUM

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37-CONTENTS	-	2	JUN 01/02	37-20-00	MP	201	JUN 01/02
37-00-00	DO	1	JUN 01/02	37-20-00	MP	202	JUN 01/02
37-00-00	DO	2	JUN 01/02	37-20-00	MP	203	JUN 01/02
37-10-10	DO	1	JUN 01/02	37-20-10	DO	1	JUN 01/02
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MAINTENANCE MANUAL

CHAPTER 37 – VACUUM

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CHAPTER 37 – VACUUM

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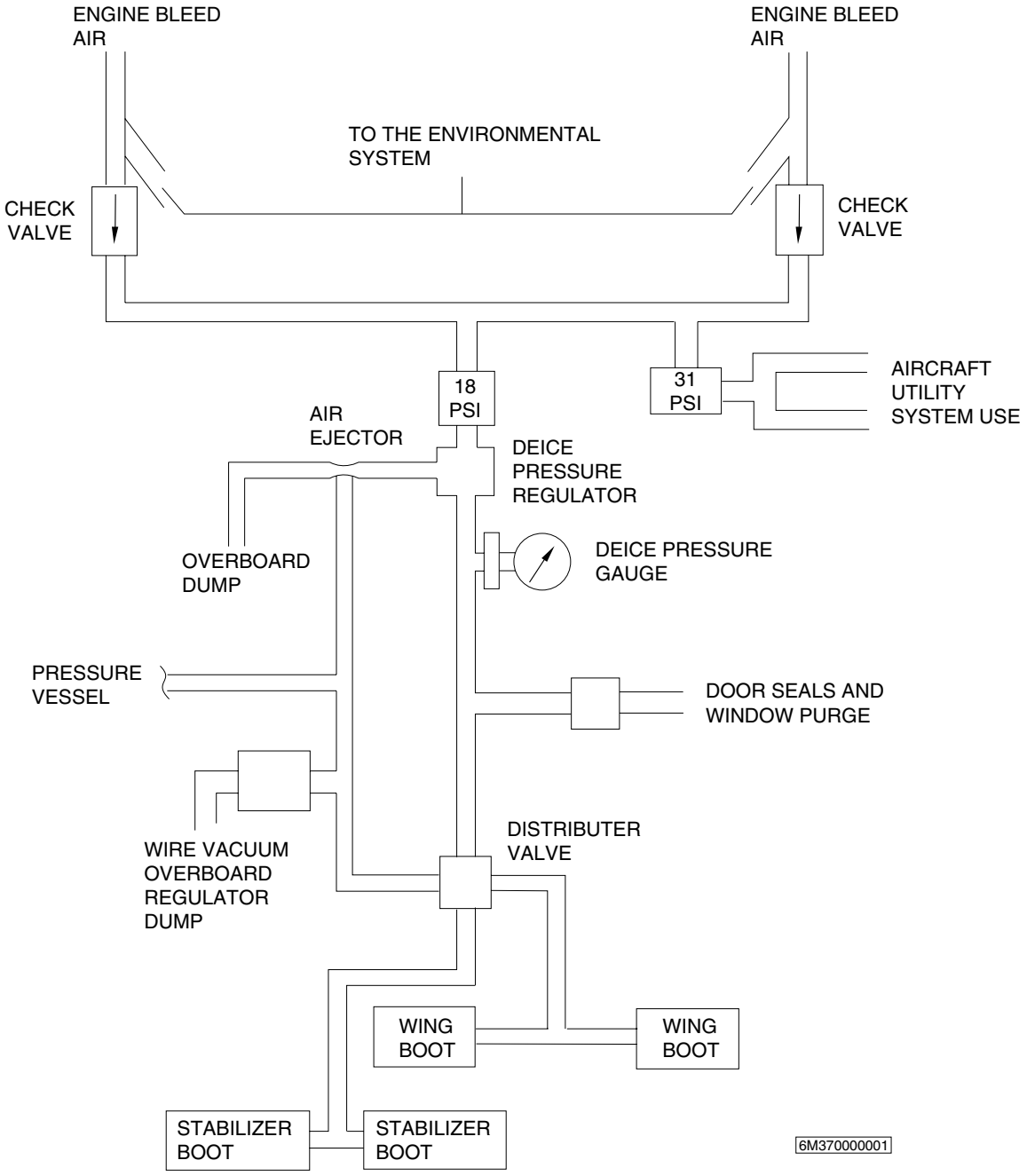
VACUUM – DESCRIPTION AND OPERATION

1. General

A. A vacuum system is installed on all SA226 series aircraft to provide vacuum for the airfoil deicer boot system and the cabin pressure control system. The principal components of the vacuum system are: the air ejector, the vacuum regulator, vacuum gauge, vacuum warning switch, distributor valve, and necessary ducting and tubing.

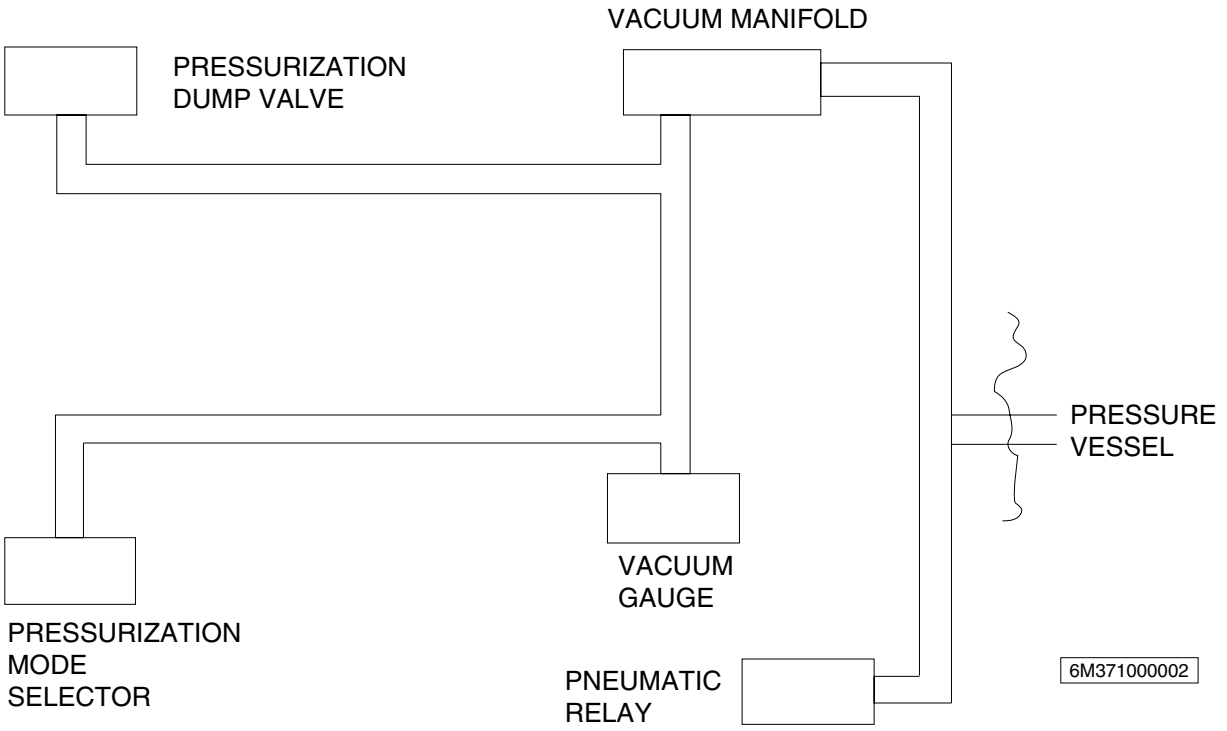


Vacuum
Figure 1

VACUUM – DESCRIPTION AND OPERATION

B. High pressure bleed air is taken from each engine plenum case and directed through check valves to a regulator valve. The regulator valve reduces and regulates the bleed air to 18 psi, and directs the regulated air to a TEE where the air stream divides. One air stream flows to the distributor valve to be available for air foil boot inflation and the other air stream flows to the air ejector where vacuum is created by venturi action. The vacuum thus created by the air ejector is ducted to a vacuum regulator where it is regulated to 4.5 hg and sent to the distributor valve to be available for airfoil boot deflation.

C. Vacuum required by the cabin pressurization system is ducted through a tee at the suction gauge.

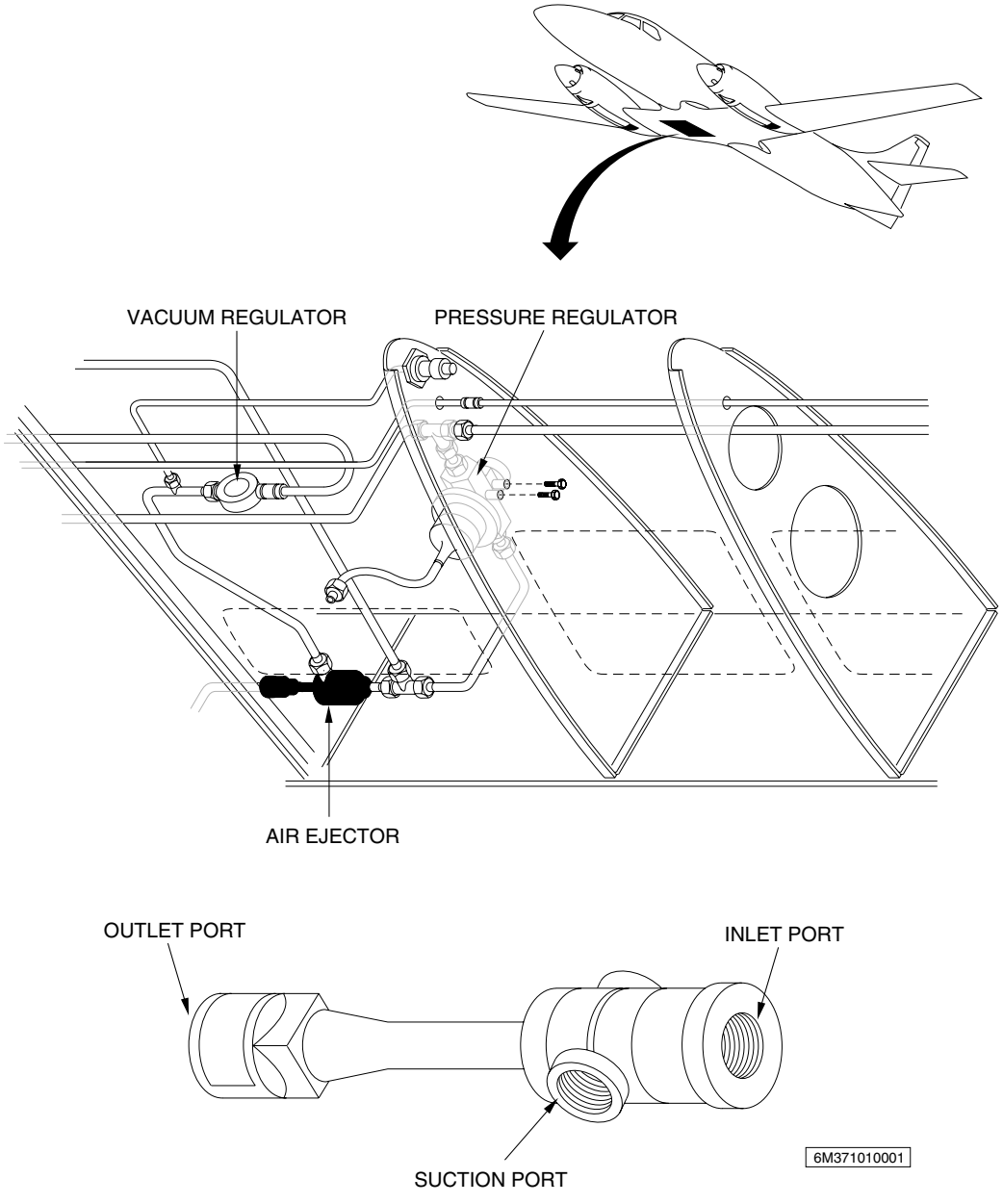


Vacuum Fuselage
Figure 2

AIR EJECTOR – DESCRIPTION AND OPERATION

1. General

A. The air ejector, located in the forward center wing section of the aircraft, is used to generate a vacuum source for hold down of the deicer boots and as required by the cabin pressurization system. The air ejector contains an inlet port, an outlet port, and a suction port, see Figure 1. Regulated 18 psi air is received through plumbing from the pressure regulator. The regulated low pressure air enters the air ejector inlet port, exits the outlet port, and is vented overboard. The air ejector produces a vacuum by venturi action. The vacuum passes through the suction port of the air ejector and is sent by plumbing to the distributor valve where it is available for deflation of the airfoil deice boots.



Air Ejector
Figure 1

AIR EJECTOR – MAINTENANCE PRACTICES

1. Removal/Installation– Air Ejector

A. Removal – Air Ejector

(1) Remove left leading edge wing to fuselage fairing.

(2) Remove lower left forward center wing section access panel.

(3) Disconnect tube assemblies from air ejector.

(4) Remove air ejector.

B. Installation – Air Ejector

(1) Install tube assemblies to air ejector. Tighten B-nuts as follows:

(a) Torque inlet and outlet end fittings 300 to 500 inch-pounds (33.90 to 56.49 N·m).

(b) Torque suction port end fittings 200 to 350 inch-pounds (22.60 to 39.55 N·m).

(2) Install lower left forward center wing section access panel.

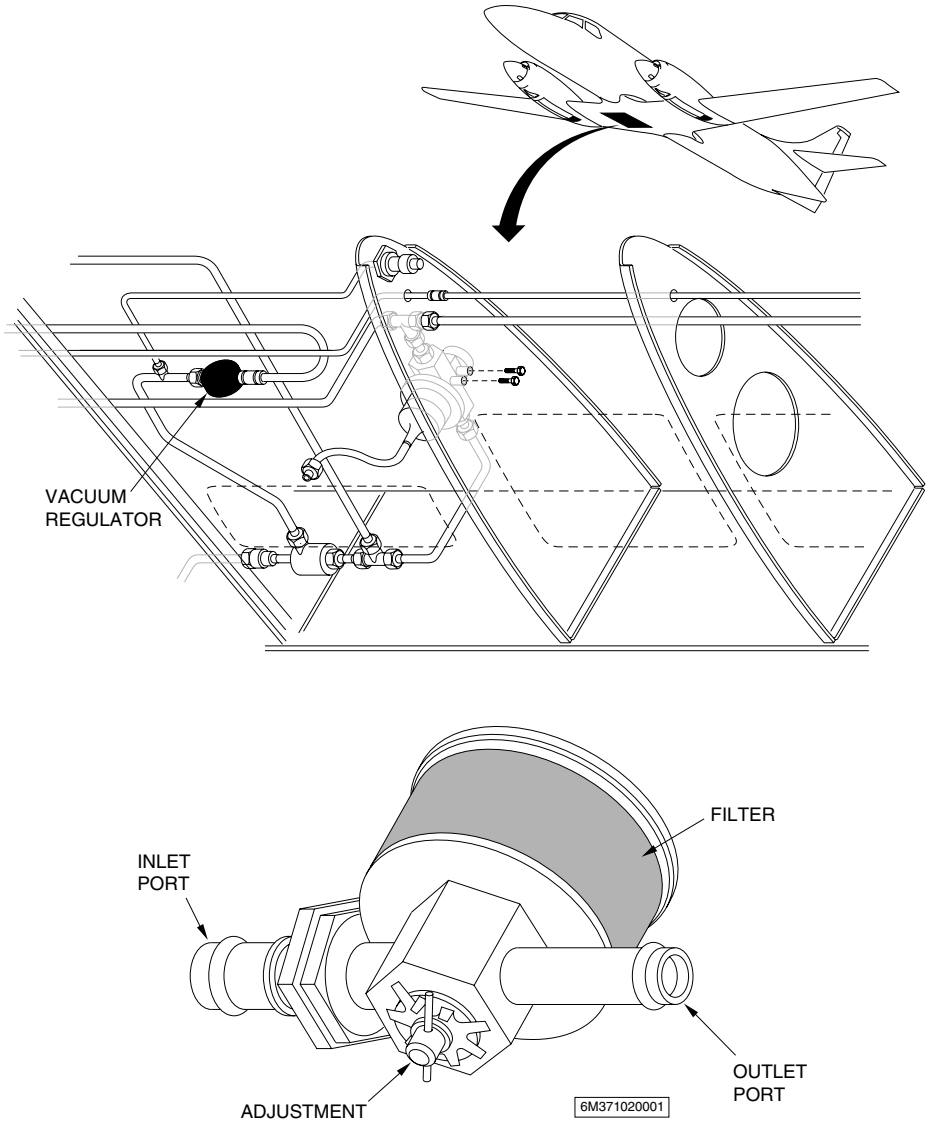
(3) Install left leading edge wing-to-fuselage fairing.

VACUUM REGULATOR – DESCRIPTION AND OPERATION

1. General

A. The vacuum regulator, located in the forward center wing section maintains vacuum pressure necessary for proper operation of the deice system. The vacuum regulator contains an inlet port, an outlet port, and a foam covered inlet valve. The inlet port of the vacuum regulator is routed through plumbing to the air ejector suction, (see Figure 1). The outlet port of the vacuum regulator is routed through plumbing to the distributor vacuum inlet. The components of the vacuum regulator are: a diaphragm, tension spring, foam filter, and adjustment screw.

B. The tension spring is set by the adjustment screw to maintain the required force on the diaphragm necessary to allow vacuum pressure to enter and exit the regulator and distributor valve. Should a higher vacuum occur in the system, an additional force is exerted on the diaphragm sufficient to allow outside air to be admitted through the foam filter. When the outside air is admitted, the vacuum pressure within the deice system is reduced to the correct 4.5 hg level



Vacuum Regulator
Figure 1

VACUUM REGULATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Vacuum Regulator

A. Removal – Vacuum Regulator

(1) Remove left forward wing to fuselage fairing.

(1) Remove left forward center wing section access panel.

(2) Disconnect hose clamps on both hoses to the vacuum regulator.

(3) Remove nut holding vacuum regulator to mounting bracket.

(4) Remove regulator.

B. Installation – Vacuum Regulator

(1) Position vacuum regulator in mounting bracket and secure with nut.

(2) Connect hose clamps on both hoses to vacuum regulator.

(3) Install left forward center wing section access panel.

(4) Install left forward wing to fuselage fairing.

C. Removal – Vacuum Regulator Filter

NOTE: The foam filter may be easily removed for cleaning or replacement with a B 3-5-1 replaceable filter from Airborne Mfg. Co., 711 Taylor St., Elyria, Ohio 4403

(1) Remove flexible foam filter by hand so as not to damage or scar regulator.

D. Installation – Vacuum Regulator Filter

(1) Position flexible foam filter over regulator, ensure that filter is placed all the way down so as not to leave space between filter and regulator.

2. Adjustment/Test – Vacuum Regulator

A. If necessary, the adjustment screw can be rotated clockwise to increase the suction of the regulator. Generally, no adjustment to the vacuum regulator is required after initial installation. If the vacuum reading changes, the most probable cause is a defective regulator, leak in a line, or a defective gauge.

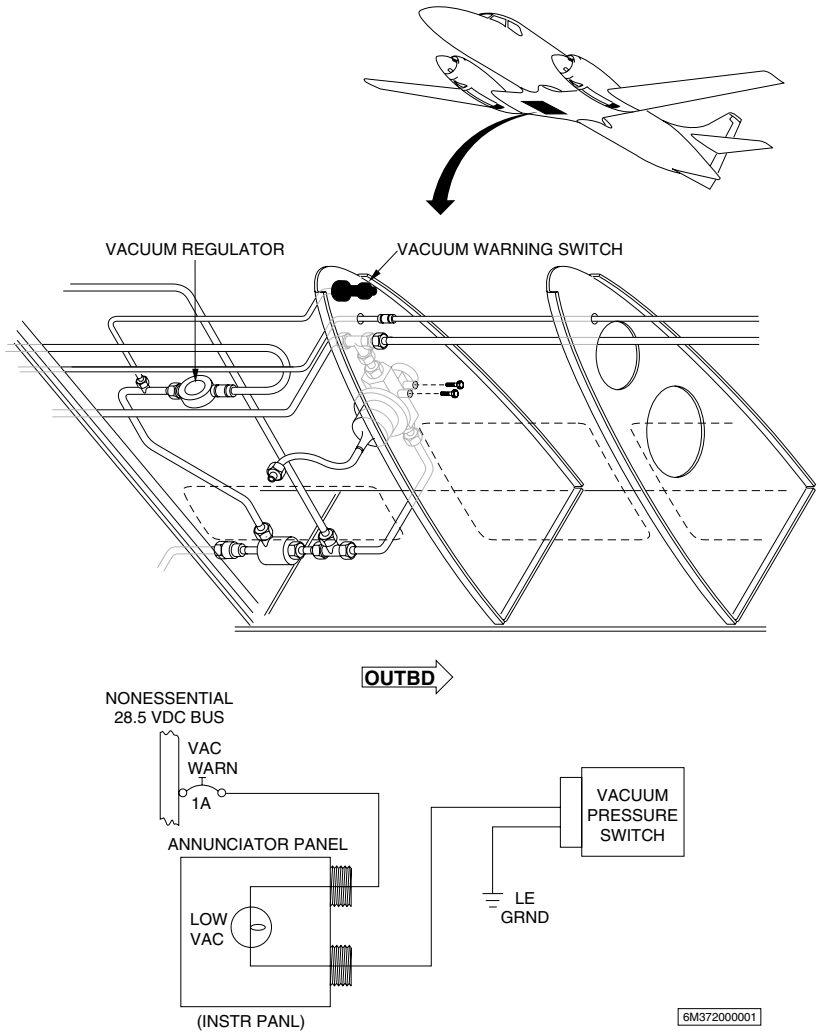
VACUUM WARNING SWITCH – DESCRIPTION AND OPERATION

1. General

A. The vacuum warning switch, located on the outboard side of the rib at W.S. 27.10 (L), is installed for the purpose of alerting the aircraft crew when a low vacuum condition in the deicing system occurs. The vacuum warning switch contains an inlet port, an electrical contact arm assembly, a diaphragm, and a pressure element. The inlet port is plumbed into the vacuum system at a point between the air ejector and the vacuum regulator.

B. When vacuum pressure drops below a set limit, the diaphragm reacts on the pressure element, completing a circuit through the contact arm assembly causing a warning light (LOW VAC) on the annunciator panel to illuminate. The vacuum warning switch is enclosed in a one and one half by a one and three fourths inch round metal housing and attached to the wing rib by two mounting studs.

C. Two types of vacuum switch have been used on the aircraft. Earlier model aircraft were equipped with a 31241A30 switch, which is no longer available. Later model aircraft are equipped with a 27-19109-003 switch. Instruction for installing later model switch in place of the earlier model unit are given under maintenance practices.



Vacuum Warning Switch
Figure 1

VACUUM WARNING SWITCH – MAINTENANCE PRACTICES

1. Removal/Installation – Vacuum Warning Switch

A. Removal – 3124-1A30 Vacuum Warning Switch

(1) Remove left front wing to fuselage fairing.

(2) Remove left forward lower wing inboard access panel.

(3) Remove left forward center wing section access panel.

(4) Disconnect electrical connection from vacuum warning switch.

(5) Remove mounting nuts securing vacuum warning switch to rib.

(6) Remove vacuum warning switch.

B. Installation – 3124-1A30 Vacuum Warning Switch

(1) Position vacuum warning switch to rib.

(2) Secure vacuum warning switch with mounting nuts.

(3) Connect electrical connection.

(4) Install and secure left forward center wing section access panel.

(5) Install and secure forward left inboard access panel.

(6) Install left front wing to fuselage fairing.

C. Removal – 27-19109-003 Vacuum Warning Switch

(1) Remove left front wing to fuselage fairing.

(2) Remove left forward lower wing inboard access panel.

(3) Remove left forward center wing section access panel.

(4) Disconnect electrical connection from vacuum warning switch.

(5) Disconnect fitting from vacuum warning switch.

(6) Cap open fitting and tube end.

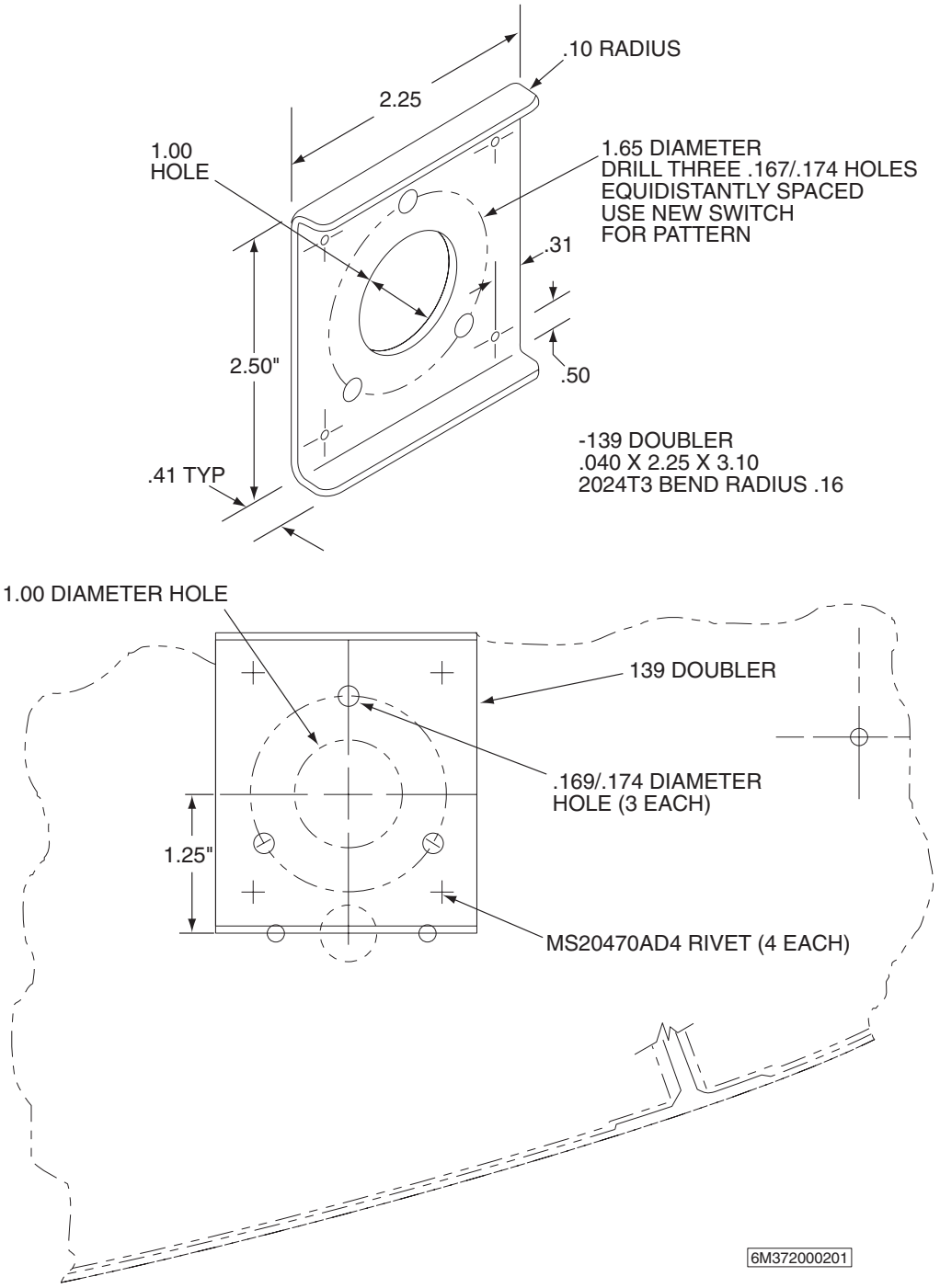
(7) Remove mounting nuts securing vacuum warning switch to rib.

(8) Remove vacuum warning switch.

D. Installation – 27-19109-003 Vacuum Warning Switch

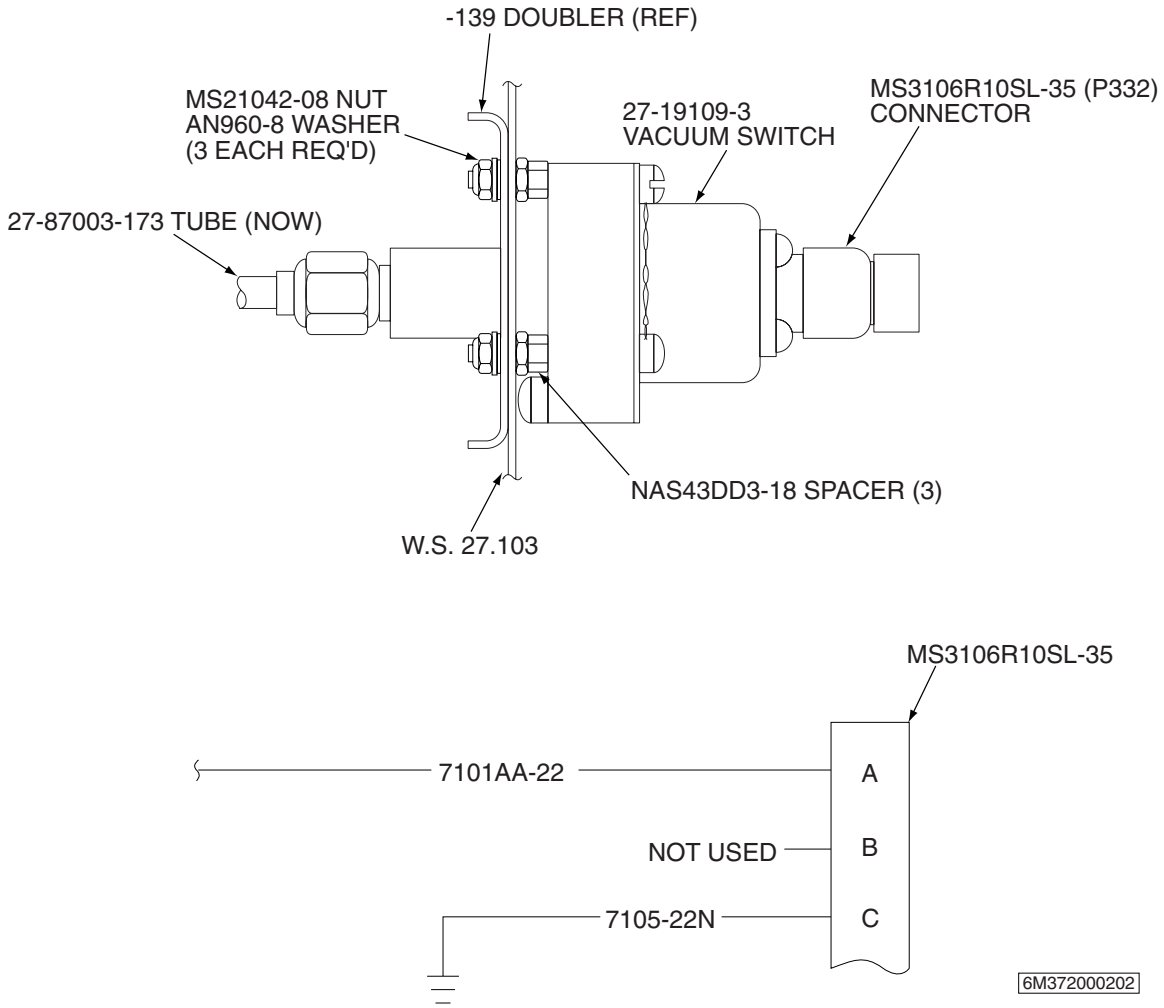
NOTE: If replacing 3124-1A30 switch with 27-19109-003 switch comply with steps (1) and (2).

(1) Using Figure 201, locate and install -139 doubler. Drill holes for new switch. Remove 27-87003-019 supply tube for vacuum warning switch and replace with 27-87003-173 tube.



Warning Switch Doubler
Figure 201

(2) Remove existing vacuum warning switch electrical connector and replace with a MS3106R10SL-35 connector as shown in Figure 202.



Vacuum Warning Switch
Figure 202

(3) Position vacuum warning switch to rib.

(4) Secure vacuum warning switch with mounting nuts.

(5) Connect fitting to vacuum warning switch.

(6) Torque fitting 40 to 65 inch-pounds (4.52 to 7.34 N·M).

(7) Connect electrical connection.

(8) Install and secure left forward center wing section access panel.

(9) Install and secure forward left inboard access panel.

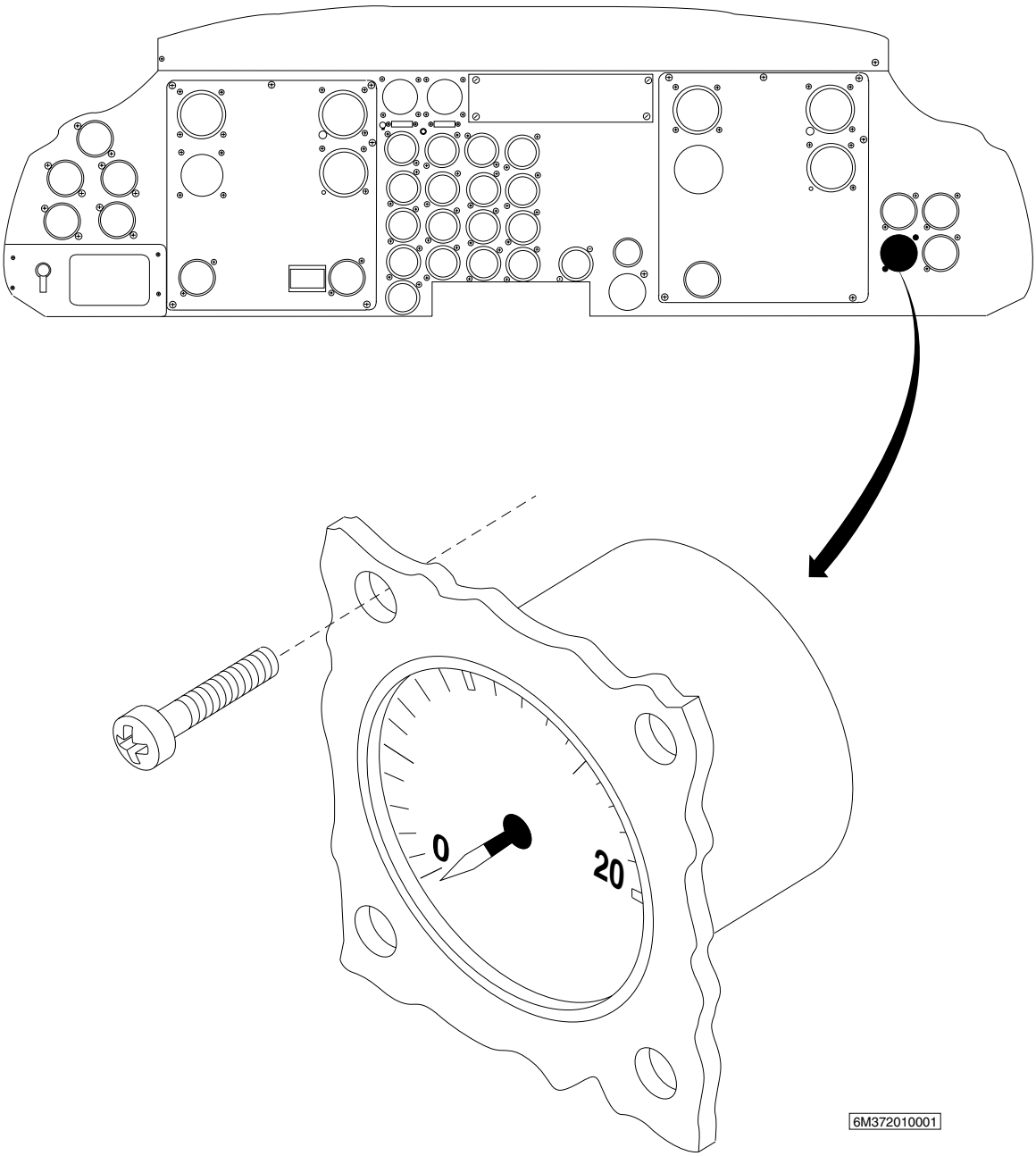
(10) Install left front wing to fuselage fairing.

(11) Operate right engine, check vacuum warning circuit operation.

VACUUM GAUGE – DESCRIPTION AND OPERATION

1. General

A. The vacuum gauge located on the copilot's side of the instrument panel allows monitoring of the vacuum system. It is connected into the deice vacuum/pressure system at a point between the air ejector and the vacuum regulator.



Vacuum Gauge
Figure 1

VACUUM GAUGE – MAINTENANCE PRACTICES

1. Removal/Installation – Vacuum Gauge

A. Removal – Vacuum Gauge

(1) Remove screws securing vacuum gauge to instrument panel.

(2) Pull gauge straight out from instrument panel.

(3) Disconnect vacuum and vent lines at back of gauge.

(4) Remove vacuum gauge.

B. Installation – Vacuum Gauge

(1) Connect vacuum, and vent lines to back of gauge.

(2) Position gauge in instrument panel.

(3) Secure vacuum gauge to instrument panel with screws.

CHAPTER 51 – STRUCTURES

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STRUCTURES – DESCRIPTION AND OPERATION

1. General

A. Fuselage and nacelles are of the semi-monocoque design and use conventional frames, ribs, stringers, and longerons. Standard methods of joining and sealing are utilized throughout with dimpled rivets on skin panels for aerodynamic smoothness.

B. Wings and stabilizers are full cantilever with conventional construction. A wet wing and the outer portion of the wing leading edge is utilized for fuel storage. The majority of fairings are of laminated fiberglass construction.

2. Definitions

A. Bonding

(1) Primary structures are riveted or fastened by screws or bolts. Wing and pressurized fuselage skins are sealed.

B. Honeycomb Structure

(1) Certain components are constructed or reinforced by honeycomb application. The radome and fairing are reinforced with a paper or cardboard hexagonal columnar cell construction bonded to the fiberglass.

C. Fiberglass Structure

(1) Fairings, dorsal fin, and radome are of fiberglass construction.

D. Painting

(1) A self-etching chromate primer is used throughout the aircraft for the prevention of corrosion. Battery compartments are painted with acid proof paint.

STRUCTURES – MAINTENANCE PRACTICES

1. Approved Repairs – Structures

NOTE: See Structural Repair Manual for procedures, materials, and repairs.

2. Removal/Installation – Structures

A. Removal – Fiberglass Fairings

(1) Remove screws from fairing. Equal number left and right side fairing.

(2) Separate the forward and aft fairing strips and remove from aircraft.

(3) Fairing strips must be stored in a safe place. They are fragile and easily damaged. Do not store on hangar floor.

B. Installation – Fiberglass Fairings

(1) Replace forward and aft portions of fillet fairing.

CAUTION: DO NOT TWIST OR DISTORT FAIRING.

(2) Install screws. Be certain washers are installed. Damage may result if the screw is too long. Plate nuts in fuselage and fuel cell are dome or sealed cap type.

(3) Tighten screws only enough to prevent shifting of fairing. Overtightening will result in cracked fiberglass.

C. Removal – Radome

(1) Remove screws from radome aft mount. Support radome.

(2) Pull forward to remove.

(3) Store in a safe protected place.

D. Installation – Radome

(1) Mount radome. Care must be taken when pushing radome over attach channel. Damage to fiberglass will occur if radome surface is not mated properly over attach surface.

(2) Align radome holes with fuselage mount holes.

(3) Install screws and countersunk skin washers.

(4) Do not overtighten screws.

3. Inspection/Check – Structures

A. Fuselage Test

(1) The completed and sealed fuselage shall be leak tested with a pressurized cabin leakage tester. The maximum permissible leak rate shall be 55 CEM at a cabin pressure differential at 5.0 psi. Generally, all critical leaks may be detected by listening for a whistling noise and by feeling with the hand. If detection by this method is inadequate, it may be necessary to apply a soap and water solution to develop bubbles. The affected areas detected shall be brush coated with 890-A 1/2 sealant until the leakage rate is reduced to no more than the aforementioned rate. Small leaks may be stopped by filleting or injecting 890-B 1/4, a quick drying sealant.

(a) Windows – Refer to Chapter 56.

(b) Removal Items – Covered in their respective chapters of this manual.

(c) Wing – Refer to Chapter 28, Fuel.

4. Cleaning/Painting – Structures

A. Cleaning – Fuselage, Wing, and Empennage

(1) Standard nonabrasive soap and water solution may be used in all cleaning applications. Avoid entry of solutions and flooding of soap or water into scoops and static vents. It is recommended scoops be plugged and vents be taped over as a precaution. Also, do not flood the horizontal stabilizer movable opening.

B. Windows

(1) The plastic windows can be cleaned using any nonacid, nonabrasive material. To prevent scratches, wash the windows carefully with plenty of soap and water, using the palm of the hand to dislodge dirt and mud. Grease, stains, and insects can be removed using a good grade of plastic cleaner and polisher. This product will also fill minor scratches and help prevent further scratching. Using light pressure apply a thin even coat of cleaner/polish with a soft clean cloth. Let dry. Wipe off. Buff with a soft, clean cloth.

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CHAPTER 52 – DOORS

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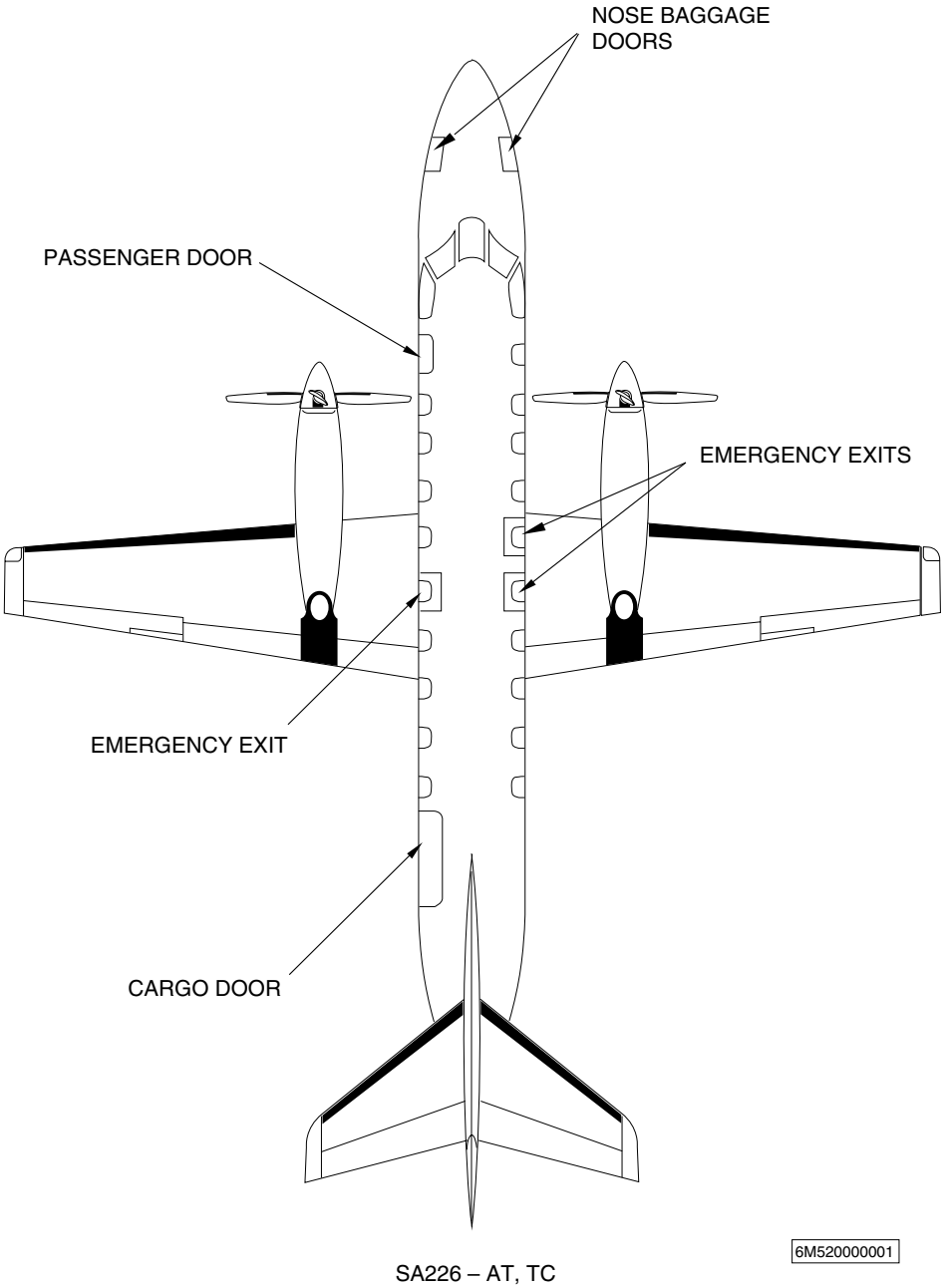
DOORS – DESCRIPTION AND OPERATION

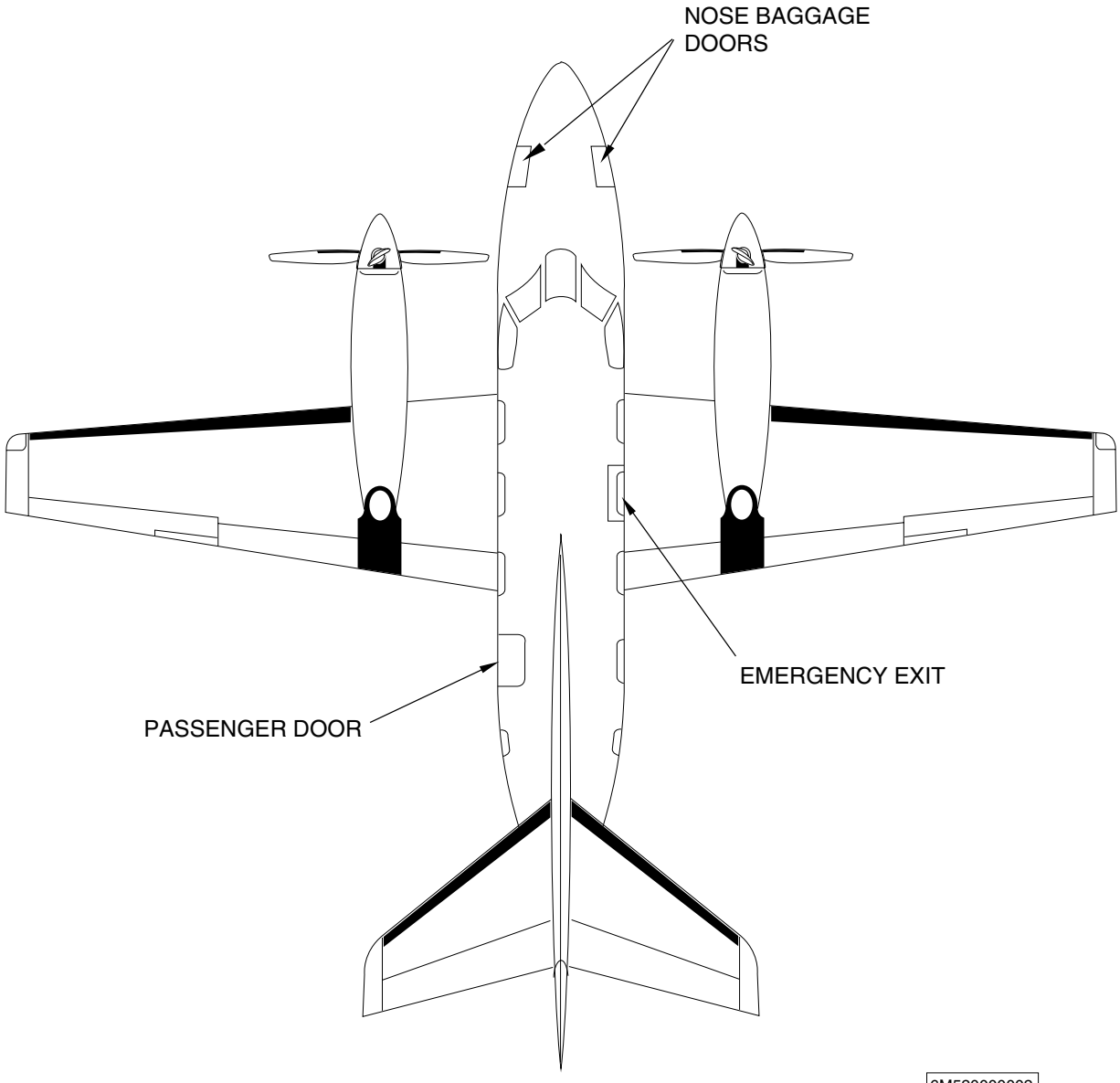
1. General

A. The Metro and Merlin IV aircraft have a forward passenger (cabin/crew) door installed on the left side of the fuselage forward of the propellers near the cockpit. The cargo door is located on the left side of the fuselage aft of the wing. The aircraft has three emergency exit doors which incorporate a standard size window. The nose baggage compartment, accessible from both the right and left sides of the aircraft, provides additional storage space.



SA226 – AT, TC Doors
Figure 1

B. The Merlin III aircraft have an aft passenger (cabin/crew) door, one emergency exit door with window, and a nose baggage compartment identical to the Metro/Merlin aircraft. A patented click-clack mechanism firmly fixes the door latches into the fuselage structure and allows structural load stresses to be carried through the door, reducing the requirement for a heavy load bearing structure around the door.

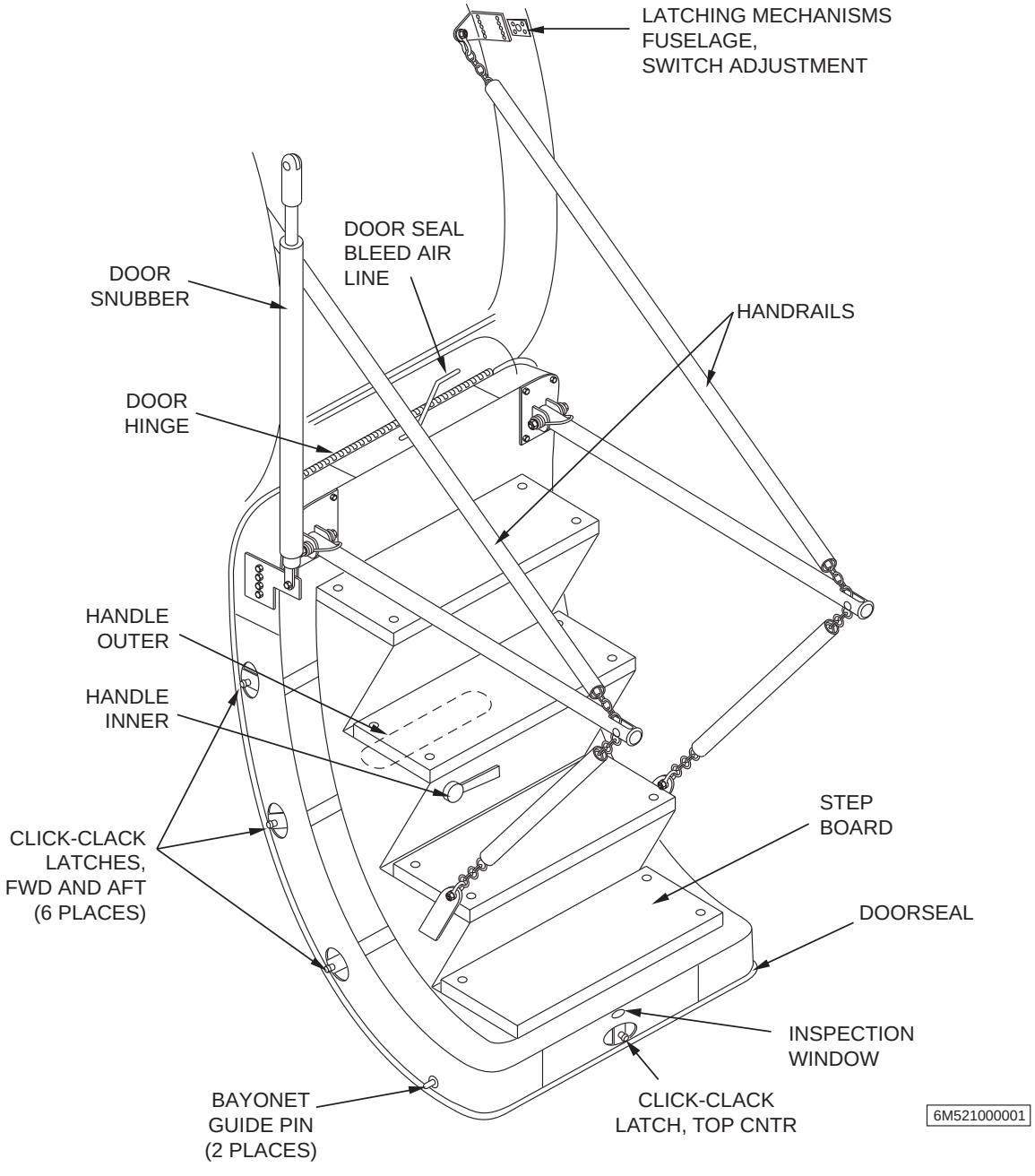


SA226 – T Doors
Figure 2

PASSENGER DOOR – DESCRIPTION AND OPERATION

1. General

A. The passenger entrance door of the SA226 – T airplane is a 25 X 53 inch air type passenger door installed on the left side of the fuselage aft of the wing. The passenger door on the SA226 – AT and TC is located on the left side of the fuselage forward of the propellers near the cockpit. It is an all aluminum structure consisting of an aluminum alloy, hydro-formed frame, and aluminum clad skin. The inner surface of the door incorporates four fixed steps.



Passenger Door
Figure 1

2. Passenger Door Latching System

A. The passenger door latching system is equipped with seven click-clack type latch assemblies plus two alignment pins (bayonets). The door latching system is operated through push-pull rods from a single door handle, which secures the door to the basic aircraft structure. These seven click-clack type latch assemblies allow the normal load paths to be routed directly across the door, reducing the amount of structure necessary. Two bayonets, at the top of the door forward and aft sides, extend initially when the door handle is rotated toward closed and locked position. The bayonets pull the door into the door opening, aligning the click-clack latches with their respective fuselage door frame receptacles, but do not carry structural loads.

B. As the door handle continues toward closed and locked position, the click-clack actuator jaws extend from the door into a door frame receptacle. Door handle movement beyond this point extends the actuator plunger shafts beyond the actuator jaws. As this extension continues, a taper at the base of the plunger shaft forces the click-clack jaws to open laterally and grip the inside of the fuselage frame receptacle, securing the door to the fuselage door frame.

C. The latching system is capable of being operated from inside or outside of the aircraft, and is equipped with a key operated locking device.

3. Passenger Door Steps

A. The passenger door incorporates an integral stairway, and requires no ground support equipment for loading. The passenger door steps are constructed of four bondalyte step boards and an aluminum frame of four risers and two side plates. The step boards are covered with vinyl aeromat and may be removed or installed individually to gain access to the inside of the door.

4. Passenger Door Handles and Safety Lock

A. The Passenger door may be opened from inside or outside by inter-connected door handles. The outside door handle is recessed into the door frame and may be locked to provide security. The inner door handle is located in the middle of the stairwell. The passenger door may be opened from the inside when the outside is locked.

B. A passenger door safety lock system operates while the aircraft is pressurized. A diaphragm actuates a safety hook, locking the latching system to prevent door from being opened while aircraft is in flight.

5. Passenger Door Seal System

A. A silicone-rubber seal is installed around the door.

(1) A door seal valve, triggered by moving the door handle to the locked position, allows bleed air from the pneumatic (deice boot) system to inflate when the aircraft is operating.

(2) A solenoid valve opens when the door is closed and allows the seal to be inflated by regulated bleed air.

B. The first three Metro aircraft, TC201-202E and Merlin IV AT002-003, used an electric valve in the left wing root to control the flow of air to the seals. Electric power is routed through the main landing gear squat switches and will not allow the seals to inflate while aircraft is on the ground.

C. The first eleven Merlin III aircraft T201-211 used non-inflatable seals installed on the fuselage cabin door frame. No maintenance to these seals is required unless the seal is damaged or deteriorated.

6. Passenger Door Snubber

A. A snubber retards the free falling action of the door when it is opened.

B. Proper procedure for closing the door is as follows:

(1) Hold the latch handle in the unlatched position while closing the door.

(2) As the door closes, ensure the handrail chains are stowed inside the aircraft.

(3) Hold the door closed and turn the handle to the latched position.

(4) When opening the door, keep a hand on the door and follow through the opening motion until the weight of the door is carried by the snubber.

7. Passenger Door Handrails

A. A vinyl covered chain serves as a handrail and as a door opening stop. Ensure that both chains are adjusted to equal length, or their stop limits will be different, and the door will tend to twist when weight is applied on the steps. Adjustments made to the chain may be made by twisting the chain at its clevis attachment or by changing the number of links.

8. Passenger Door Unlatch Warning System

A. Door warning switches, wired in parallel, are installed in each of the door frame receptacles. A red warning light in the annunciator panel illuminates if any one of the latches is not FULLY engaged and remains illuminated until all latches are secure.

PASSENGER DOOR – MAINTENANCE PRACTICES

1. Removal/Installation – Passenger Door

A. Removal – Passenger Door

(1) Position door cradle outside aircraft to support door when door is released from supporting mechanisms. Ensure cradle has suitable surface to avoid damage to door finish or material.

(2) Close and secure passenger door.

(3) Remove interior floor covering inside passenger door.

(4) Working from inside aircraft, with passenger door closed, remove snubber bolts from bracket at fuselage frame.

(5) Disconnect handrail chain assemblies from door frame bracket.

(6) Holding handrail chain, open door slowly and lower door to cradle.

(7) Remove cabin floorboard immediately adjacent to door opening.

CAUTION: IMMEDIATELY INSTALL CAPS OR PLUGS ON OPEN LINES TO PREVENT CONTAMINATION.

(8) Disconnect door seal inflation tube. Cap open lines.

NOTE: Access to door seal inflation tube may be made either through access doors in the door step assembly, or through lightning holes in the cabin floor.

(9) Withdraw seal inflation tube.

(10) With door in cradle, remove door hinge pin as follows:

(a) Using a small drift pin and hammer, drive hinge pin 6.0 to 8.0 inches (15.24 to 20.32 cm) out of hinge assembly.

(b) Grasp the extended part of hinge pin with clamping pliers (vise-grips).

(c) Remove hinge pin by striking clamping pliers with mallet.

NOTE: An alternate method may be used for removing hinge pin, but only if hinge halves are to be replaced. Chuck the extended part of the hinge pin in drill, start drill motor at low RPM, while pulling hinge pin out.

B. Installation – Passenger Door

(1) Pre-position door on suitable cradle prior to installation. Ensure cradle will provide full weight support and has suitable surface to avoid damage to door finish or materials.

(2) Check door and fuselage hinge condition. If either hinge half is loose or damaged, tighten or replace as necessary.

(3) Align and mate door and fuselage hinge halves.

(a) Insert hinge pin using vise-grips as IAW **1.**, A., (10).

NOTE: Install new hinge pin if old hinge pin shows signs of wear.

(b) Peen ends to secure hinge pin.

(4) Install door seal inflation tube.

NOTE: Access to door seal inflation tube may be made either through access doors in the door step assembly, or through lightning holes in the cabin floor.

(a) Remove protective caps or plugs from lines.

(b) Connect door seal inflation tube.

(5) Install floorboards immediately adjacent to door opening.

(6) Close door slowly and secure.

(7) Connect handrail chain assemblies to fuselage door frame brackets.

(8) Connect snubber assembly from inside with door closed.

(9) Install interior floor covering inside passenger door.

(10) Slowly open and close door several times to ensure proper operation and a secure closing.

(11) Perform water leak test IAW Adjustment/Test – Passenger Door Seal.

NOTE: Door seal should prevent entry of water into aircraft even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(12) Perform pressure check IAW Inspection/Check – Passenger Door Seal.

2. Preparation/Assembly – Passenger Door

A. Parts Required – New Passenger Door

(1) Door assembly.

(2) Hinge assembly and hinge pin.

(3) Bayonets (2).

(4) Click-clacks (7).

(5) Teflon Guides (minimum 4).

(6) Eccentric bushings and retainers.

(7) Faceplates (9).

(8) Chains and handrails.

(9) Door seal.

(10) Snubber.

B. Tools Required – New Passenger Door

(1) Reamer 0.630 inch.

(2) ZT-535C pancake drill.

(3) SAT 1223 bushing set.

(4) Door stand/cradle to support weight of door, with suitable surface to avoid damage to door finish and/or material.

(5) Web fuselage strap and cinch.

(6) Transfer punches and drill bits No. 10, No. 21, No. 30, and No. 40, with drill stop.

(7) Countersink with stop – No. 10 and No. 21.

(8) Open end wrenches, box end wrenches, and 1/4 inch drive sockets, sizes, 3/16, 7/32, 1/4, 3/8, 7/16, and a 1/4 universal.

(9) Drill motor and rivet gun (4X), bucking bars including a 6.0 to 8.0 inch (15.24 to 20.32 cm) offset for rivet gun.

C. Consumable Materials Required – New Passenger Door

(1) EP-890 B 1/2 or B2 sealer.

(2) RTV 3145 silicone sealant.

(3) Tri-flow (aerosol), 903 Dri Slide (liquid), MIL-L-23398 Molykote (aerosol) or NATO #S-749 lubricants.

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(4) Methyl Propyl Ketone (MPK).

(5) Masking Tape – one and two inch width.

(6) Hinge oil.

D. Assembly Step 1 – New Passenger Door

(1) The objective of this phase of assembly is to align, center, and install a new passenger door in the fuselage.

CAUTION: AIRCRAFT MUST BE LEVEL. IF AIRCRAFT REQUIRES MOVEMENT AFTER WORK HAS STARTED, IT MUST BE RETURNED TO A LEVEL WORKING AREA. AIRCRAFT MUST TRAVEL AT LEAST TWENTY (20) FEET (SIX [6] METERS) IN A STRAIGHT LINE, WITH NOSE WHEEL STRAIGHT, BEFORE WORK CAN RESUME.

(2) Park aircraft on level surface. DO NOT place aircraft on jacks or other support(s).

(3) Remove emergency exit window to provide alternate entrance/exit (SA226 – T).

(4) Check fuselage hinge condition, If hinge is loose, tighten or replace as necessary.

(5) Before positioning door into fuselage, ensure the following conditions exist:

NOTE: Passenger door should be pre-positioned on suitable cradle or suitable work bench prior to installation. Ensure cradle will provide full weight support and has suitable surface to avoid damage to door finish or materials.

(a) Bayonet guide pins – NOT INSTALLED.

(b) Bayonet latches – NOT INSTALLED.

(c) Top click-clack – NOT INSTALLED.

(d) Eccentric bushings in fuselage receptacles – NOT INSTALLED.

(e) Door hinge – NOT INSTALLED.

(f) All sealer and bondo – REMOVED FROM FUSELAGE AND DOOR FRAMES.

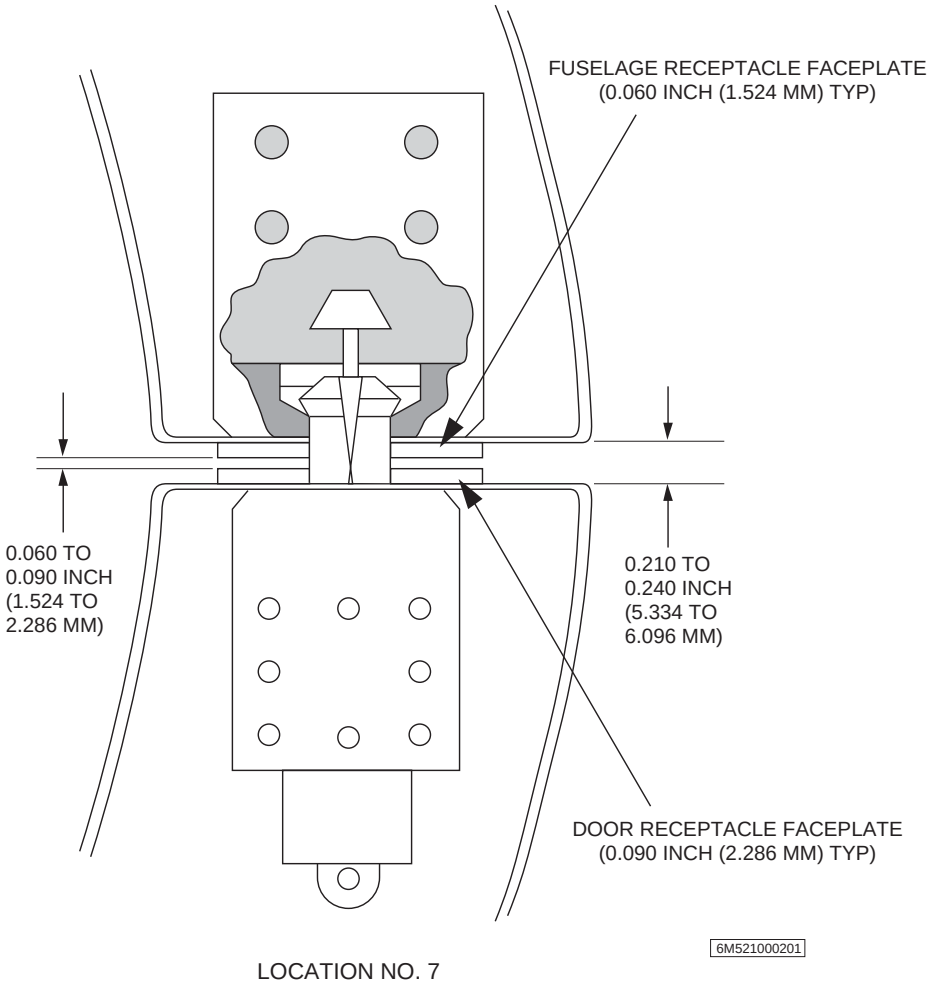
(g) Faceplates on door and fuselage – INSTALLED.

(6) Temporarily position passenger door in fuselage.

(7) Set height of new door in fuselage. Place a 0.060 to 0.090 inch (1.524 to 2.286 mm) spacer between door faceplate and fuselage faceplate at position 7 (Figure 201).

(8) Optimum spacing between door frame and fuselage frame is 0.220 inch (5.558 mm).

NOTE: Spacing tolerance of 0.210 to 0.240 inch (5.334 to 6.096 mm) allows for proper adjustment of side latches.



LOCATION NO. 7

Passenger Door Spacing
Figure 201

(9) Note clearance between forward and aft door faceplates and fuselage faceplates.

(10) Center door between forward and aft door faceplates, and fuselage door frame faceplates.

(11) Fabricate shims to same dimensions as Teflon guide. Use aluminum material of sufficient thickness to obtain desired clearance.

(12) Shim Teflon guides to provide positive contact with fuselage door frame.

(13) Make reference marks on door skin (forward and aft) to fuselage skin to aid in maintaining height positioning.

(14) Remove passenger door. Place in cradle.

(15) Position and install door handle assembly, bayonets, click-clacks, and push-pull tubes.

(16) Check proper operation of door handle, bayonets, and click-clacks while door is on door cradle.

(a) Ensure handle operates freely.

(b) Lubricate bayonets and click-clacks.

(17) Position assistant inside aircraft to inspect general action and location of bayonets and click-clacks.

(18) Temporarily install door.

(19) Check height positioning IAW step **2.**, D., (7).

(20) Check reference marks IAW step **2.**, D., (13) on door skin (forward and aft) to ensure door is centered.

NOTE: Each time door is removed and installed, check reference marks to ensure door is at proper height.

(21) Ensure click-clacks will pass through fuselage receptacles (bushings not installed).

(22) Remove door. Place on door cradle.

(23) Install bushings in fuselage position 1 and 2 (see Figure 201). Ensure bushings are on dead center.

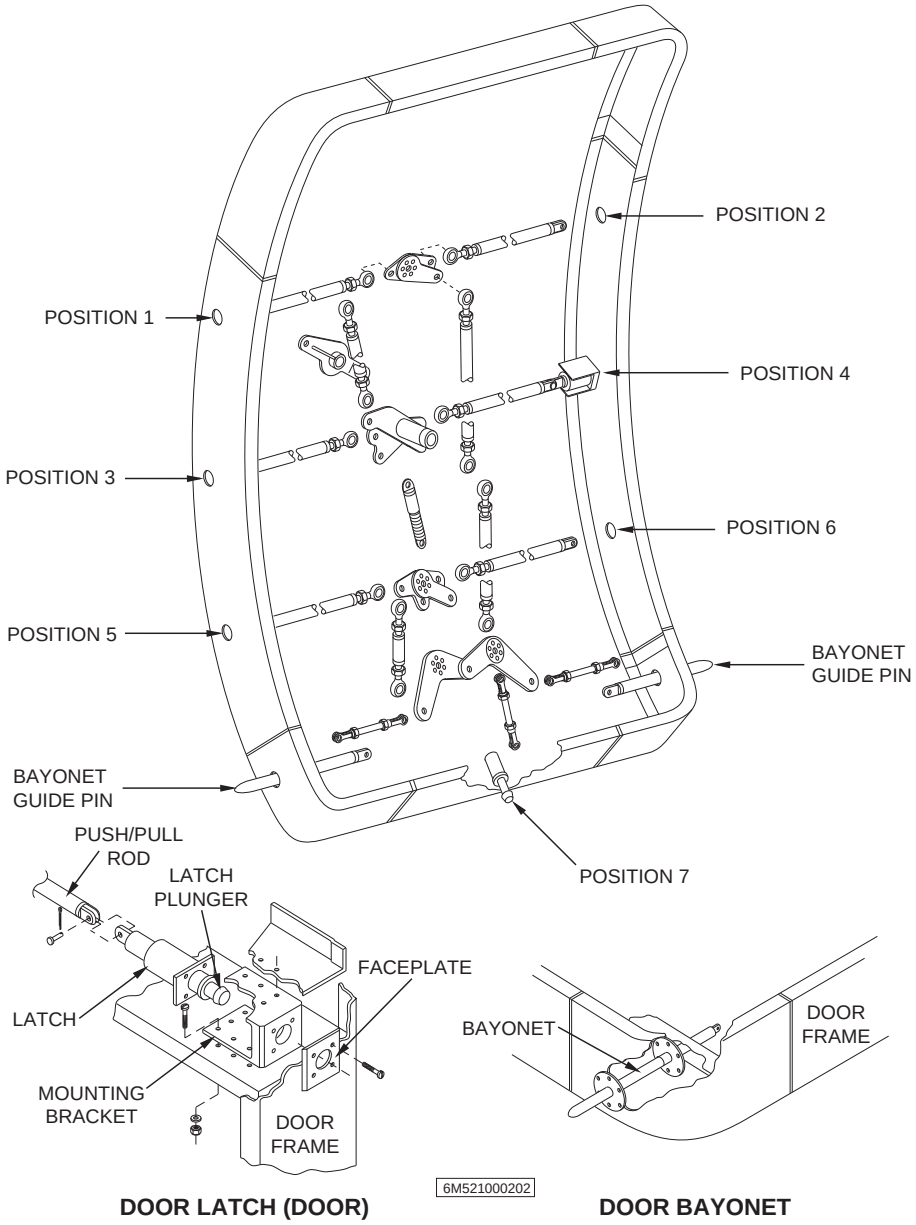
NOTE: Refer to SA226 series Illustrated Parts Catalog (IPC) for bushing sizes and descriptions.

(24) Position and install door carefully, so as to not damage handle mechanism. Gently attempt to latch door in place.

(25) Have assistant inside aircraft observe click-clack action and for any movement of the door as click-clacks attempt to engage.

NOTE: Due to untrimmed skin, gentle pressure from outside at click-clack locations may assist in engagement.

(26) If door can be latched, proceed to next step. If door cannot be latched, remove door and adjust receptacle/bushing causing the most problems.



Passenger Door – Click-Clack Positions
Figure 202

CAUTION: DO NOT ALLOW DOOR TO OPEN AGAINST CLECO FASTENERS.

(27) After latches 1 and 2 are installed and operating properly, install temporary hinge and secure using at least six (6) Cleco fasteners.

(28) Temporarily install door.

(a) Proceed to bushing positions 3 and 4, then 5 and 6 (Figure 202).

(b) Repeat steps **2.**, D., (23), (24), and (25), until all bushings are located.

(29) Trim door skin at lower corners.

(30) With door latched:

(a) Check that all Teflon door guides touch fuselage frame.

(b) Add additional Teflon door guides as necessary to ensure spacing between faceplates is equal.

(c) Ensure all six click-clacks are fully engaged.

E. Assembly Step 2 – New Passenger Door

(1) Objective of the following steps is to obtain initial skin contour, through receptacle adjustments, in preparation for installing permanent door hinge.

(2) Trim fore and aft skin. With door latched and click-clacks engaged, check door contour.

(3) Slowly open door and leave open.

(4) Apply one inch wide masking tape around perimeter of fuselage door cutout.

NOTE: Tape must exactly match edge of fuselage skin.

(5) Carefully close and latch door.

(6) Apply two inch wide masking tape around door frame. Tape edge must precisely match outboard edge of tape installed during step **2.**, E., (4).

(7) With a razor or knife, follow edge of door and cut top layer of tape.

(8) Open door and trim skin; follow edge of remaining tape.

(9) Use a Vixon file and finish trimming fore and aft to maximum clearance of 0.0313 inch (0.794 mm).

(10) Begin with click-clack position 1 (Figure 202) and alternate from side to side. As contour requires and click-clack interference indicates, exchange or replace bushings as required.

NOTE: Refer to SA226 Series, Illustrated Parts Catalog (IPC) for bushing sizes and descriptions.

(11) Begin with bushing at the three o'clock position.

(a) Alternate between three o'clock to one o'clock, to five o'clock positions, until proper combinations of skin contour and latch operation are determined.

(b) For forward latches, the preferred position is between one o'clock to five o'clock. This allows for future adjustment.

(c) For aft latches, the seven o'clock to one o'clock position is preferred.

(12) Progress in bushing size(s) as necessary.

(13) Select bushing, and rotate until all six latches are operating.

(a) A tight fit with all six latches operating easily, can be obtained by applying pressure in the middle of the door to stimulate click-clack reaction.

(b) The click-clacks will pull the door in place before latch engagement.

(14) Remove temporary hinge.

CAUTION: HINGE LOCATING IS CRITICAL TO PROPER FIT OF DOOR. TIGHT FITTING HOLES ARE REQUIRED.

(15) With door closed, install permanent hinge and hinge pin.

(a) Match door hinge half to fuselage hinge half.

(b) Begin in center and alternate sides working outboard.

1 Use a No. 21 transfer punch to transfer holes from door to skin, to new hinge half.

2 Step drill from No. 40 to No. 21.

CAUTION: DO NOT ALLOW DOOR TO OPEN AGAINST CLECO FASTENERS.

3 Insert Cleco fasteners in each hole, as holes are drilled.

(c) Countersink and debur hinge holes.

1 Apply sealant to both sides of hinge.

2 Install and rivet hinge to door.

NOTE: Cleco fasten holes before riveting.

F. Assembly Step 3 – New Passenger Door

(1) The objective is to locate bayonet, and click-clacks in position 7 (see Figure 202), and install door seal.

(2) Trim lower corners as follows:

(a) Apply one inch wide masking tape to lower forward and aft corners of fuselage.

1 Align edge of tape exactly with edge of fuselage skin.

2 Close and latch door.

(b) Slight adjustments of click-clacks may be necessary to ensure door contour is maintained.

(c) Apply two inch wide masking tape around door frame. Tape edge must precisely match to the outboard edge of tape applied in step **2.**, F., (2), (a).

(d) With a razor or knife, follow edge of door and cut tape.

(e) Open door and trim skin.

1 Use a Vixon file to trim skin.

2 After initial trim, alternately check door and trim to ensure proper skin clearance.

(3) Locate bayonets as follows:

(a) Close door and try bayonets in holes.

(b) Check alignment. If necessary, polish bayonet faceplates until each bayonet will slip in and out with moderate pressure.

(c) Open door.

1 Connect Bayonets.

2 Ensure proper operation.

NOTE: It may be necessary to relocate faceplates.

(4) Install door seal.

(a) Drill 0.250 inch (6.350 mm) hole in door frame, lower aft corner, in line with lightning hole for seal inflation tube.

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(b) Clean edge of door frame with MPK where seal attaches.

(c) Clean new seal thoroughly with MPK to remove shipping preservatives.

CAUTION: EXCESS SEALER BETWEEN SKIN AND SEAL WILL CAUSE CONTOUR DEVIATION AND UNSATISFACTORY OPERATION OF DOOR. USE CARE NOT TO SPREAD SILICONE SEALANT OR TO ALLOW IT TO CONTACT SKIN SURFACE. IMMEDIATELY DISPOSE OF RAGS OR TOWELS USED TO CLEAN UP EXCESS SILICONE SEALANT.

(d) Run a bead of Dow Corning RTV 3145 clear approximately 0.125 inch (3.175 mm) wide all the way around door at the seal seat.

CAUTION: SEAL MUST BE SLID ALONG SKIN INTO CORNERS. DO NOT SLIDE SEAL ALONG FRAME INTO CORNERS.

(e) Beginning at inflation tube, install seal all the way around door.

1 Using a sliding action, work from edge of skin into the structure and seal seat.

2 When seal is in place, ensure no twists exist in seal around door.

(f) Check contour of ridged sealing area of door seal in area of splices (should not exceed three splices). Ensure seal contour does not deviate by more than -0.020 inch (-0.508 mm) from contour on either side of splice(s).

(g) Remove caps from open lines. Connect door seal inflation tube to bleed air tubing.

(h) When the seal is completely installed, carefully close door.

1 Use web strap around fuselage and passenger door to help draw door in.

2 Allow adhesive to dry in place (approximately two hours).

(5) Trim top skin, making certain upper door skin and corners clear fuselage skin.

(a) Ensure upper forward and aft Teflon guides contact fuselage frame.

(b) If top of door skin is excessively loose, shim as necessary. If too tight, remove shims. Slightly concave door skin is acceptable.

(c) Complete skin riveting, or shim as required, before locating upper click-clack latch (position 7).

(6) Install upper click-clack in position 7 (Figure 201, and 202) as follows:

(a) Install tool set SAT 1233-2 into receptacle from door to fuselage.

(b) Install receptacle into passenger door with SAT 1233-2 inserted.

1 Secure SAT 1233-2 to SAT 1233-1 with 0.250 inch (6.350 mm) bolt.

2 Tighten bolt until dimension of approximately 0.110 to 0.140 inch (2.778 to 3.572 mm) is obtained from faceplate to door frame.

(c) Drill each hole in the door receptacle with ZT-535C pancake drill.

1 Alternate from side to side of receptacle while drilling.

2 Install hardware as each hole is drilled.

(d) Close door and latch. Complete remaining riveting in door structure.

(e) Check click-clacks for final adjustment. Lubricate click-clacks with Tri-Flow (aerosol), 903 Dri-Slide (liquid), MIL-L-23398 Molykote (aerosol), or NATO #S-749 Lubricants.

(f) Close door. Install steps to door.

(g) Install snubber and door seal inflation tube.

(h) Perform water leak test IAW Adjustment/Test – Passenger Door Seal. No leakage is allowed.

NOTE: Door seal should prevent entry of water into aircraft even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(i) Perform pressure check IAW Inspection/check – Passenger Door Seal.

1 If pressurization leaks occur, slide a wide tip common screwdriver between seal and frame. Gently loosen seal in affected area.

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2 Clean area with MPK.

3 Apply additional RTV 3145 sealer (approximately 0.250 inch (6.350 mm) wide to affected area.

4 Close door immediately after sealer is applied.

5 Allow sealer to dry for minimum of two hours.

6 Open door and trim away excess sealer.

7 Perform water leak test and pressure check.

(7) Install handrail chain assemblies to door frame brackets.

(8) Install cabin floorboards.

(9) Install interior around door.

3. Inspection/Check – Passenger Door

A. Inspect Passenger Door as follows:

(1) Inspect structure and skin for wrinkles, cracks, surface scratches, dents, loose rivets, corrosion, or any indication of structural irregularity.

(2) Check operation of latching system.

(a) Check upper and lower latching rods and rod ends for alignment, wear, general condition, security, and operation.

(b) Check click-clacks for nicks, burrs, alignment, general condition, and operation.

(c) Check inner and outer door handles for adjustment, fit, security, and operation.

(3) Inspect door, hinge, steps, handrail chains, and interior covering for security and general condition.

(4) Check plastic inspection cover ports for general condition and security.

(5) Check door seal and seal supply line for punctures, cracks, kinks, leaks, security, and general condition.

(6) Inspect, clean, and lubricate latching mechanisms.

(7) Check Teflon guide blocks for general condition and security.

PASSENGER DOOR LATCHING SYSTEM – MAINTENANCE PRACTICES

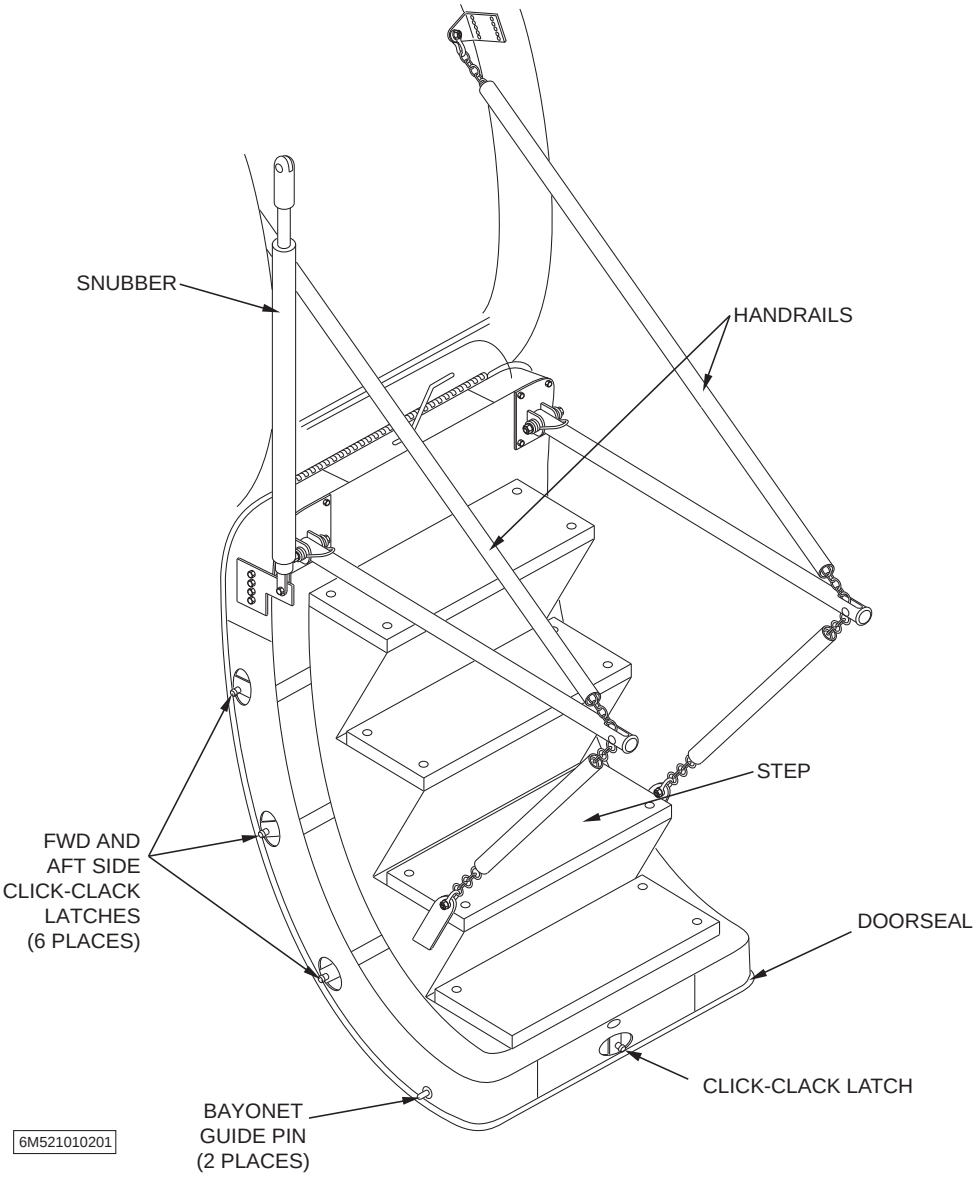
1. Servicing – Passenger Door Latching System

A. Lubrication – Passenger Door Bayonets and Click-Clacks

(1) Lubricate bayonets and/or click-clacks with Tri-Flow (aerosol), 903 Dri-Slide (liquid), MIL-L-23398 Molykote (aerosol), or NATO #S-749 lubricants.

NOTE: Lubricate at regular inspection intervals. To reduce dust and dirt contamination, do not use lubricants that leave an oily or greasy coating.

(2) The extended tip of plunger and bayonets or click-clacks to surface of faceplate, should always be well lubricated.



Passenger Door Latching System
Figure 201

B. Cleaning – Passenger Door Bayonets and Click-Clacks

(1) Use dry compressed air to blow contaminants from bayonets and click-clacks.

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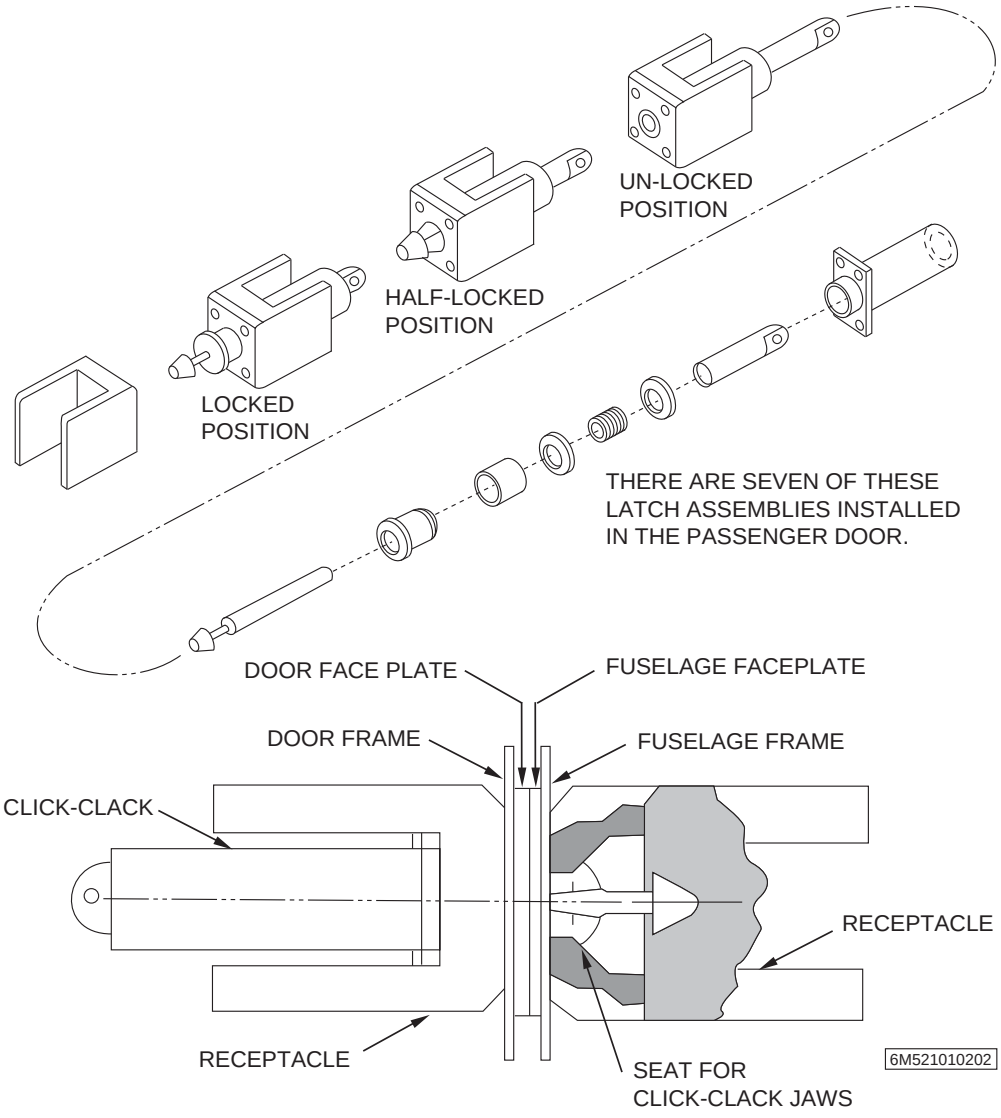
(2) Clean bayonets and/or click-clacks with a clean cloth using MPK or Varsol (Division of EXXON - G27H-10-2-80, E1 0835).

(3) Use dry compressed air to dry bayonets and/or click-clacks.



Click-Clack, Faceplate, and Receptacle
Figure 202

2. Removal/Installation – Passenger Door Latching System

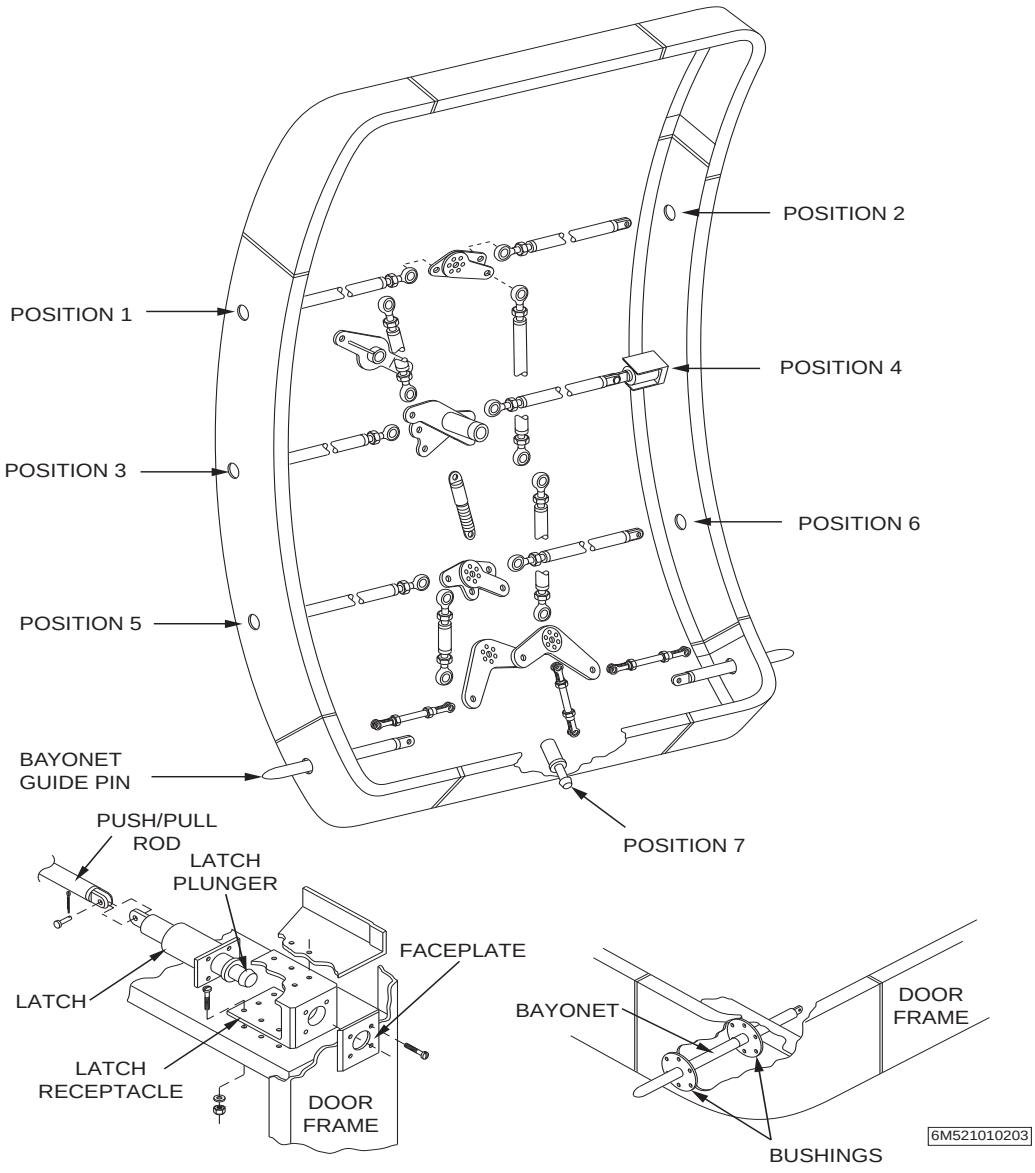
NOTE: Patented click-clack latches and bayonet alignment pins are used in passenger door receptacles.

A. Removal – Door Latches (Figures 201, 202, and 203)

(1) Open passenger door.

(2) Remove upholstery from step boards as necessary.

(3) Remove screws securing step boards and remove step boards.



Passenger Door Click-Clack Latching system
Figure 203

(4) Remove inner door handle.

(5) Remove screws securing faceplate to door frame and remove faceplate.

(6) Ensure click-clack actuator is fully retracted.

(7) Disconnect click-clack actuator from push-pull rod.

(8) Withdraw click-clack actuator from fitting assembly and remove from door.

B. Installation – Door Latches (Figures 201, 202, and 203)

(1) Open passenger door.

(2) Begin installation with click-clack actuator fully retracted.

(3) Insert click-clack actuator into fitting assembly.

(4) Connect click-clack actuator to push-pull rod.

(5) Install faceplate to door frame and secure with screws.

(6) Make initial adjustments to click-clack latch IAW 3. A. Passenger Door Latch Mechanism Adjustment/Test, this section.

(7) Install inner door handle.

(8) Cycle door handle to ensure proper operation of click-clack actuator.

(9) Make final adjustments to click-clack latch IAW 3. A. Passenger Door Latch Mechanism Adjustment/Test, this section.

(10) Install step board and secure with screws.

(11) Install upholstery to step boards as required.

C. Removal – Bayonet Guide Pins (Figure 203)

(1) Open passenger door.

(2) Remove upholstery from bottom door step.

(3) Remove screws securing bottom step board and remove step board

(4) Ensure bayonet guide pin is fully retracted.

(5) Disconnect bayonet guide pin from push-pull rod.

(6) Remove bayonet guide pin from guide pin fitting assembly.

D. Installation – Bayonet Guide Pins (Figure 203)

(1) With passenger door open and bayonets fully retracted, insert bayonet guide pin into guide fitting assembly.

(2) Connect bayonet guide pin to push-pull rod.

(3) Adjust bayonet guide pin as required IAW 3. A. Passenger Door Latch Mechanism Adjustment/Test, this section

(4) Install and secure bottom step board.

(5) Install upholstery to bottom door step.

E. Removal – Fuselage Receptacles (Figure 204)

(1) Open passenger door and remove interior covering as required to gain access to affected receptacles located in fuselage door frame.

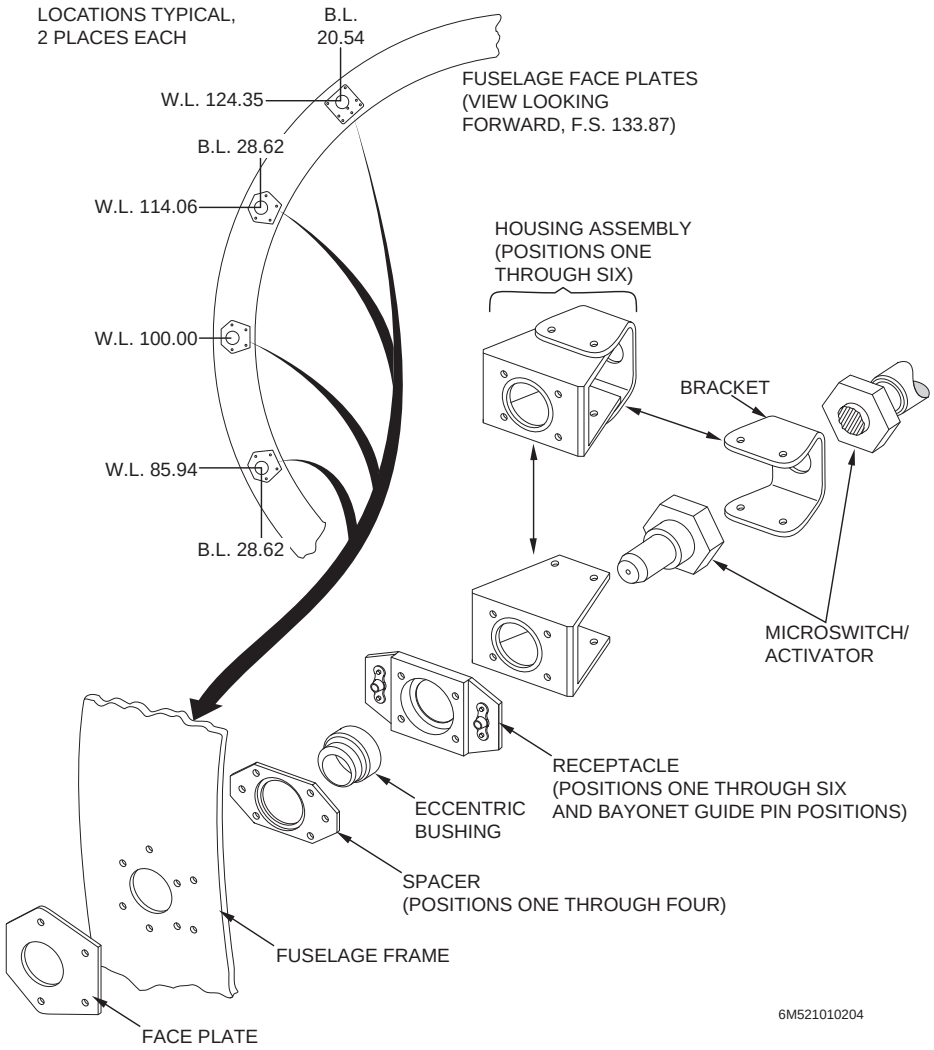
(2) Remove passenger door unlock warning switches for affected receptacle(s) (see 52-70-10).

(3) Remove screws and nuts securing retainer or fillers, bushing, and faceplate to fuselage door frame.

NOTE: Use care when removing screws and nuts to avoid dropping fillers and/or nuts behind retainer.

(4) Remove faceplate, bushing (side receptacles only) and retainer or fillers.

(5) Remove screws, washers and nuts attaching receptacle to fuselage frame and remove receptacle.



Fuselage Receptacles
Figure 204

F. Installation – Fuselage Receptacles (Figure 204)

(1) Position receptacle in place behind fuselage frame. Install screws, washers and nuts securing receptacle to fuselage frame.

NOTE: Use care to avoid dropping fillers and/or nuts.

(2) Position faceplate, bushing (side receptacles only) and retainer or fillers in place and secure with screws and nuts.

(3) Install door warning micro switch and actuator for affected receptacle(s) (see 52-70-10).

(4) Close door and actuate locking mechanism. Ensure that latches fully engage into receptacles and microswitch assemblies.

(5) Perform 3. A. Passenger Door Latch Mechanism Adjustment/Test, this section.

(6) Install interior covering.

3. Adjustment/Test – Passenger Door Latching System

A. Passenger Door Latch Mechanism Adjustment/Test

(1) With the passenger door open and handle in unlocked position, verify tips of bayonet guide pins are flush with the surface of their respective door frame bushing.

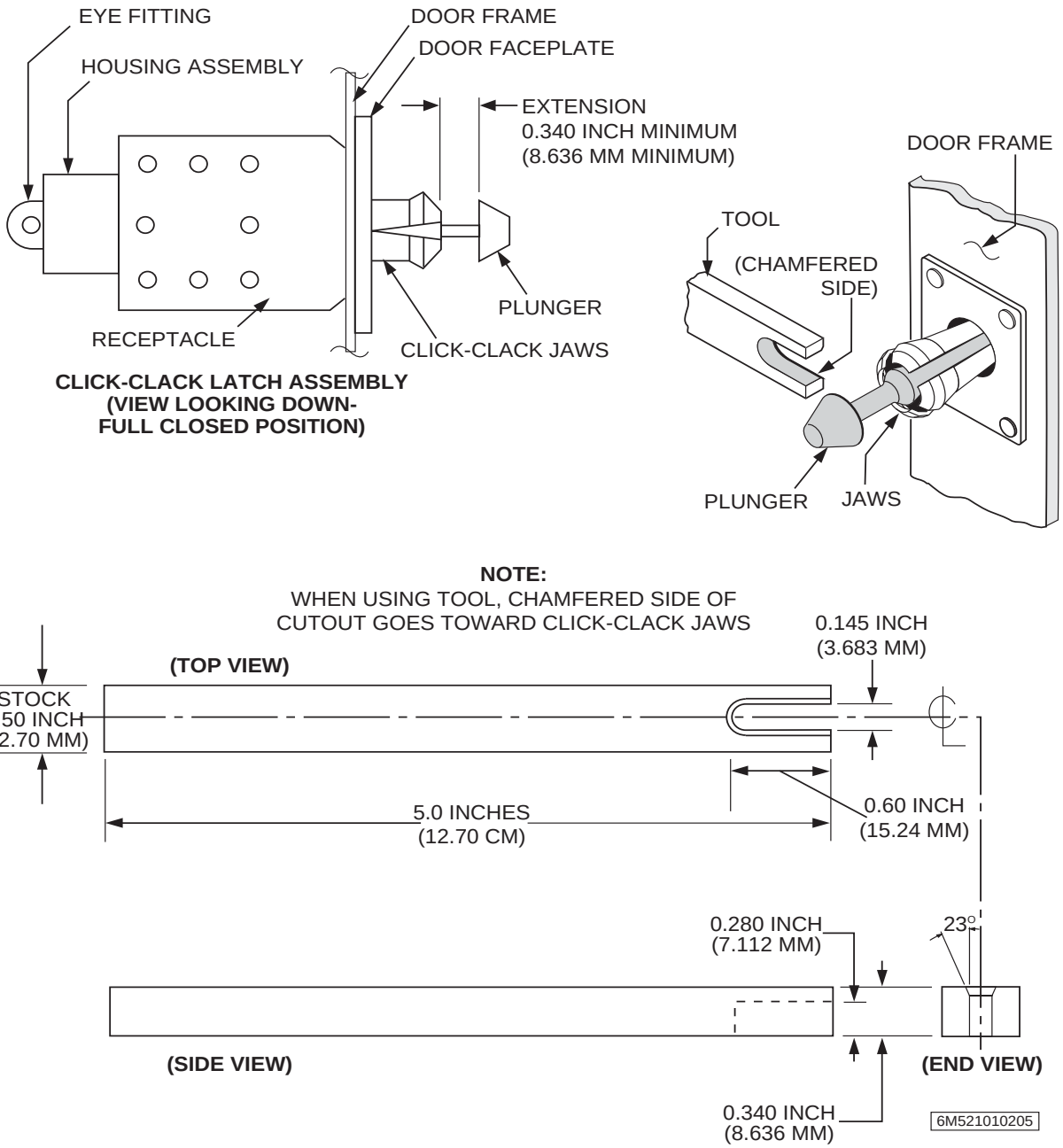
(2) With the passenger door open, conduct the following inspection on each latch to assure full expansion of the latch actuator jaws.

(a) Rotate the door handle to the door closed position.

(b) Refer to Illustrated Tool and Equipment Manual or Figure 205 for Go/No-Go Gauge P/N SAT1102.

(c) Using "go/no-go" gauge, verify the distance from inside of actuator plunger face to edge of the actuator jaws is not less than 0.34 inches (8.63 mm) (see Figure 205).

(d) Adjust each actuator as necessary to gain a minimum of 0.34 inches by varying the length of its connecting push-pull rod.



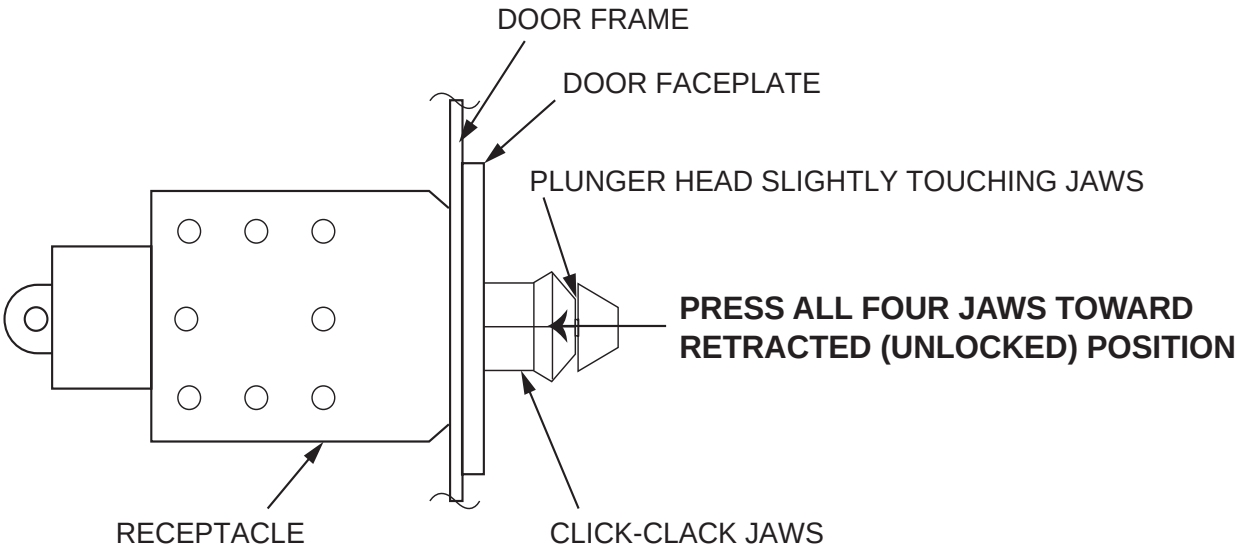
Click-Clack Latch Actuator Adjustmentt
Figure 205

(3) With passenger door open and the handle in LOCKED position, rotate handle back toward OPEN position until the plunger head lightly touches and holds all four jaws together (see Figure 206). Hold handle in this position.

(a) Press jaws of each latch actuator toward retracted position (UNLOCKED).

(b) Rotate actuator jaws and repeat at several positions around plunger.

(c) If any jaw can be pushed inward by exerting finger pressure, reject latch actuator.



Click-Clack Latch Actuator Jaw Integrity Test
Figure 206

(4) Using an inside micrometer, or equivalent, measure and record the inside diameter of each receptacle (or bushing if installed) located in the passenger door frame.

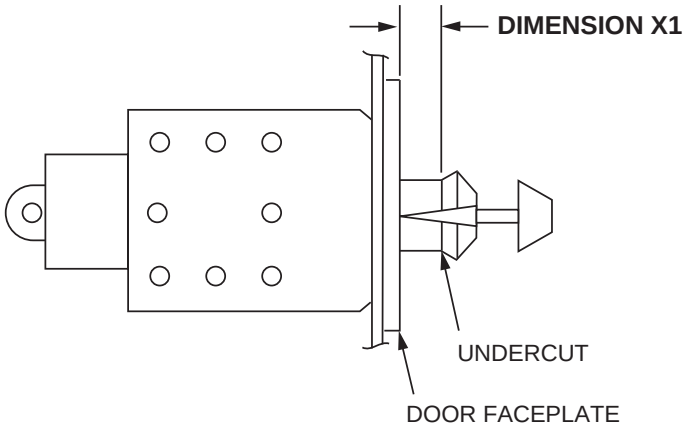
(a) Measure across the hole in at least three directions to check for roundness. The widest dimension must be used to compare with the maximum allowable dimensions provided in Table 201.

(b) If any receptacle is oversized, replace with an airworthy part.

(c) Determine the X2 value from Table 201 for each receptacle (or bushing) based on inside diameter measured. Record X2 value in Table 202.

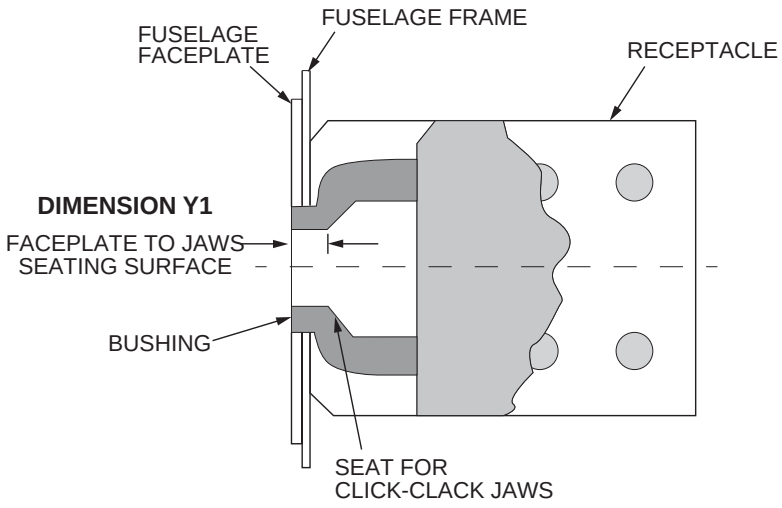
(5) To assure proper adjustment of the latch actuator jaws into the door frame receptacle, three additional measurements are necessary:

(a) The first measurement (Dimension X1) is the dimension from the door face plate surface to the undercut on the actuator jaws with the door open and handle in LOCKED position (see Figure 207). Measure and record as Dimension X1 in Table 202 for each latch position.



Dimension X Measurement
Figure 207

(b) The second measurement (Dimension Y1) is the dimension from the fuselage frame receptacle face plate surface to the actuator jaws seating surface inside the fuselage frame receptacle with door open (see Figure 208). Measure and record as Dimension Y1 in Table 202 for each latch position receptacle.



Dimension Y1 Measurement
Figure 208

POSITION	TYPE	LIMIT (INSIDE DIAMETER
Bottom	Receptacle Only (No Bushing)	0.670 inches (17.018 mm)
Side	Receptacle With Bushing	0.700 inches (17.78 mm)
Side	Receptacle Only (No Bushing)	0.690 inches (17.526 mm)

X2 FOR ENLARGED RECEPTACLE/BUSHING HOLES								
RECEPTACLE/BUSHING DIAMETER								
Inches	0.630	0.640	0.650	0.660	0.670	0.680	0.690	0.700
Millimeters	16.002	16.256	16.510	16.764	17.018	17.272	17.526	17.780
X2								
Inches	0.000	0.006	0.012	0.018	0.024	0.030	0.036	0.042
Millimeters	0.000	0.1524	0.3048	0.4572	0.6096	0.7620	0.9144	1.0668

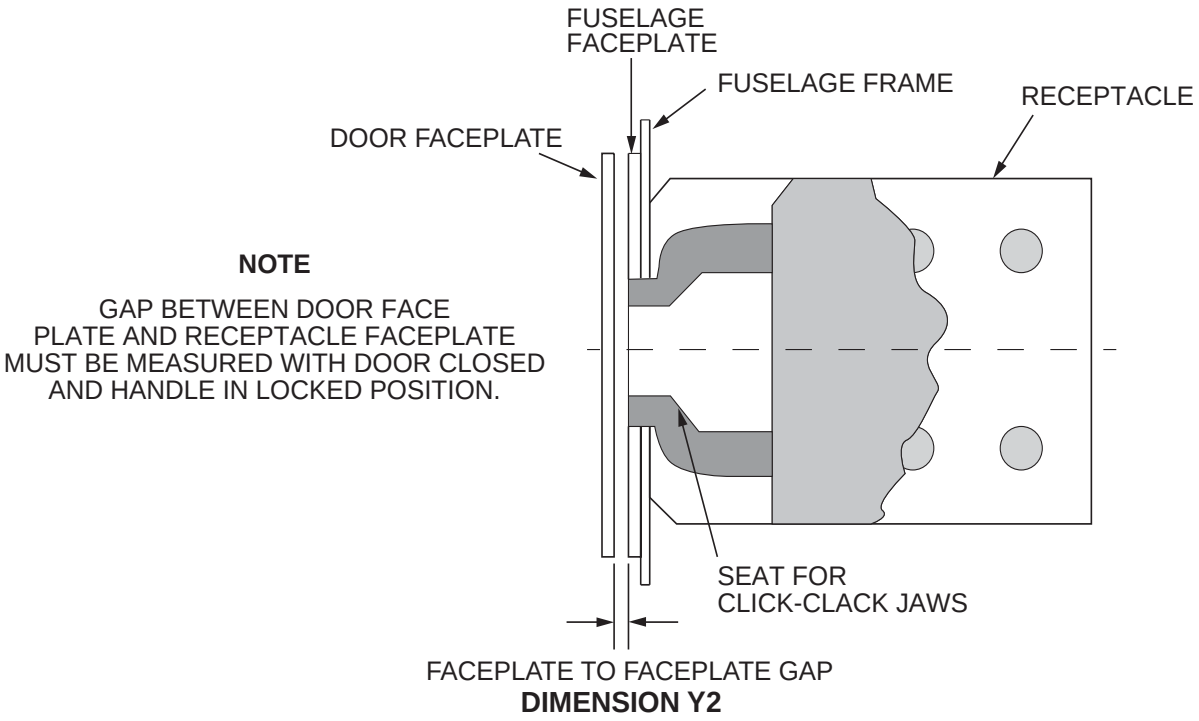
Door Receptacle/Bushing Wear Limits and X2 Value
Table 201

NOTE: When replacing bushings, check dash number marked on bushing. If number has an S, contact M7 Aerospace Field Service Engineering for replacement bushing, (reference MRBSR-512 to Service Engineering).

LATCH POSITION ►	FWD TOP	FWD CNTR	FWD LWR	AFT TOP	AFT CNTR	AFT LWR	TOP
X1							
X2							
X = X1 + X2							
Y1							
Y2							
Y = Y1 + Y2							
Is X ≥ Y? (Yes/No)							

Latch Engagement Dimensions
Table 202

(c) The third measurement (Dimension Y2) is the gap (taken with a feeler gauge with door closed and handle in locked position) between the surface of the fuselage frame face plate and the door frame face plate. Measure and record as Dimension Y2 in Table 202 for each latch position (see Figure 209).



Dimension Y2 Measurement
Figure 209

(6) In Table 202 for each door latch position:

(a) Add X1 to X2 and enter sum as X dimension.

(b) Add Y1 to Y2 and enter sum as Y dimension.

(c) Determine if X dimension is equal to or greater than dimension Y and enter Yes or No as applicable.

(d) Dimension X must be equal to or greater than dimension Y. If Dimension X is less than dimension Y, replace the fuselage receptacle/bushing with an airworthy part and repeat above measurements to obtain X and Y dimensions.

4. Inspection/Check – Passenger Door Latching System

A. Inspect Passenger Door Latching System as follows:

(1) Check passenger door latching mechanisms for general condition, security, and proper operation.

(a) Inspect bellcrank assemblies, push-pull rods, and rod ends for general condition, alignment, wear, and security.

(b) Check click-clack latches and bayonet guide pins for adjustment, nicks, burrs, alignment, general condition, and proper operation.

(c) Clean and lubricate click-clack latches and bayonet guide pins.

(2) Inspect door faceplates for evidence of deformation.

(3) Inspect door structure for cracks, loose rivets, and general condition.

(4) Check plastic inspection covers for condition and security.

(5) Check door handles for condition and security.

(6) Inspect, clean and lubricate lock mechanism on door handle.

(7) Inspect fuselage frame receptacle openings and faceplates for condition and evidence of pulling out toward door.

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PASSENGER DOOR STEPS – MAINTENANCE PRACTICES

1. Removal/Installation – Passenger Door Steps

A. Removal – Door Steps (Figure 201)

(1) Position cradle to support door when door is released from supporting mechanisms. Cradle should have suitable surface to avoid damage to door finish or material.

(2) Open passenger door.

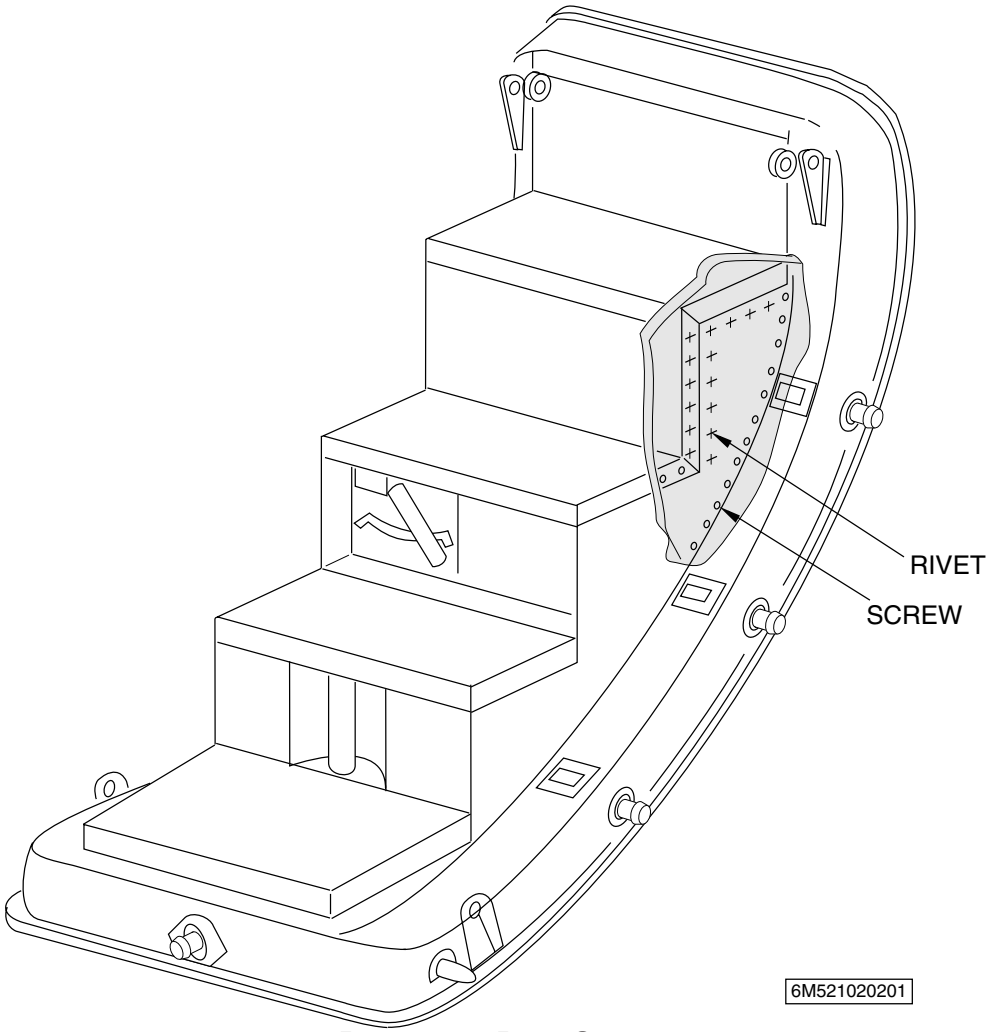
(3) Remove door IAW Removal/Installation – Passenger Door.

(4) Remove upholstery from door steps.

(5) Remove screws in step boards and remove boards.

(6) Drill out rivets in risers and inside rails. Remove screws from side rails as necessary.

(7) Remove door step assembly from passenger door.



Passenger Door Steps
Figure 201

B. Installation – Door Steps (Figure 201)

(1) Install door step assembly to passenger door.

(a) Align and rivet side rails to door. Install screw as applicable.

(b) Align and rivet risers to side rails.

(2) Secure step boards with screws.

(3) Install upholstery.

(4) Install door to aircraft IAW Removal/Installation – Passenger door.

(5) Perform water leak test IAW Adjustment/Test – Passenger Door Seal.

NOTE: Door seal should prevent entry of water into aircraft even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

2. Inspection/Check – Passenger Door Steps

A. Inspect passenger door steps as follows:

(1) Inspect step surfaces for cuts, cracks, scratches, dents, security, and corrosion.

(2) Inspect general condition of carpet for wear, loose edges, tears, or gouges.

PASSENGER DOOR HANDLES AND SAFETY LOCK – MAINTENANCE PRACTICES

1. Removal/Installation – Passenger Door Handles and Safety Lock

A. Removal – Inner Handle (Figure 202)

(1) Remove retaining pin (item 1), and washer and nut (item 2) securing inner door handle to adapter assembly.

(2) Remove inner door handle.

NOTE: If passenger door is to be closed, an alternate method of entering/exiting aircraft should be available.

B. Installation – Inner Handle (Figure 202)

(1) With passenger door open, position inner door handle for installation.

(2) Install retaining pin (item 1), and washer and nut (item 2) securing door handle to adaptor assembly.

(3) With passenger door open, operate door handle several times to ensure proper operation.

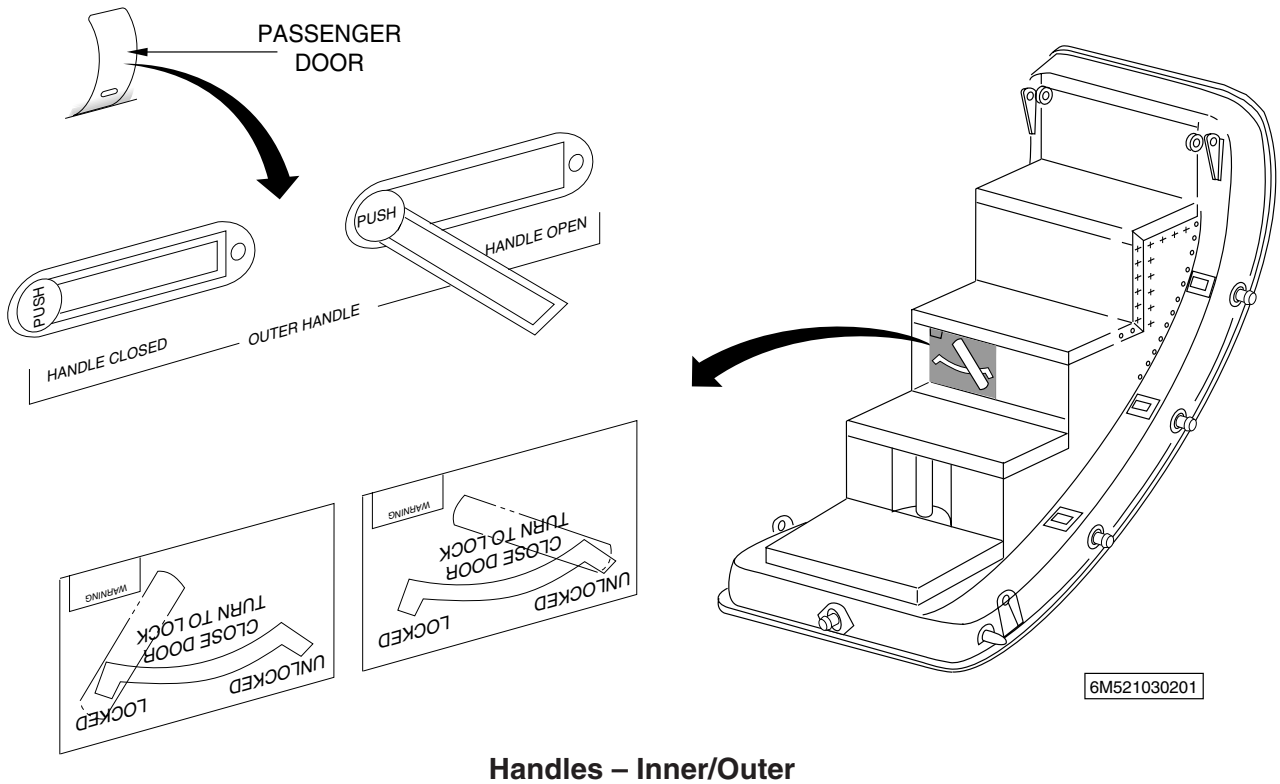
(4) Open and close passenger door slowly several times to ensure proper operation.

C. Removal – Outer Handle (Figure 202)

(1) Open passenger door.

(2) Remove upholstery from door step assembly.

(3) Remove screws securing top two step boards and remove boards.



Handles – Inner/Outer
Figure 201

PASSENGER DOOR HANDLES AND SAFETY LOCK – MAINTENANCE PRACTICES

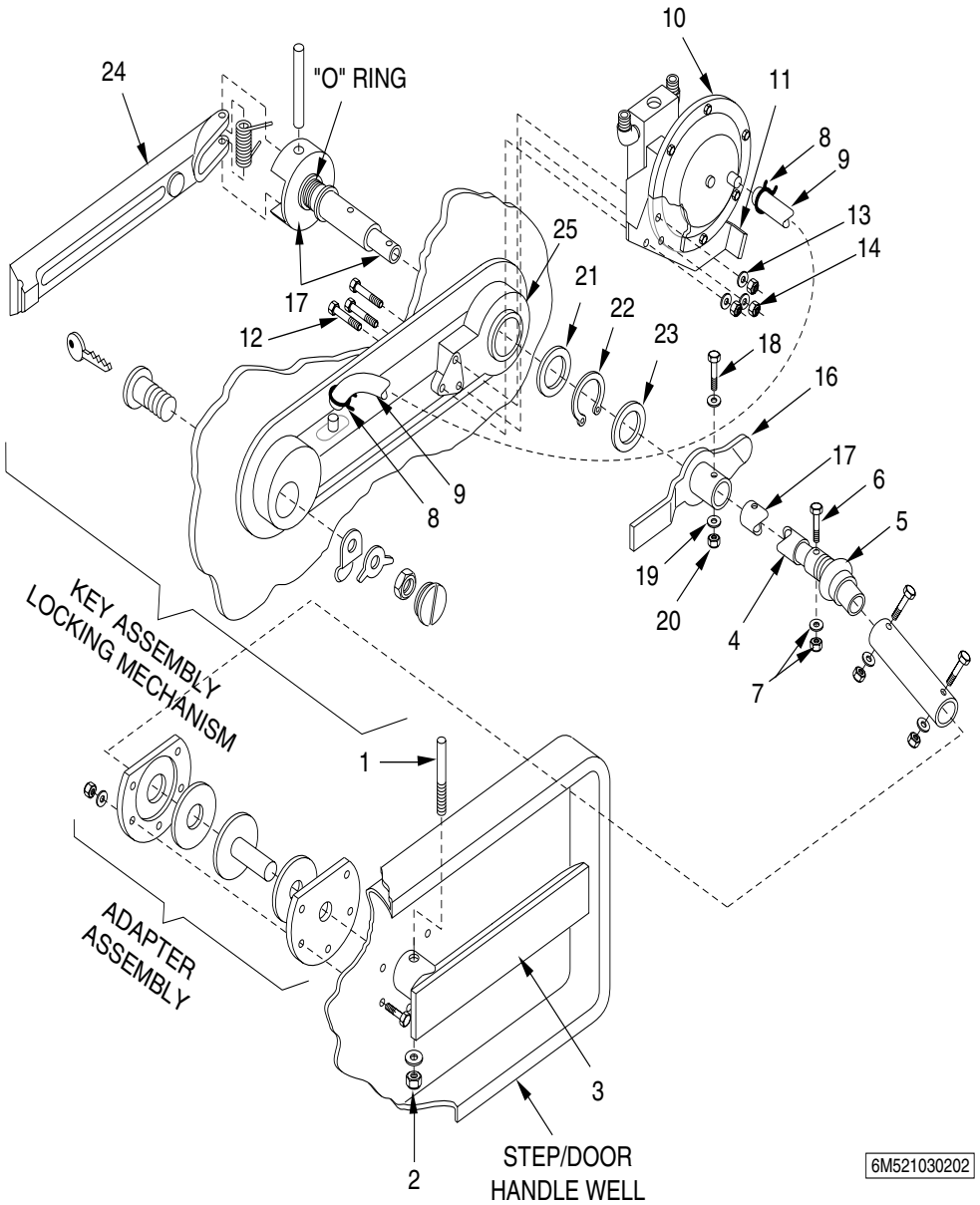
(4) Remove inner door handle (item 3) IAW Removal – Inner Handle.

(5) Disconnect shaft assembly (item 4) from universal joint assembly (item 5) by removing bolt, washer, and nut (items 6 and 7).

(6) Remove door step assembly IAW Removal/Installation – Passenger Door Steps.

CAUTION: IMMEDIATELY INSTALL CAPS OR PLUGS ON OPEN LINES TO PREVENT CONTAMINATION.

(7) Remove clamp (item 8) and disconnect tubing (item 9) from door seal safety lock assembly (item 10). Cap open tubing.



Door Handles – Internal Assembly
Figure 202

(8) Remove door seal safety lock and door stop (item 11) assemblies by removing screws, washers, and nuts (items 12, 13, and 14).

(9) Disconnect and remove push-pull rod assemblies from bellcrank assembly (item 16).

(10) Disconnect shaft assembly (item 4) from bellcrank assembly and actuator assembly (item 17) by removing screw, washer, and nut (items 18, 19, and 20).

CAUTION: LOCK OUTER DOOR HANDLE AND SECURE TO HOUSING WITH KEY TO PREVENT ACCIDENTAL RELEASE AND POTENTIAL DAMAGE TO OUTER DOOR HANDLE ASSEMBLY.

(11) Remove bellcrank assembly from outer door handle actuator.

(12) Remove snap ring and washers (items 21, 22, and 23) from outer door handle actuator assembly.

(13) Unlock, withdraw, and remove actuator/outer door handle assemblies from housing.

D. Installation – Outer Handle (Figure 202)

(1) Position and install outer door handle (item 24) actuator (item 17) assemblies through housing (item 25). Secure with washers and snap ring (items 21, 22, and 23).

(2) Install bellcrank assembly (item 16) to outer door handle actuator.

(3) Connect shaft assembly (item 4) to bellcrank and actuator assemblies. Secure with screw, washer, and nut (items 18, 19, and 20).

(4) Install and connect push-pull rod assembly to bellcrank assembly.

(5) Install door seal safety lock (item 10) and door stop (item 11) assemblies. Secure with screws, washers, and nuts (items 12, 13, and 14).

(6) Remove cap from tubing (item 9). Install tubing to door seal safety lock assembly at inner door cover assembly. Secure with clamp (item 8).

(7) Install door step assembly IAW Removal/Installation – Passenger Door Steps.

(8) Connect shaft assembly (item 4) to universal joint assembly (item 5) with bolt, washer, and nut (items 6 and 7).

(9) Install inner door handle (item 3) IAW Installation – Inner Handle.

(10) Operate door handle several times to ensure proper functioning.

(11) Install top two step boards.

(12) Install upholstery to door step assembly.

E. Removal – Passenger Door Safety Lock (Figure 202)

(1) Complete steps **1.**, C., (1) through (7) to gain access to door safety lock assembly (item 10).

(2) Remove screws, washers, and nuts (items 12, 13, and 14) holding door seal safety lock assembly and door stop assembly to outer handle housing (item 25).

(3) Remove door seal safety lock and door stop assemblies from door.

(4) Remove door seal safety lock assembly from door stop assembly IAW Safety Lock Diaphragm Assembly, this section.

F. Installation – Passenger Door Safety Lock (Figure 202)

(1) Assemble door seal safety lock assembly (item 10) to door stop safety lock assembly (item 11). Position for installation to outer door handle housing (item 25).

(2) Install door seal safety lock and door stop safety lock assemblies to housing. Secure with screws, washers, and nuts (items 12, 13, and 14).

(3) Check valve in door seal safety lock for adjustment IAW Para. **2.** Adjustment/Test – Passenger Door Safety Lock.

(4) Complete steps **1.**, D., (7) through (13)

2. Adjustment/Test – Passenger Door Safety Lock

A. Perform the following tests after installing lock assembly to door.

NOTE: Door handle must be in locked position.

(1) Apply 9.60 psi (0.675 kg/cm^2) differential pressure to lock assembly pan by applying vacuum to pan inlet. Verify no excessive distortion or permanent set.

(2) With increasing pressure, apply 1.50 psi (0.105 kg/cm^2) to lock assembly and hold. Verify door mechanism is locked and door handle is restrained from moving to unlocked position.

(3) Reduce pressure to 0.50 psi (0.035 kg/cm^2). Verify door handle will unlock.

NOTE: Do not set 1.50 psi (0.105 kg/cm^2) with decreasing pressure or the 0.50 psi (0.035 kg/cm^2) with increasing pressure.

3. Inspection/Check – Passenger Door Handles and Safety Lock

A. Inspection/Check – Safety Lock System

(1) Open passenger door.

(2) Remove upholstery and step boards as necessary to gain access to safety lock.

(3) Close passenger door.

WARNING: ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF PERSONNEL AND OBJECTS DURING ENGINE RUNS.

(4) Run engines IAW AFM and build up cabin pressure. With an inspection mirror, check safety lock for proper operation.

NOTE: Refer to AFM for engine ground run procedures.

B. Inspection/Check – Handles

(1) Inspect components of mechanisms and assemblies for corrosion, wear, signs of failure, and security.

(2) Inspect all supporting structures for security, corrosion, wear, and signs of failure.

PASSENGER DOOR SEAL VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – Passenger Door Seal Valve

A. Removal – Door Seal Valve (Figure 201)

(1) Open passenger door.

(2) Remove retaining pin, washer, and nut from inner door handle. Remove inner door handle.

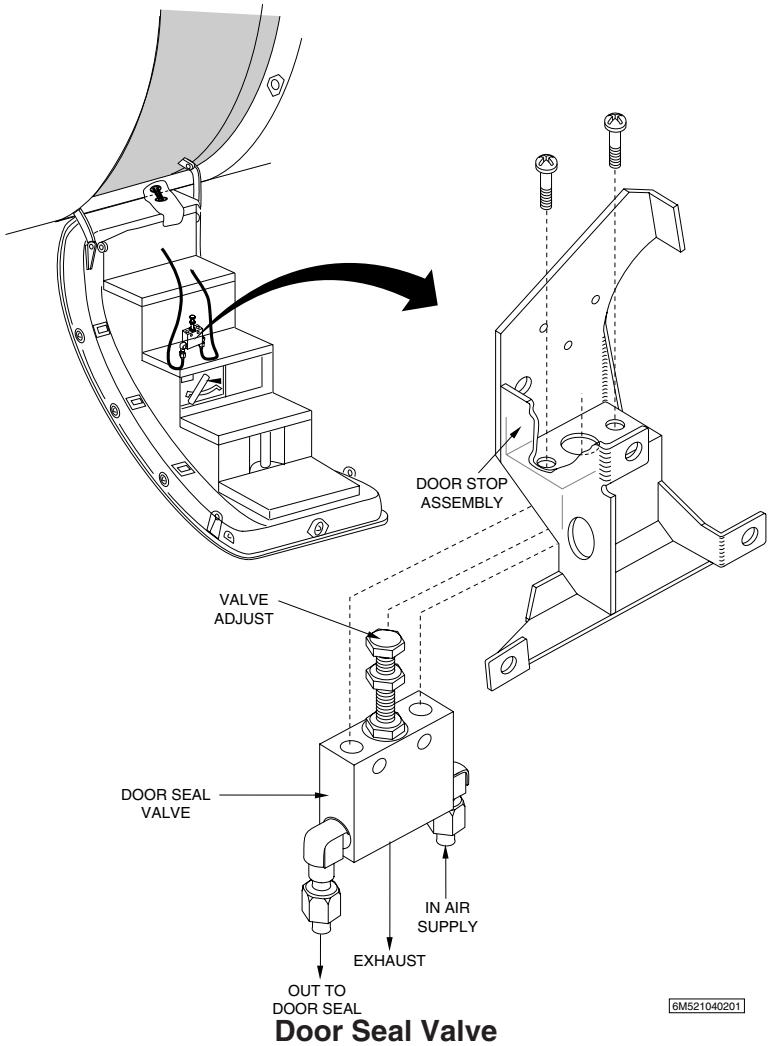
NOTE: Ensure an alternate entry/exit is available in event door becomes closed.

(3) Remove upholstery from upper two step boards.

(4) Remove upper two step boards.

CAUTION: IMMEDIATELY INSTALL CAPS OR PLUGS ON OPEN LINES TO PREVENT CONTAMINATION.

(5) Remove clamp and disconnect tubing from door safety lock assembly at inner door cover assembly. Cap tubing.



Door Seal Valve
Figure 201

(6) Remove clamps and disconnect tubes from door seal valve. Cap open tubing.

(7) Remove screws securing door seal valve to door stop assembly.

(8) Remove door seal valve from door stop assembly.

B. Installation – Door Seal Valve (Figure 201)

(1) With passenger door open, position new door seal valve to door stop assembly. Ensure valve is properly positioned for correct tube connections.

(2) Install door seal valve to door stop assembly. Secure with screws.

(3) Remove caps from tubing. Connect tubes to door seal valve. Secure with clamps.

(4) Check door seal valve for adjustment IAW Para. **2.** Adjustment/Test – Passenger Door Seal.

(5) Remove cap from tubing. Connect tube to door safety lock assembly. Secure with clamp.

(6) Install upper two step boards.

(7) Install upholstery on two upper step boards.

(8) Install inner door handle. Secure with pin, washer, and nut.

2. Adjustment/Test – Passenger Door Seal Valve

A. Adjustment – Door Seal Valve (Figure 201)

(1) Open passenger door.

(2) Remove upholstery and step boards as necessary to gain access to door seal valve IAW **1.**, A. – Removal – Door Seal Valve

NOTE: Ensure an alternate entry/exit is available in event door becomes closed.

(3) Close passenger door.

CAUTION: DO NOT APPLY MORE THAN 18 PSI (1.27 KG/CM^2) PRESSURE.

(4) Apply 15 psi (1.05 kg/cm2) air pressure to upstream side of door seal valve.

WARNING: ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF PERSONNEL AND OBJECTS DURING ENGINE RUN.

(a) As an alternate source of air pressure from an external source, one engine may be operated.

NOTE: Refer to AFM for engine ground run procedures.

(5) While rotating inner door handle to LOCKED position, observe door seal to ensure seal is inflating.

(6) Adjust door seal valve so that seal inflates at approximately 90° handle travel.

(7) Check to ensure door seal deflates before inner door handle reaches full UNLOCKED.

(8) Install step boards and upholstery to step boards.

PASSENGER DOOR SEAL – MAINTENANCE PRACTICES

1. Servicing – Passenger Door Seal

A. Lubricate door seal at regular intervals using Tri-Flow (aerosol), Dri-Slide (liquid), MIL-L-23398 (aerosol), or NATO #S-749.

NOTE: To preclude seal deterioration and reduce dust and dirt contamination, do not use lubricants that leave an oily or greasy coating.

2. Removal/Installation – Passenger Door Seal

A. Removal – door seal.

(1) Position cradle outside aircraft to support door when door is released from supporting mechanism. Cradle should have suitable surface to avoid damage to door finish or material.

(2) Close and secure passenger door.

(3) Remove interior floor covering inside passenger door.

(4) Working from inside of aircraft, with passenger door closed, remove snubber bolts from bracket at fuselage frame.

(5) Disconnect handrail chain assemblies from door frame bracket.

(6) Holding handrail chain, open door slowly and lower door to cradle.

(7) Remove cabin floorboard immediately adjacent to door opening.

CAUTION: IMMEDIATELY INSTALL CAPS OR PLUGS ON OPEN LINES TO PREVENT CONTAMINATION.

(8) Locate and disconnect door seal inflation tubing at door seal. Cap open lines.

(9) Remove old seal from door frame.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(10) After old seal is removed, clean door frame area thoroughly with MPK.

B. Installation – door seal.

(1) Position door cradle outside aircraft.

(2) Clean new door seal thoroughly with MPK to remove shipping preservatives. Ensure door frame area, where seal is to be installed, is thoroughly clean prior to installation of new door seal. Clean area with MPK, if necessary.

CAUTION: EXCESS SEALER BETWEEN SKIN AND SEAL WILL CAUSE CONTOUR DEVIATION AND UNSATISFACTORY OPERATION OF DOOR. USE CARE NOT TO SPREAD SILICONE SEALANT OR ALLOW IT TO CONTACT SKIN SURFACE. IMMEDIATELY DISPOSE OF RAGS OR TOWELS USED TO CLEAN UP EXCESS SILICONE SEALANT.

(3) Run a bead of Dow Corning RTV 3145 clear sealer, approximately 0,125 inch (3.175 mm)

wide all the way around door at seal seat.

PASSENGER DOOR SEAL – MAINTENANCE PRACTICES

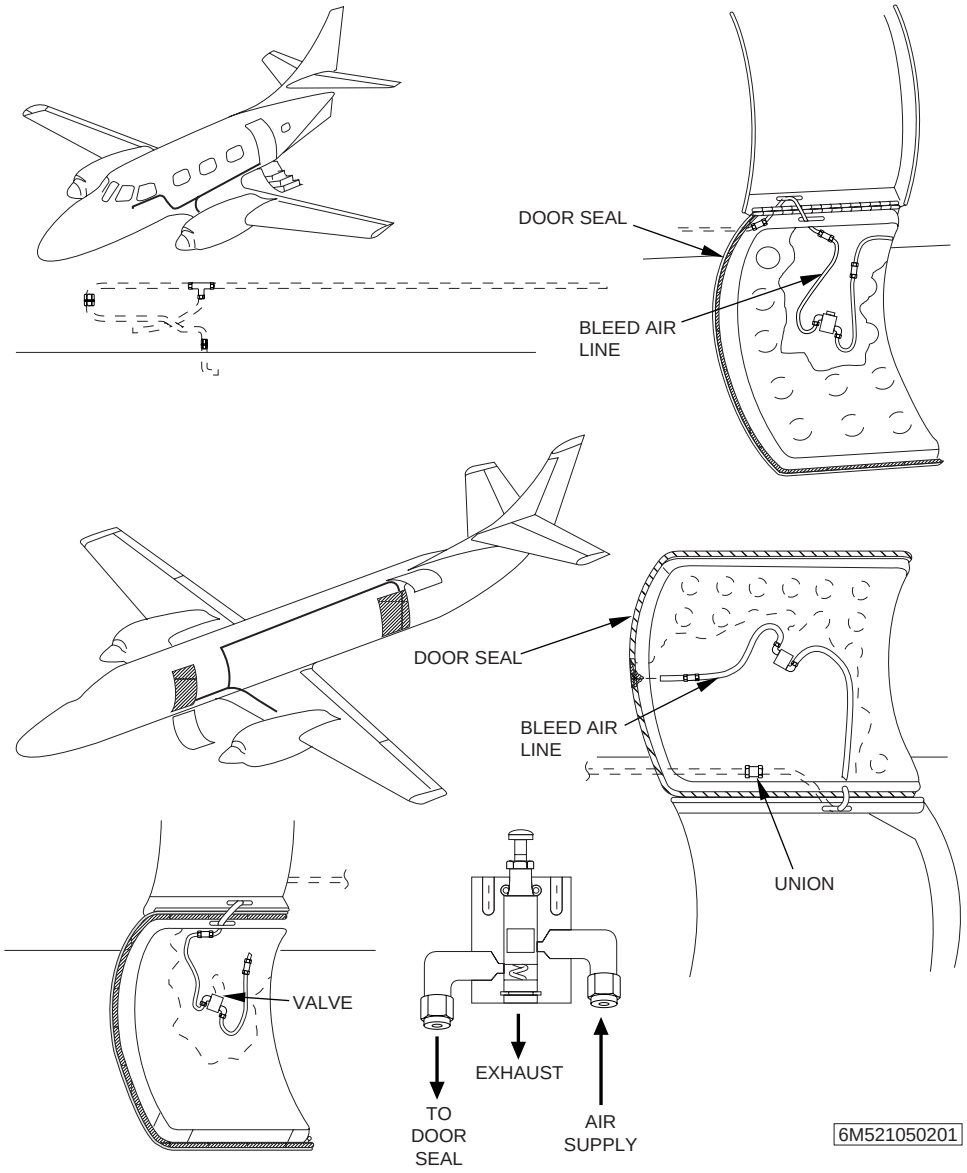
CAUTION: SEAL MUST BE SLID ALONG SKIN INTO CORNERS. DO NOT SLIDE SEAL ALONG FRAME INTO CORNERS.

(4) Beginning at inflation tube, install seal all the way around door.

(a) Using a sliding action, work from edge of skin into structure and seal seat.

(b) When seal is in place, ensure no twists exist in seal around door.

(5) Check contour of ridged sealing area of door seal in the area of splices (should not exceed three splices). Ensure contour does not deviate by more than -0.020 inch (-0.508 mm) from the contour on either side of splice(s).



(6) When the seal is completely and satisfactorily installed, carefully close door. Allow seal to dry in place (approximately two hours).

(7) When seal is dry, open door carefully.

(8) Connect handrail chain assemblies to door frame brackets.

(9) Open door.

(10) Remove caps from tubing.

(11) Connect bleed air tubing and seal inflation tube.

(12) Install upper two step boards.

(13) Install cabin floorboard.

(14) Install upholstery to upper two step boards.

(15) Install interior floor covering inside passenger door.

(16) Close door. Check alignment of door seal to fuselage frame.

WARNING: ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF PERSONNEL AND OBJECTS DURING ENGINE RUN.

(17) Check seal for inflation and leaks IAW Para. 3. – Adjustment/Test – Passenger Door Seal. To provide pressure, operate one engine.

NOTE: Refer to AFM for engine ground run procedures.

3. Adjustment/Test – Passenger Door Seal

A. Pressurization Leak – Door Seal

(1) Check rubber seal for signs of wear, deterioration, proper contact, and pressurization leaks.

NOTE: Leaks normally appear as stains on the fuselage skin around door opening.

(2) If seal is checked and a leak is detected, seal should be shimmed IAW **3.**, B. Water Leak Test – Door Seal.

B. Water Leak Test – Door Seal

(1) Accomplish test with fuselage unpressurized.

NOTE: Door seal should prevent entry of water into aircraft even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(2) Position an assistant with a flashlight and marking pencil inside aircraft at passenger door.

(a) Use a flashlight to determine water seepage during test.

(b) Identify and mark water entry point(s) with marking pencil while water test is being conducted.

PASSENGER DOOR SEAL – MAINTENANCE PRACTICES

(3) Close and latch passenger door.

(4) Use water hose, insecticide sprayer, or similar device to dispose water under pressure.

(5) With nozzle approximately 2.0 to 3.0 inches (5.08 to 7.62 cm) from door, apply water completely around door edge for several minutes.

(6) Open passenger door. Check for evidence of seepage. No water leakage is allowed.

(7) If test indicates door is not sealing properly, peel seal away from door frame in area(s) of water leak.

(a) Use thin pieces of neoprene rubber to shim between door and seal.

(b) Secure seal using RTV 3145 silicone sealant or equivalent. Do not add more than necessary.

4. Inspection/Check – Passenger Door Seal

A. Inspect passenger Door Seal as follows:

(1) Check passenger door seal and door seal supply for security and general condition.

(2) Check contour of ridged sealing area of door seal, in area of splices, to ensure contour does not deviate by more than -0.020 inch (-0.508 mm) from the contour on either side of each splice.

PASSENGER DOOR SNUBBER – MAINTENANCE PRACTICES

1. Servicing – Passenger Door Snubber

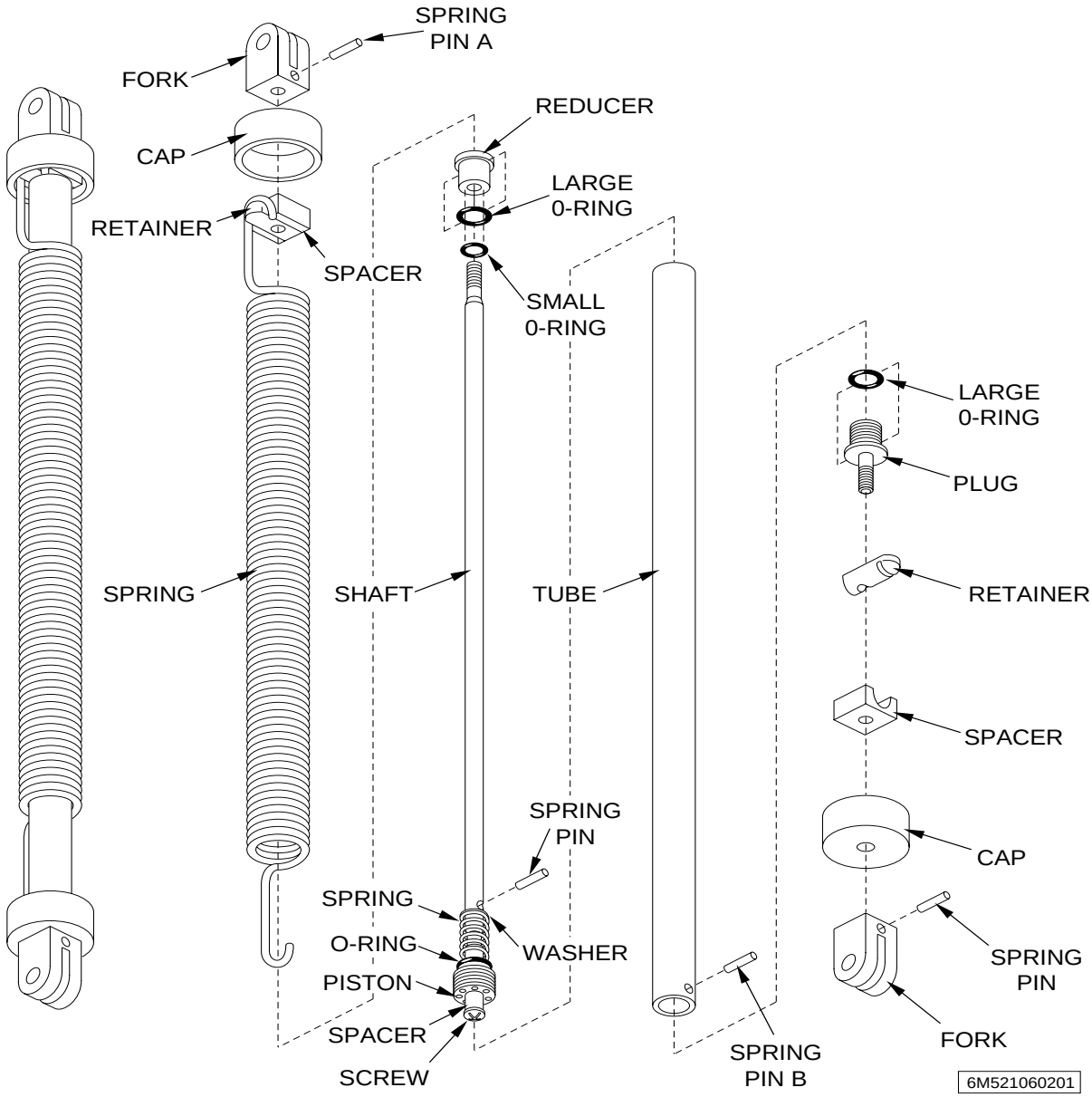
A. Replenishing Snubber Hydraulic Fluid

(1) Obtain Dow Corning 200 Fluid, 5 cs viscosity, clear hydraulic fluid and methyl propyl ketone (MPK).

(2) Install placards visible to outside and inside of aircraft at door area. Label placards: SNUBBER NOT INSTALLED. DO NOT DROP DOOR.

(3) Remove door snubber from aircraft.

(4) Partially close door to remove weight from the snubber.



Snubber – Servicing
Figure 201

(5) Disconnect upper end of snubber by removing screw from bracket located on door frame.

(6) Disconnect lower end of snubber by removing screw from bracket assembly on passenger door.

(7) Remove upholstered cover from snubber.

(8) Disassemble door snubber.

(9) Place snubber on table so that pin B is at the bottom.

(10) Using a punch, remove spring A from fork.

(11) Unscrew fork and remove cap, spacer, and retainer.

(12) Using pliers, remove reducer from tube.

(13) Remove shaft assembly from tube.

(14) Install new O-ring on piston.

(15) Empty tube of any remaining hydraulic fluid and clean inside of tube with MPK.

(16) Pour approximately two (2) ounces of new Dow Corning 200 fluid into tube.

(17) Insert shaft assembly into tube until 1 1/8 inches of the shaft extends above the tube. Work shaft in and out a few times. This will prevent any air pockets from forming within the tube.

(18) Fill tube with Dow Corning 200 fluid to 3/8 inches from top of tube.

(19) Install new O-rings to reducer. Small O-ring fits inside reducer; large O-ring fits over outside of reducer.

(20) Install reducer and tighten.

(21) Install retainer and spacer. Stretch spring slightly and hook the spring-end over the retainer.

(22) Install cap and fork.

(23) Install new spring pin through fork.

(24) Install upholstered cover.

(25) Install snubber to passenger door.

(26) Open and close door to ensure proper operation.

(27) Remove temporarily installed placards.

2. Removal/Installation – Passenger Door Snubber

NOTE: The following removal/installation procedures apply to single or dual snubber installation. If aircraft is equipped with dual snubber, repeat procedures for second snubber.

A. Removal – Passenger Door Snubber

(1) Partially close door to remove weight from snubber.

(2) Disconnect upper end of snubber by removing screw from handrail bracket.

(3) Disconnect lower end of snubber by removing screw from bracket assembly located on passenger door.

NOTE: Install placards visible to outside and inside of aircraft at door area. Label placards: SNUBBER NOT INSTALLED, DO NOT DROP DOOR.

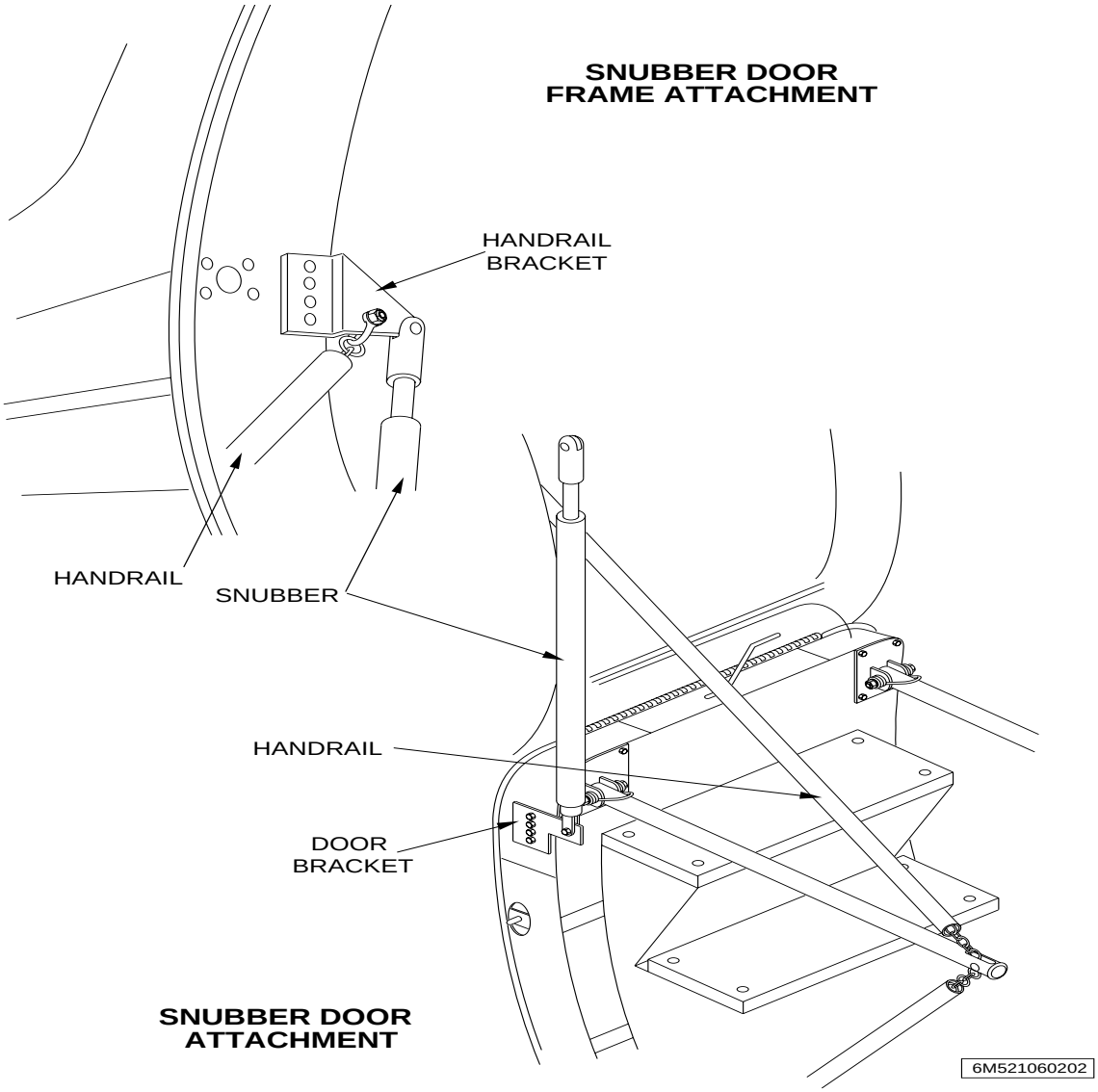
B. Installation – Passenger Door Snubber

(1) Install bottom of snubber to bracket assembly on passenger door.

(2) With door partially closed, install top of snubber to handrail bracket.

(3) Operate door several times to ensure proper functioning.

(4) Remove temporarily installed placards.



Passenger Door – Snubber Attachment
Figure 202

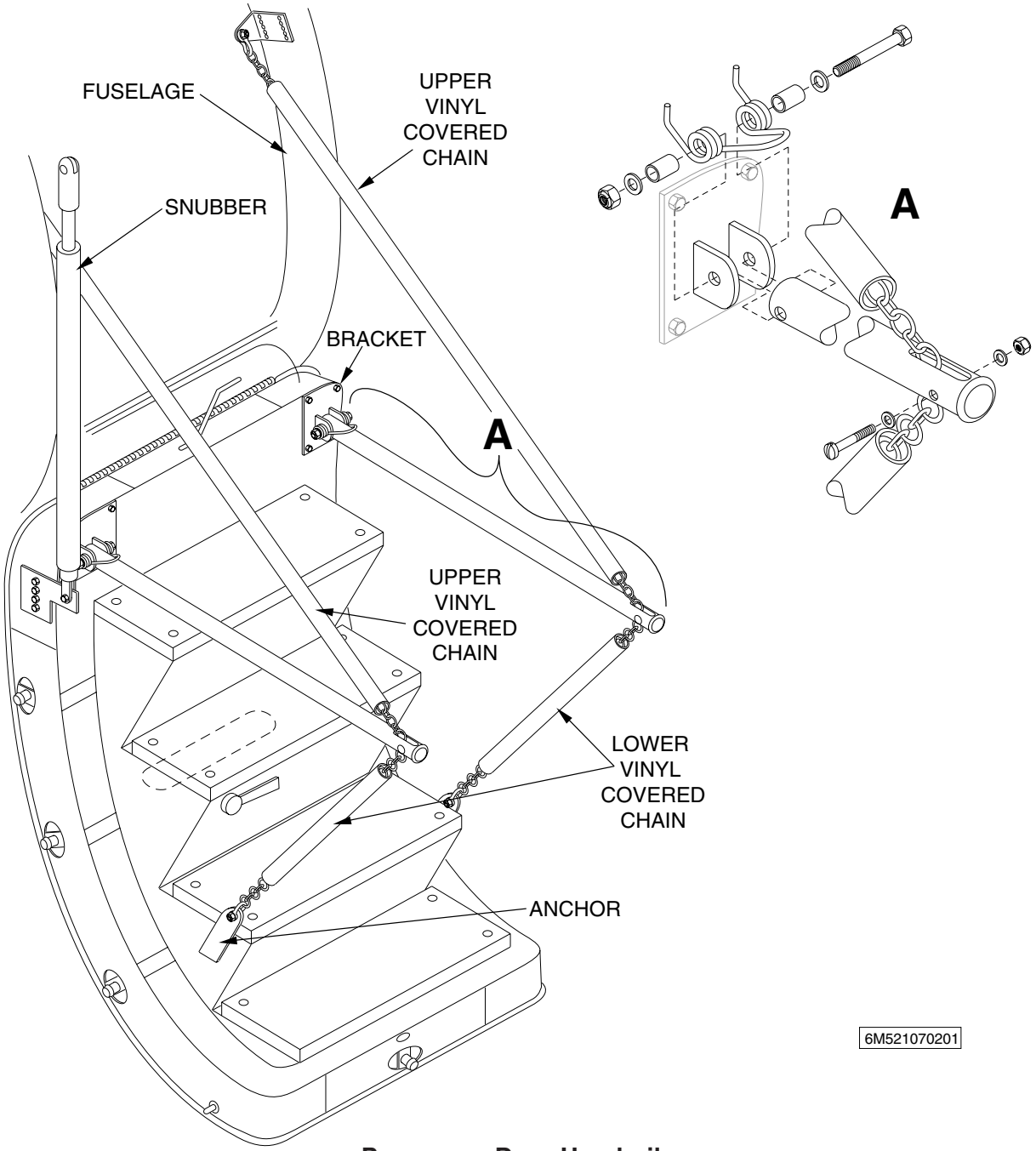
PASSENGER DOOR HANDRAILS – MAINTENANCE PRACTICES

1. Removal/Installation – Passenger Door Handrails

A. Removal – Passenger Door Handrails

(1) Partially close door to remove weight from snubber.

(2) Place support under door to relieve weight.



Passenger Door Handrails
Figure 201

(3) Install temporary placards at passenger door visible inside and outside of aircraft. Label placards: HANDRAILS NOT INSTALLED. DO NOT DROP DOOR.

(4) Disconnect chain at lower door anchor, and remove.

(5) Remove vinyl cover from chain.

(6) Disconnect upper end of lower chain from support.

(7) Disconnect upper chain from support assembly.

(8) Remove vinyl cover from chain.

(9) Disconnect upper chain from fuselage.

(10) Disconnect support assembly by removing bolt at bracket.

B. Installation – Passenger Door Handrails

(1) Install support assembly to bracket on passenger door.

(2) Connect upper chain to fuselage.

(3) Install vinyl cover on chain.

(4) Connect upper and lower chain to support assembly.

(5) Install vinyl cover on lower chain.

(6) Connect lower chain to lower anchor on passenger door.

(7) Remove temporary placards.

(8) Remove support holding door weight.

2. Adjustment/Test – Passenger Door Handrails

NOTE: The full weight of the door and loads applied to the door must be carried by the chains.

A. Test proper chain adjustment by standing on the lowest step of the door and checking the chains for slack. The chains should each carry an equal amount of the load applied to the door.

B. Adjustment may be made by twisting the chains or, if necessary, removing a link. Maladjusted chains will transmit loads to the snubber, that may lead to failure of the snubber attachments.

3. Inspection/Check – Passenger Door Handrails

A. Check passenger door handrails as follows:

(1) Check handrail chains, brackets, and anchors for general condition and security.

EMERGENCY EXITS – DESCRIPTION AND OPERATION

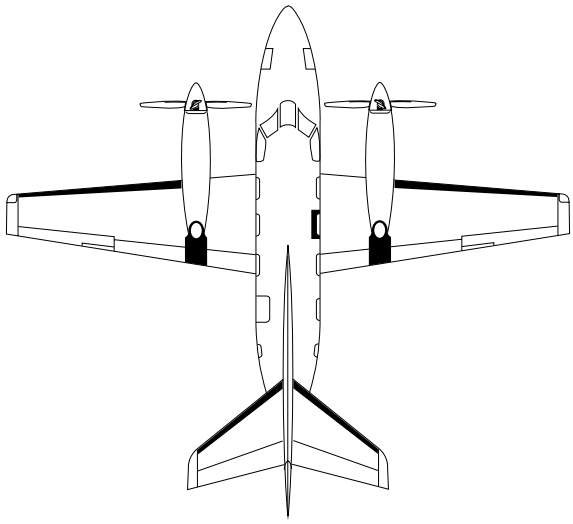
1. General

A. SA226 series aircraft emergency exits differ in location, size, and latching mechanisms.

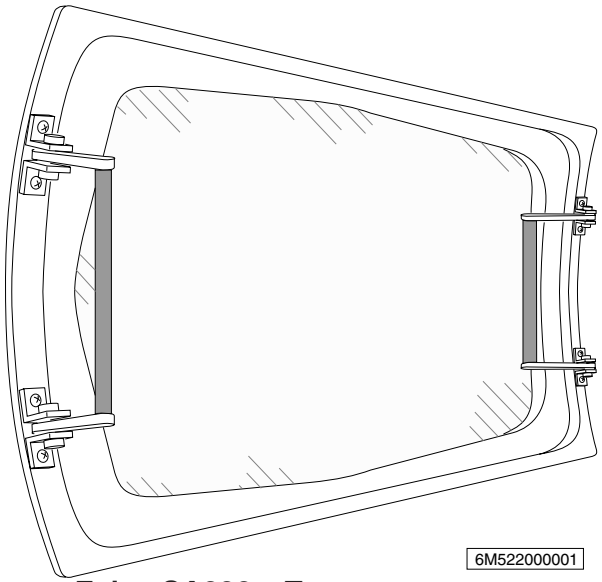
2. SA226 – T Aircraft

A. SA226 – T aircraft are equipped with one plug type emergency exit. It is approximately 19 inches (48.26 cm) wide by 26 inches (66.04 cm) high, and is located on the right side, over the wing. Cabin pressure is exerted against the hatch and holds it against the frame. The exit opens into the cabin.

B. Each hatch is held in the locked and sealed position by two over centering cams connected to the handle. The hatch base is recessed to fit over two guide pins. The over centering cams force two latch levers to engage in the fuselage latch cutouts. Springs on the cams assist in the locking operation and in holding the cams over center.



SA226 – T



Emergency Exit – SA226 – T
Figure 1

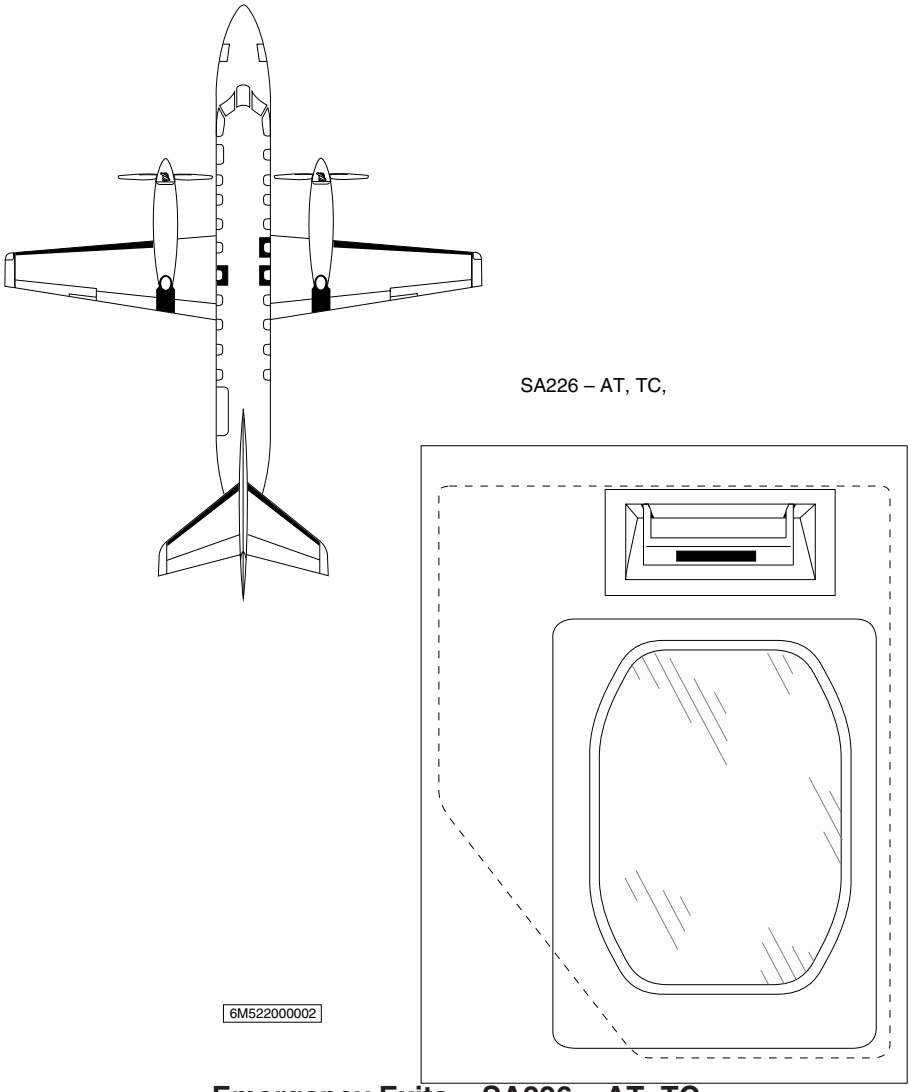
3. SA226 – AT and TC Aircraft

A. SA226 – AT, and TC aircraft are equipped with three plug type emergency exits. Each emergency exit incorporates a standard size cabin window.

(1) The emergency exits are approximately 20 inches (50.80 cm) wide by 28 inches (71.12 cm) high, and are opened by internal latches. Each latch is labeled to show method of operation, and is recessed to prevent inadvertent release.

(2) Two emergency exits are located over the right wing between stringers 4 and 8, and between F.S. 249.71 and 271.27, and F.S. 281.51 and 304.07. The third emergency exit is located over the left wing between stringers 4 and 8, and between F.S. 281.51 and 304.07. All emergency exits open into the cabin.

B. The forward passenger door and aft cargo door may also be used as emergency exits during ground operations.



Emergency Exits – SA226 – AT, TC
Figure 2

EMERGENCY EXITS – MAINTENANCE PRACTICES

1. Removal/Installation – Emergency Exits

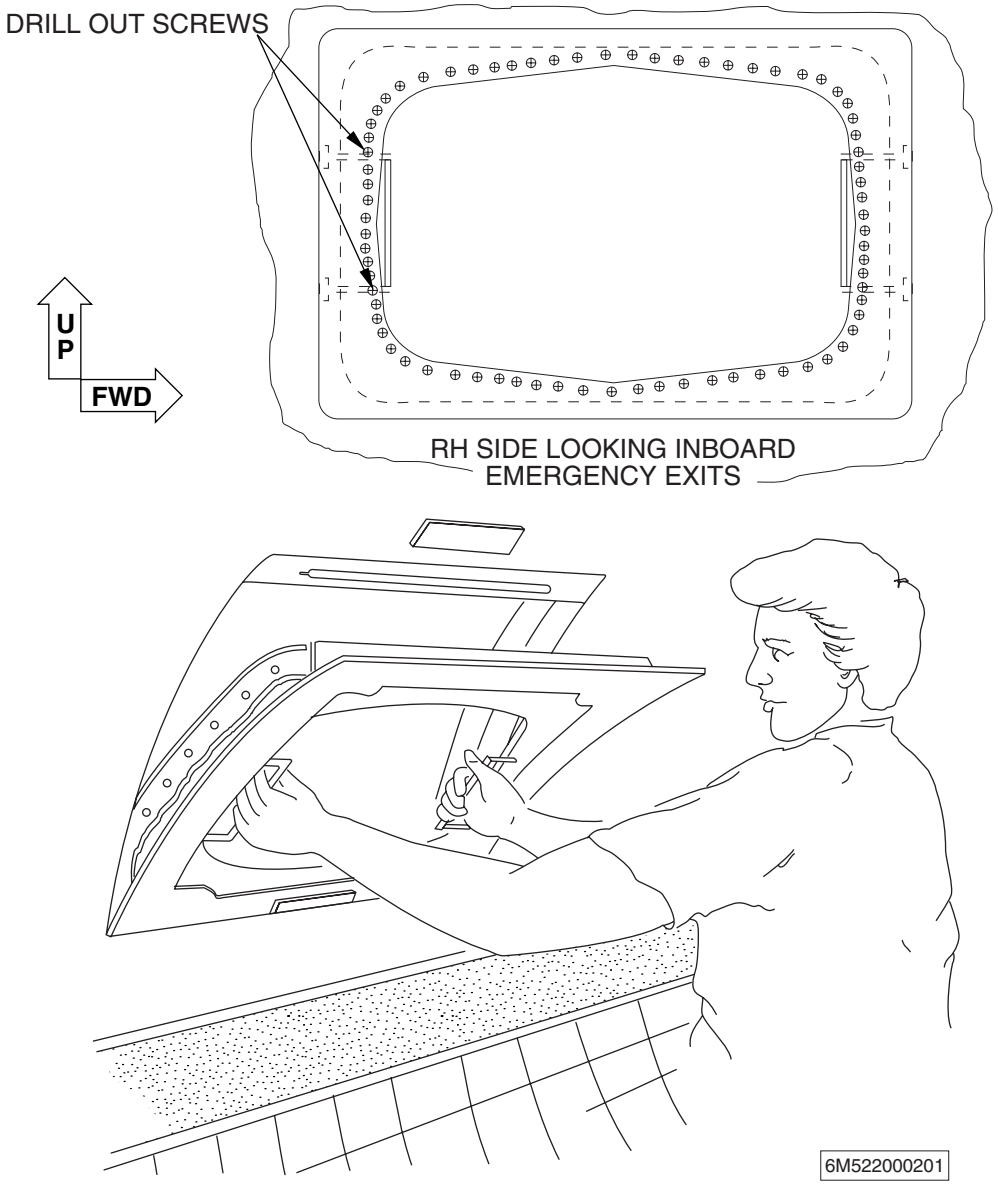
A. Removal – SA226 – T Emergency Exits (Figure 201)

(1) Pull both emergency exit hatch handles inward.

(2) Tilt top of emergency exit hatch into cabin.

(3) Lift emergency exit hatch clear of frame pins.

(4) Remove emergency exit hatch.



Emergency Exit - Removal – SA226 – T
Figure 201

B. Installation – SA226 – T Emergency Exits (Figure 201)

(1) Position emergency exit hatch for installation into opening.

(2) Insert emergency exit hatch. Mate assembly base with fuselage frame guide pins.

(3) Push emergency exit hatch outboard into closed position.

(4) Push both emergency exit handles in to engage latches.

C. Removal – SA226 – AT, TC Emergency Exits (Figure 202)

(1) Pull handle inboard and up to release emergency exit hatch.

(2) Pull top of emergency exit hatch inboard.

(3) Lift emergency exit hatch to clear fuselage frame guide pins.

(4) Remove emergency exit hatch.

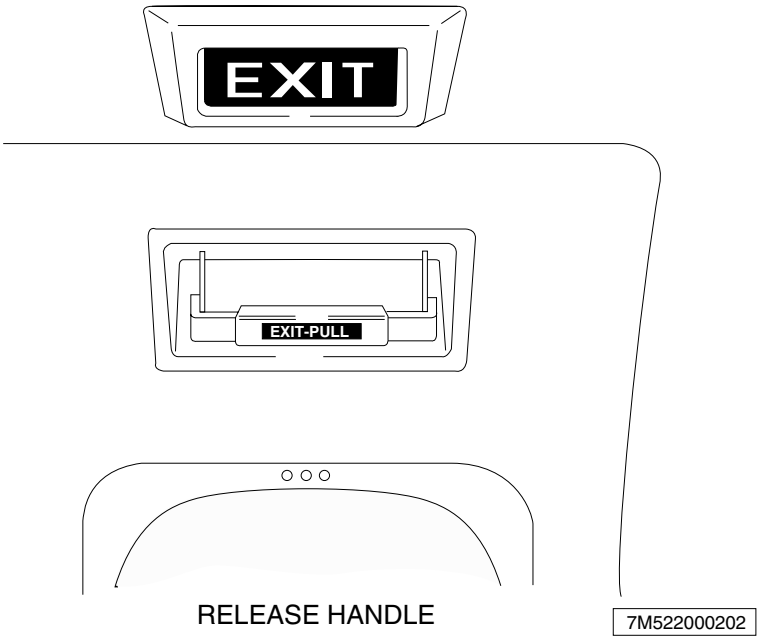
D. Installation – SA226 – AT, TC Emergency Exits (Figure 202)

(1) Position emergency exit hatch for installation into opening.

(2) Insert emergency exit hatch. Mate assembly base with fuselage frame guide pins.

(3) Push top of emergency exit hatch outboard into closed position.

(4) Push emergency exit handle down to engage latches.



Emergency Exit - Removal – SA226 – AT, TC
Figure 202

E. Removal – Emergency Exit Hatch Bulb Seal

(1) Remove screws securing bulb seal and retainer to emergency exit hatch.

(2) Remove bulb seal and retainer.

F. Installation – Emergency Exit Hatch Bulb Seal

(1) Position bulb seal and retainer to hatch with bulb seal between retainer and hatch.

(2) Ensure bulb seal splice is on bottom of hatch.

(3) Align holes in retainer to match holes in hatch.

(4) Carefully drill one hole through bulb seal and install screw.

(5) Following procedure in **1.**, F., (4) drill remaining holes and secure with screws.

G. Emergency entrance to aircraft (SA226 – T Only

NOTE: Should the passenger door latching mechanism malfunction with the door closed, emergency entrance to the aircraft is made from the outside through the escape hatch on the right side of the fuselage.

(1) Removal Procedure:

(a) Drill the heads off the two screws directly above the aft handle arms as shown in Figure 201. Do not use a screwdriver to remove the screws.

(b) Drive the screw shanks inward and using small drifts, force the arms of the handle forward and force the locking dogs to the unlocked position.

(c) Push the escape hatch inward to open.

(2) Installation Procedure:

(a) Install new screws and seal installation with sealant prior to reinstalling escape hatch.

(b) Mate hatch base with fuselage frame guide pins, push hatch into closed position.

(c) Push handle in to engage latches.

NOTE: It is advisable to remove the escape hatch or to station an assistant inside the aircraft when making adjustments and/or performing operational tests on the cabin door latching mechanism.

2. Adjustment/Test – Emergency Exits

A. Water Leak Test – Emergency Exit Hatch Seal

(1) Perform water leak test with fuselage unpressurized.

NOTE: Seals around emergency exits should prevent entry of water into aircraft even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(2) Position an assistant inside aircraft with flashlight and marking pencil.

(a) Use flashlight to detect water seepage during test.

(b) Identify and mark water entry point(s) with marking pencil.

(3) Close passenger and cargo doors during water test to prevent water from entering aircraft.

(4) Use water hose, insecticide sprayer, or similar device to dispense water under pressure.

(5) With nozzle approximately two (2) to three (3) inches from emergency exit, apply water completely around edge of emergency exit for several minutes.

(6) Check for evidence of seepage around emergency exit hatch. No water leakage is allowed.

(7) If test indicates emergency exit hatch is not sealed properly, make repairs as outlined in Para. **4.** Approved Repairs – Emergency Exits.

3. Inspection/Check – Emergency Exits

A. Inspect emergency exits as follows:

(1) Check emergency exit hatches for proper functioning, latching mechanism for wear, corrosion, cracks, and springs for tension.

(2) Inspect emergency exit hatch structure for cracks, scratches, dents, loose rivets, corrosion, or structural integrity.

(3) Check emergency exit hatch seal for wear, deterioration, and proper contact.

4. Approved Repairs – Emergency Exits

A. If water leak test indicates that a water leak is present, make repairs as follows:

(1) Remove emergency exit hatch.

(2) Leaks may be stopped by applying a brush coat of MIL-S-8802 sealant to fuselage cutout in area marked by assistant.

(3) Allow sealant to dry a minimum of 24 hours before installing emergency exit hatch.

(4) Install emergency exit hatch and perform leak test.

(5) If emergency exit hatch continues to leak, remove hatch and proceed as follows:

(a) Remove bulb seal retainer screws from area marked by assistant.

(b) Using thin pieces of neoprene rubber, manufacture a shim of same width and length as seal retainer. Shim thickness must be determined by trial and error.

NOTE: Do not add more shim, Excessive shims may cause emergency exit hatch to be difficult to install/operate.

(c) Drill holes to match seal retainer.

(d) Apply a coat of sealant to both surfaces of shim.

(e) Install shim between emergency exit hatch and bulb seal.

(f) Install seal retainer screws through seal retainer, seal, and shim to emergency exit hatch.

(g) Install emergency exit hatch.

NOTE: Sealant should be allowed to dry a minimum of 24 hours before installing emergency exit hatch in aircraft.

(h) Perform water leak test.

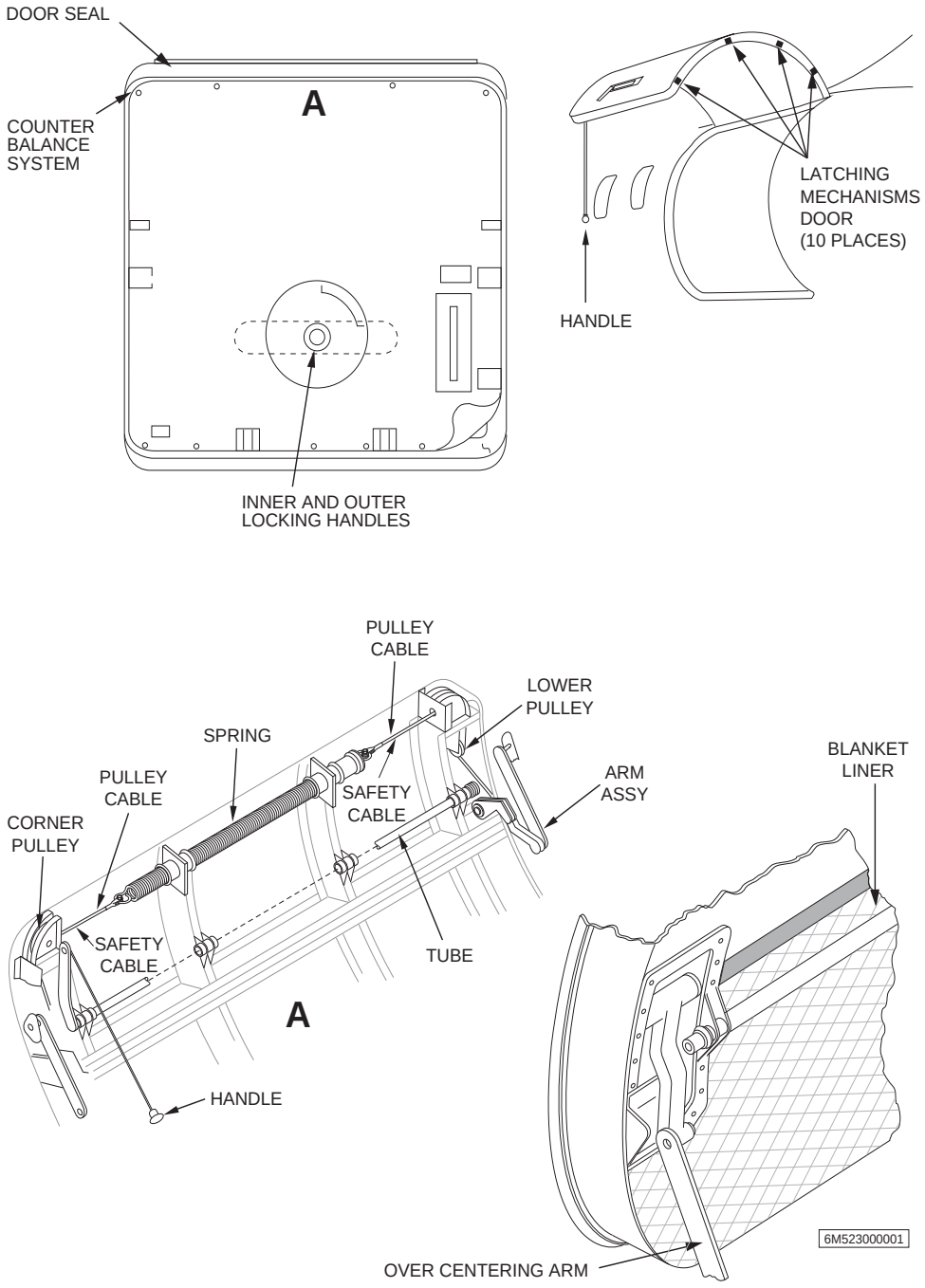
(i) If necessary, repeat shimming procedure until water leak is stopped.

B. Should structural defects be detected (e. g. cracks, loose rivets, corrosion, ect.), refer to SA226/227 Series Structural Repair Manual for repair procedures.

MAIN CARGO DOOR – DESCRIPTION AND OPERATION

1. General

A. A main cargo door on SA226 – AT and TC airplanes, located on the aft left section of the fuselage is 53 inches (134.62 cm) high by 51.25 inches (130.18 cm) wide. It is hinged across the top and open out and up. The door is furnished with an assist spring and an overcenter mechanism for holding it in the open position.



Main Cargo Door
Figure 1

MAIN CARGO DOOR – DESCRIPTION AND OPERATION

2. Main Cargo Door Structure

A. The structure of the cargo door is made of high grade aluminum. The different frames in this door are intercostals, corner frames, and belt frames.

3. Main Cargo Door Seal Valve System

A. A silicone seal is located around the outer frame of the door. The purpose of the seal is to aid in pressurizing the cabin. The solenoid door seal valve is triggered when the handle is moved to the locked position. The seal inflates with 18 psi (1.27 kg/cm^2) bleed air from the pneumatic system.

4. Main Cargo Door Latching System

A. The cargo door latching system provides a door lock for security. The folding door handle turns the main bellcrank, which actuates push-pull rods. The rods move eight click-clacks and two aligning bayonets into and out of their receptacles.

B. Eight receptacles are installed in the fuselage structure around the cargo door. Each receptacle provides a seating surface for the latch (click-clack) jaws, and forms the structural carry-through from the fuselage structure to the door. The six receptacles on the side of the door are equipped with steel bushings. These bushings may be rotated to aid in alignment of the door.

C. An overcenter rod assembly holds the main bellcrank and connected latching mechanism in the closed position by spring tension.

5. Main Cargo Door Counterbalance

A. A counterbalance spring is used to overcome door weight, and an overcentering arm links to hold it at its fully opened position. Pulleys attached to handle which hangs down along the aft frame, move the arm from overcenter and allow the door to be closed. Tension increases on the spring in the counterbalance as the door closes, and prevents the door from free falling.

6. Main Cargo Door Handle – Inner/Outer

A. The cargo door may be opened from the inside or outside by interconnected handles. The outside handle in the closed position, is recessed into the door frame, and may be locked to provide security.

MAIN CARGO DOOR – MAINTENANCE PRACTICES

1. Removal/Installation – Main Cargo Door

A. Removal – Main Cargo Door

(1) Position door cradle outside aircraft to provide full weight support for cargo door when door is removed from aircraft. Use a cradle with a suitable surface to avoid damage to door finish or material.

(2) Open cargo door.

(3) Remove interior covering from cargo door to gain access to seal inflation tube.

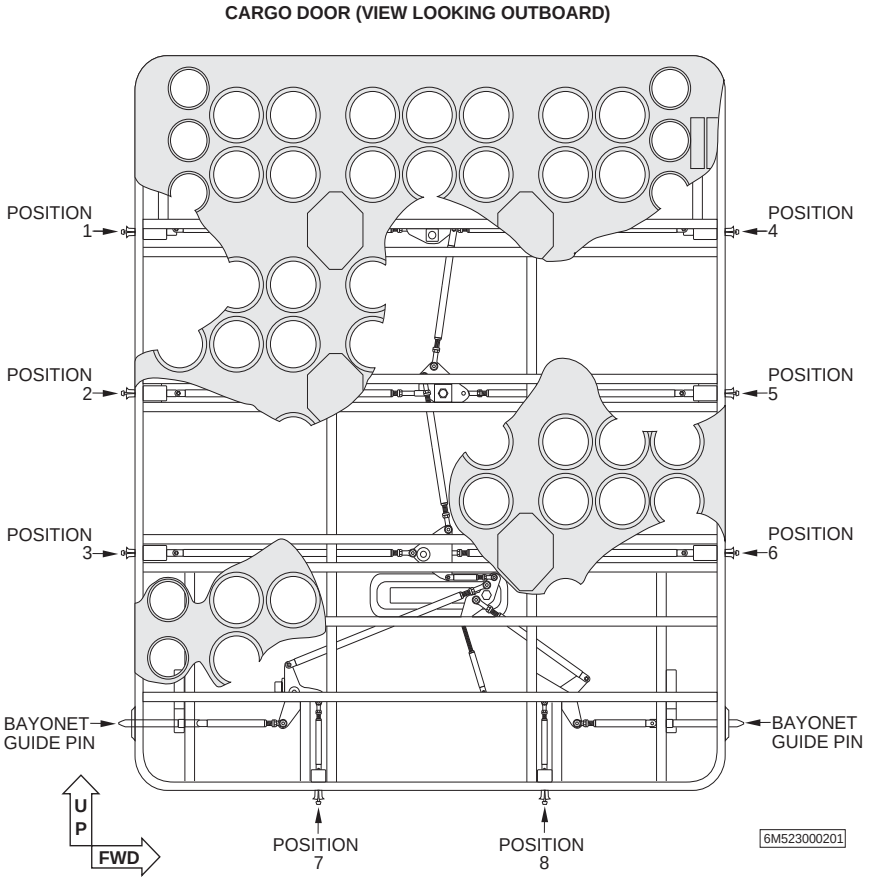
(4) Remove headliner next to door opening to gain access to door seal inflation tube.

CAUTION: IMMEDIATELY INSTALL CAPS OR PLUGS ON OPEN LINES TO PREVENT CONTAMINATION.

(5) Disconnect door seal inflation tube at union, and withdraw tube through door facing. Cap open lines.

(6) Have two people support cargo door in open position, and remove aft and forward bolt connecting each overcentering arm to the fuselage.

NOTE: Continued support of cargo door will be needed until door is removed from aircraft.



Main Cargo Door – Removal/Installation
Figure 201

(7) With door in cradle, remove hinge pin as follows:

(a) Using a small drift pin and hammer, drive hinge pin 6.0 to 8.0 inches (15.24 to 20.32 cm) out of hinge assembly.

(b) Grasp extended part of hinge pin with clamping pliers (vise-grips).

(c) Remove hinge pin from the hinge assembly by striking vise-grips with a mallet.

NOTE: An alternate method may be used for removing hinge pin, but only if hinge halves are to be replaced. Chuck extended part of hinge pin in drill, start drill motor at low RPM, while pulling hinge pin out.

(8) Remove cargo door, and place on support cradle.

B. Installation – Main Cargo Door

(1) Place cargo door on door cradle.

(2) Check fuselage and door hinge condition. If either hinge is half loose or damaged, tighten or replace.

(3) With help of two people, raise the door into position for installation.

(4) Align and mate door and fuselage hinge halve.

(a) Insert hinge pin using drill motor.

NOTE: Install new hinge pin if old hinge pin shows signs of wear.

(b) Peen ends of hinge pin to secure.

(5) Support cargo door in open position and install aft and forward bolt connecting each overcentering arm.

(6) Remove caps from lines. Connect door seal inflation tube at union.

(7) Check cargo door for proper opening and closing. Open and close door several times to ensure proper operation.

(8) Perform water leak test IAW Adjustment/Test – Main Cargo Door Seal Valve System.

NOTE: Door seal should prevent entry of water into aircraft even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(9) Perform pressure check IAW Adjustment/Test – Main Cargo Door Seal Valve System.

(10) Install headliner.

(11) Install interior covering to cargo door as required.

2. Preparation/Assembly – Main Cargo Door

A. Parts required – New Main Cargo Door

(1) Door assembly.

(2) Hinge and hinge pin.

(3) Bayonets (2).

(4) Click-Clacks (8).

(5) Teflon guides (minimum 4).

(6) Eccentric bushings and retainers.

(7) Faceplates.

(8) Nylon assist handle.

(9) Door seal.

B. Tools required – New Main Cargo Door

(1) 0.630 inch reamer.

(2) ZT-535C pancake drill.

(3) SAT 1223 bushing set.

(4) Door cradle.

(5) Web fuselage strap and cinch.

(6) Transfer punches and drill bits No. 10, No. 21, No. 30, and No. 40, with drill stop.

(7) Countersink with stop – No. 10 and No. 21.

(8) Open end, box end wrenches and 1/4 inch drive sockets; 3/16, 7/32, 1/4, 3/8, 7/16, and a 1/4 universal.

(9) Drill motor and rivet gun (4X), bucking bars including six - eight inch offset.

C. Consumable Materials Required – New Main Cargo Door

(1) EP-890 B 1/2 or B2 sealer.

I (2) Methyl propyl ketone (MPK).

(3) Masking tape – one inch and two inch wide.

(4) Hinge oil.

(5) RTV 3145 Silicone sealant.

(6) Tri-Flow (aerosol), 903 Dri-Slide (liquid), MIL-L23398 Molycote (aerosol), or NATO #S-749 Lubricants.

D. Assembly Step I – New Main Cargo Door

NOTE: The objective of this phase of assembly is to align and center the new main cargo door.

CAUTION: AIRCRAFT MUST BE LEVEL. IF AIRCRAFT HAS TO BE MOVED AFTER WORK HAS STARTED, IT MUST BE ROLLED BACK INTO A LEVEL WORKING AREA HAVING TRAVELED AT LEAST 20 FEET IN A STRAIGHT LINE WITH NOSE WHEEL STRAIGHT.

(1) Park aircraft on level surface. DO NOT place aircraft on jacks or support.

(2) Remove emergency exit window to provide alternate entrance.

(3) Check condition of fuselage and door hinge halves. If hinge half is loose or damaged, tighten or replace.

(4) Before positioning the door into the fuselage, ensure the following conditions exist:

(a) Bayonet guide pins – NOT INSTALLED.

(b) Click-Clacks – NOT INSTALLED.

(c) Eccentric bushings in fuselage receptacles – NOT INSTALLED.

(d) Door hinge – NOT INSTALLED.

(e) Faceplates on door and fuselage – INSTALLED.

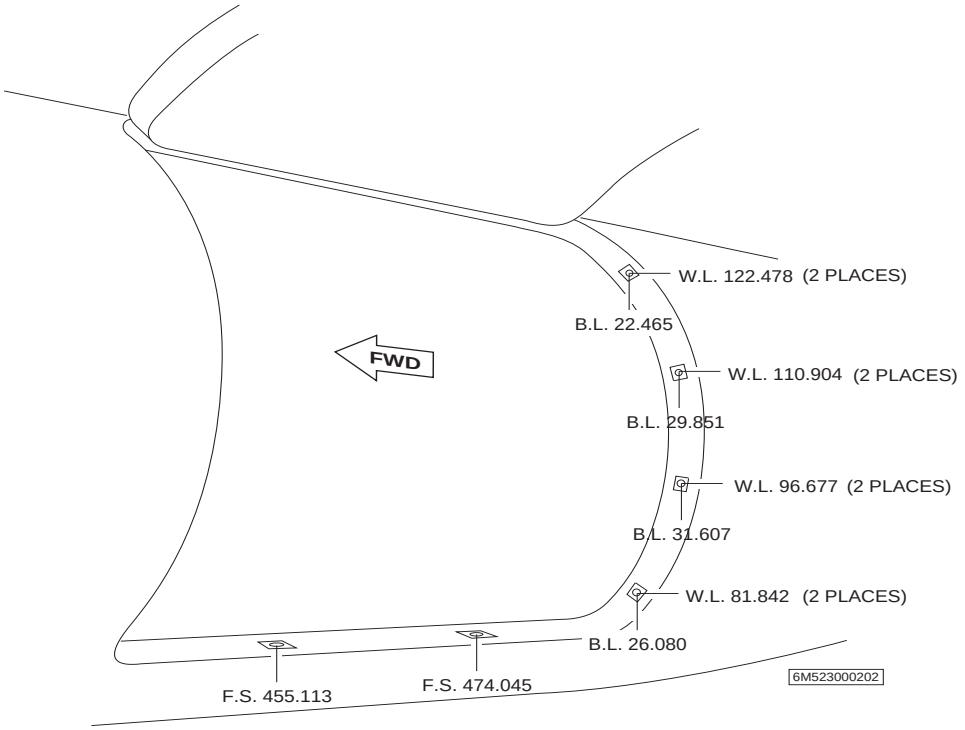
(f) All sealer and bondo – REMOVED FROM FUSELAGE FRAME.

(5) Temporarily position main cargo door in fuselage.

(6) Set height of new door in fuselage by placing 0.225 to 0.230 inch (5.72 to 5.84 mm) spacer between door faceplate and fuselage faceplate at location numbers 7 and 8 (Figure 201).

(7) Optimum spacing between door frame and fuselage frame is 0.220 inch (5.558 mm). Spacing tolerance of 0.210 to 0.240 inch (5.334 to 6.096 mm) allows for proper adjustment of side latches.

(8) Note clearance between forward and aft door faceplates and fuselage faceplates.



Main Cargo Door – Recptacle Location
Figure 202

(9) Center door between door faceplates and fuselage door frame.

(10) Fabricate shims to same dimensions as Teflon guide. Use aluminum material of sufficient thickness to obtain desired clearance.

(11) Shim Teflon guides to provide positive contact with fuselage door frame.

(12) Make reference mark on forward and aft door skin to fuselage skin to aid in maintaining height positioning.

(13) Remove main cargo door and place in cradle or suitable work bench.

(14) Install door handle, click-clacks, bellcranks, and push-pull tubes.

(15) Check proper operation of door handle, click-clacks, and extension of plunger while door is on door cradle.

(a) Note freedom of handle operation.

(b) Lubricate latches at this time.

(16) Position assistant inside aircraft to inspect general action and location of click-clacks.

(17) Temporarily install door on aircraft.

(18) Check height positioning IAW steps **2.**, D., (6) and (7).

(19) Check reference marks on forward and aft door skin to ensure door is centered.

NOTE: Each time door is removed and reinstalled, check reference marks to verify door is at proper height.

(20) Ensure the click-clacks will pass through the empty fuselage receptacles (bushings not installed).

(21) Remove door from aircraft. Place door on cradle.

(22) Make initial installation of bushings. Ensure bushings are on dead center in fuselage receptacles (see Figure 202).

NOTE: See SA226 Series Illustrated Parts Catalog (IPC) for bushing sized/descriptions.

(23) Do not damage operating mechanisms. Position and install cargo door carefully. Gently attempt to latch door into place.

(24) Have assistant observe click-clack actions and note any door movements as the click-clacks attempt to engage.

NOTE: Due to untrimmed skin, gentle pressure from outside the aircraft at click-clack locations may assist in engagement.

(25) If door can be easily latched using only moderate pressure, proceed to next assembly step.

(a) If door cannot be easily latched, remove door. Adjust/replace receptacle/bushing causing the most problems.

(b) Temporarily install door. Proceed to adjust receptacles/bushings in additional positions.

(c) Repeat steps **2.**, D., (25), (a) and (b) until all receptacles/bushings are properly positioned/located.

CAUTION: DO NOT ALLOW DOOR TO OPEN AGAINST CLECO FASTENERS.

(26) After door latches are installed and operating properly, install temporary hinge. Secure temporary hinge by using minimum of six (6) Cleco fasteners.

(27) Trim skin at upper corners.

(28) With door shut and latches engaged:

(a) Ensure all Teflon door guides contact fuselage door frame.

(b) Install additional Teflon door guides (as necessary) to ensure equal spacing between faceplates.

(c) Ensure all click-clacks are fully engaged.

E. Assembly Step 2 – New Main Cargo Door

NOTE: The objective is to obtain initial skin contour, through receptacle adjustments, in preparation for installing door hinge.

(1) Trim forward and aft skin as necessary. With door closed and click-clacks engaged, check door contour.

(2) Slowly open door and leave open.

(3) Apply one inch masking tape around perimeter of fuselage door cutout.

NOTE: Tape must exactly match edge of fuselage skin.

(4) Carefully close and latch door.

(5) Using two inch masking tape, apply tape around door frame. Tape edge must precisely match to outboard edge of tape installed during step **2.**, E., (3).

(6) With a razor or knife, follow edge of door cutting tape.

(7) Open door and trim skin; follow edge of remaining tape.

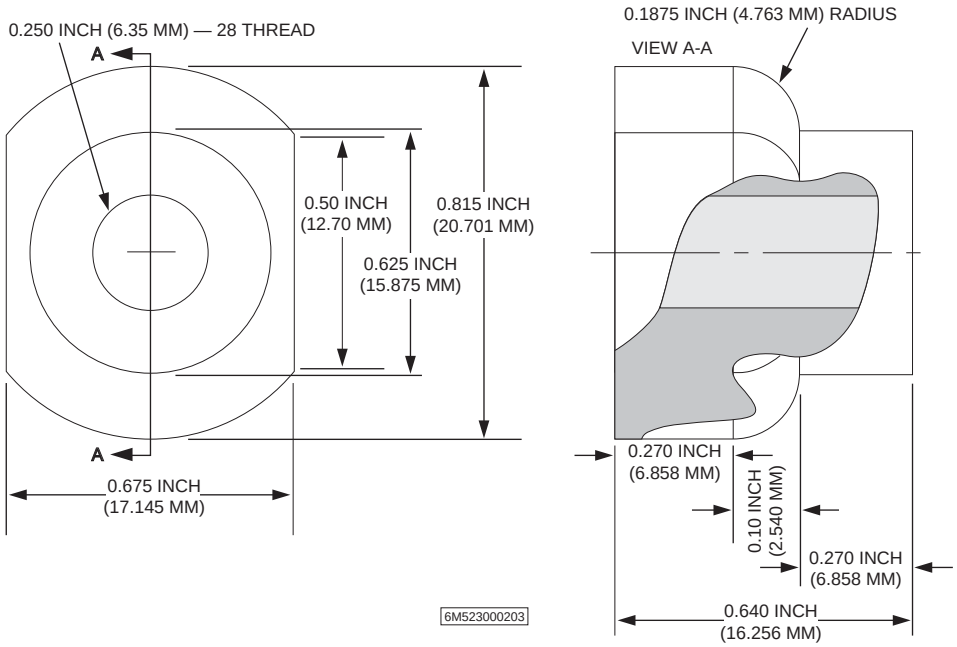
(8) After initial trim alternately check door and trim to ensure proper skin clearance. Use a Vixon file to trim skin.

(9) Begin with click-clack position 2. Alternate between positions 2, 1, and 3, and work down. Exchange or replace bushings (as necessary) to ensure proper skin contour. click-clack operation, and door alignment are obtained.

(10) Move to position 5 and alternate between position 5, 4, and 6 until proper combination of skin contour, click-clack operation, and door alignment are obtained.

NOTE: This alignment procedure will allow for future adjustment.

(11) Complete all riveting of inside skin at this time.



Main Cargo Door – SAT 1233-1 Latch Installation tool
Figure 203

(12) Install bottom click-clacks in positions 7 and 8.

(a) Starting with position 7, install receptacle into cargo door. Install tool set SAT 1233-1 (Figure 203) into fuselage receptacle.

NOTE: The SAT 1233-1 tool is used to align the receptacle from the door to the fuselage. Tighten until a grip length of 0.140 inch (3.556 mm) is obtained. Measurement is from faceplate to door frame.

(b) As each hole is drilled with a ZT535C pancake drill, install hardware with tool set inserted.

(c) Alternate from side to side of receptacle while drilling.

(d) Remove bushing set. Install click-clack and faceplate. Adjust click-clack actuating rod to achieve the required dimension of 0.304 inch and lubricate.

(13) Check click-clacks for final adjustment, and lubricate with Tri-Flow (aerosol), 903 Dri-Slide (liquid), MIL-L23398 Molykote (aerosol), or NATO #S-749.

NOTE: To install bottom click-clack in position 8, repeat steps **2.**, F., (12) and (13).

(14) With door closed, install permanent hinge and hinge pin.

(a) Match door hinge half to fuselage hinge half.

(b) Begin in center and alternate from side to side, working in outboard direction.

1 Use a No. 21 transfer punch to transfer holes from door to new hinge half.

2 Step drill from No. 40 to No. 21.

CAUTION: DO NOT ALLOW DOOR TO OPEN AGAINST CLECO FASTENERS.

3 As holes are drilled, insert Cleco fastener in each hole.

(c) Countersink and deburr hinge holes.

1 Apply sealant to both side of hinge.

2 Install and rivet hinge to door.

NOTE: Cleco fasten holes before riveting.

F. Assembly Step 3 – New Main Cargo Door

NOTE: The objective is to locate bayonet guide pins and install door seal

(1) With door closed, try bayonets in holes. Align bayonets to faceplates on fuselage. Check alignment. If necessary, polish bayonet faceplates until bayonet(s) will slip in and out with only moderate pressure.

(2) With door open, install bayonets in door.

(3) Adjust bayonets as necessary to ensure proper operation.

NOTE: It may be necessary to adjust location of faceplates.

(4) Install door seal.

(a) Drill 0.250 inch hole in forward door frame for installation of door seal tube.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(b) Use MPK to clean edge of door to door frame where seal seats.

(c) Use MPK to clean new seal thoroughly.

(d) Check seal when in place to verify no twists exist in seal around door.

CAUTION: EXCESS SEALER BETWEEN SKIN AND SEAL WILL CAUSE CONTOUR DEVIATION AND UNSATISFACTORY OPERATION OF DOOR. USE CARE NOT TO SPREAD SILICONE OR TO ALLOW IT TO CONTACT SKIN SURFACE. IMMEDIATELY DISPOSE OF RAGS OR TOWELS USED TO CLEAN UP EXCESS SILICONE SEALANT.

(e) Remove seal and run a bead of RTV 3145 sealant approximately 0.125 inch (3.175 mm) wide all the way around the door at seal seat.

CAUTION: SEAL MUST BE SLID ALONG SKIN INTO CORNERS. DO NOT SLIDE SEAL ALONG FRAME INTO CORNERS.

(f) Beginning at inflation tube, install seal all the way around door.

1 Use a sliding action and work from edge of skin into structure and seal seat.

2 When seal is in place, ensure no twists exist in seal around door.

(g) Check contour of ridged sealing area of door seal in area of splices (should not exceed three [3] splices). Ensure seal contour does not deviate by more than -0.020 inch (-0.508 mm) from contour on either side of splice(s).

(h) When seal is completely installed, carefully close door.

1 Use web strap around the fuselage and cargo door to help draw door in.

2 Allow sealer to dry in place for approximately two (2) hours.

(5) If necessary, trim bottom corners to ensure door skin and corners clear fuselage frame.

(a) Tape lower forward and aft corners of fuselage with one inch masking tape. Align edge of tape with edge of door cutout.

(b) Close and latch door.

(c) Tape over door edge using two inch masking tape. Precisely match outboard edge of tape IAW step **2.**, F., (5), (a).

(d) With a razor or knife, follow edge of door and cut top layer of tape.

(e) Open door and trim skin; follow edge of remaining tape.

(f) After initial trim, use a Vixon file and finish trimming forward and aft corners.

(6) Remove caps from lines.

(7) Connect door seal inflation tubing to bleed air tubing.

(8) Perform water leak test IAW Adjustment/Test – Main Cargo Door Seal.

NOTE: Door seal should prevent entry of water into aircraft even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(9) Perform pressurization check IAW Adjustment/Test – Main Cargo Door Seal.

(a) If seal leaks around door, slide a wide tip common screwdriver between seal and frame and gently loosen seal in affected area.

(b) Apply additional RTV 3145 sealer approximately 0.250 inch (6.350 mm) wide.. Close door immediately.

(c) Allow sealer to dry for two (2) hours.

(d) Open door and trim any surplus sealer.

(e) Perform additional water leak test and pressure check.

(10) Check alignment, proper closing and opening. Open and close door several times to ensure proper operation.

(11) Replace cargo door upholstery (inner panel).

3. Inspection/Check – Main Cargo Door

A. Inspect main cargo door as follows:

(1) Inspect cargo door and door hinge for general condition and security.

(2) Check operation of latching mechanisms.

(a) Inspect latching bellcrank assemblies, push-pull rods, and rod ends for general condition, alignment, wear, and security.

(b) Check click-clack latches and bayonet guide pins for adjustment, nicks, burrs, alignment, general condition, and operation.

(c) Check inner and outer handles for adjustment, fit, security, and operation.

(d) Check safety lock for freedom of operation, security, and general condition.

(3) Check, clean, and lubricate door lock mechanisms.

(4) Inspect seal and seal supply line for punctures, cracks, kinks, leaks, security, and general condition.

(5) Inspect door structure for cracks, loose rivets, and general condition.

(6) Check Teflon door guide blocks for general condition and security.

MAIN CARGO DOOR SEAL VALVE SYSTEM – DESCRIPTION AND OPERATION

1. General

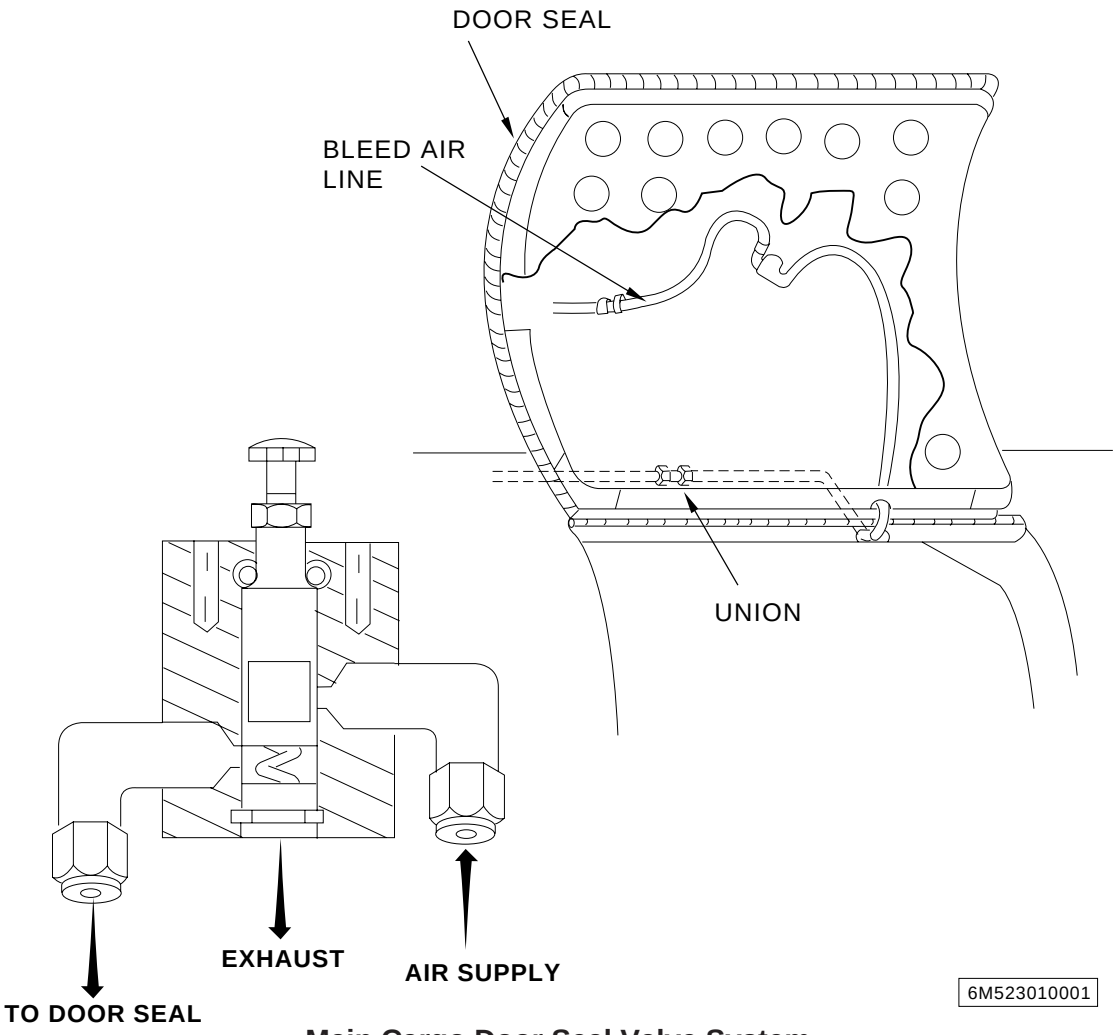
A. A silicone door seal is located around the outer frame of the door. A solenoid valve is triggered when the door handle is moved to the locked position. The seal inflates with 18 psi (1.27 kg/cm2) bleed air from the pneumatic system (see Chapter 36). When handle is moved to the open position, the valve blocks the flow of air to the seal, allowing air already in the seal to escape overboard. This deflates the seal to permit opening and closing of the door with the engines running.

2. Cargo Door Seal Valve

A. The door seal valve is actuated by the door handle. As the door handle is turned the valve is triggered and air is routed to the door seal.

3. Cargo Door Seal

A. The door seal is an inflatable tube, installed around the door to prevent air leaks and to maintain cabin pressure.



Main Cargo Door Seal Valve System
Figure 1

MAIN CARGO DOOR SEAL VALVE SYSTEM – MAINTENANCE PRACTICES

1. Removal/Installation – Main Cargo Door Seal Valve System

A. Removal – Main Cargo Door Seal Valve

(1) Open cargo door.

(2) Remove cargo door interior covering.

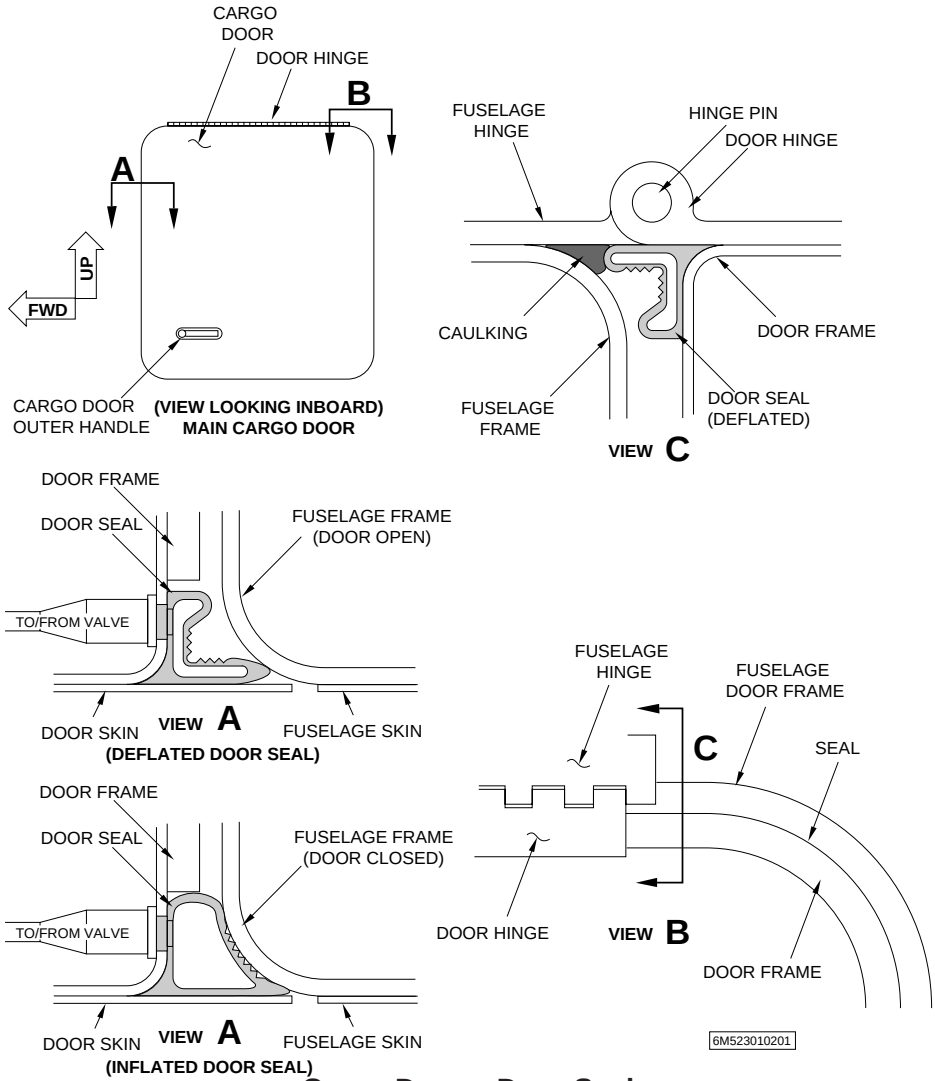
(3) Locate seal valve at upper bellcrank assembly for click-clack positions 1 and 4.

CAUTION: IMMEDIATELY INSTALL CAPS OR PLUGS ON OPEN LINES TO PREVENT CONTAMINATION.

(4) Disconnect tubing from valve. Cap open lines.

(5) Remove screws and washers securing valve to bracket.

(6) Remove seal valve from cargo door.



Cargo Door – Door Seal
Figure 201

B. Installation – Main Cargo Door Seal Valve

(1) Position seal valve for installation at upper bellcrank assembly for click-clack positions 1 and 4.

(2) Secure valve to bracket with screws and washers.

(3) Remove caps from lines. Install tubing to valve.

(4) Check door seal valve for adjustment IAW Adjustment/Test, this section.

(5) Install covering to cargo door.

C. Removal – Main Cargo Door Seal

(1) Open cargo door.

(2) Remove interior covering from cargo door.

(3) Locate and disconnect bleed air line from fuselage to door. Cap open lines.

(4) Have two people support cargo door in open position and remove aft and forward bolts connecting each overcentering arm to the fuselage.

NOTE: Continue to support door through door seal removal and replacement.

CAUTION: IMMEDIATELY INSTALL CAPS OR PLUGS ON OPEN LINES TO PREVENT CONTAMINATION.

(5) Locate and disconnect tubing from door seal valve to door seal (inside, upper, forward area of cargo door). Cap open lines.

(6) Remove old seal from door frame.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(7) Clean area thoroughly with methyl propyl ketone (MPK).

D. Installation – Main Cargo Door Seal

(1) Open cargo door.

(2) Clean new seal thoroughly with MPK to remove shipping preservatives.

(3) Clean door frame area where seal is to be installed thoroughly with MPK prior to installation of new door seal.

CAUTION: EXCESS SEALER BETWEEN SKIN AND SEAL WILL CAUSE CONTOUR DEVIATION AND POOR OPERATION OF DOOR. USE CARE NOT TO SPREAD SILICONE SEALANT OR ALLOW IT TO CONTACT SKIN SURFACE. IMMEDIATELY DISPOSE OF TOWELS OR RAGS USED TO CLEAN UP SILICONE SEALANT.

(4) Working a small area at a time, install new door seal using Dow Corning RTV 3145 Silicone Rubber Sealant.

(a) Run a bead of sealant approximately 0.125 inch (3.175 mm) wide around cargo door at seal seat.

CAUTION: SEAL MUST BE SLID ALONG SKIN INTO CORNERS. DO NOT SLIDE SEAL ALONG FRAME INTO CORNERS.

(b) Beginning at inflation tube, install seal all the way around door.

1 Using a sliding action, work from edge of skin into door structure and seal seat.

2 When seal is in place around door, ensure no twists exist in seal.

3 Check contour of ridged sealing area of door seal in area of splice (should not exceed three [3] splices). Ensure contour does not deviate by more than -0.020 inch (-0.508 mm) from the contour on either side of splice(s).

(5) When seal is completely installed, carefully close door. Allow seal to dry in place for a minimum of two (2) hours.

(6) Open door carefully. Provide support to keep door in open position.

(7) Remove caps from lines.

(8) Connect line from door seal to door seal valve and secure.

(9) Connect bleed air line from fuselage to door and secure.

(10) Install aft and forward bolts connecting overcentering arm to fuselage.

(11) Close door and check for alignment of door seal to fuselage frame.

(12) Check seal for inflation and leaks IAW Adjustment/Test, this section.

NOTE: Allow sealant to dry for 24 hours before pressurizing airplane.

WARNING: ENSURE PROPELLERS AND TAILPIPE AREAS ARE CLEAR OF PERSONNEL AND OBJECTS DURING ENGINE RUN.

(a) Operate one engine as a pressure source.

NOTE: Refer to AFM for engine ground run procedures.

(13) Install interior covering to cargo door.

2. Adjustment/Test – Main Cargo Door Seal Valve System

A. Water Leak Test – Main Cargo Door Seal

NOTE: This test is to be conducted with fuselage unpressurized.

Door seal should prevent entry of water into aircraft even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(1) Use water hose, insecticide sprayer, or similar device that dispenses water under pressure.

(2) Position an assistant with a flashlight and marking pencil inside the aircraft at the cargo door.

(3) Instruct assistant to determine point(s) of seepage during test, and mark water entry point(s) with a marking pencil.

(4) With nozzle approximately 2.0 to 3.0 inches (5.08 to 7.62 cm) from door, apply water completely around door edge for several minutes.

(5) Open cargo door. Check for evidence of seepage. No water leakage is allowed.

(6) If door seal is not sealing properly, peel seal away from door frame in area of water leak.

(a) Use thin pieces of neoprene rubber to shim door and seal.

(b) Secure seal using RTV 3145 silicone sealant or equivalent. Do not add more than necessary as it will make door difficult to close.

B. Pressurization Leak Test – Main Cargo Door Seal

(1) Open cargo door.

(2) Check rubber seal for signs of wear, deterioration, proper contact, and pressurization leaks.

NOTE: Visible evidence of leaks may appear as stains on fuselage skin around door opening.

(3) If a pressurization leak is detected or suspected, shim seal IAW step **2.**, A., (7).

C. Adjustment – Main Cargo Door Seal

(1) Open cargo door.

(2) Remove interior covering from cargo door to gain access to door seal valve.

(3) Close cargo door.

CAUTION: DO NOT APPLY MORE THAN 18 PSI (1.2654 KG/CM^2) PRESSURE.

(4) Apply 15 psi (1.0545 kg/cm^2) air pressure to upstream side of door seal valve.

WARNING: ENSURE PROPELLERS AND TAILPIPE AREAS ARE CLEAR OF PERSONNEL AND OBJECTS DURING ENGINE RUN.

(a) Operate one engine as a pressure source.

NOTE: Refer to AFM for engine ground run procedures.

(5) While rotating inner door handle to LOCKED position, observe to ensure seal is inflating

(a) Adjust door seal valve so that seal inflates at approximately 90° travel while rotating inner door handle.

(6) Check to ensure door seal deflates before inner door handle reaches full UNLOCKED.

(7) Install interior covering to cargo door.

3. Inspection/Check – Main Cargo Door Seal Valve System

A. Inspect main cargo door seal valve as follows:

(1) Inspect cargo door seal and door seal supply for security and general condition.

(2) Check contour of ridged sealing area of door seal in area of splice(s). Ensure contour does not deviate by more than -0.020 inch (-0.508 mm) from contour on either side of splice(s).

MAIN CARGO DOOR LATCHING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The latching mechanism on the cargo door consists of a door lock, folding door handle, bellcranks, overcenter rod assembly, push-pull rods, bayonet guide pins, and click-clacks. There are eight click-clack latches, and two aligning guide pins on the cargo door. As the door handle is turned towards the closed position, the bayonet guide pins and click-clacks move outward into their respective receptacles. The latching mechanism can be operated from inside or outside of the aircraft.

2. Lock Assembly

A. The lock assembly is located in the door handle housing and consists of a key, cam, washer, lock tab, hex-nut plug assembly, and a plug. Its primary purpose is to lock the aircraft for security.

3. Handle Assemblies

A. The cargo door may be opened from the inside or outside by interconnecting handles. The outside handle, in the closed position, is recessed into the door frame and may be locked to provide security.

4. Bellcranks

A. There are six bellcranks on the cargo door. The main bellcrank is attached directly to the door handle. Connecting rods join the main bellcrank to satellite bellcranks. The bellcranks change rotational motion to forward and aft movement. The bellcranks are strategically centered throughout the door.

5. Overcenter Rod Assembly

A. The overcenter rod assembly holds the main bellcrank and connected latching mechanism in the closed position by spring tension. The door will open only when the handle is rotated with sufficient force to opposed the spring tension, thus preventing vibration from opening the door inflight.

6. Push-Pull Rods

A. There are sixteen push-pull rods installed in the cargo door. They connect the bellcranks together and link the bellcranks to the click-clacks and bayonets.

7. Bayonet Guide Pins

A. Two bayonets, at the bottom of the door forward and aft sides, extend initially when the door handle is rotated toward closed and locked position. The bayonets pull the door into the door opening, aligning the click-clack latches with their respective fuselage door frame receptacles, but do not carry structural loads

8. Click-Clack Latch Assemblies

A. There are eight click-clack latch assemblies in the cargo door. These click-clack latch assemblies consist of an actuator assembled within a door frame mounted receptacle.

B. As the door handle continues toward closed and locked position, the click-clack actuator jaws extend from the door mounted receptacle into a fuselage door frame mounted receptacle. Door handle movement beyond this point extends the actuator plunger beyond the actuator jaws. As this extension continues, a taper at the base of the plunger shaft forces the click-clack jaws to open laterally and grip the inside of the fuselage frame receptacle, securing the door to the fuselage door frame.

1. Servicing – Main Cargo Door Latching System

A. Lubrication – Cargo Door Latches

(1) Lubricate click-clack and bayonet type latches with Tri-Flow (aerosol), Dri-Slide (liquid), MIL-PRF-23398 (aerosol) or NATO # S-749.

NOTE: Lubricate click-clacks and bayonets at scheduled maintenance intervals. To help prevent contamination from dust and dirt, do not use lubricants that leave an oily or greasy coating.

(2) Ensure extended tip of plungers and bayonets to surface of faceplate, are always well lubricated.

B. Cleaning – Cargo Door Latches

(1) Use dry compressed air to blow any contaminants from bayonets and click-clacks.

WARNING: METHYL PROPYL KETONE (MPK) AND VARSOL ARE EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK AND VARSOL CONSTITUTE A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(2) Clean bayonets and click-clacks using a clean cloth and Methyl Propyl Ketone (MPK) or Varsol (Division of Exxon).

(3) Use dry compressed air to dry bayonets and click-clacks.

2. Removal/Installation – Main Cargo Door Latching System

A. Removal – Cargo Door Latch Assembly (Figure 201)

NOTE: When the latch assembly actuator has failed, it must be replaced. It is not permissible to repair or replace individual pieces of the actuator. For the lower two cargo door latch assemblies on Post SB 226-52-008 aircraft, it is not necessary to replace the latch assembly receptacle when replacing the latch assembly actuator.

(1) Open cargo door.

(2) Remove door upholstery.

(3) Remove bolt, nut, and washer connecting push-pull rod assembly to latch assembly eye fitting.

(4) Remove screws from faceplate, and remove latch actuator from latch receptacle.

(5) Remove nuts, washers and screws from latch assembly receptacle and remove latch receptacle from door.

B. Installation – Cargo Door Latch (Figure 201)

NOTE: When the latch assembly actuator has failed, it must be replaced. It is not permissible to repair or replace individual pieces of the actuator. For the lower two cargo door latch assemblies on Post SB 226-52-008 aircraft, it is not necessary to replace the latch assembly receptacle when replacing the latch assembly actuator.

(1) Position latch receptacle in place on cargo door frame and secure receptacle with screws, washers and nuts.

(2) Position latch actuator and faceplate in place on door frame and secure with screws.

(3) Secure latch actuator eye fitting to push-pull rod with bolt, washer, and nut.

(4) Perform 3. A. Cargo Door Latch Mechanism Adjustment/Test, this section.

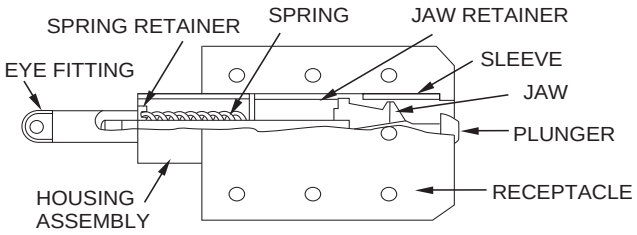
(5) Install door upholstery.

C. Removal – Cargo Door Lock Assembly (Figure 202)

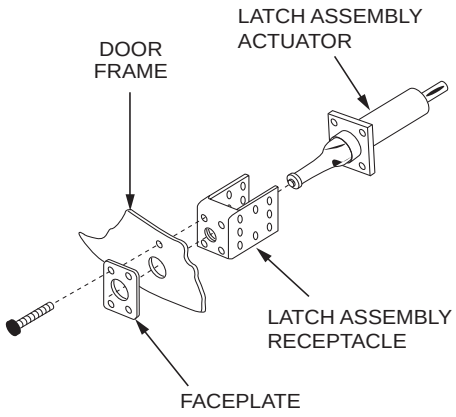
(1) Remove door upholstery to gain access to inside of cargo door.

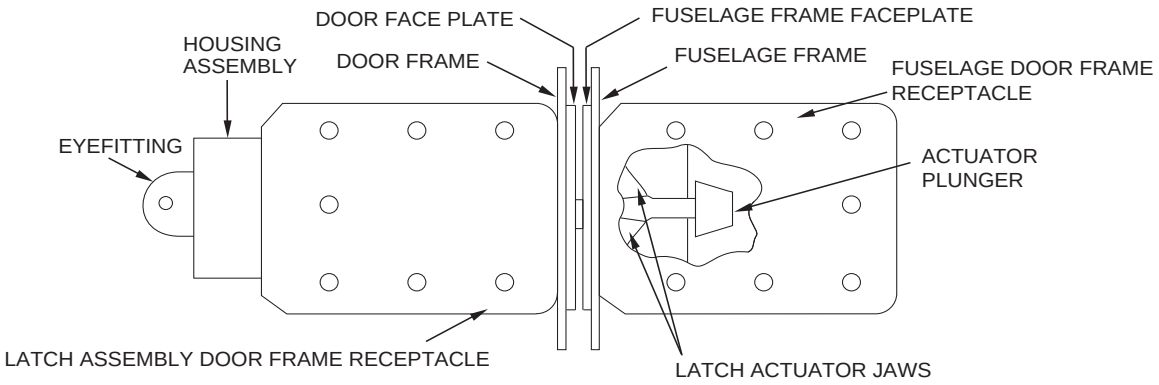
(2) Remove plug on back of door handle housing.

(3) Remove nut and lock washer attaching lock to door handle housing. Remove cam and lock.



(TOP VIEW) CLICK-CLACK ASSEMBLY - OPEN POSITION





CLICK-CLACK/RECEPTACLES (TOP VIEW - CLOSED POSITION)

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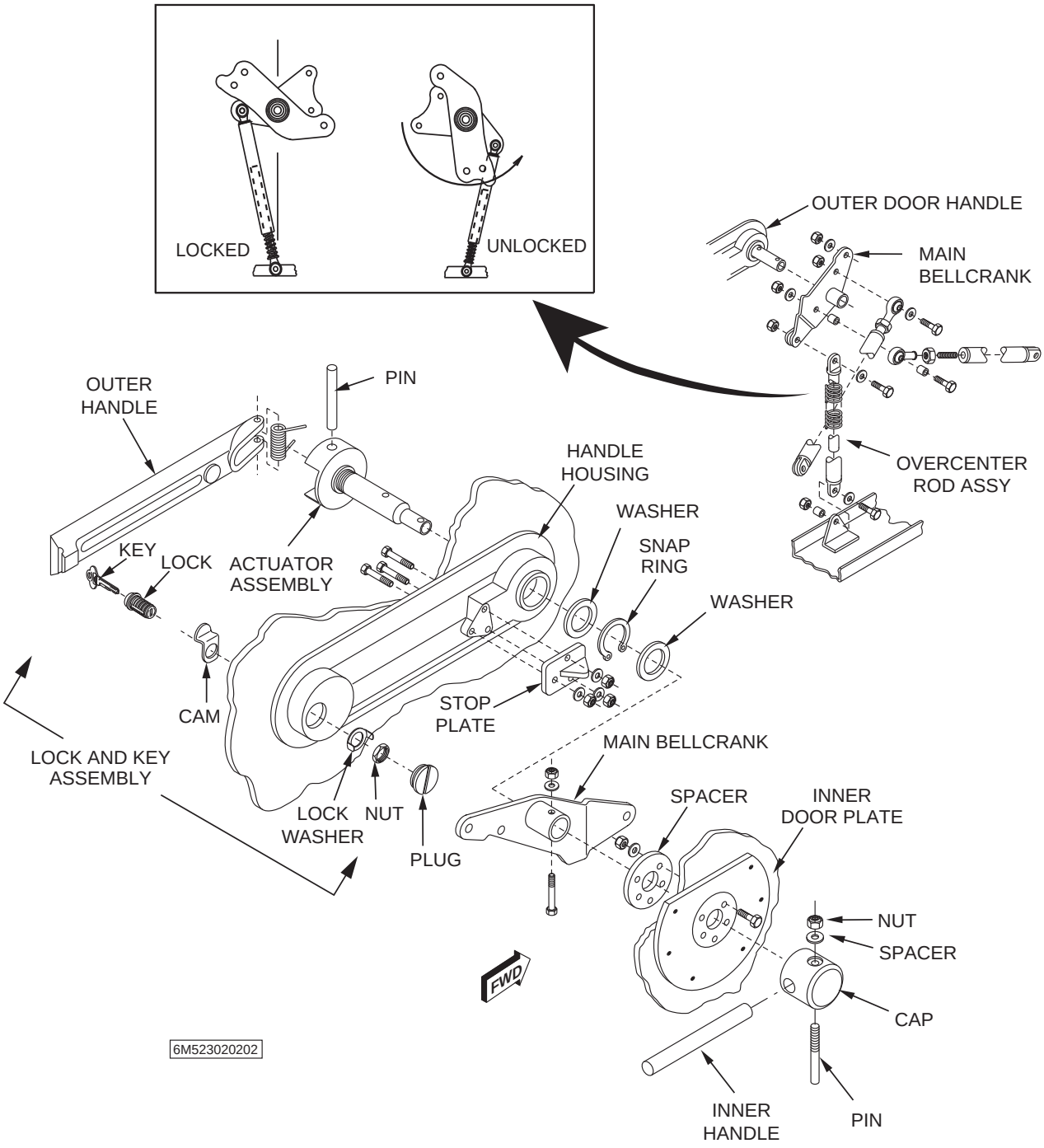
Cargo Door – Click-Clack Latch Assembly Figure 201



Cargo Door Handles – Inner/Outer
Figure 202

D. Installation – Cargo Door Lock Assembly (Figure 202)

(1) Position lock and cam onto door handle housing. Install lock washer and secure lock with nut. Bend lock washer tab over nut.

(2) Install plug into door handle housing.

(3) Insert key into lock assembly and check lock operation.

(4) Install door upholstery.

E. Removal – Cargo Door Fuselage Receptacles (Figure 203)

(1) Open cargo door. Leave door latching mechanism in full UNLOCKED position.

(2) Remove cargo compartment liners as necessary to gain access to affected receptacle(s) located in the fuselage door frame.

(3) Remove door warning micro switch and actuator for affected receptacle(s) (see 52-70-20).

(4) Remove screws and nuts securing retainer or fillers, bushing, and faceplate to fuselage door frame.

NOTE: Use care when removing screws and nuts to avoid dropping fillers and/or nuts behind retainer.

(5) Remove faceplate, bushing (side receptacles only) and retainer or fillers.

(6) Remove screws, washers and nuts attaching receptacle to fuselage frame.

(7) Remove receptacle from fuselage.

F. Installation – Cargo Door Fuselage Receptacles (Figure 203)

(1) Position receptacle in place behind fuselage frame.

(2) Install screws, washers and nuts securing receptacle to fuselage frame.

NOTE: Use care to avoid dropping fillers and/or nuts.

(3) Position faceplate, bushing (side receptacles only) and retainer or fillers in place and secure with screws and nuts.

(4) Install door warning micro switch and actuator for affected receptacle(s) (see 52-70-20).

(5) Close door and actuate locking mechanism. Ensure that latches fully engage into receptacles and microswitch assemblies.

(6) Perform 3. A. Cargo Door Latch Mechanism Adjustment/Test, this section.

(7) Install cargo compartment liners.

G. Removal – Cargo Door Bellcranks (Figure 204)

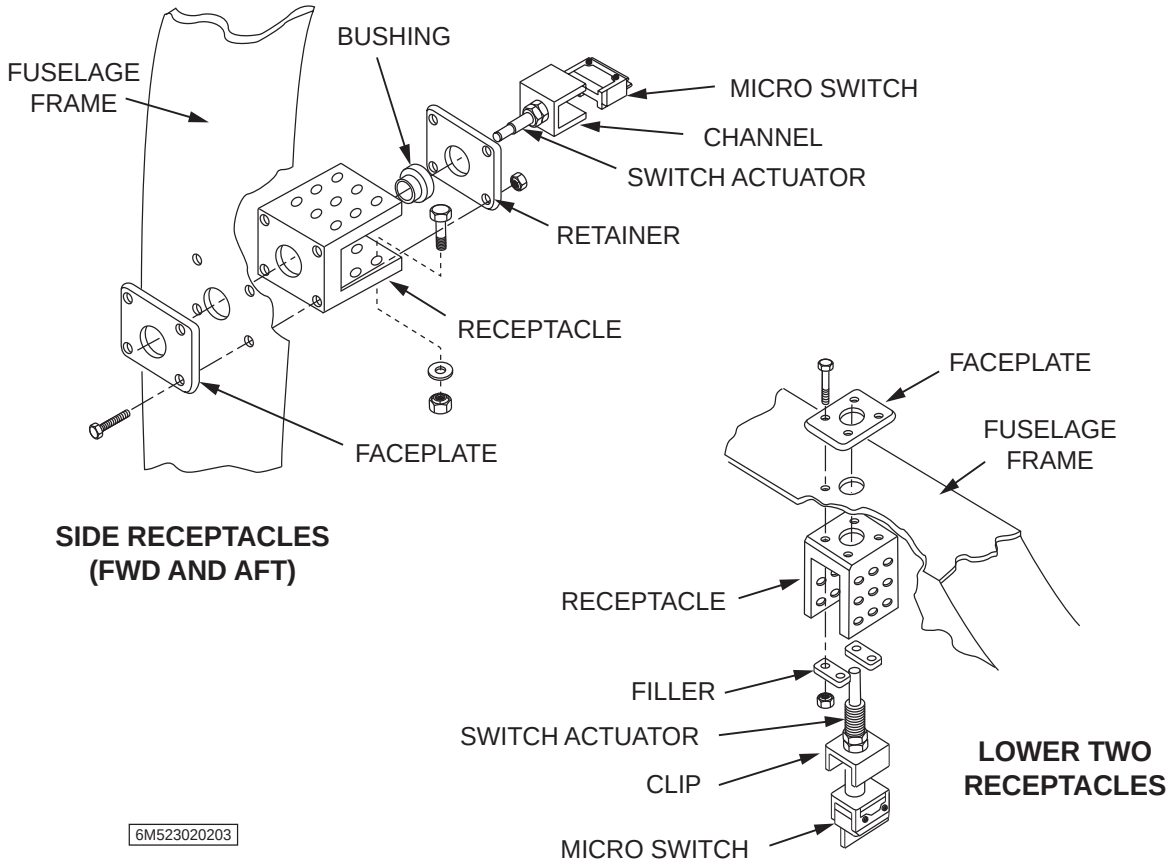
(1) Remove door upholstery as necessary to gain access to bellcrank.

(2) Remove bolts, washers (or spacers), and nuts securing push-pull rod assemblies to bellcrank assemblies.

(3) Remove bolts, washers (or spacers), and nut securing bellcrank.

(4) Remove bellcrank.

CARGO DOOR FUSELAGE FRAME RECEPTACLE INSTALLATION (TYPICAL)



Latching System – Fuselage Frame Receptacles
Figure 203

H. Installation – Cargo Door Bellcranks (Figure 204)

(1) Position bellcrank for installation.

(2) Secure bellcrank with bolt, washer (or spacer), and nut. Install safety wire as appropriate.

(3) Install push-pull rods to bellcrank. Secure with bolts, washers (or spacers), and nuts.

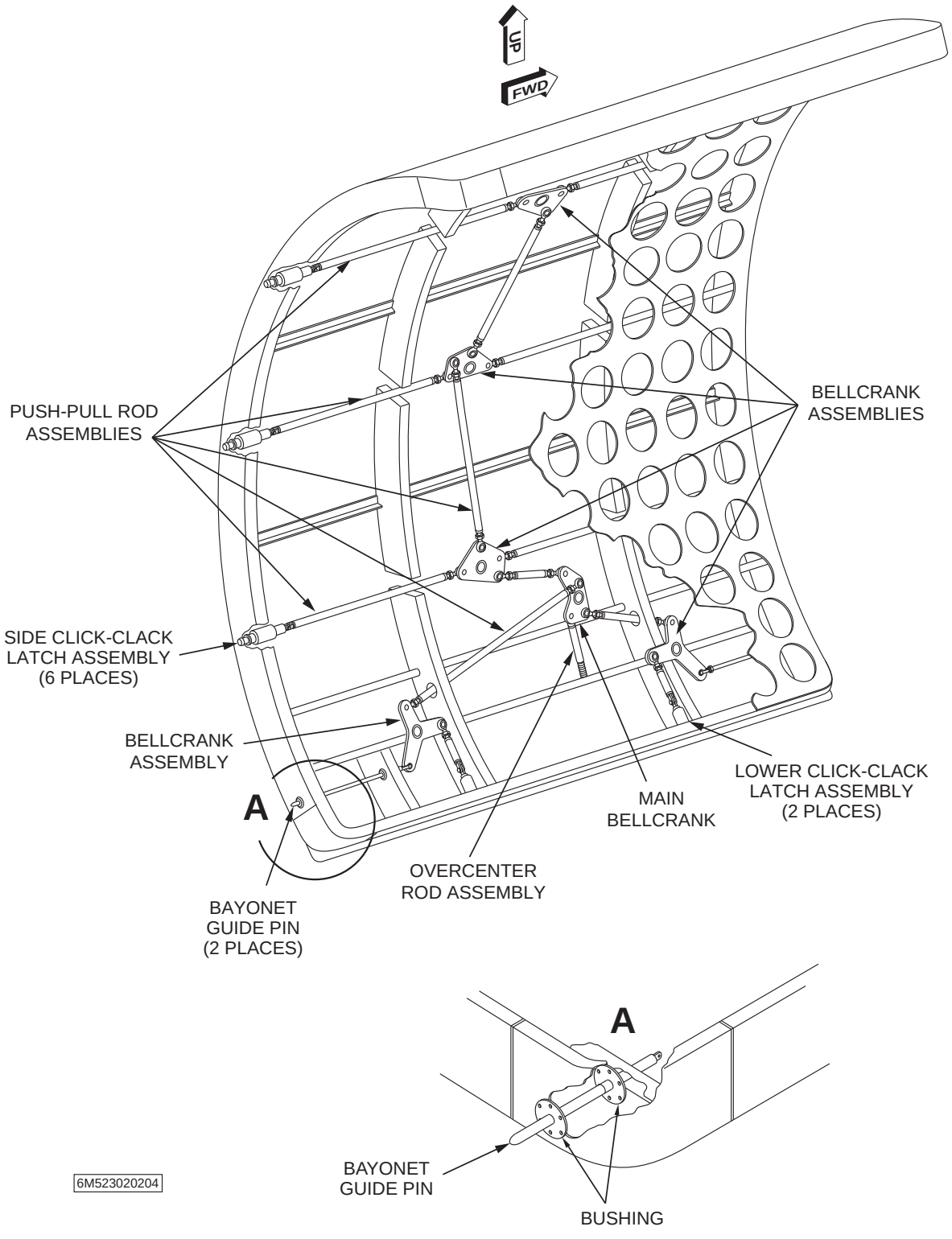
(4) Perform 3. A. Cargo Door Latch Mechanism Adjustment/Test, this section.

(5) Install door upholstery.

I. Removal – Cargo Door Overcenter Rod Assembly (Figure 204)

NOTE: The overcenter rod assembly is located just aft of bellcrank/push-pull rod assemblies to click-clack latch in position 8.

(1) Remove door upholstery as necessary to gain access to overcenter rod.



Bellcrank Assemblies/Push-Pull Rods/Overcenter Rod Assembly
Figure 204

CAUTION: OVERCENTER ROD ASSEMBLY IS SPRING LOADED. SECURE WITH ONE HAND WHEN REMOVING ASSEMBLY FROM BELLCRANK.

(2) Remove screw, washer, and nut securing overcenter rod assembly to bellcrank.

(3) Remove bolt, washer (or spacer), and nut securing overcenter rod to bracket on clip stringer.

(4) Remove overcenter rod assembly.

J. Installation – Cargo Door Overcenter Rod Assembly (Figure 204)

(1) Position overcenter rod for installation.

(2) Install overcenter rod to bracket on clip stringer. Secure overcenter rod with bolt, washer (or spacer), and nut.

(3) Connect overcenter rod assembly to bellcrank. Secure with bolt, washer, and nut.

(4) Turn handle several times to ensure proper operation.

(5) Install door upholstery.

K. Removal – Cargo Door Push-Pull Rods (Figure 204)

(1) Remove door upholstery to gain access to push-pull rods.

(2) Remove bolt, washer (or spacer), and nut from push-pull rod at bellcrank end.

(3) Disconnect push-pull rod at latching mechanism as follows:

(a) For positions 1 through 6, remove bolt, washer, and nut connecting push-pull rod to click-clack assembly.

(b) At bayonet guide pin positions, remove bolt, washer, and nut connecting push-pull rod to bayonet guide pins.

(c) For positions 7 and 8, remove nut and cotter pin connecting push-pull rod to click-clack latch.

(4) Remove push-pull rod from door.

L. Installation – Cargo Door Push-Pull Rod Assemblies (Figure 204)

(1) Position push-pull rod in cargo door.

(2) Connect push-pull rod to door latching mechanisms as follows:

(a) For positions 1 through 6, connect push-pull rod to click-clack latch assembly using bolt, washer, and nut.

(b) For bayonet guide pin positions, connect push-pull rod to guide pin bayonet. Secure with bolt, washer, and nut .

(c) For positions 7 and 8, connect push-pull rod to click-clack latch assembly. Secure with nut and cotter pin

(3) Connect push-pull rod with bolt to bellcrank. Secure with bolt, washer, (or spacers), and nut.

(4) Use handle to open and close door latching assemblies several times. Ensure proper operation of bellcranks, push-pull rods, and latching mechanisms.

(5) Perform 3. A. Cargo Door Latch Mechanism Adjustment/Test, this section.

(6) Install door upholstery.

M. Removal – Cargo Door Bayonet Guide Pins (Figure 204)

(1) Remove door upholstery to gain access to bayonet guide pin assembly.

(2) Remove bolt, washer, and nut connecting push-pull rod to bayonet guide pin.

(3) Withdraw bayonet guide pin from bushings.

N. Installation – Cargo Door Bayonet Guide Pins (Figure 204)

(1) Position and insert bayonet guide pin into bushings.

(2) Install push-pull rod to bayonet guide pin. Secure with bolt, washer, and nut.

(3) Operate door handle several times checking for proper alignment and operation of bayonet guide pins.

(4) Install door upholstery.

3. Adjustment/Test – Cargo Door Latch Mechanism

A. Cargo Door Latch Mechanism Adjustment/Test

NOTE: The following procedure duplicates an Airworthiness Limitation Item (ALI) procedure in the Airworthiness Limitations Manual (ALM). This procedure shall not be changed without prior FAA approval of the same change to the ALI procedure in the ALM.

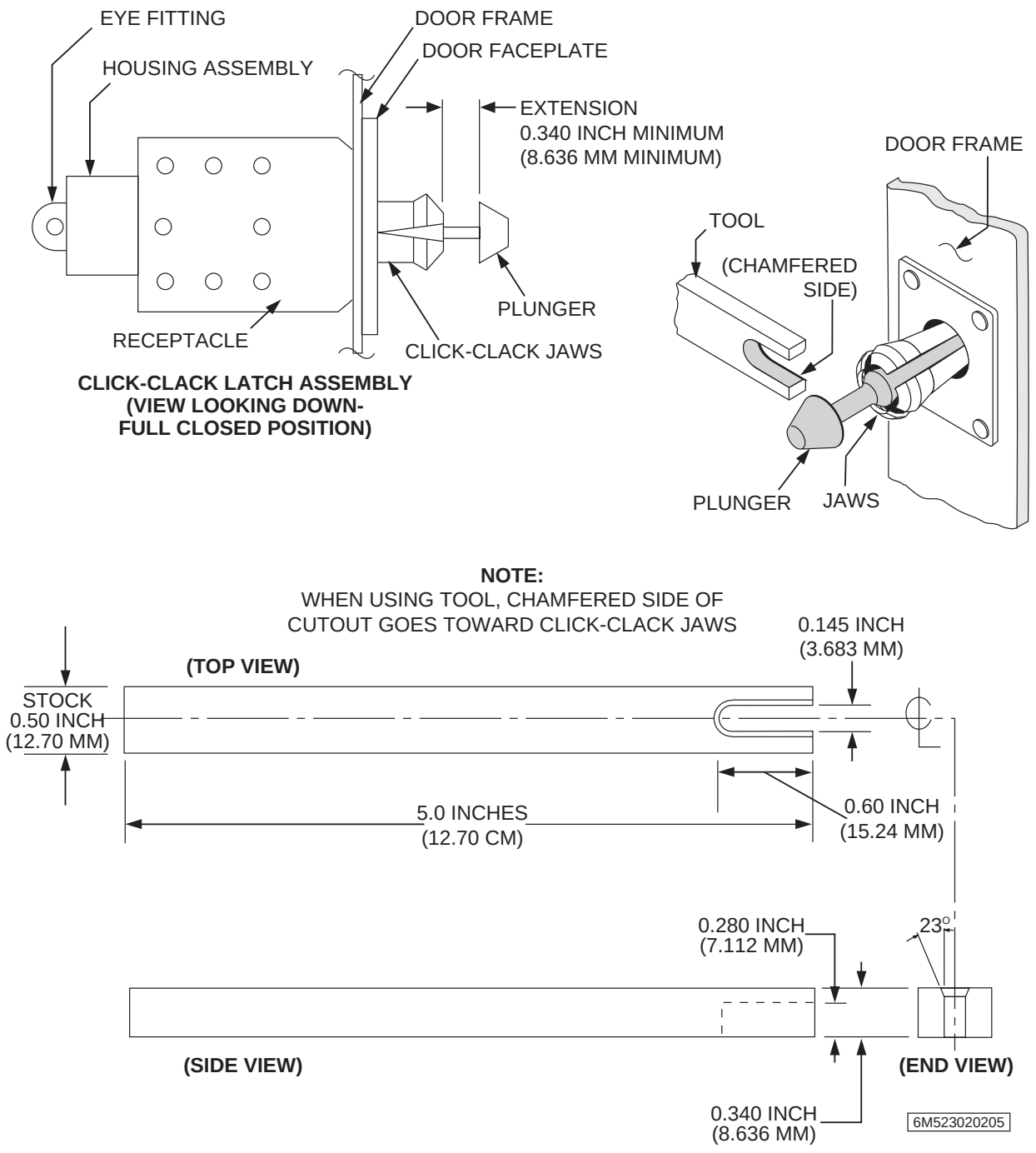
(1) With the cargo door open, conduct the following inspection to assure full expansion of the latch actuator jaws.

(a) Move the door handle to the door closed position.

(b) Refer to Illustrated Tool and Equipment Manual or Figure 205 for Go/No-Go Gauge P/N SAT1102.

(c) Using "go/no-go" gauge, verify the distance from inside of actuator plunger face to edge of the actuator jaws is not less than 0.34 inches (8.63 mm) (see Figure 205).

(d) Adjust each actuator as necessary to gain a minimum of 0.34 inches by varying the length of its connecting push-pull rod.



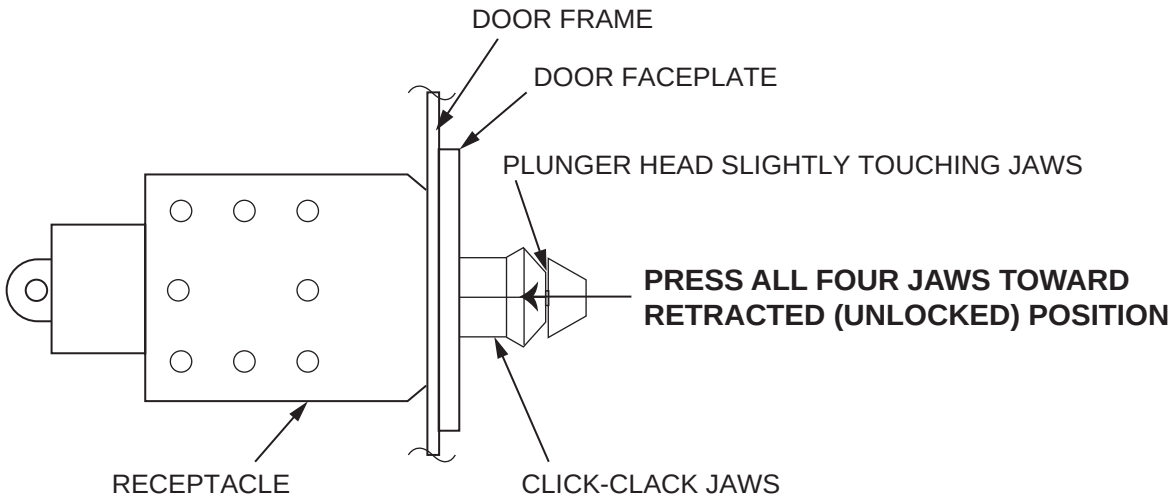
Click-Clack Latch Actuator Jaws to Plunger Measurement
Figure 205

(2) With cargo door open and the handle in LOCKED position, rotate handle back toward OPEN position until the plunger head lightly touches and holds all four jaws together (see Figure 206). Hold handle in this position.

(a) Press jaws of each latch actuator toward retracted position (UNLOCKED).

(b) Rotate actuator jaws and repeat at several positions around plunger.

(c) If any jaw can be pushed inward by exerting finger pressure, reject latch actuator.



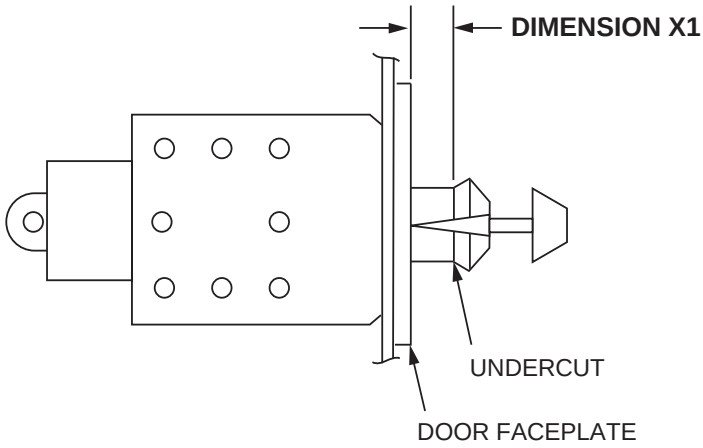
Click-Clack Latch Actuator Jaw Integrity Test
Figure 206

(3) Using an inside micrometer, or equivalent, measure and record the inside diameter of each receptacle (or bushing if installed) located in the fuselage door frame.

(a) Measure across the hole in at least three directions to check for roundness. The widest dimension must be used to compare with the maximum allowable dimensions provided in Table 201.

(b) If any receptacle is oversized, replace with an airworthy part.

(c) Determine the X2 value from Table 201 for each receptacle (or bushing) based on inside diameter measured. Record X2 value in Table 202.

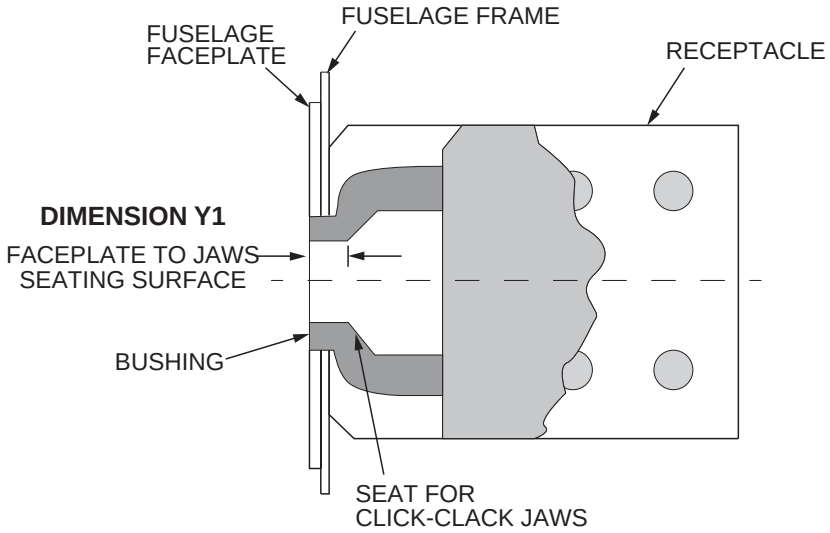


Dimension X Measurement
Figure 207

(4) To assure proper adjustment of the latch actuator jaws into the door frame receptacle, three additional measurements are necessary:

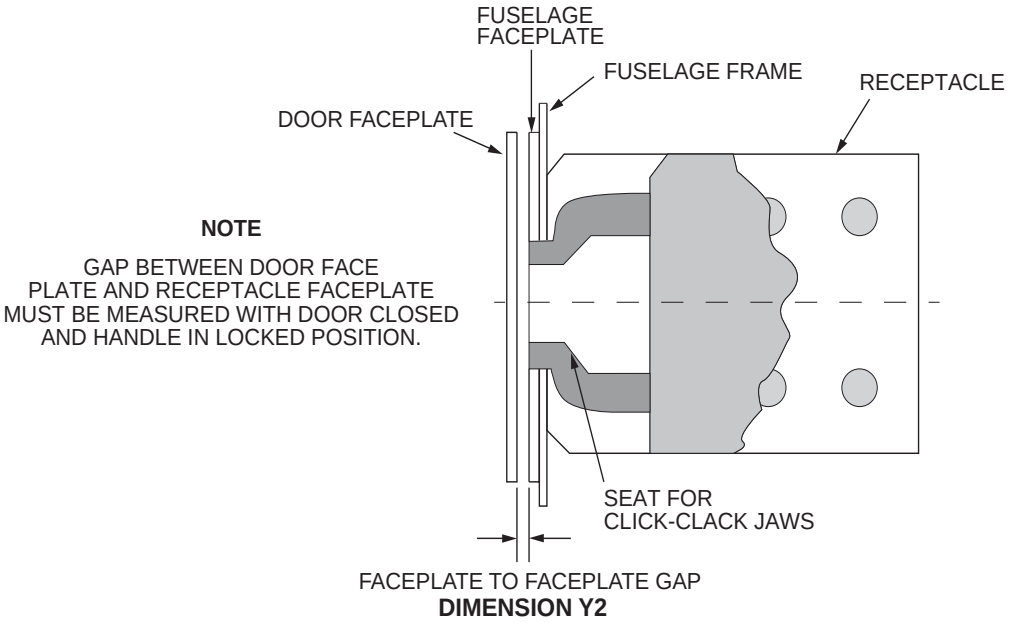
(a) The first measurement (Dimension X1) is the dimension from the door face plate surface to the undercut on the actuator jaws with the door open and handle in LOCKED position (see Figure 207). Measure and record as Dimension X1 in Table 202 for each latch position.

(b) The second measurement (Dimension Y1) is the dimension from the fuselage frame receptacle face plate surface to the actuator jaws seating surface inside the fuselage frame receptacle with door open (see Figure 208). Measure and record as Dimension Y1 in Table 202 for each latch position receptacle.



Dimension Y1 Measurement
Figure 208

(c) The third measurement (Dimension Y2) is the gap (taken with a feeler gauge with door closed and handle in locked position) between the surface of the fuselage frame face plate and the door frame face plate. Measure and record as Dimension Y2 in Table 202 for each latch position (see Figure 209).



Dimension Y2 Measurement
Figure 209

(5) In Table 202 for each door latch position:

(a) Add X1 to X2 and enter sum as X dimension.

(b) Add Y1 to Y2 and enter sum as Y dimension.

(c) Determine if X dimension is equal to or greater than dimension Y and enter Yes or No as applicable.

(d) Dimension X must be equal to or greater than dimension Y. If Dimension X is less than dimension Y, replace the fuselage receptacle/bushing with an airworthy part and repeat above measurements to obtain X and Y dimensions.

POSITION	TYPE	LIMIT (INSIDE DIAMETER
Bottom	Receptacle Only (No Bushing)	0.670 inches (17.018 mm)
Side	Receptacle With Bushing	0.700 inches (17.78 mm)
Side	Receptacle Only (No Bushing)	0.690 inches (17.526 mm)

X2 FOR ENLARGED RECEPTACLE/BUSHING HOLES								
RECEPTACLE/BUSHING DIAMETER								
Inches	0.630	0.640	0.650	0.660	0.670	0.680	0.690	0.700
Millimeters	16.002	16.256	16.510	16.764	17.018	17.272	17.526	17.780
X2								
Inches	0.000	0.006	0.012	0.018	0.024	0.030	0.036	0.042
Millimeters	0.000	0.1524	0.3048	0.4572	0.6096	0.7620	0.9144	1.0668

Door Receptacle/Bushing Wear Limits and X2 Value
Table 201

NOTE: When replacing bushings, check dash number marked on bushing. If number has an S, contact M7 Aerospace Field Service Engineering for replacement bushing, (reference MRBSR-512 to Service Engineering).

LATCH POSITION ►	FWD TOP	FWD CNTR	FWD LWR	AFT TOP	AFT CNTR	AFT LWR	Bottom FWD	Bottom AFT
X1								
X2								
X = X1 + X2								
Y1								
Y2								
Y = Y1 + Y2								
Is X ≥ Y? (Yes/No)								

Latch Engagement Dimensions
Table 202

B. Door Frame Repair (Figure 210)

(1) Drill 5/32 relief hole in door frame approximately 0.32 inches (0.81 mm) from outer radius.

(2) Slot out from relief hole through radius of frame.

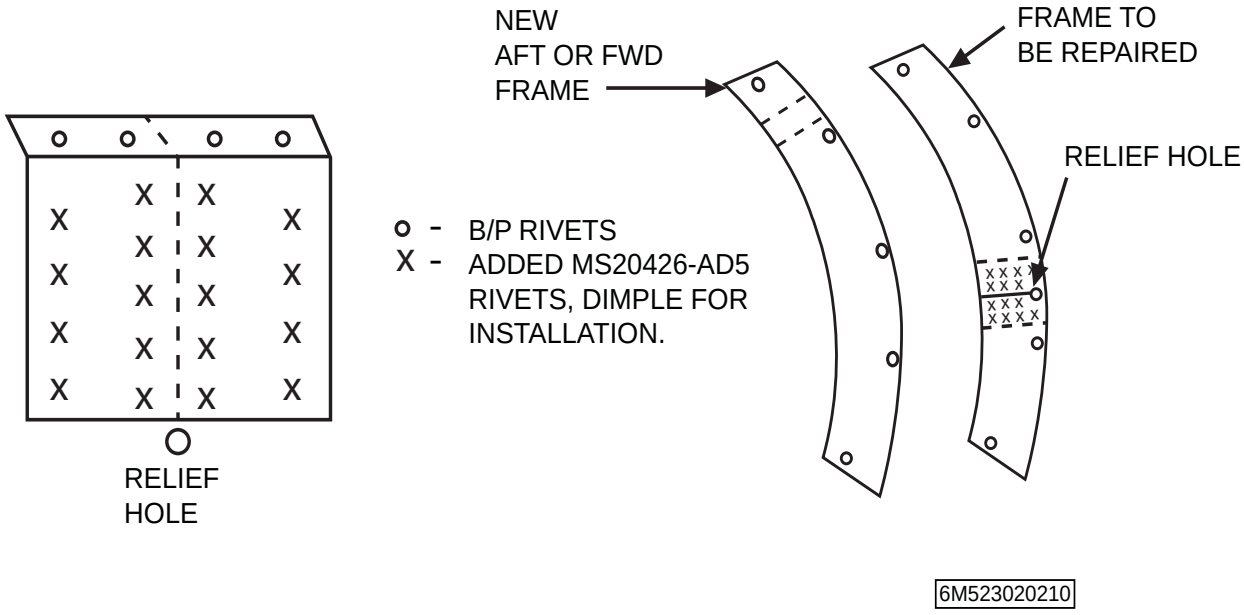
(3) Close door and reposition to best possible contour.

(4) Splice with an equal size, type, and thickness material.

NOTE: Material needed – new aft or forward frame.

(5) Assemble door to a maximum of 0.060 inches (1.524 mm) step and relocate latch assembly in door if required, see **3.**, D. Misaligned Click-Clack Repair.

(6) Fill slot and relief hole flush with Devcon F and polish.



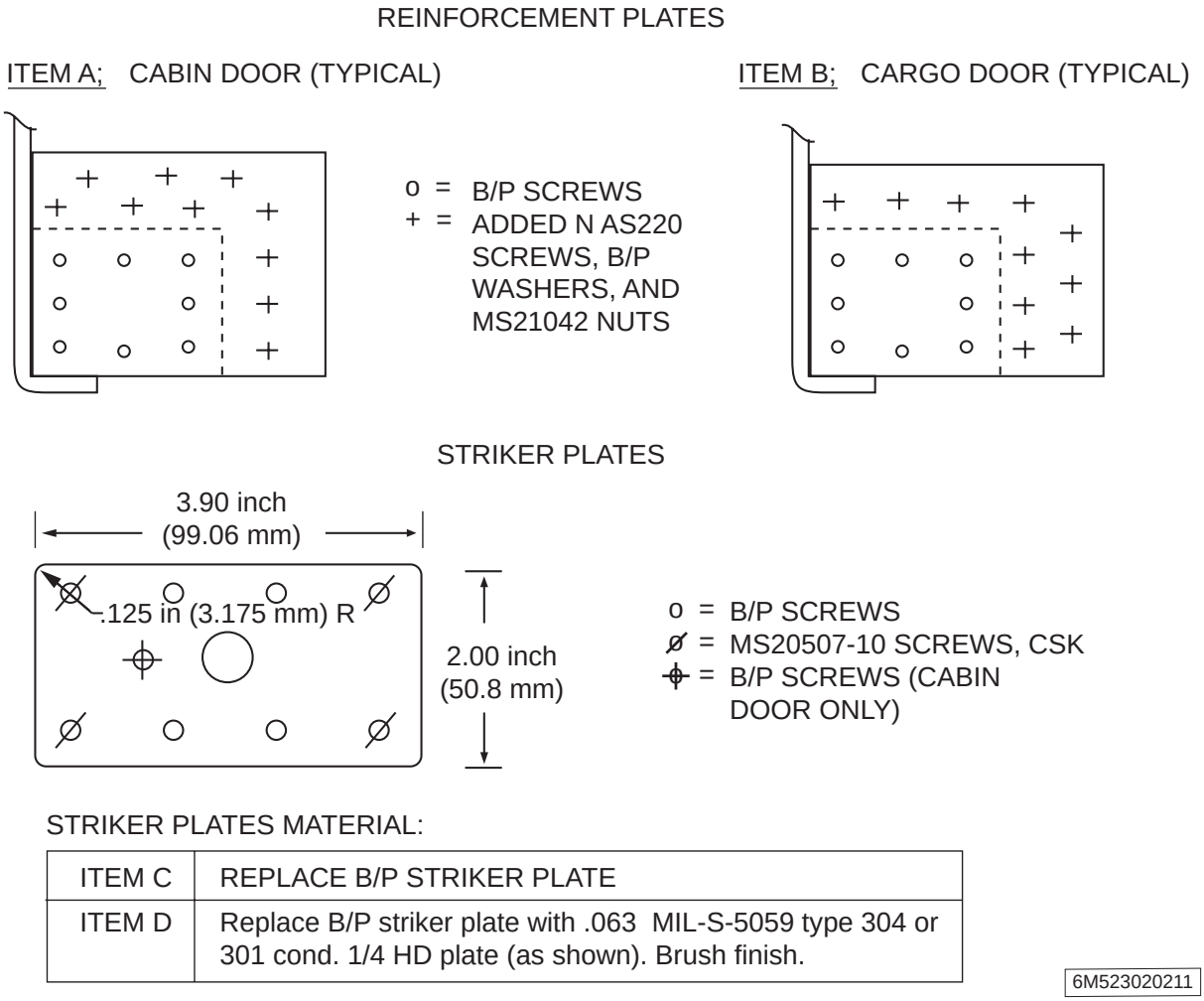
Door Frame Repair
Figure 210

C. Misaligned Click-Clack Repair

(1) Disassemble as required.

(2) Relocate applicable click-clack assemblies on door to match fuselage frame.

(3) Manufacture and install primed 0.040 inch (1.016 mm) thick 2024-T3 reinforcement plate(s) and or/applicable striker plate(s) as shown in Figure 211.



STRIKER PLATES MATERIAL:

ITEM C	REPLACE B/P STRIKER PLATE
ITEM D	Replace B/P striker plate with .063 MIL-S-5059 type 304 or 301 cond. 1/4 HD plate (as shown). Brush finish.

6M523020211

Reinforcement/Striker Plates
Figure 211

4. Inspection/Check – Main Cargo Door Latching System

A. Perform the following inspection checks on the cargo door latching system.

(1) Check cargo door latching mechanisms for general condition, security, and proper operation.

(a) Inspect bellcrank assemblies, push-pull rods, and rod ends for general condition, alignment, wear, and security.

(b) Check click-clack latches and bayonet guide pins for adjustment, nicks, burrs, alignment, general condition, and proper operation.

(c) Clean and lubricate click-clack latches and bayonet guide pins.

(2) Inspect door structure for cracks, loose rivets, and general condition.

(3) Perform 3.A. Cargo Door Latch Mechanism Adjustment/Test in this section.

(4) Correct any exposed discrepancies prior to the next flight.

MAIN CARGO DOOR COUNTERBALANCE – MAINTENANCE PRACTICES

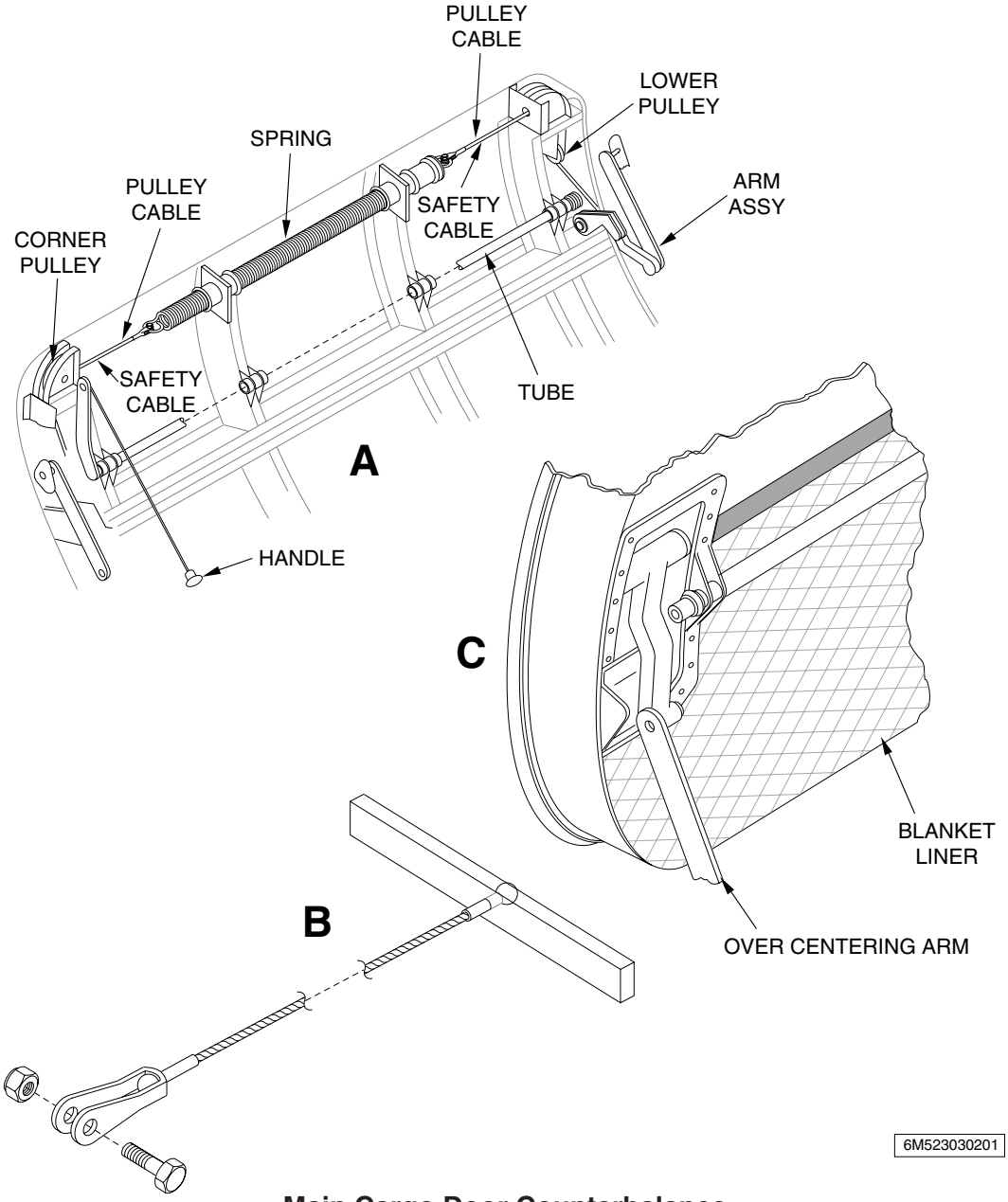
1. Removal/Installation – Main Cargo Door Counterbalance

A. Removal – Cargo Door Counterbalance Spring (Figure 201)

(1) Open cargo door.

(2) Secure overcentering arms by looping tie strap or similar device through arm stop bracket and around arm.

(3) Remove upper one-third of door upholstery to gain access to spring.



Main Cargo Door Counterbalance
Figure 201

(4) At either end of spring, cut tie straps holding safety cable and pulley cable together.

NOTE: Manufacture a spring pulling tool as shown in Figure 201, inset B.

(5) While assistant stretches spring using spring pulling tool to release tension, unfasten end of safety cable at spring. (Safety cable is the short cable fastened to the door frame).

(6) Unfasten pulley cable from spring, and slowly release spring.

(7) Unfasten both cables from other end of spring, and remove spring through guides.

B. Installation – Cargo Door Counterbalance Spring (Figure 201)

(1) Position spring through spring guides, and connect both cable ends to spring. Insert pulley cable onto spring first.

(2) Safety the cable ends onto spring by inserting two small tie straps through cable clevises.

(3) While assistant stretches spring connect cables (pulley cable first) to other end of spring, safety with tie straps IAW step **1.**, B., (2).

(4) Install door upholstery.

(5) Remove tie straps from arm and brackets.

C. Removal – Cargo Door Counterbalance Pulley Cable (Figure 202)

(1) Remove cargo door, IAW Removal– Main Cargo Door.

(2) Perform steps **1.**, A., (3) through (6).

(3) Remove access plate at corner of door.

(4) Remove two cable guides from corner pulley bracket.

(5) Unfasten cable end from arm assembly, and remove cable through pulleys.

D. Installation – Cargo Door Counterbalance Pulley Cable (Figure 202)

(1) Position cable through pulley, and insert cable guides at corner pulley.

(2) Connect cable to arm assembly.

(3) While assistant stretches counterbalance spring, connect cable to spring.

(4) Connect safety cable to spring, and safety with tie straps.

(5) Install access plate at corner.

(6) Install door upholstery.

(7) Install cargo door IAW Installation – Main Cargo Door.

E. Removal – Counterbalance Pulley (Corner Pulley) (Figure 202)

(1) Remove cargo door, IAW Removal– Main Cargo Door.

(2) Perform steps **1.**, A., (3) through (6).

(3) Remove access plate at corner of door.

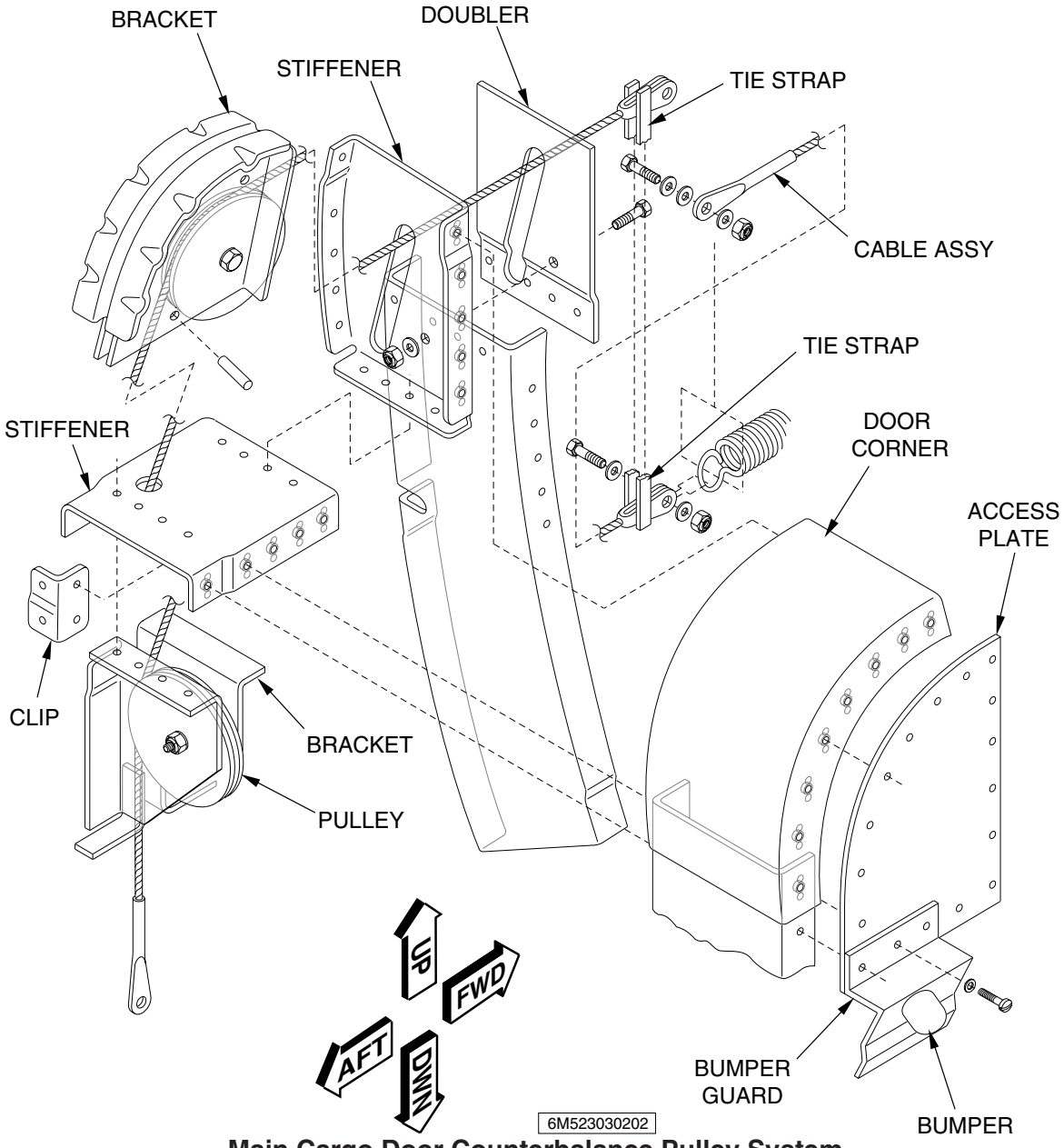
(4) Remove two cable guides from corner pulley bracket.

(5) Remove bolt and washer securing pulley, and remove pulley through hole in inner skin.

F. Installation – Counterbalance Pulley (Corner Pulley) (Figure 202)

(1) Position pulley in place, and position cable onto pulley.

(2) Secure pulley with bolt and washer.



Main Cargo Door Counterbalance Pulley System
Figure 202

(3) Insert pulley guides into pulley bracket.

(4) While assistant stretches counterbalance spring, connect cable to spring.

(5) Connect safety cable to spring, and safety with tie straps.

(6) Install access plate at corner.

(7) Install door upholstery.

(8) Install cargo door IAW Installation – Main Cargo Door.

G. Removal – Counterbalance Pulley (Lower Pulley) (Figure 202)

(1) Remove cargo door, IAW Removal– Main Cargo Door.

(2) Perform steps **1.**, A., (3) through (6).

(5) Remove bolt and washer securing pulley, and remove pulley through arm assembly access opening.

H. Installation – Counterbalance Pulley (Lower Pulley) (Figure 202)

(1) Position pulley in place, and position cable onto pulley.

(2) Secure pulley with bolt and washer.

(3) While assistant stretches counterbalance spring, connect cable to spring.

(4) Connect safety cable to spring, and safety with tie straps.

(5) Install door upholstery.

(6) Install cargo door IAW Installation – Main Cargo Door.

MAIN CARGO DOOR HANDLES – MAINTENANCE PRACTICES

1. Removal/Installation – Main Cargo Door Handles

A. Removal – Cargo Door Outer Handle (Figure 201)

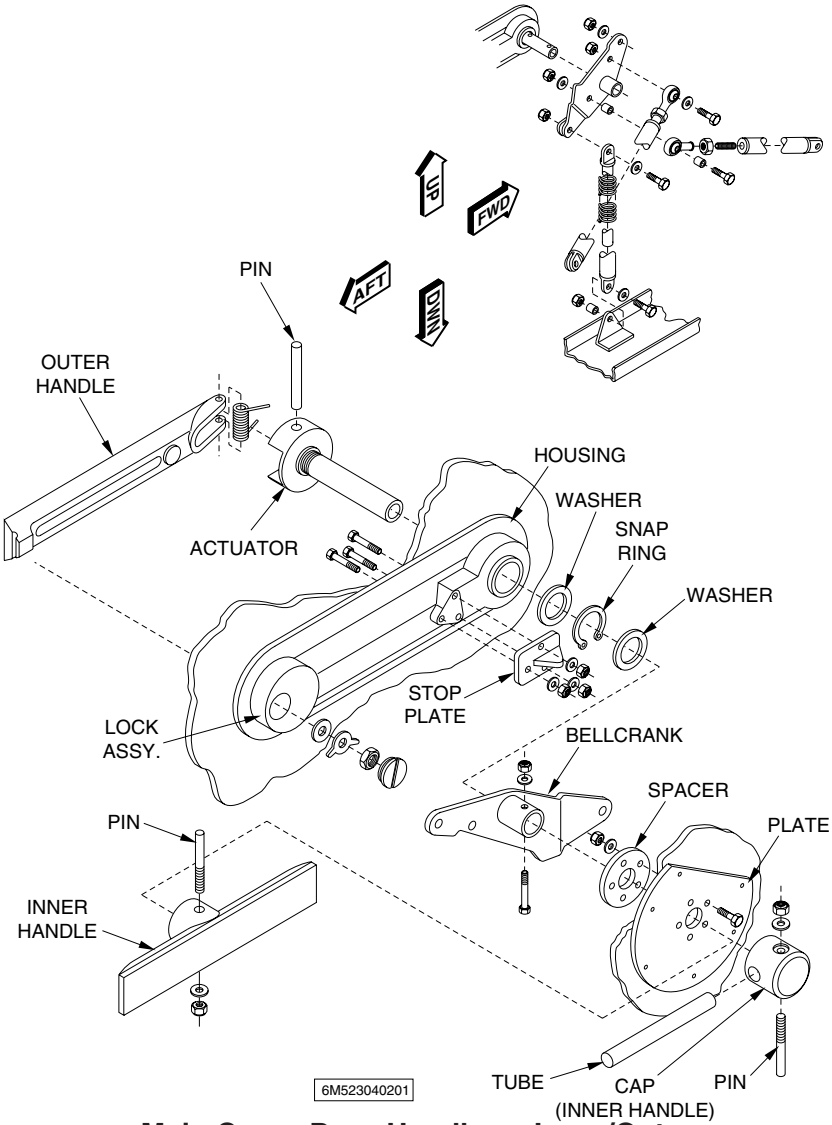
(1) Open cargo door.

(2) Lock outer door handle to immobilize door handle actuator assembly and bellcrank assemblies.

NOTE: Locking outer door handle will prevent handle assembly from accidentally being dropped when assemblies are disconnected.

(3) Remove upholstery/blanket liner from cargo door.

(4) Remove cap or inner handle (whichever applies to aircraft) by removing retaining pin, spacer, and nut.



Main Cargo Door Handles – Inner/Outer
Figure 201

(5) Remove push-pull rod assemblies attached to bellcrank:

(a) Remove bolt, spacers, and nut connecting push-pull rod assembly to bellcrank assembly for position 8 click-clack latch.

(b) Remove bolt, spacers, and nut connecting push-pull rod assembly to bellcrank assembly for position 7 click-clack latch.

(c) Remove bolt, washer, spacers, and nut connecting push-pull rod assembly to bellcrank assembly for positions 3 and 6 click-clack latches and bayonet guide pins.

CAUTION: OVERCENTERING CAM ROD ASSEMBLY IS SPRING LOADED. SECURE WITH ONE HAND WHILE REMOVING ASSEMBLY FROM BELLCRANK.

(d) Remove screw, washer, and nut connecting overcentering cam rod assembly from bellcrank assembly.

(6) Remove tapered pin, spacer, and nut securing bellcrank assembly to outer door handle actuator assembly.

(7) Remove bellcrank from outer door handle actuator.

(8) Remove outer washer, snap ring, and inner washer from outer door handle actuator.

(9) Unlock outer door handle. Remove outer door handle assembly and actuator assembly from housing.

B. Installation – Cargo Door Outer Handle (Figure 201)

(1) Install outer door handle and actuator assemblies through housing.

(2) Lock outer door handle to immobilize door handle actuator assembly and bellcrank assemblies.

(3) Secure outer door handle and actuator assemblies with washer, snap ring, and inner washer.

(4) Install bellcrank to outer door handle actuator. Secure with tapered pin, spacer, and nut.

(5) Install push-pull rod assemblies to bellcrank assembly:

(a) Install push-pull rod assembly to bellcrank assembly for positions 3 and 6 click-clack latches and bayonet guide pins. Secure with bolt, washer, spacers, and nut.

(b) Install push-pull rod assembly to bellcrank assembly for position 7 click-clack latch. Secure with bolt, spacers, and nut.

(c) Install push-pull rod assembly to bellcrank assembly for position 8 click-clack latch. Secure with bolt, spacers, and nut.

CAUTION: OVERCENTERING CAM ROD ASSEMBLY IS SPRING LOADED. SECURE WITH ONE HAND WHILE REMOVING ASSEMBLY FROM BELLCRANK.

(d) Install overcentering cam rod assembly to bellcrank assembly. Secure with screw, washer, and nut.

(6) Install cap or inner door handle (whichever applies to aircraft). Secure with retaining pin, spacer, and nut.

(7) Unlock outer door handle. Operate handle mechanism several times before closing door to check for smooth operation.

(8) Check click-clack latches for proper adjustment. Adjust latches IAW Adjustment/Test – Cargo Door Latching System, this section.

(9) Install blanket liner/upholstery to cargo door.

C. Removal – Cargo Door Assist Handle (Figure 202)

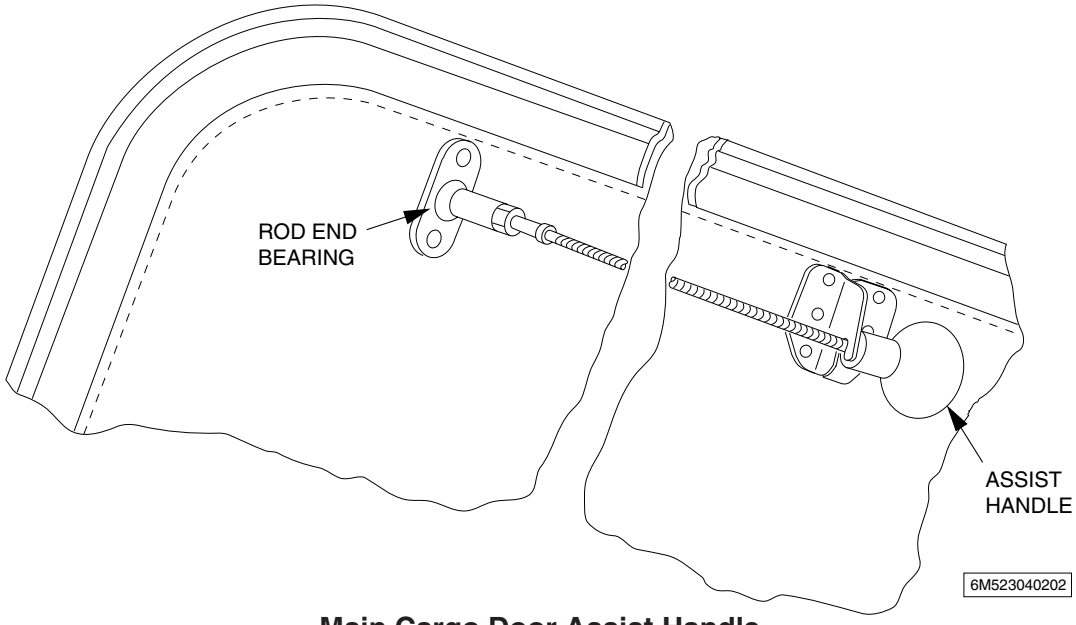
(1) Remove attaching hardware from rod end bearing.

(2) Remove handle assembly from cargo door.

D. Installation – Cargo Door Assist Handle (Figure 202)

(1) Secure handle assembly with attaching hardware.

(2) Open cargo door. Lift door up and down to check security of handle.



Main Cargo Door Assist Handle
Figure 202

2. Inspection/Check – Main Cargo Door Handles

A. Inspect main cargo door handles as follows:

(1) Check inner and outer door handles for adjustment, fit, corrosion, wear, security, and operation.

(2) Check safety lock for freedom of operation, security, and general condition.

(3) Check bellcrank and push-pull rod assemblies for alignment, wear, general condition, and operation.

(4) Check door assist handle for general condition and security.

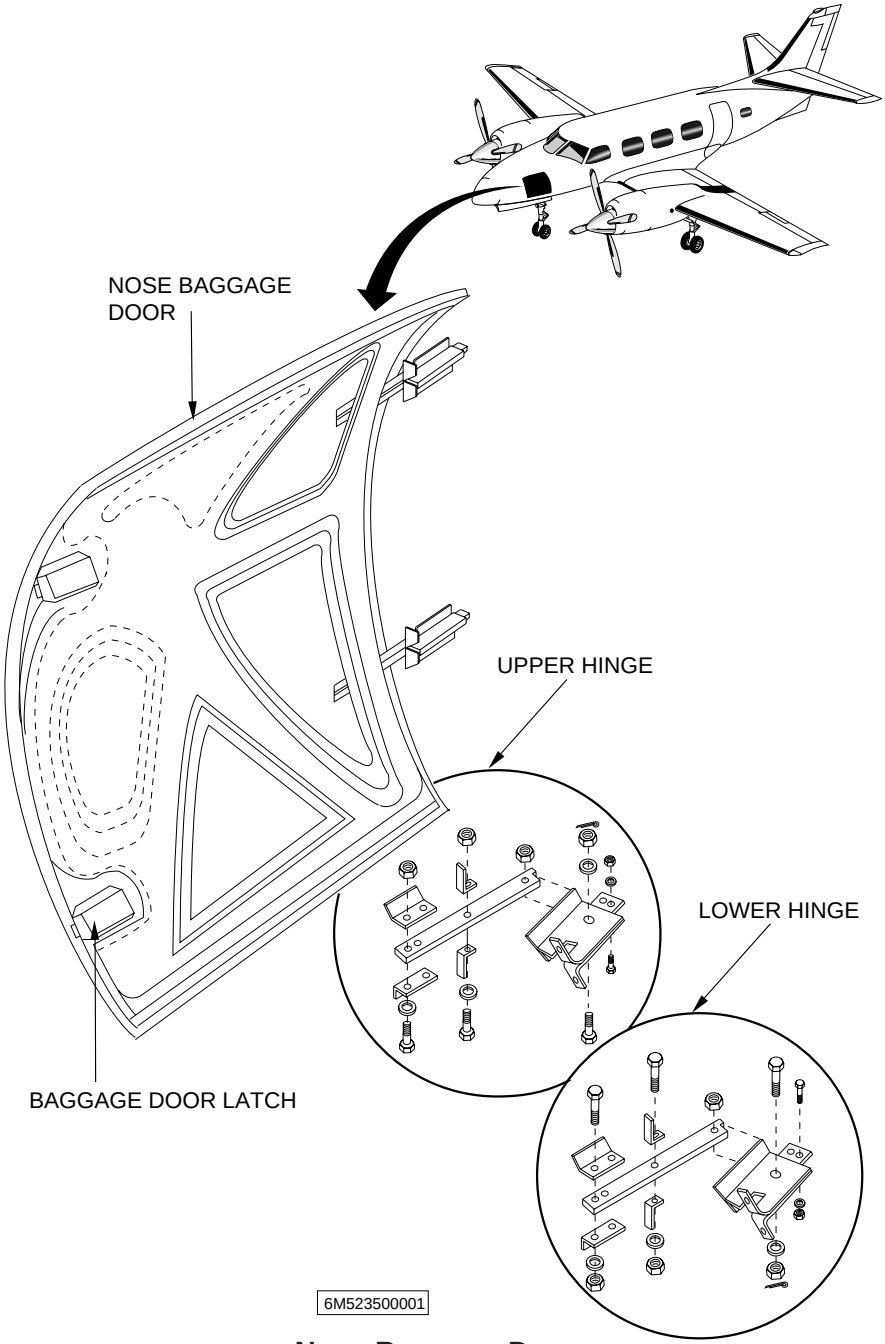
(5) Check door structure for cracks, loose rivets, corrosion, and general condition.

MAINTENANCE MANUAL

NOSE BAGGAGE DOORS – DESCRIPTION AND OPERATION

1. General

A. The nose baggage compartment on the SA226 – T, AT, and TC aircraft is located in the unpressurized nose section. Hinged doors, located on each side of the nose section between F.S. 33.0 and 51.3, allow access to the avionics and baggage area. Each door, approximately 18 inches wide X 26 inches high (45.72 cm wide X 66.04 cm high), has a key operated lock to provide security for the baggage compartment.



Nose Baggage Doors
Figure 1

NOSE BAGGAGE DOORS – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Baggage Doors

A. Removal – Nose Baggage Doors

(1) Unlock and open nose baggage door.

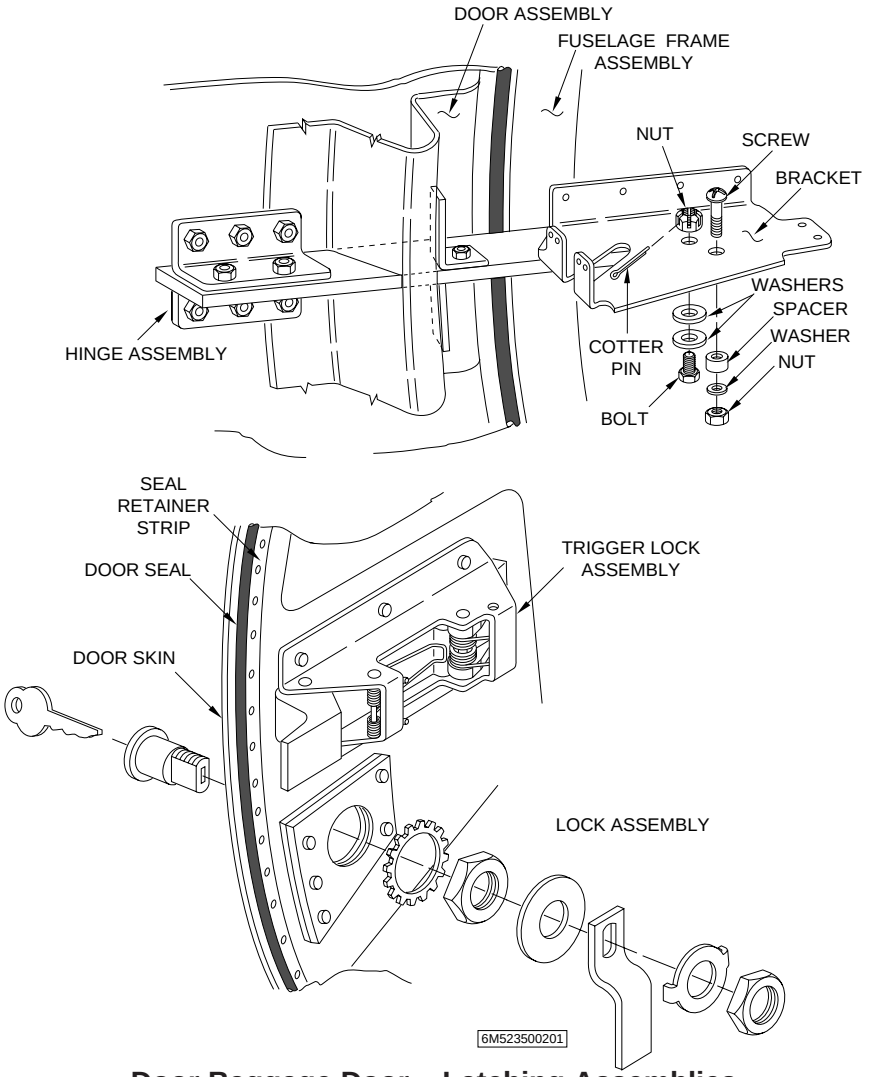
CAUTION: SUPPORT NOSE BAGGAGE DOOR DURING REMOVAL PROCEDURE TO AVOID STRUCTURAL DAMAGE TO HINGE, FUSELAGE, OR DOOR ASSEMBLIES.

(2) Remove cotter pin, nut, washers, and bolt from lower hinge at fuselage frame.

(3) Remove cotter pin, nut, washers, and bolt from upper hinge at fuselage frame.

NOTE: Upper hinge bolt is installed/removed with bolt head down.

(4) Remove door. Place door on a suitable workbench or cradle to avoid damage to finish or material.



Door Baggage Door – Latching Assemblies
Figure 201

B. Installation – Nose Baggage Doors

(1) Position door for installation.

(2) Mate hinges to brackets on fuselage frame forward of baggage door.

CAUTION: SUPPORT DOOR DURING INSTALLATION TO AVOID DROPPING OR DAMAGING ASSEMBLIES.

(3) Install bolt with head down, washers, nut, and cotter pin through upper bracket and hinge.

(4) Install bolt, washers, nut, and cotter pin through lower bracket and hinge.

(5) Open and close door several times to ensure proper operation and door alignment with fuselage.

(6) Conduct water leak test IAW Adjustment/Test, this section.

NOTE: Door seal should prevent entry of water into nose baggage compartment even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(7) Adjust door and door seal as necessary to ensure proper alignment and fit.

C. Removal – Nose Baggage Door Seal

(1) Open nose baggage door.

(2) Drill out rivets securing seal and retaining strips to door.

(3) Remove seal and retaining strips.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(4) Clean area thoroughly with MPK to ensure old sealant and rubber are removed.

D. Installation – Nose Baggage Door Seal

(1) Clean new seal with MPK to remove shipping preservatives.

(2) Ensure door frame where seal is to be installed, is clean. Use MPK to clean door frame prior to installing new seal.

(3) Sandwich door seal between door and retaining strips.

(4) Secure door seal and retaining strips with rivets.

(5) Sealant may be applied around hinge openings to prevent water leaks. Trim excess sealant to prevent hinge binding.

(6) Open and close door several times to ensure proper alignment. Adjust IAW Adjustment/Test, this section.

(7) Perform water leak test IAW Adjustment/Test, this section.

2. Adjustment/Test – Nose Baggage Doors

A. Water Leak Test – Nose Baggage Doors

(1) Position an assistant with a flashlight and marking pencil inside nose baggage compartment.

(a) Use flashlight to determine water seepage during test.

(b) Mark water entry point(s) with marking pencil.

NOTE: Door seals should prevent entry of water into aircraft nose baggage compartment even when parked. If water is leaking into aircraft, water leak test will determine point(s) of entry.

(2) Close and latch baggage doors.

(3) Use water hose, insecticide sprayer, or similar device that dispenses water under pressure.

(4) With nozzle approximately two (2) to three (3) inches (5.08 to 7.62 cm) from door, apply water completely around door edge foe several minutes.

(5) Open baggage door and check for seepage. No water leakage is allowed.

(6) If test indicates door is not sealing properly, proceed as follows:

(a) Close door and check contour fit with fuselage.

(b) Check for proper alignment and general condition of door seal.

1 If door seal is not sealing properly, peel seal away from door frame area. Use thin pieces of neoprene rubber to shim door and seal. Secure seal with RTV 3145 silicone sealant, or equivalent. Do not add an excessive amount of sealant.

2 Sealant (3M Brand, Super Weather-strip Adhesive) may be applied around the hinge opening(s) to prevent water seepage. Trim excess sealant to prevent hinge binding.

3. Inspection/Check – Nose Baggage Doors

A. Inspect Nose Baggage Doors as follows:

(1) Inspect left and right nose baggage doors for general condition and security.

(2) Inspect all door hinges, handles, and locks for operation, security, and general condition.

(a) Check handles for adjustment, fit, security, and operation.

(b) Inspect, clean, and lubricate door lock mechanisms.

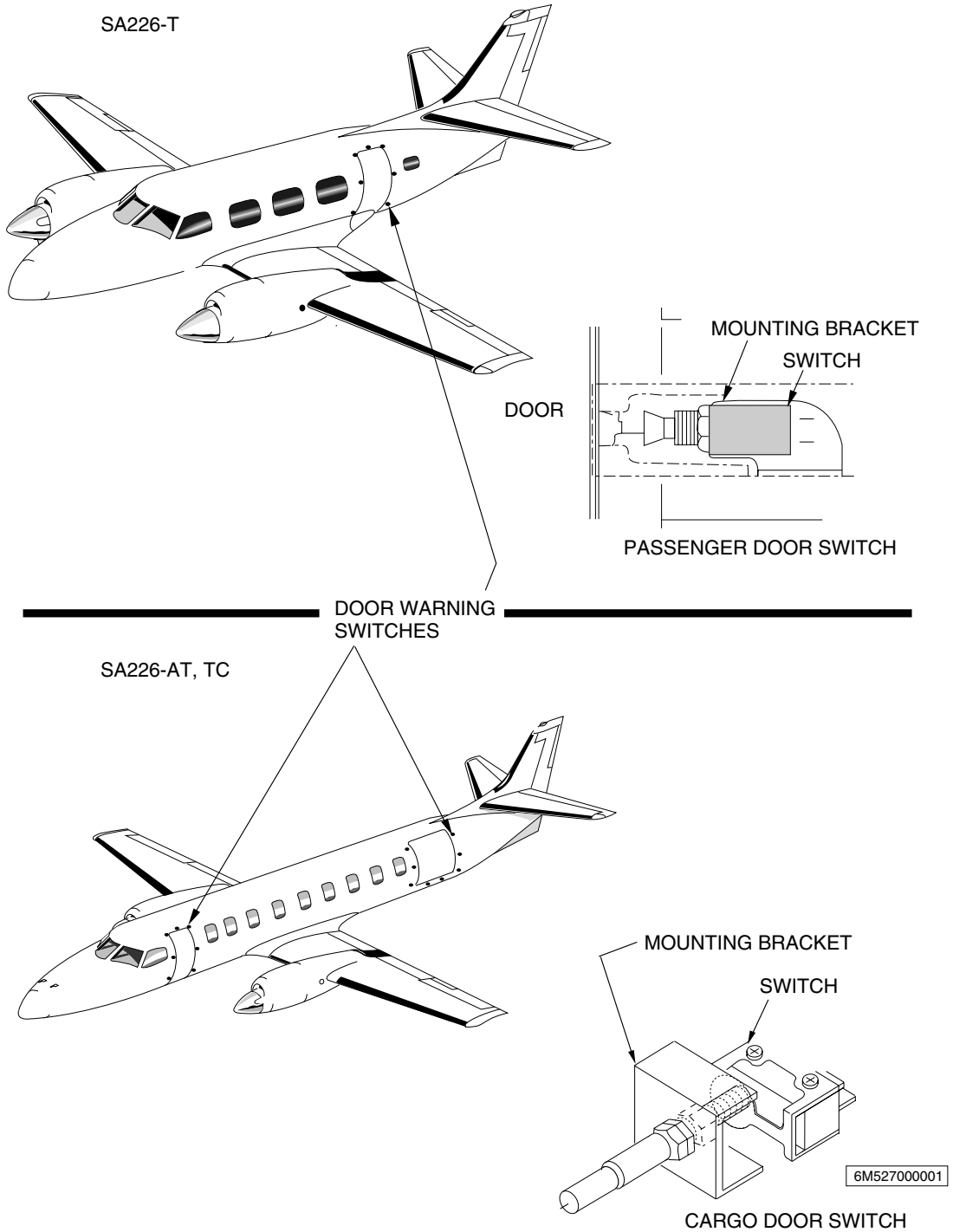
(3) Check door seal for general condition and security.

(4) Check doors and surrounding structures for cracks, loose rivets, and general condition.

DOOR UNLOCK WARNING SWITCHES – DESCRIPTION AND OPERATION

1. General

A. Door warning light microswitches are installed in passenger and main cargo door fuselage receptacles.



Door Unlock Warning Switches
Figure 1

DOOR UNLOCK WARNING SWITCHES – DESCRIPTION AND OPERATION

(1) Microswitches provide a ground for warning lights until latches are extended. Microswitches are opened by extending click-clack and bayonet latches. Failure of a latch to full extend will cause a warning light to illuminate on the cockpit annunciator panel.

(2) All microswitches are wired in parallel and will cause annunciator warning light(s) to remain illuminated until all latch assemblies are secure.

2. Passenger Door Unlock Warning Switches

A. A microswitch is installed in all seven latch assemblies on the passenger door. If any one of the latches has not extended sufficiently to open its respective microswitch, a red warning light on the annunciator panel will illuminate and remain on until each latch is secure.

3. Main Door Unlock Warning Switches

A. Serial numbers TC – 398-419 have a cargo door warning light circuit with warning lights on the RH console instead of at the cabin door sill. These aircraft have combination cargo door warning light and test circuit microwitches (two sets of contacts) installed in each door frame receptacle.

(1) One set of contacts provides a ground for the door warning light on the annunciator panel and a ground for the door warning light on the copilots instrument panel via the warning light relay.

(2) The other set of contacts operates a resistive circuit for the normal test switch green light on the right console. A microswitch operated by the door handle bellcrank is common to both the door unsafe and the switch normal circuits. A test jack is provided on the right console to troubleshoot the switch normal circuit with an ohmmeter.

PASSENGER DOOR UNLOCK WARNING SWITCHES – MAINTENANCE PRACTICE

1. Servicing – Passenger Door Unlock Warning Switches

A. Any microswitch/actuator assembly showing a tendency to stick in a depressed (closed) position should be removed, cleaned, and lubricated using Dow-Corning DC-4 silicone, or equivalent. Replace microswitch assemblies that do not respond to servicing.

2. Removal/Installation – Passenger Door Unlock Warning Switches

A. Removal – Passenger Door Unlock Warning Switches

(1) Ensure electrical power is OFF.

(2) Remove fuselage frame interior trim surrounding door, as necessary, to gain access to each switch.

NOTE: The passenger door has three (3) microswitches on each side and one (1) microswitch located at the top of the door as shown in Figure 201.

(3) Remove locknut, washer, and nut securing microswitch actuator assembly to fuselage frame and housing assembly.

(4) Depress actuator towards microswitch assembly. Withdraw actuator through mounting hole. Release actuator and withdraw microswitch assembly from fuselage.

(5) Disconnect electrical leads from microswitch assembly.

B. Installation – Passenger Door Unlock Warning Switches

(1) Ensure electrical power is OFF.

(2) Connect electrical leads to microswitch assembly.

(3) Position microswitch assembly to access opening in fuselage for installation.

(4) Depress actuator towards microswitch assembly. Insert assembly into access opening. Release and allow actuator to extend through mounting hole in fuselage frame and housing assembly.

(5) Install nut (finger tight), washer, and locknut (finger tight) securing microswitch assembly to fuselage frame and housing assembly.

(6) Turn electrical power ON.

(7) Place microswitch actuator against tip of click-clack latch. Adjust microswitch assembly IAW Adjustment/Test – Passenger Door Unlock Warning Switches.

(8) When microswitch assembly has been properly adjusted, secure nut, washer, and locknut.

(9) Install interior trim to fuselage frame surrounding passenger door.

3. Adjustment/Test – Passenger Door Unlock Warning Switches

A. Adjustment – Passenger Door Unlock Warning Switches

NOTE: Do not adjust microswitches until latch assemblies have been checked for positive engagement in receptacles IAW 52-10-10 Passenger Door Latch Mechanism Adjustment/Test.

(1) To gain access to switches, remove necessary interior trim panel(s).

(2) Close door, and put door handle in LOCKED position.

(3) Adjust each individual switch as follows:

(a) Verify wiring is connected to switch per Aircraft Wiring Manual 52-70-01.

(b) Select BATTERY switches to ON.

(c) With all cabin door warning switches other than the switch being adjusted depressed, verify CABIN DOOR warning light on annunciator panel extinguishes when switch is depressed and illuminates when switch is extended.

(d) With jam nuts securing affected microswitch to bracket loose, move microswitch until plunger of microswitch contacts latch tip. CABIN DOOR warning light on annunciator panel should be illuminated.

(e) Turn jam nuts to move microswitch towards latch until CABIN DOOR warning light on annunciator panel extinguishes.

(f) To compensate for pressurization loads on door structure in-flight, adjust the microswitch an additional 0.15 inch (3.81 mm) towards latch (maximum adjustment tolerance).

(g) Tighten jam nuts on microswitch.

(4) Verify any initial movement of the cabin door handle from the LOCKED position will illuminate the CABIN DOOR warning light.

(5) If switch is adjusted per step (3) and light does not extinguish, check adjustment of the other cabin door warning switches. If problem persists, isolate and correct defective wiring or replace switch(es) as required.

(6) Select BATTERY switches to OFF.

(7) Install removed trim panel(s).

B. Test – Passenger Door Unlock Warning Microswitches

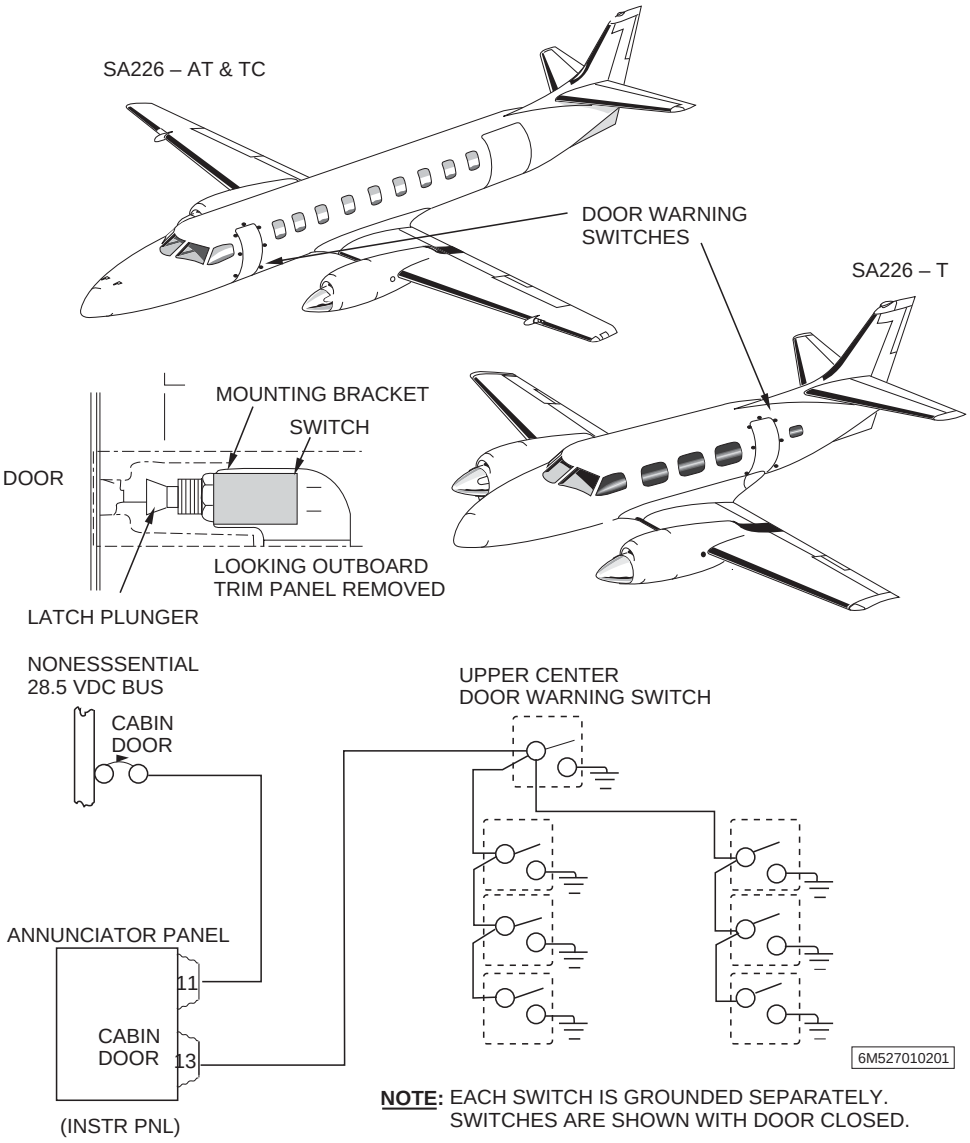
(1) Check door warning light operation.

(a) With door open, manually depress all door warning microswitch actuators. Check to ensure CABIN DOOR warning light on annunciator panel extinguishes.

(b) Selectively release and depress each microswitch actuator.

1 With all actuators depressed, check to ensure release of an individual actuator causes illumination of the door warning light.

2 Move each actuator in and out several times. Check that each actuator does not stick in a depressed position. Replace defective microswitches.



Passenger Door Warning System
Figure 201

4. Inspection/Check – Passenger Door Unlock Warning Switches

A. Inspect passenger door unlock warning microswitches as follows:

(1) Check operation of passenger door warning system/lights.

(2) Check each passenger door warning microswitch assembly individually to ensure warning lights illuminate.

(3) Check components for corrosion, wear, or signs of failure.

(4) Check support structures for security, corrosion, cracks, and wear.

CARGO DOOR UNLOCK WARNING SWITCHES – MAINTENANCE PRACTICES

1. Servicing – Cargo Door Unlock Warning Switches

A. Any microswitch/actuator assembly showing a tendency to stick in a depressed (closed) position should be removed, cleaned, and lubricated using Dow-Corning DC-4 silicone, or equivalent. Replace microswitch assemblies that do not respond to servicing.

2. Removal/Installation – Cargo Door Unlock Warning Switches

A. Removal – Cargo Door Unlock Warning Microswitches

(1) Ensure electrical power is OFF.

(2) Remove fuselage frame interior trim surrounding door, as necessary, to gain access to microswitch access opening(s).

NOTE: Eight (8) microswitches are located in the passenger door. Three (3) microswitches are on each side and two (2) microswitches are located at the bottom of the door (positions 7 and 8).

(3) Remove locknut, washer, and nut securing microswitch actuator assembly to fuselage frame and housing assembly.

(4) Depress actuator towards microswitch assembly. Withdraw actuator through mounting hole. Release actuator and withdraw microswitch assembly from fuselage.

(5) Disconnect electrical leads from microswitch assembly.

B. Installation – Cargo Door Unlock Warning Microswitches

(1) Ensure electrical power is OFF.

(2) Connect electrical leads to microswitch assembly.

(3) Position microswitch assembly to access opening in fuselage for installation.

(4) Depress actuator towards microswitch assembly. Insert assembly into access opening. Release and allow actuator to extend through mounting hole in fuselage frame and housing assembly.

(5) Install nut (finger tight), washer, and locknut (finger tight) securing microswitch assembly to fuselage frame and housing assembly.

(6) Turn electrical power ON.

(7) Place microswitch actuator against tip of click-clack plunger. Adjust microswitch assembly IAW 3. A. Adjustment – Cargo Door Unlock Warning Switches, this section.

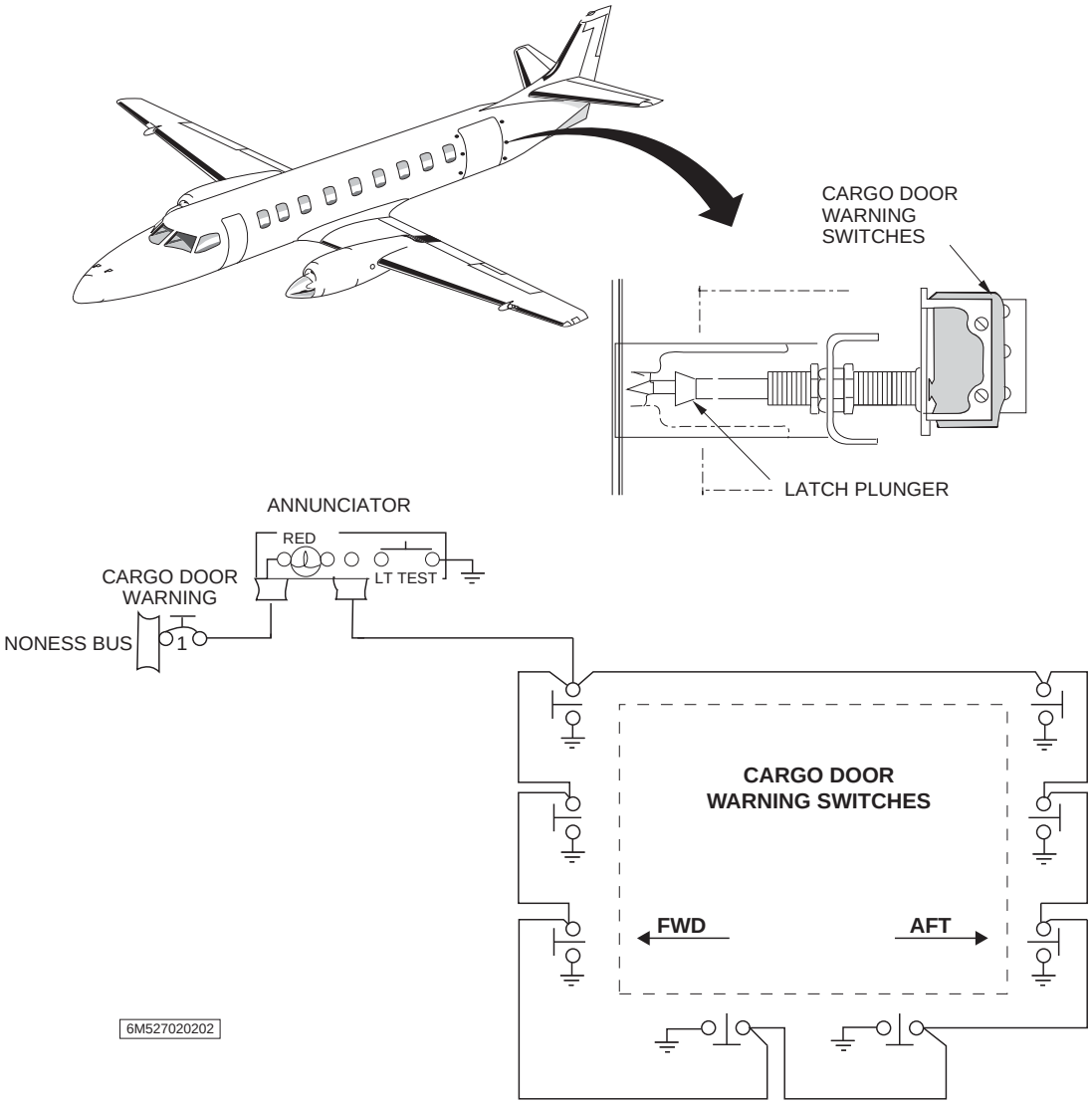
(8) When microswitch assembly has been properly adjusted, secure nut, washer, and locknut.

(9) Install interior trim/covering to fuselage frame surrounding passenger door.

C. Removal – Cargo Door Handle Position Switch

(1) Ensure electrical power is OFF.

(2) Remove handle and blanket liner from interior cargo door.



Cargo Door Wiring Diagram
TC – 201-397, AT – 001-397
Figure 201

(3) Remove screws securing microswitch assembly. Microswitch assembly is located adjacent to bellcrank assembly.

(4) Remove microswitch assembly.

(5) Disconnect electrical leads to microswitch assembly.

D. Installation – Cargo Door Handle Position Switch

(1) Ensure electrical power is OFF.

(2) Connect electrical leads to microswitch assembly.

(3) Position microswitch assembly adjacent to bellcrank assembly.

(4) Secure microswitch assembly with screws.

(5) Perform functional check of cargo door handle position switch.

(6) Install interior cargo door handle and blanket liner.

3. Adjustment/Test – Cargo Door Unlock Warning Switches

A. Adjustment – Cargo Door Unlock Warning Switches

NOTE: Do not adjust microswitches until latch assemblies have been checked for positive engagement in receptacles IAW 52-30-20 Cargo Door Latch Mechanism Adjustment/Test.

(1) To gain access to switches, remove necessary interior trim panels surrounding door.

(2) Close door, and put door handle in LOCKED position.

(3) Adjust each individual switch as follows:

(a) Verify wiring is connected to switch per Aircraft Wiring Manual 52-70-02.

(b) Select BATTERY switches to ON.

(c) With all cargo door warning switches other than the switch being adjusted depressed, verify CARGO DOOR warning light on annunciator panel extinguishes when switch is depressed and illuminates when switch is extended.

(d) With jam nuts securing affected microswitch to bracket loose, move microswitch until plunger of microswitch contacts latch tip. CARGO DOOR warning light on annunciator panel should be illuminated.

(e) Turn jam nuts to move microswitch towards latch until CARGO DOOR warning light on annunciator panel extinguishes.

(f) To compensate for pressurization loads on door structure in-flight, adjust the microswitch an additional 0.15 inch (3.81 mm) towards latch (maximum adjustment tolerance).

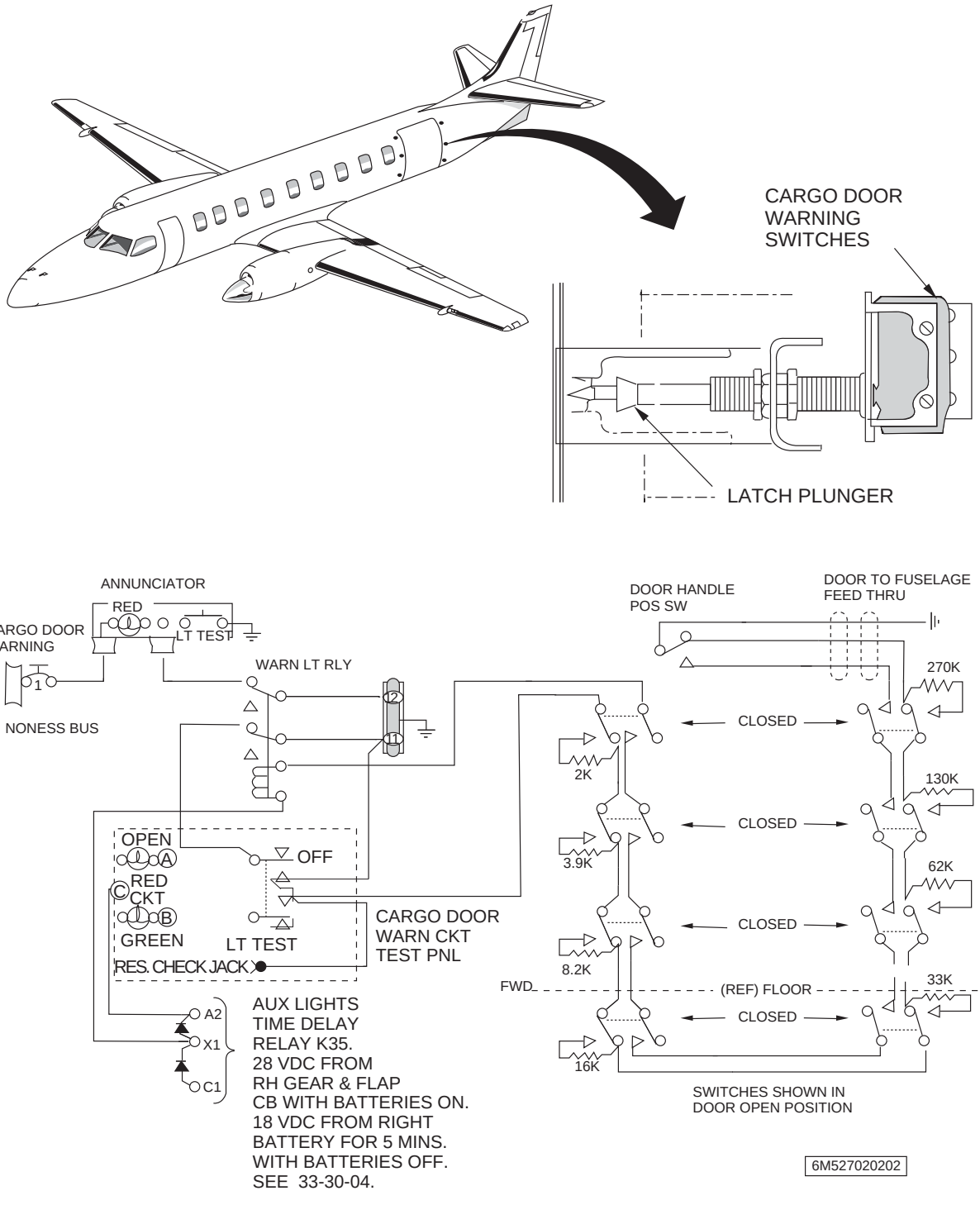
(g) Tighten jam nuts on microswitch.

(4) Verify any initial movement of the cargo door handle from the LOCKED position will illuminate the CARGO DOOR warning light.

(5) If switch is adjusted per step (3) and light does not extinguish, check adjustment of the other cargo door warning switches. If problem persists, isolate and correct defective wiring or replace switch(es) as required.

(6) Select BATTERY switches to OFF.

(7) Reinstall removed interior trim panels.



Cargo Door Wiring Diagram
TC – 397-419
Figure 202

B. Test – Cargo Door Unlock Warning Switches

(1) Check operation of DOOR UNSAFE red warning light and SWITCHES NORMAL green light as follows:

NOTE: Without battery power, the annunciator panel CARGO DOOR warning light will not illuminate when cargo door is UNLOCKED and in the closed position. It should be illuminated if either battery switch is ON.

(a) With cargo door LOCKED, hold test switch in LAMP TEST position. Verify DOOR UNSAFE and SWITCHES NORMAL lights illuminate.

(b) With cargo door LOCKED, hold test switch in SWITCH TEST position. Verify DOOR UNSAFE and SWITCHES NORMAL lights illuminate.

(c) With cargo door handle rotated out of LOCKED position enough to activate door handle position microswitch, verify DOOR UNSAFE light illuminates.

(d) Hold test switch in SWITCH TEST position. Verify SWITCHES NORMAL light does not illuminate.

(e) With cargo door open, verify DOOR UNSAFE light remains illuminated.

(f) With cargo door open, hold test switch in SWITCH TEST position. Verify SWITCHES NORMAL light illuminates.

C. Adjustment – Cargo Door Handle Position Switch

(1) Remove cargo door inner handle and blanket liner/interior covering to gain access to cargo door handle microswitch.

CAUTION: TO AVOID STRUCTURAL DAMAGE, USE CARE TO ENSURE CARGO DOOR IS NOT ALLOWED TO CLOSE AGAINST FUSELAGE WITH LATCHES EXTENDED.

(2) Open cargo door far enough to permit click-clack latches and guide pin bayonets to clear fuselage frame when extended.

(3) Temporarily install inner cargo door handle. Rotate handle to LOCKED position.

(4) Loosen mounting screws securing microswitch assembly.

(5) Adjust microswitch assembly position. Set microswitch actuator against bellcrank assembly. Verify microswitch actuator is depressed when door handle is in LOCKED position.

(6) Tighten screws securing microswitch assembly.

(7) Perform test for door handle microswitch IAW instructions, this section.

(8) Install blanket liner/interior covering and inner cargo door handle.

D. Test – Cargo Door Handle Position Microswitch

(1) Position cargo door handle in LOCKED position.

(2) Locate TEST JACK on copilot's console forward of RIGHT ESSENTIAL BUS in CARGO DOOR WARNING & TEST panel.

(a) Measure resistance at TEST JACK.

(b) Verify resistance is infinity.

(3) Rotate cargo door handle to UNLOCKED position.

(4) Measure resistance at TEST JACK. Verify resistance is zero (o) (GROUND).

(5) An alternate test procedure is as follows:

(a) Turn electrical power ON.

(b) Locate TEST SWITCH in CARGO DOOR WARNING & TEST panel on copilot's console forward of RIGHT ESSENTIAL BUS.

1 Position switch to SWITCH TEST.

2 Ensure DOOR UNSAFE – SWITCHES NORMAL light does not illuminate.

3 Rotate cargo door handle to UNLOCKED position.

4 Ensure DOOR UNSAFE – SWITCHES NORMAL light illuminates.

4. Inspection/Check – Cargo Door Unlock Warning Switches

A. Cargo Door Unlock Warning Switches (TC 201-397)

(1) Check components for corrosion, wear, or signs of failure.

(2) Check support structure for security, corrosion, cracks, and wear.

(3) Check door warning light operation.

(a) With door open, manually depress all door warning switches. Verify the DOOR WARNING light in the annunciator panel extinguishes.

(b) Selectively release and depress each warning switch.

(c) With all other switches depressed, check that releasing any individual switch causes illumination of the door warning light.

(d) Actuate each switch several times while checking for any tendency for the switch to stick in the depressed position.

NOTE: On the SA226 – TC and AT aircraft, it is not possible to check the cargo door from inside the airplane when the structural bulkhead divider is installed between the passenger cabin and the aft cargo compartment. Therefore, just inside the passenger entrance door of the aircraft is a cargo door warning light and switch with which cargo door latching can be checked prior to entering the cockpit. If any one of the click-clack latches is not sufficiently engaged to break the ground to its microswitch, the warning light will illuminate. When the switch is moved to the TEST position, a ground is supplied directly to the warning light to check the continuity of the lamp. The door warning light circuit receives power from the nonessential DC bus when that bus is powered, or from the auxiliary battery when power is not available from the nonessential bus.

B. Cargo Door Unlock Warning Switches (TC 398-419)

(1) Check components for corrosion, wear, or signs of failure.

(2) Check support structure for security, corrosion, cracks, and wear.

(3) Check door warning light operation.

(a) With door open, manually depress all door warning switches. Verify the DOOR WARNING light in the annunciator panel extinguishes.

(b) Selectively release and depress each warning switch.

(c) With all other switches depressed, check that releasing any individual switch causes illumination of the door warning light.

(d) Actuate each switch several times while checking for any tendency for the switch to stick in the depressed position.

NOTE: On the SA226 – TC and AT aircraft, it is not possible to check the cargo door from inside the airplane when the structural bulkhead divider is installed between the passenger cabin and the aft cargo compartment. Therefore, just inside the passenger entrance door of the aircraft is a cargo door warning light and switch with which cargo door latching can be checked prior to entering the cockpit. If any one of the click-clack latches is not sufficiently engaged to break the ground to its microswitch, the warning light will illuminate. When the switch is moved to the TEST position, a ground is supplied directly to the warning light to check the continuity of the lamp. The door warning light circuit receives power from the nonessential DC bus when that bus is powered, or from the auxiliary battery when power is not available from the nonessential bus.

(4) Activate auxiliary light circuit by cycling a battery switch to ON and then OFF, or by pressing auxiliary lights switch in nose baggage compartment (left aft door frame).

NOTE: As long as a battery or generator is on, the auxiliary light circuit remains active since the five minute timer is defeated.

(a) Ensure circuit remains active for a minimum of five (5) minutes.

NOTE: Without battery power, the annunciator panel CARGO DOOR warning light will not illuminate when the cargo door is UNLOCKED in the closed position. It should be illuminated if either battery switch is ON.

LANDING GEAR DOORS – DESCRIPTION AND OPERATION

I. General

A. Landing gear doors consist of a left and right door for each of the three landing gear. The doors are constructed of hydro-pressed aluminum pans, formers, angles, and skin, welded and riveted together. The doors are hinge mounted to each side of the wheel well forward of the gear assembly. A spade door is mounted aft of the main gear. The doors are flush fitted and completely enclose the landing gear when the gear is retracted. Refer to Chapter 32 for gear door actuator maintenance practices.

2. Main Landing Gear Doors

A. The left and right main landing gear doors are spring loaded overcenter to the open position when the gear is extended. They are mechanically closed by linkage from the door to the door actuator hook when the gear retracts. The door actuator hook is actuated by a roller mounted on the drag brace which engages or disengages the hook of the door actuator as the gear retracts or extends. The spade door, attached to the gear strut by linkage, mechanically opens and closes with extension and retraction of the gear.

3. Nose Landing Gear Doors

A. The left and right nose landing gear doors are spring loaded overcenter to the open position when the gear is extended. They are mechanically closed by linkage from the door to the door actuator hook when the gear retracts. The door actuator hook is actuated by a roller mounted on the gear strut which engages or disengages the hook of the door actuator as the gear retracts or extends.

MAIN LANDING GEAR DOORS – MAINTENANCE PRACTICES

1. Removal/Installation – Main LandingGear Doors

A. Removal – Door Actuator

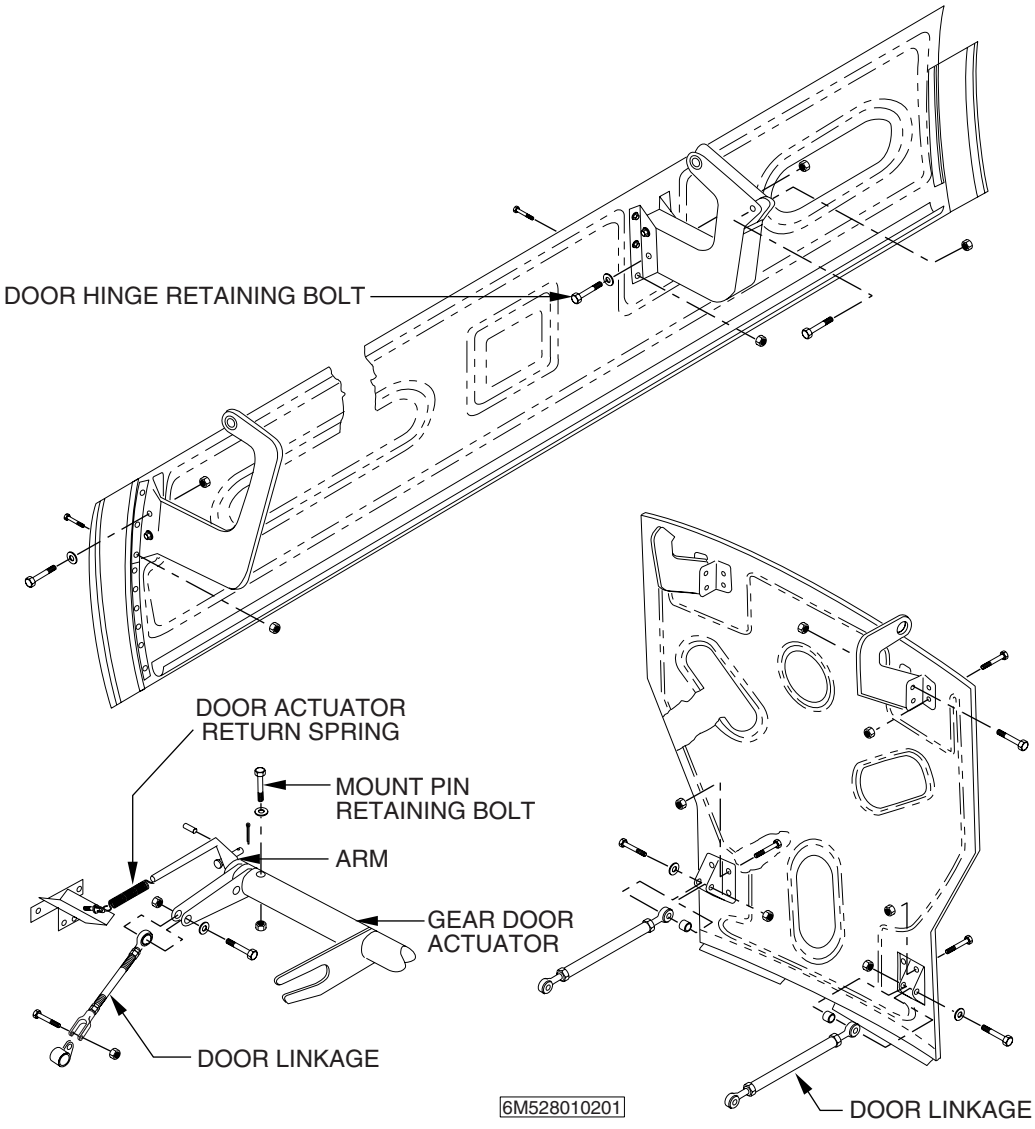
(1) Remove electrical wiring and clamp installations from the door actuator and stow away.

(2) Disconnect door uplock return springs by detaching from door actuator assembly arms.

(3) Disconnect door linkage assemblies from left and right hand doors.

(4) Remove retaining bolts from door actuator assembly mount pins.

(5) Move mount pins inboard and lower door actuator assembly.



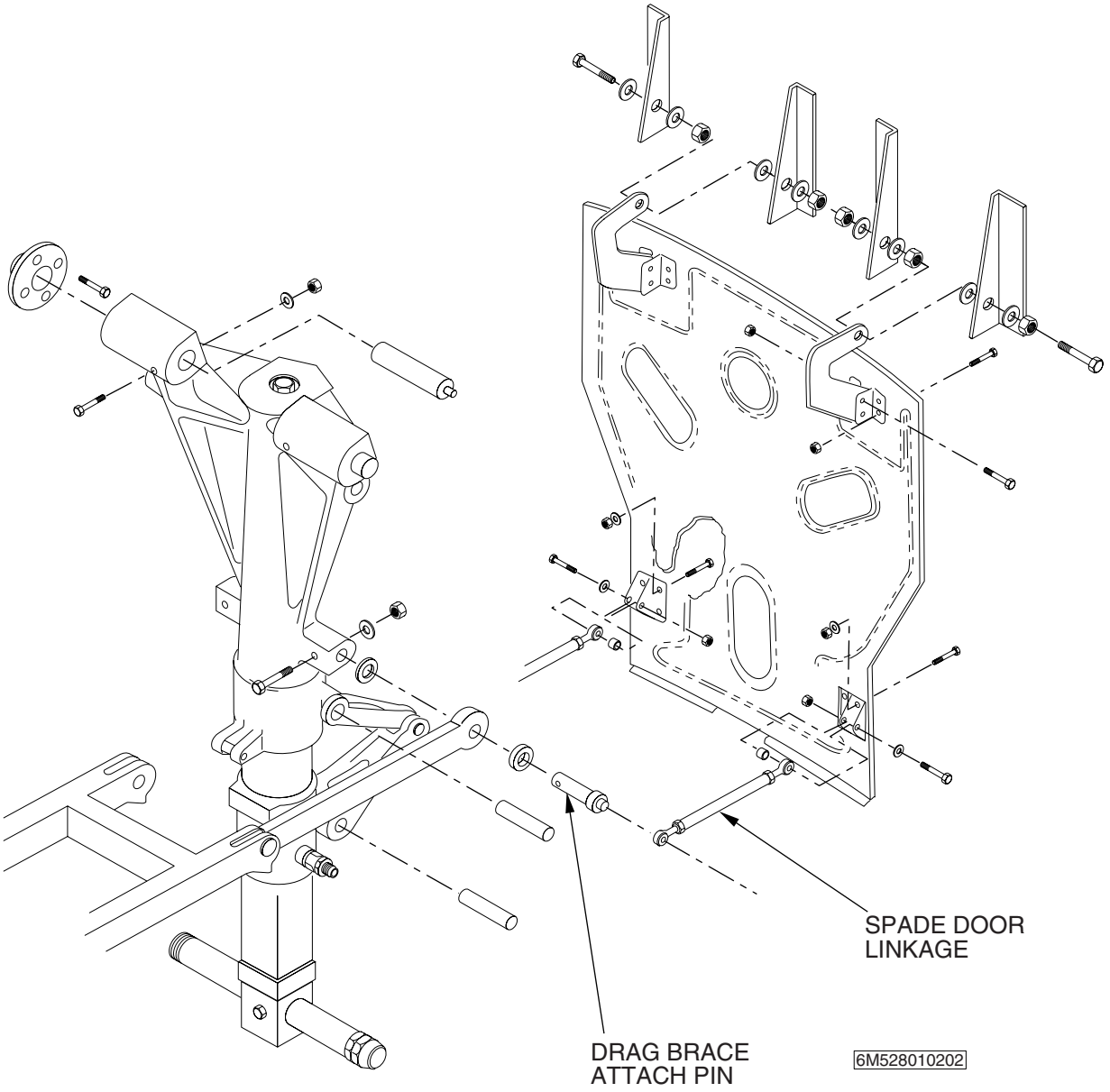
Main Landing Gear Doors
Figure 201

B. Installation – Door Actuator

(1) Position hinge pins in actuator torque tube.

(2) Position actuator assembly between landing gear mounts. Align hinge pins with mount bushings.

NOTE: If hinge pin bushing is loose in landing gear mount, secure bushing with loctite 271.



Main Gear Door and Strut
Figure 202

(3) Temporarily slide hinge pins into mounting bushings. Push actuator torque tube to one side and determine gap (on opposite side) between mount bushing face and shoulder of actuator tube. Maintain 0.040 to 0.060 inch (1.016 to 1.524 mm) lateral movement of torque tube.

(4) Back drill hinge pins 0.190 to 0.194 diameter hole through actuator. Remove actuator and hinge pins. Deburr holes.

(5) Apply grease MIL-G-21164 to hinge pin bushings. Slide hinge pins into torque tube. Position actuator torque tube between strut mounts.

(6) Slide hinge pins into bushings aligning holes in pins with holes in torque tube.

(7) Secure hinge pins to actuator torque tube with bolts, washers, and nuts. Torque nuts 30 to 40 inch-pounds (3.39 to 4.52 N·m).

(8) Install door linkage assemblies.

(9) Install door actuator return springs attaching them to door actuator assembly arms.

(10) Install electrical wiring and clamps onto door actuator assembly.

(11) Perform landing gear functional check as described in LANDING GEAR RIGGING PROCEDURES, Chapter 32, section 32-30-00.

C. Removal – Doors

(1) Remove retaining bolts and disconnect door linkage from left and right hand door actuator. Remove retaining nuts and disconnect door linkage from drag brace attach pin (spade door assembly).

(2) Remove bolts, washers, and nuts at hinge.

(3) Remove door.

D. Installation – Doors

(1) Position door and install all hinge bolts, retaining washers, and nuts.

(2) Connect door linkages and install retaining bolts, washers, nuts, and cotter pins.

(3) Adjust gear doors for proper fit and flushness (see LANDING GEAR RIGGING PROCEDURES, Chapter 32, section 32-30-00).

NOSE LANDING GEAR DOORS – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Gear Doors

A. Removal – Door Actuator

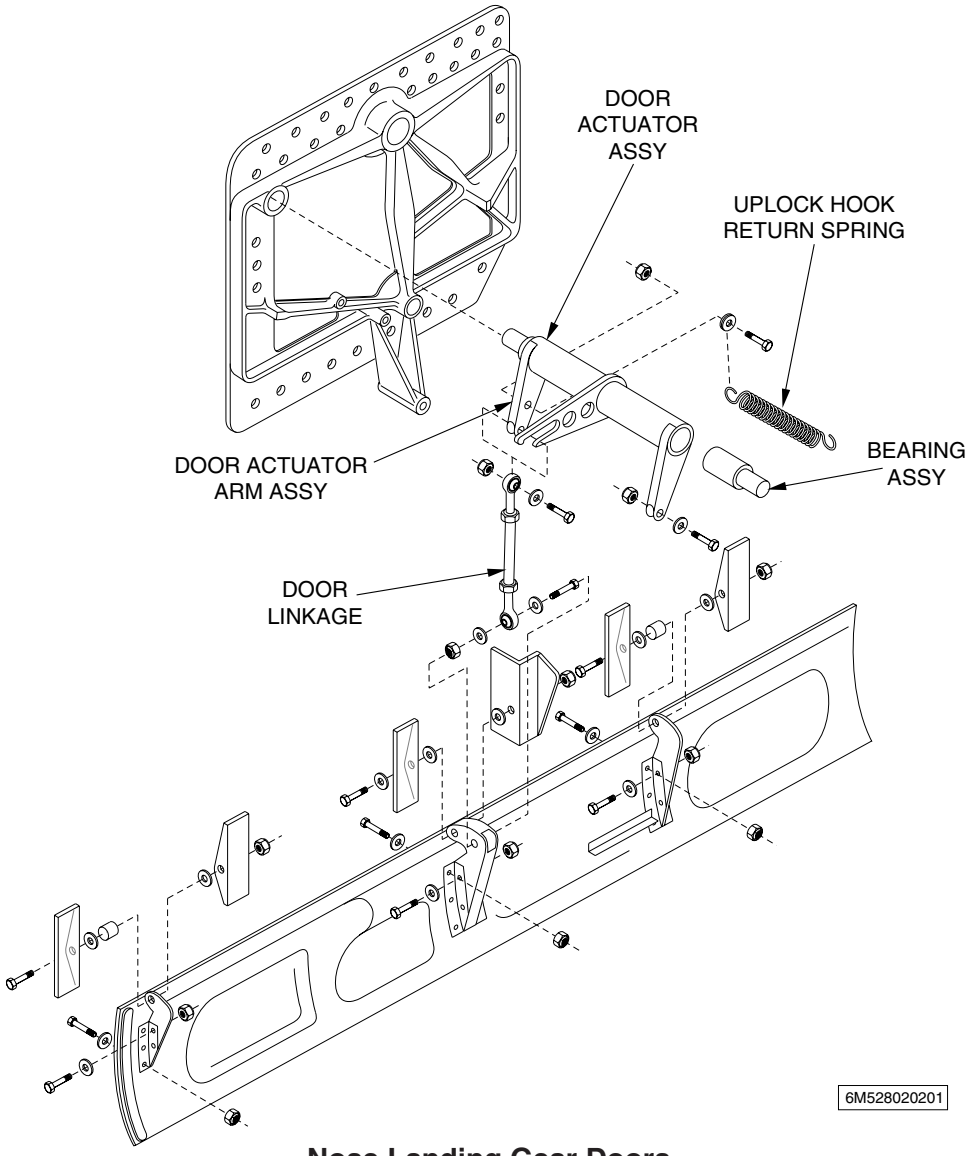
(1) Remove electrical wiring and clamp installations from door actuator assembly and stow away.

(2) Remove retaining bolts from door actuator arm assemblies securing uplock hook return springs. Detach springs from assembly.

(3) Disconnect door linkage assemblies from left and right doors.

(4) Remove retaining bolts securing door actuator mounting pins.

(5) Slide mount pins into actuator tube and lower door actuator assembly.



Nose Landing Gear Doors
Figure 201

B. Installation – Door Actuator

(1) Position door actuator assembly into mounting location with gear door arms facing forward.

(2) Apply grease MIL-G-21164 to mount pin bushings. Slide mount pins into bushings. Align mount pin retaining bolt holes of actuator with holes in mount pins on a previously installed assembly. Temporarily install two (2) each AN960-816L washers on each side of actuator tube between actuator and gear drag brace mount.

(3) Push actuator torque tube to one side and determine gap between pin assembly shoulder and drag brace mount. Maintain 0.030 to 0.050 inch (0.762 to 1.270 mm) total lateral clearance. Install retaining bolts, washers, and nuts. Torque nuts 20 to 25 inch-pounds (2.26 to 2.82 N·M).

(4) The four AN washers temporarily installed in step **1.**, B., (2) will position actuator tube at 0.015 inch (0.381 mm) spacing on each side. Push actuator hinge pins into bushings.

(5) With 0.190 to 0.194 drill bit, back drill hinge pins through actuator tube.

(6) Remove actuator and hinge pins. Remove AN washers and deburr holes.

(7) Reinstall actuator and hinge pins without AN washers. Install bolts and check actuator tube for lateral clearance of 0.030 to 0.050 inch (0.762 to 1.270 mm).

(8) Install retaining bolts, washers, and nuts. Torque 20 to 25 inch-pounds (2.26 to 2.82 N·M).

(9) Install door linkage assemblies.

(10) Install uplock hook return springs to arm assemblies. Install retaining bolt, washer, and nut.

NOTE: Uplock hook return spring tension should be 60, ± 3 pounds when stretched one (1) inch.

(11) Install electrical wiring and clamps onto door actuator assembly.

(12) Perform landing gear functional check IAW LANDING GEAR RIGGING PROCEDURES, Chapter 32, section 32-30-00.

C. Removal – Doors

(1) Remove retaining bolts, washers, and nuts and disconnect door linkage from door actuator assembly.

(2) Remove retaining bolts, washers, and nuts at each door hinge.

(3) Remove door.

D. Installation – Doors

(1) Position door and install all hinge retaining bolts. washers, and nuts.

(2) Connect door linkages and install bolts, washers, nuts, and cotter pins.

(3) Rig gear doors for proper fit IAW LANDING GEAR RIGGING PROCEDURES, Chapter 32, section 32-30-00.

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MAINTENANCE MANUAL

CHAPTER 53 – FUSELAGE

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MAINTENANCE MANUAL

CHAPTER 53 – FUSELAGE

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CHAPTER 53 – FUSELAGE

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FUSELAGE – DESCRIPTION AND OPERATION

1. General (SA226-AT and SA226-TC)

A. The SA226–AT and TC fuselage is an aluminum alloy, semi-monocoque structure, pressurized between F.S. 69.310 and 631.310. It is assembled in three basic sections;

(1) The forward section - F.S. 0.000 to F.S. 126.060.

(2) The center or constant section - F.S. 126.060 to F.S. 493.247.

(3) The tail section - F.S. 493.247 to F.S. 631.310.

B. The structure consists of conventional transverse frames, longerons, bulkheads, longitudinal floor beams, door frames, and skin assembly.

C. The lower surface of the wing center section passes through the passenger compartment where it is sealed and contoured to inner fuselage structure by sheet metal structure. Wing-to-fuselage contour is formed by fiberglass fairing.

D. A series of glands made of tubing and filled with sealer provide routing for cables, lines, and wiring through the pressure vessel to the unpressurized wing section.

E. A radome is attached to fuselage at F.S. 00.000 and extends forward to F.S. -28.940. A tail cone is attached to F.S. 631.310 and extends aft to F.S. 659.975.

F. The fuselage is covered by aluminum clad stressed skin. Skin sheets are bonded and/or riveted to longitudinal stringers, which are riveted to fuselage frames. Doublers, internally attached over fuselage skin joints, are sealed with sealant to effect a dry skin joint seal.

2. General SA226-T

A. The SA226–T model airplane is of the same construction design as the AT and TC model airplanes. The T model is pressurized between F.S. 69.310 and F.S. 424.560. Forward section extends rearward from F.S. 0.000 to F.S. 126.060, center section from F.S. 126.060 to F.S. 273.810, and tail section from F.S. 273.810 to F.S. 424.560.

AERODYNAMIC FAIRINGS – DESCRIPTION AND OPERATION

1. General

A. Aerodynamic fairings are auxiliary members or structures designed to reduce drag. They are either metal or fiberglass fillets shaped and formed to fair in with the contour of the airplane.

B. Fairings installed on the fuselage consist of nose section radome, wing center section to fuselage fillets, and empennage fillets.

2. Radome

A. The radome is made of fiberglass and honeycomb. It is attached to airframe with screws at F.S. 00.00.

AERODYNAMIC FAIRINGS – MAINTENANCE PRACTICES

1. Removal/Installation – Aerodynamic Fairings

A. Removal – Wing to Fuselage Fairing

(1) Remove screws and washers.

NOTE: Due to 3 sizes of screws in fairing, a suitable method should be used to identify screws by size. One method is to apply tape along rows or at end of rows with same size screws and indicate screw size to use during installation of fairing.

Washers are not used where the aft belly skin overlaps fairing.

(2) Remove forward and aft fairing sections.

B. Installation – Wing to Fuselage Fairing

(1) Position forward section of fairing on wing.

CAUTION: ENSURE NUTPLATE THREADS ARE NOT DAMAGED WHILE ALIGNING HOLES.

(2) Align holes in fairing, fuselage, and wing using punch, awl, or similar tool.

CAUTION: THREE DIFFERENT SIZE SCREWS ARE USED TO SECURE FAIRING TO WING, FUSELAGE, AND AFT BELLY SKIN. ENSURE THE SAME SIZE SCREWS ARE RETURNED TO HOLES TO PREVENT DAMAGE TO NUTPLATES.

(3) Install one screw and washer at leading edge of wing and forward hole in fuselage.

NOTE: Do not tighten screws until forward and aft sections of fairing are in place.

(4) Install screws and washers from forward to aft alternating, one to fuselage and one to wing.

(5) Install screws and washers from leading edge of wing and fuselage to bottom of wing and fuselage.

(6) Position aft section of fairing on wing.

NOTE: The aft belly skin overlaps bottom aft section of fairing.

CAUTION: ENSURE NUTPLATE THREADS ARE NOT DAMAGED WHILE ALIGNING HOLES.

(7) Align holes in fairing, fuselage, wing, and aft belly skin with punch, awl, or similar tool.

(8) Install screws and washers in 3 holes between 2 rows of holes on top of wing.

NOTE: Washers are not used where the aft belly skin overlaps fairing.

(9) Install 4 screws and washers at joint of forward and aft fairing on top of wing.

(10) Install remaining screws and washers aft of joint, alternating from wing to fuselage.

(11) Tighten all screws on forward and aft sections of fairing.

C. Removal – Dorsal Fin Fairing

(1) Remove screws and washers.

(2) Remove fairing.

D. Installation – Dorsal Fin Fairing

(1) Position fairing.

CAUTION: ENSURE NUTPLATE THREADS ARE NOT DAMAGED WHILE ALIGNING HOLES.

(2) Align holes in fairing and dorsal fin with punch, awl, or similar tool.

(3) Install screws and washers in first hole (RH/LH side) at front of fairing.

NOTE: Do not tighten screws until all screws are in place.

(4) Install next screw and washer alternating (1 screw and washer) from side to side along bottom of fairing.

(5) Install screws and washers in top, aft of fairing.

(6) Tighten all screws.

RADOME – MAINTENANCE PRACTICES

1. Removal/Installation – Radome

A. Removal – Radome

(1) Supporting radome by hand, remove the attaching screws around the radome, or release toggle latches at 2, 4, 8, and 10 o'clock positions.

(2) On those aircraft equipped with lightning diverter straps, proceed as follows:

(a) Locate bond strap wing nuts at approximately 3 and 9 o'clock positions on bulkhead at F.S. 0.00. Remove RTV sealant from wing nuts.

(b) Unscrew wing nuts and remove bond strap from terminal post. Remove thin washers from posts.

(3) Remove radome.

B. Installation – Radome

(1) Position radome next to bulkhead at F.S. 0.00

(2) On those aircraft equipped with lightning diverter straps, proceed as follows:

(a) Put thin washers on terminal post. Place bond straps onto terminal post at 3 and 9 o'clock positions.

(b) Tighten wing nuts. Seal wing nuts with RTV732 sealant.

(3) Position radome against fuselage aligning the screw holes and install the screws. On those aircraft that use toggle latches align toggle latches with fuselage cutouts. Secure radome by locking toggle latches in place.

C. Build-Up Installation

(1) If a new radome is to be installed, perform the following steps:

(a) Place a strip of 1 inch masking tape around the forward edge of the fuselage to protect the painted surface when trimming.

(b) Place radome onto fuselage and trim excess material from the radome as required. Radome is transparent to allow forward edge of fuselage to be seen when trimming.

(2) For aircraft that use screws to attach the radome perform the following steps:

(a) Place a 1 inch strip of masking tape around edge of radome.

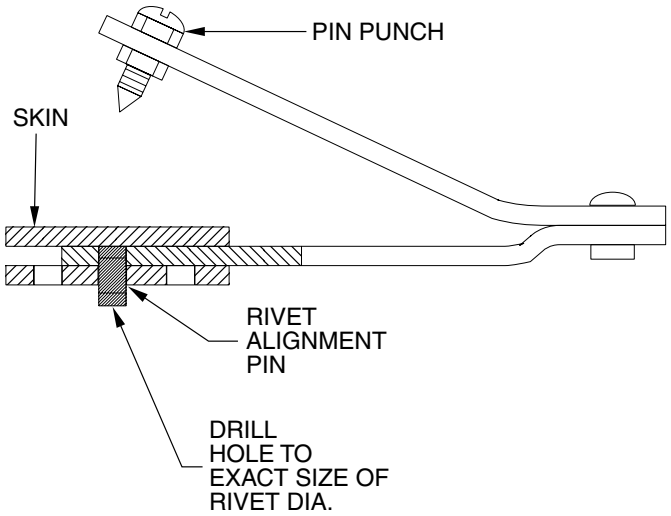
(b) Position radome against fuselage and using a hole duplicator (see Figure 201), locate and mark one hole opposite sides of the radome.

(c) Remove radome, using a compass mark the remaining holes. Drill #10 holes and countersink, using a 100 degree countersink.

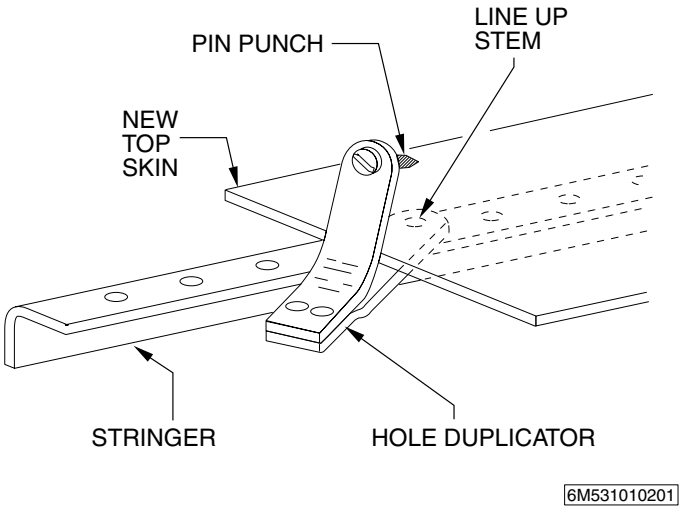
(d) Ensure that the rubber seal is installed around the inside of radome, 0.80 of an inch from edge.

1 Cement seal to the radome using 1300L. Position seal retainer onto seal and rivet retainer and seal to radome.

A HOLE FINDER



B HOLE FINDER USED TO DUPLICATE HOLES



Hole Finder or Duplicator
Figure 201

(e) Prepare surfaces for painting and paint.

(f) Position radome against fuselage aligning screw holes. Secure radome in place with screws.

(3) For aircraft that use toggle latches perform the following steps:

(a) Place a strip of 1 inch masking tape around the forward edge of the fuselage to protect the painted surface when trimming.

(b) Place radome onto fuselage and trim excess material from the radome as required. Radome is transparent to allow forward edge of fuselage to be seen when trimming.

(c) Using a scale or straight edge align the latch positions and draw an outline of the latches.

(d) Cut out the two top latch areas and rivet the latches into place.

(e) Holding radome in place, latch the upper latches, and draw an outline of lower latches.

(f) Remove radome. Cut out two lower latch areas and rivet latches into place.

(g) Ensure that the rubber seal is installed around the inside of the radome, 0.80 of an inch from edge.

1 Cement seal to the radome using 1300L. Position seal retainer onto seal and rivet retainer and seal to radome.

2 Fill void between seal and radome at toggle latch assemblies using PRO-SEAL 890.

(h) Install radome and check for fit.

(i) Remove radome from aircraft, prepare surfaces for painting and paint.

(2) Position radome against fuselage aligning toggle latches with fuselage cutouts. Secure radome by locking toggle latches in place.

(3) On those aircraft equipped with lightning diverter straps, proceed as follows:

(a) Put thin washers on terminal posts. Place bond straps onto terminal posts at 3 and 9 o'clock positions.

(b) Tighten wing nuts. Seal wing nuts with RTV732 sealant.

(4) Position radome against fuselage aligning toggle latches with fuselage cutouts. Secure radome by locking toggle latches in place.

MAIN FRAME – DESCRIPTION AND OPERATION

1. General

A. Frames, bulkheads, stringers and beams form the primary airplane structures.

(1) Frames are aluminum alloy, conventional formed "C" frames, of 2024-0, heat treated to T4 hardness and spaced approximately 15 inches (38.10 cm) apart.

(2) The forward bulkhead (F.S. 69.310) is constructed of a frame and web, reinforced by vertical beams, stiffeners and angles. The forward bulkhead is the forward end of the pressure vessel.

(3) The aft bulkhead (F.S. 631.310 on SA226-AT/TC and F.S. 424.560 on SA226-T) is a flat plate constructed of 0.125 inch aluminum alloy and two "U" channels. The aft bulkhead is secured to the fuselage frame with bolts and is the aft end of the pressure vessel.

(4) Stringers are aluminum alloy extruded bulb angles riveted to the fuselage frames and skin.

(5) The longitudinal floor beams are made up of "Z" channels and flat webs and secured to the fuselage frames and skin with rivets.

AUXILIARY STRUCTURE – DESCRIPTION AND OPERATION

1. General

A. Description

(1) The auxiliary fuselage structure consists of secondary structural members that are not part of the fuselage main frame. Included are floors and fixed partitions.

B. Floors

(1) Fuselage floors are made of aluminum alloy sheets reinforced by extrusions. The balsa core (bondolite) panels secured to and supported by longitudinal floor beam structures. Flooring is divided into five compartments; flight, passenger, center aisle, baggage, and cargo.

(2) The flight compartment floor is aluminum alloy sheets. The floor contains pilot and copilot seat tracks and cutouts for mounting pedestal and control columns. Eight removable panels provide access to the control columns, engine control cables, hydraulic lines and pedestal controls. Floor panels are secured with screws and clipnuts.

(3) The passenger compartment floor is made of bondolite and secured to the aircraft structure with screws and washers. This section is convertible to serve as a cargo compartment. Tie downs are installed in this section and used in conjunction with fittings for folding passenger seats.

(4) The center aisle floorboards are bondolite. The floor is installed in ten sections. Floorboards are secured with camlocks.

(5) The baggage compartment floorboards are of bondolite. Floorboards are installed in five sections and secured with screws and washers.

(6) The standard SA226 cargo compartment floor has bondolite over the sheet flooring. The floor is installed in four sections. The cargo compartment floor is made of gillfloor 5007A, consisting of top and bottom fiberglass skins with an end grain balsa core. The floor is installed in three sections and secured with screws and washers.

NOSE BAGGAGE COMPARTMENT FLOORBOARDS – MAINTENANCE PRACTICES

1. Removal/Installation – Nose Baggage Compartment Floorboards

A. Removal – Nose Baggage Compartment Floorboards

(1) Remove P-12 Aluminum backed tape, screws, and washers.

(2) Remove floorboards.

(3) Remove screws and protective strips from center floorboard only.

B. Installation – Nose Baggage Compartment Floorboards

(1) Position floorboards in baggage area.

CAUTION: DO NOT TRIM OR DRILL CARBON FIBER COMPOSITE FLOORBOARDS INSIDE AIRCRAFT. SEAL CUT-OUTS, SCREW HOLES AND FLOORBOARD EDGES WITH NUVASIL 91 OR RTV 3145.

(2) Secure floorboards with screws and washers.

(3) Install P-12 aluminum backed tape along floorboards edge.

(4) Secure center floorboard with existing screws and protective strips.

FLIGHT COMPARTMENT FLOOR PANELS – MAINTENANCE PRACTICES

1. Removal/Installation – Flight Compartment Floor Panels

A. Removal – Flight Compartment Floor Panels

(1) Remove pilot/copilot seat track forward pins.

(2) Remove pilot/copilot seats IAW Chapter 25.

(3) Remove flight compartment carpeting.

(4) Remove screws securing pilot/copilot footwarmer and remove footwarmer.

(5) Remove screws securing pilot/copilot forward panel and remove panel.

(6) Remove screws securing panel around control column.

(7) Remove panel by lifting in direction of pedestal.

(8) Remove screws securing center panel and remove panel.

(9) Remove screws securing pilot/copilot seat tracks and remove seat tracks.

(10) Remove screws securing pilot/copilot panel and remove panel.

(11) Remove screws securing center aft panel and remove panel.

B. Installation – Flight Compartment Floor Panels

(1) Install clipnuts on flight compartment supporting frames.

(2) Install pilot/copilot panel and seat tracks securing with existing hardware.

(3) Install center aft panel securing with existing hardware.

(4) Install center panel securing with existing hardware.

(5) Install panel around control column securing with existing hardware.

(6) Install pilot/copilot forward panel securing with existing hardware.

(7) Install pilot/copilot footwarmer securing with existing hardware.

(8) Install floor carpeting.

(9) Install pilot/copilot seats IAW Chapter 25.

(10) Install pilot/copilot seat tracks forward pins and secure with new cotter pins.

PASSENGER COMPARTMENT FLOORBOARDS – MAINTENANCE PRACTICES

1. Removal/Installation – Passenger Compartment Floorboards

A. Removal – Passenger Compartment Floorboards

(1) Loosen jam nuts on passenger seat legs.

(2) Remove passenger seats as required to remove cabin floorboards.

(3) Remove aft cabin door actuator hinge cover. (L/H forward floorboard only).

(4) Remove carpeting as required to remove cabin floorboards.

(5) Remove screws and washers.

B. Installation – Passenger Compartment Floorboards

CAUTION: DO NOT TRIM OR DRILL CARBON FIBER COMPOSITE FLOORBOARDS INSIDE AIRCRAFT. SEAL CUT-OUTS, SCREW HOLES AND FLOORBOARD EDGES WITH NUVASIL 91 OR RTV 3145.

(1) Position floorboards in cabin area.

(2) Secure cabin floorboards with screw and washers.

(3) Install carpeting.

(4) Install aft cabin door actuator hinge cover. (L/H forward floorboard only).

(5) Install passenger seats and tighten jam nuts on legs.

2. Removal/Installation – Center Aisle Floorboards

A. Removal – Center Aisle Floorboards

NOTE: Following procedures apply to lightweight floorboards only.

(1) Remove screws from passenger door hinge cover and remove cover.

(2) Remove carpeting as required to remove center aisle floorboards.

(3) Remove screws and washers from center aisle floorboards.

(4) Remove floorboards.

B. Installation – Center Aisle Floorboards

CAUTION: DO NOT TRIM OR DRILL CARBON FIBER COMPOSITE FLOORBOARDS INSIDE AIRCRAFT. SEAL CUT-OUTS, SCREW HOLES AND FLOORBOARD EDGES WITH NUVASIL 91 OR RTV 3145.

(1) Position floorboards in center aisle area.

(2) Secure floorboards with screws, washers, and sleeves.

(3) Install passenger door hinge cover and secure.

(4) Install carpeting.

CARGO COMPARTMENT FLOORBOARDS – MAINTENANCE PRACTICES

1. Removal/Installation – Cargo Compartment Floorboards

A. Removal – Cargo Compartment Floorboards

(1) Remove forward cargo bulkhead.

(2) Remove screws and washers.

(3) Remove tie downs.

(4) Remove aft floorboard.

(5) Remove center floorboard.

(6) Remove remaining two floorboards as required.

B. Installation – Cargo Compartment Floorboards

CAUTION: DO NOT TRIM OR DRILL CARBON FIBER COMPOSITE FLOORBOARDS INSIDE AIRCRAFT. SEAL CUT-OUTS, SCREW HOLES AND FLOORBOARD EDGES WITH NUVASIL 91 OR RTV 3145.

(1) Position outer two floorboards.

(2) Position center floorboard.

(3) Position aft floorboard.

(4) Position tie downs, secure with screws and washers.

(5) Secure floorboards with screws and washers.

(6) Install forward cargo bulkhead.

FUSELAGE DRAINS – DESCRIPTION AND OPERATION

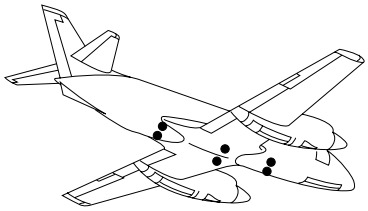
1. General

A. Fuselage Drains

(1) To prevent accumulation of water in the fuselage belly, three pairs (six total) of fuselage drains are located in the fuselage belly skin at F.S. 157, F.S. 435 and F.S. 497 on AT, and TC aircraft, and at F.S. 133, F.S. 186 and F.S. 263 on T aircraft (see Figure 201). Drains are located on left and right side of aircraft centerline, in pairs.

(2) Each drain consists of a rubber diaphragm, a retainer ring, and a 0.190 inch diameter hole in the fuselage skin. The diaphragm is located directly above the hole and held in place by the retainer ring.

(3) When the aircraft is pressurized, the diaphragm is pressed against the hole, preventing pressurized air from escaping. When the aircraft is at ambient pressure, natural spring action of rubber causes the diaphragm to unseat, allowing trapped water to drain through the holes.



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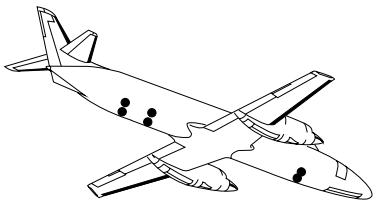

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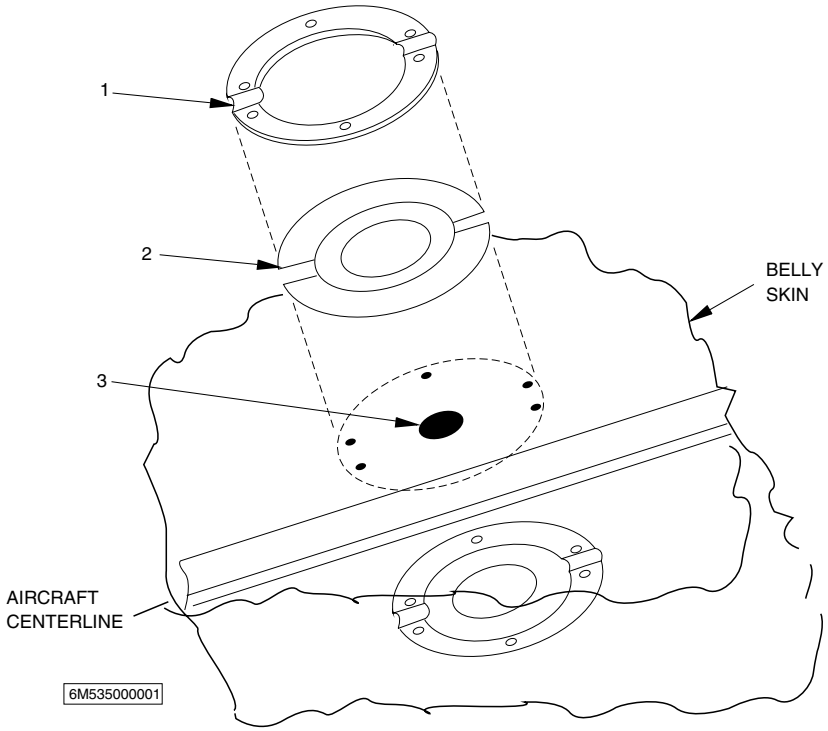


Fuselage Drains
Figure 1

FUSELAGE DRAINS – MAINTENANCE PRACTICES

1. Removal/Installation – Fuselage Drains (Figure 1)

A. Removal – Fuselage Drains

NOTE: Perform the following procedures if inspection check reveals damaged diaphragm and/or retainer ring.

(1) Remove floorboards above drain (item 3).

(2) Drill out rivets securing retainer ring (item 1) and diaphragm (item 2) to fuselage skin. Remove retainer ring and diaphragm.

(3) Clean area around drain thoroughly.

B. Installation – Fuselage Drains

(1) Coat upper and lower mating surface of new diaphragm (item 2) with sealant.

CAUTION: BE CAREFUL NOT TO GET SEALER INTO SLOTS. SLOTS ARE DRAIN CHANNELS.

(2) Position diaphragm (item 2) over drain hole (item 3) with slots aligned forward and aft, parallel with the aircraft centerline.

(3) Position retainer ring (item 1) over diaphragm (item 2) with retainer beads centered over slots of diaphragm.

(4) Secure retainer ring and diaphragm to fuselage skin with rivets.

(5) Verify drain allows water to exit when aircraft is at ambient air pressure.

(6) Pressurize aircraft per instruction in Chapter 21. Verify diaphragm closes.

(7) Install floorboards and hardware.

2. Inspection/Check – Fuselage Drains

A. Inspection – Fuselage Drains

(1) Remove floorboards.

(2) Vacuum debris and waste particles from belly of aircraft around drain.

NOTE: Be particular around forward and aft edges of retainer ring drain slots.

(3) Remove foreign objects and obstructions.

(4) Visually inspect diaphragm for punctures, and for indications of rubber deterioration or inflexibility.

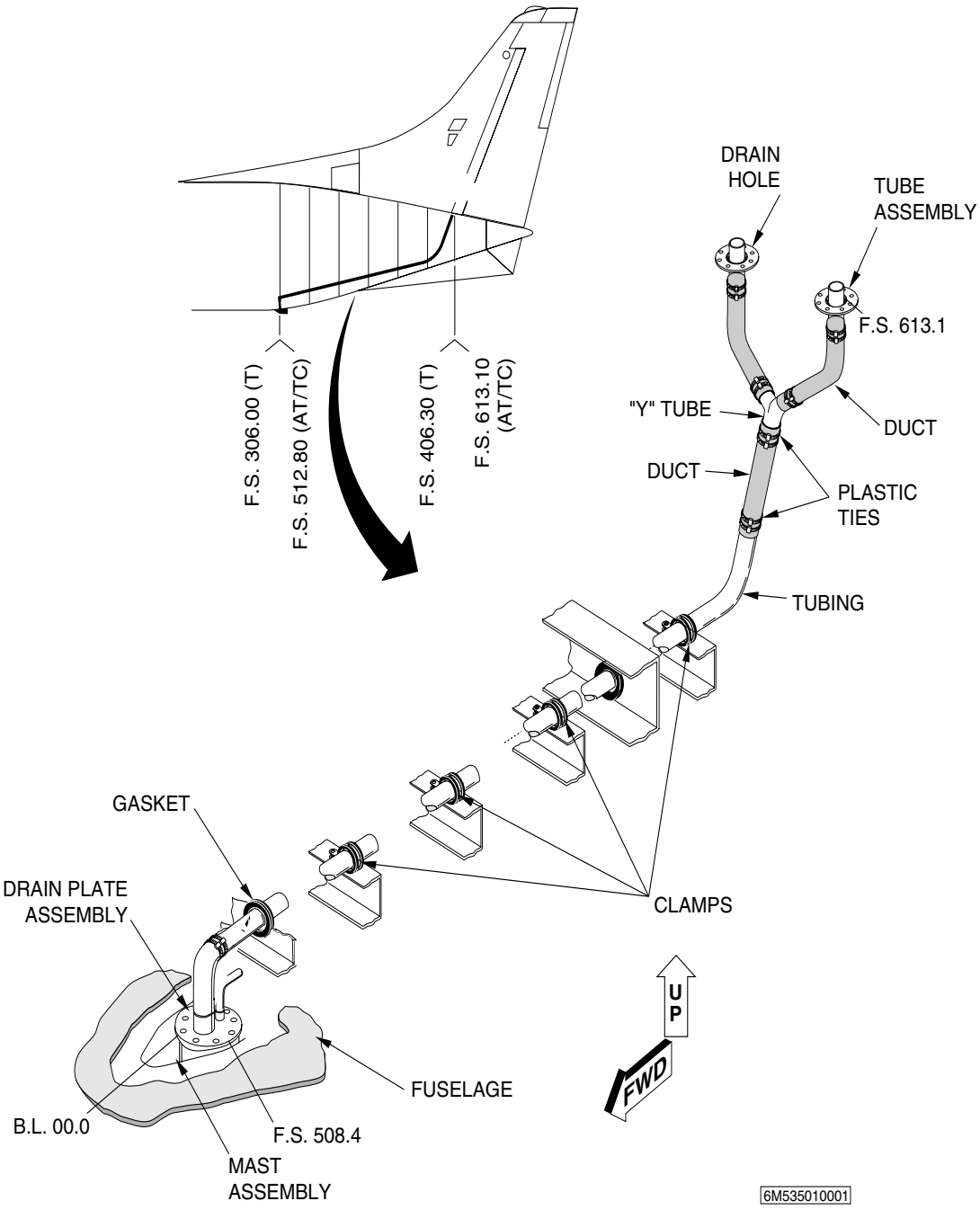
(5) If drain is damaged, replace according to Removal/Installation procedures.

(6) Verify drain allows water to exit when aircraft is at ambient air pressure.

AFT FUSELAGE DRAIN – DESCRIPTION AND OPERATION

1. General

A. A drain system is installed to prevent accumulation of water under the vertical fin (see Figure 1). Two drain holes with attached tube assemblies are located at F.S. 613.1 to left and right of fuselage centerline. Water drains through the holes into tube and duct, and overboard through the drain mast at F.S. 508.4.



Aft Fuselage Drain
Figure 1

AFT FUSELAGE DRAIN – MAINTENANCE PRACTICES

1. Removal/Installation – Aft Fuselage Drain

A. Removal – Drain Mast Assembly (Figure 1)

(1) Remove floorboards above drain mast assembly.

(2) Disconnect tubing from drain tube.

(3) Drill out rivets attaching drain mast assembly to fuselage skin.

(4) Remove drain mast assembly.

B. Installation – Drain Mast Assembly (Figure 1)

(1) Shank/wet seal drain mast plate during installation IAW Structural Repair Manual.

(2) Position drain mast assembly and secure to fuselage skin with rivets.

NOTE: The 0.38 inch (9.5 mm) diameter tube on the exhaust plate assembly is plugged and to be used with customer option items.

(3) Connect tubing and secure with clamp.

(4) Verify water drains and check for leaks.

(5) Pressurize aircraft IAW Chapter 21.

(6) Install floorboards.

C. Removal – Drain Tubes and Ducts (Figure 1)

(1) Remove floorboards or insulation above tubes or ducts.

(2) Remove ties or clamps from items to be replaced.

(3) Remove tubes or ducts.

D. Installation – Drain Tubes and Ducts (Figure 1)

CAUTION: TUBE ROUTING MUST HAVE POSITIVE SLOPE FROM SCOOP TO UPPER SKIN OF NOT LESS THAN 10%.

(1) Install tubes or duct.

(2) Secure with ties or clamps.

(3) Verify water drains and check for leaks.

(4) Pressurize aircraft IAW Chapter 21.

(5) Install floorboards and insulation.

E. Removal – Tube Assembly Plate (Figure 1)

(1) Remove ties from duct to tube assembly.

(2) Disconnect duct from tube assembly.

(3) Drill out rivets from drain plate.

(4) Remove drain plate.

F. Installation – Tube Assembly Plate (Figure 1)

(1) Position tube assembly plate.

(2) Secure plate to fuselage skin with rivets.

(3) Connect duct to tube assembly and secure with ties.

(4) Verify that water drains and check for leaks.

(5) Pressurize aircraft and IAW Chapter 21.

2. Inspection/Check – Aft Fuselage Drain

A. Inspection – Aft Fuselage Drain

(1) Inspect drain mast for damage, corrosion, and security. Ensure drain holes are not blocked.

(2) Inspect tubes, ducts, tube assembly, and drain plates inside aircraft for signs of damage and leakage.

(3) Inspect drain holes in top of fuselage for debris and obstructions.

(4) Verify water drains through system.

(5) If damage or leaks are found, replace items IAW Removal/Installation Procedures.

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CHAPTER 54 – NACELLES

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CHAPTER 54 – NACELLES

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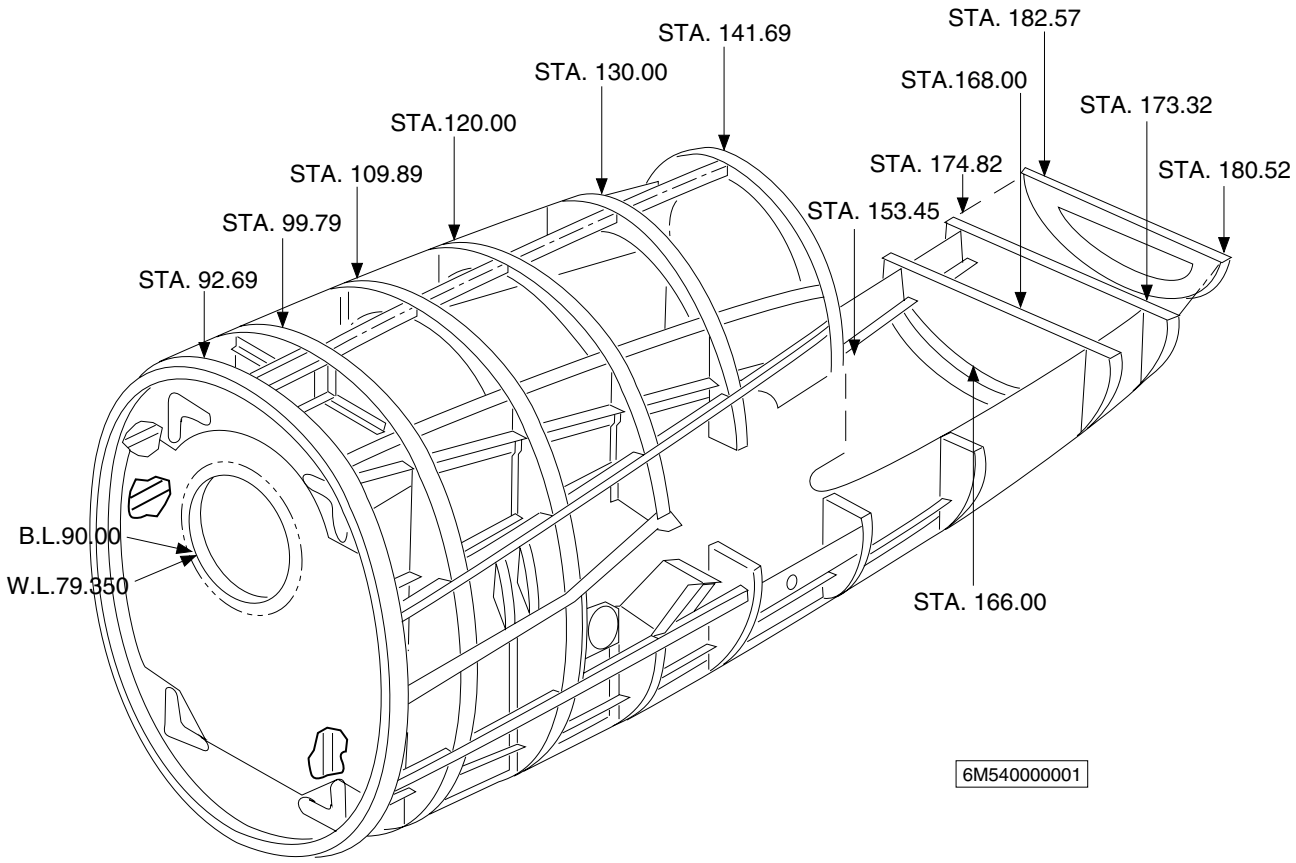
NACELLES – DESCRIPTION AND OPERATION

1. General

A. Two nacelles mounted on the one-piece wing are composed of covered framework and serve, primarily, as mounting bases for aircraft engines. Secondarily, they afford a streamlined housing for main landing gear when aircraft is in flight. Components of numerous aircraft systems are mounted in the nacelle.



Nacelle and Firewall
Figure 1

MAIN FRAME – DESCRIPTION AND OPERATION

1. General

A. Description

(1) The nacelle primary structure consists of a firewall, supporting frames, stringers, and keelson webs.

B. Frames

(1) There are ten frames in the nacelle center section. Frames are formed of aluminum alloy sheet metal parts. Frames are attached to the keelson webs and wing with rivets.

C. Stringers

(1) Stringers are formed from extruded aluminum 2024-T3511 bulb angle and riveted to frames to form a supporting framework for skin sheets.

D. Keelson Web

(1) Keelson webs are primary load carrying members made from formed aluminum alloy sheet metal parts. The landing gear main attaching points are located at the rear of the keelson webs.

E. Firewall

(1) The firewall is a formed frame made of stainless steel. An aluminum doubler is used to stiffen the firewall.

PLATES AND SKIN – DESCRIPTION AND OPERATION

1. General

A. The nacelles are made in four pieces, formed of .032 2024-T4 aluminum alloy sheet. The sheet is then riveted to the frame work with standard AN426 rivets.

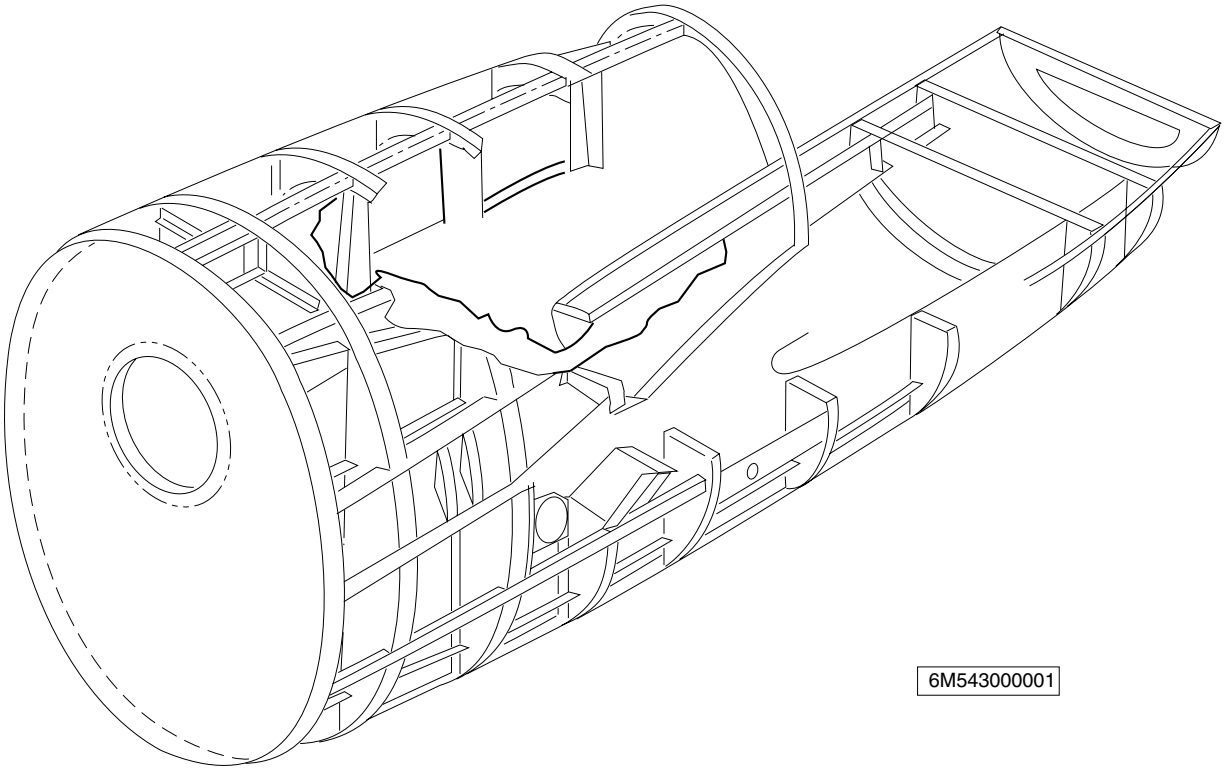
B. Access Covers

NOTE: There are seven access doors/panels on each of the nacelles.

(1) Ice Light Panel

(a) The ice light panel is located on the outboard side of the nacelle in front of the wing leading edge. It is secured with screws.

(b) The ice light and ice light bracket and the connection for the deicer boot are accessible through this panel.



Nacelle Structure
Figure 1

C. Hydraulic Cylinder Cam roller Door

NOTE: The first round door in the nacelle under the wing is used to gain access to the retainer pin on the hydraulic cylinder cam and roller pivot pin, left and right side.

(1) Inspection Door

(a) The second door in the lower skin of the nacelle is used for inspection of the structure only. This door is located in both the outboard and inboard sides of the nacelle.

(2) Landing Gear Trunnion

(a) Inspection of the landing gear trunnion is accomplished by removing the last oval door on the outboard side and the oblong door on the inboard side of the nacelles.

(3) Bottom Transmitter Access

(a) Access to the fuel, oil, and torque transmitters is gained by removing the door on the center of the nacelle forward of the wheel well.

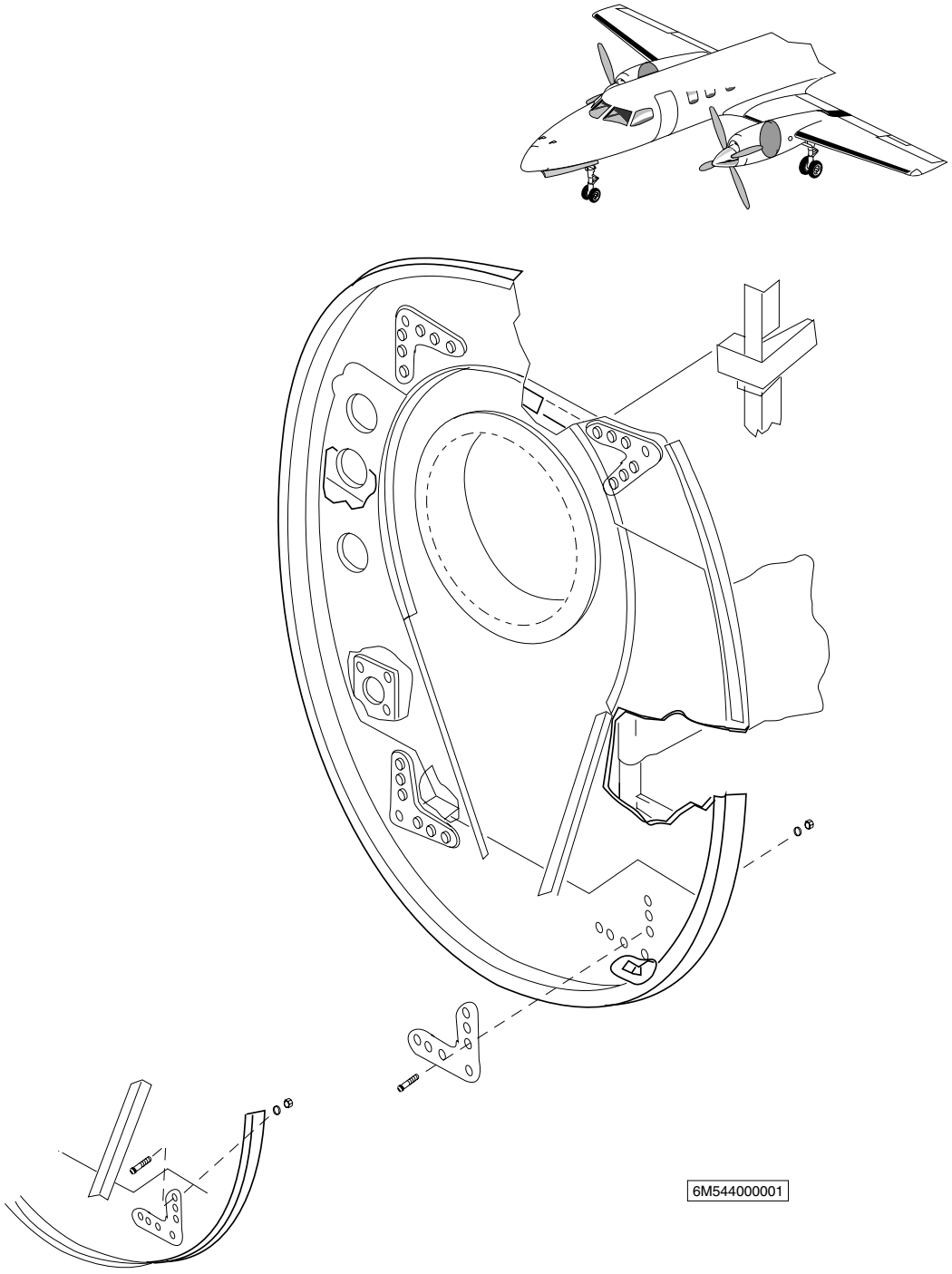
(4) Hydraulic Service

(a) Servicing of the hydraulic package is accomplished by opening the access door on the outboard side of the left nacelle above the hydraulic fluid indicator, see Chapter 12.

ATTACH FITTINGS – DESCRIPTION AND OPERATION

1. General

NOTE: Numerous fittings are located in and around the nacelle to secure parts of aircraft systems installed in, or transiting, the nacelle structure.



Firewall
Figure 1

ATTACH FITTINGS – DESCRIPTION AND OPERATION

A. Hydraulic Reservoir Attachment

(1) The hydraulic reservoir is attached by two brackets at the front section of the left nacelle. The brackets are attached by two AN3 bolts.

B. Hydraulic Tubing

(1) Hydraulic tubing in the left nacelle is held in place by a series of hard wood clamps attached to the keelson web by No. 10 bolts.

C. Electrical Panel Right Nacelle

(1) The electrical relays located in the right nacelle are mounted on a sheet metal panel. The panel is attached to the firewall and keelson webs with No. 8 screws.

D. APU Connection

(1) On early aircraft, the APU connection on the outboard front side of the right nacelle is retained by a double channel in the nacelle structure, and covered by a hinged, spring-loaded door.

MZ AEROSPACE
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MAINTENANCE MANUAL

CHAPTER 55 – STABILIZERS

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MAINTENANCE MANUAL

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STABILIZERS – DESCRIPTION AND OPERATION

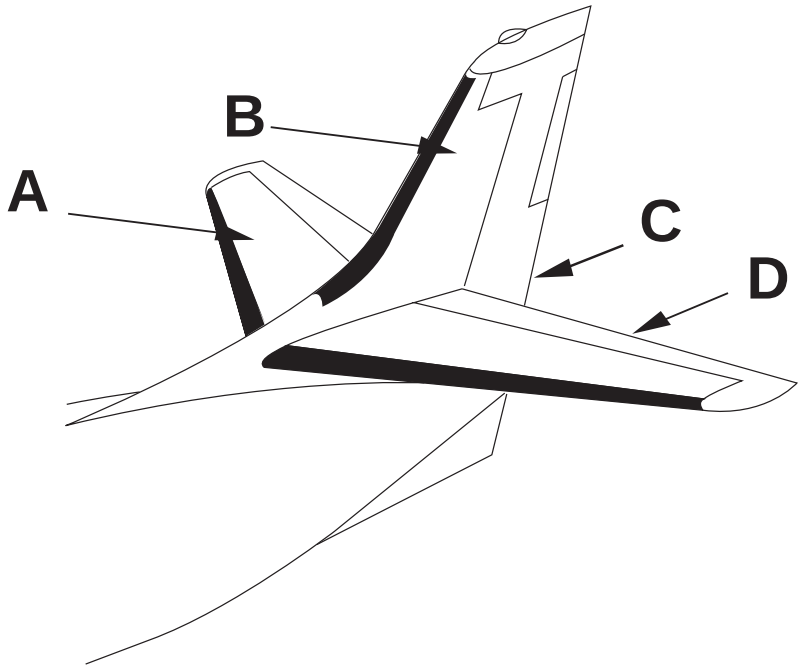
1. General

A. SA226 Series aircraft are equipped with conventional horizontal and vertical stabilizers, elevators and rudder. The fixed vertical stabilizer is attached to the fuselage. The horizontal stabilizer, mounted to the vertical stabilizer above the fuselage, is pivoted so the angle of incidence can be electrically adjusted to trim the aircraft in flight. The elevators are attached to the rear of the horizontal stabilizer. The rudder is attached to the rear of the vertical stabilizer. For removal/installation procedures see Chapter 27.

B. The major components for stabilizers are; ribs, stringers, spars, and skin with various other components pertaining to each stabilizer.

2. Horizontal Stabilizer

A. The horizontal stabilizer is a metal cantilever design structure, consisting of left and right units bolted together at the fuselage centerline. The horizontal stabilizer is attached to the vertical stabilizer with a large trunnion bolt. The unit pivots around the trunnion to change the angle of incidence of the stabilizer. The stabilizer is moved by two motor operated jack screws (pitch trim actuator) mounted on the rear top of the fuselage. The pitch trim actuator is controlled by the pilot or copilot with a trim switch on the control wheel.



A Horizontal Stablizer

B Vertical Stablizer

C Rudder

D Elevator

6M550000001

Stabilizers
Figure 1

STABILIZERS – DESCRIPTION AND OPERATION

3. Vertical Stabilizer

A. The vertical stabilizer is a metal cantilever design structure that consists of front and rear spars attached to two canted bulkheads in the fuselage.

B. The stabilizer structure consists of two spars with a built-up web, truss type ribs, and spanwise stringers, skin assemblies with removable fairings, and tip.

4. Rudder

A. The rudder is attached to the rear of the vertical stabilizer by two hinges, a torque tube, and bellcrank assembly. The rudder is equipped with a movable trim tab actuated by a chain driven actuator housed in the vertical stabilizer. A lead counterbalance is incorporated at the top of the rudder forward of the hinge line.

5. Elevator

A. The elevator frame is made of 2024-T4 aluminum alloy. The skin is 0.025 inch (0.635 mm) thick 2024-T3 aluminum alloy riveted to the frame. Balance of the elevator is accomplished by adding lead weights at the outboard forward end of the hinge line. The elevator has three hinges on each side. Two hinges are attached to the rear of the horizontal stabilizer and the third is part of the center splice plate of the rear spar.

6. Attach Fittings

A. The attach fittings used on the vertical and horizontal stabilizers are routed out parts. They are attached to the airframe with standard aircraft hardware and are removable to allow removal of the stabilizers.

(1) Fittings, AFT, Spar Splice – The fitting serves the dual purpose of a spar splice and support for the elevator bellcrank. The AFT side is the third pivot point for the elevator. It is attached to the AFT spars with 12 bolts on each side.

(2) Fitting, Horizontal Stabilizer Pivot Point – There are two fittings on the stabilizers. They are used to support the stabilizer and act as a pivot point for a trunnion bolt. This part is not removable.

(3) Arm Assembly, Actuator Support – This dual purpose fitting serves as a spar splice and a support for the trim actuator. It is attached with standard aircraft hardware.

(4) Splice Plate – There are two splice plates on the rear spar; one on the top and one on the bottom, attached with ten bolts each. They are made of 0.250 inch (6.350 mm) thick aluminum alloy.

(5) Plate, Front Spar – The front spar is spliced, top and bottom, with splice plates made of 0.250 inch (6.350 mm) thick aluminum alloy(and attached with 16 bolts.

B. Vertical Stabilizer – The vertical stabilizer is attached through the spar and is not dependent of any casting or fittings. It is secured with bolts through the spars and bulkheads in the fuselage.

HORIZONTAL STABILIZER – DESCRIPTION AND OPERATION

1. Horizontal Stabilizer Components

A. Ribs – The stabilizer ribs are formed parts made of 2024-T3 aluminum.

B. Stringers – The stringers are made of extruded bulb angle. They are attached to the ribs to form a frame for the skin attachment.

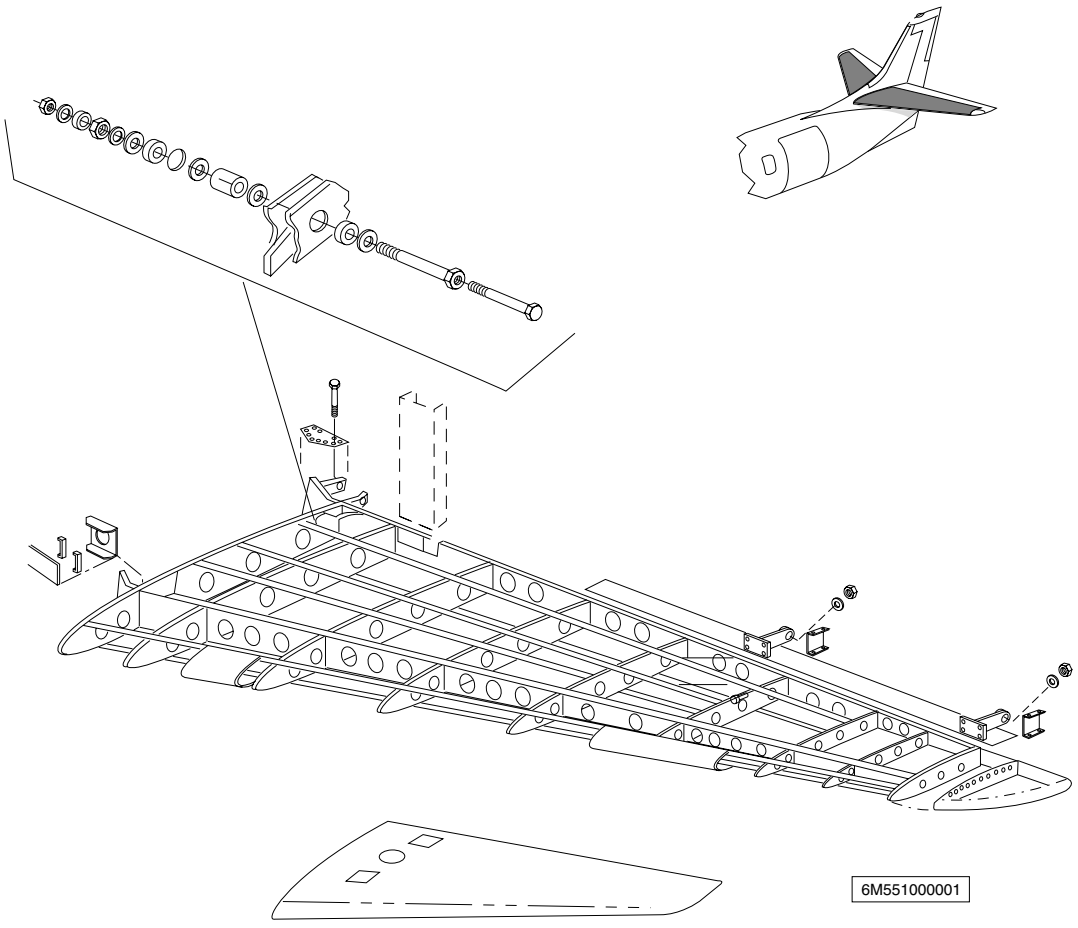
C. Spar – The spar is a built-up type with the spar web constructed by hydroforming metal heat treated to 2024-T4. There are two such spars in the stabilizers.

D. Skin – The skin is made in four sections; top and bottom forward, and top and bottom aft. All horizontal stabilizer skins are made from 2024-T3 0.032 inch Aluminum. It is attached to the frame work with rivets.

E. Deicing Boots – The horizontal stabilizer is equipped with rubber inflatable boots made by B.F. Goodrich. The boots are installed with EC-1300L glue.

F. Tip Fairings – The horizontal stabilizers have a tip made of aluminum alloy and is attached with screws.

G. Doors – There are six inspection doors, attached with screws, on the horizontal stabilizer in the lower inboard side.



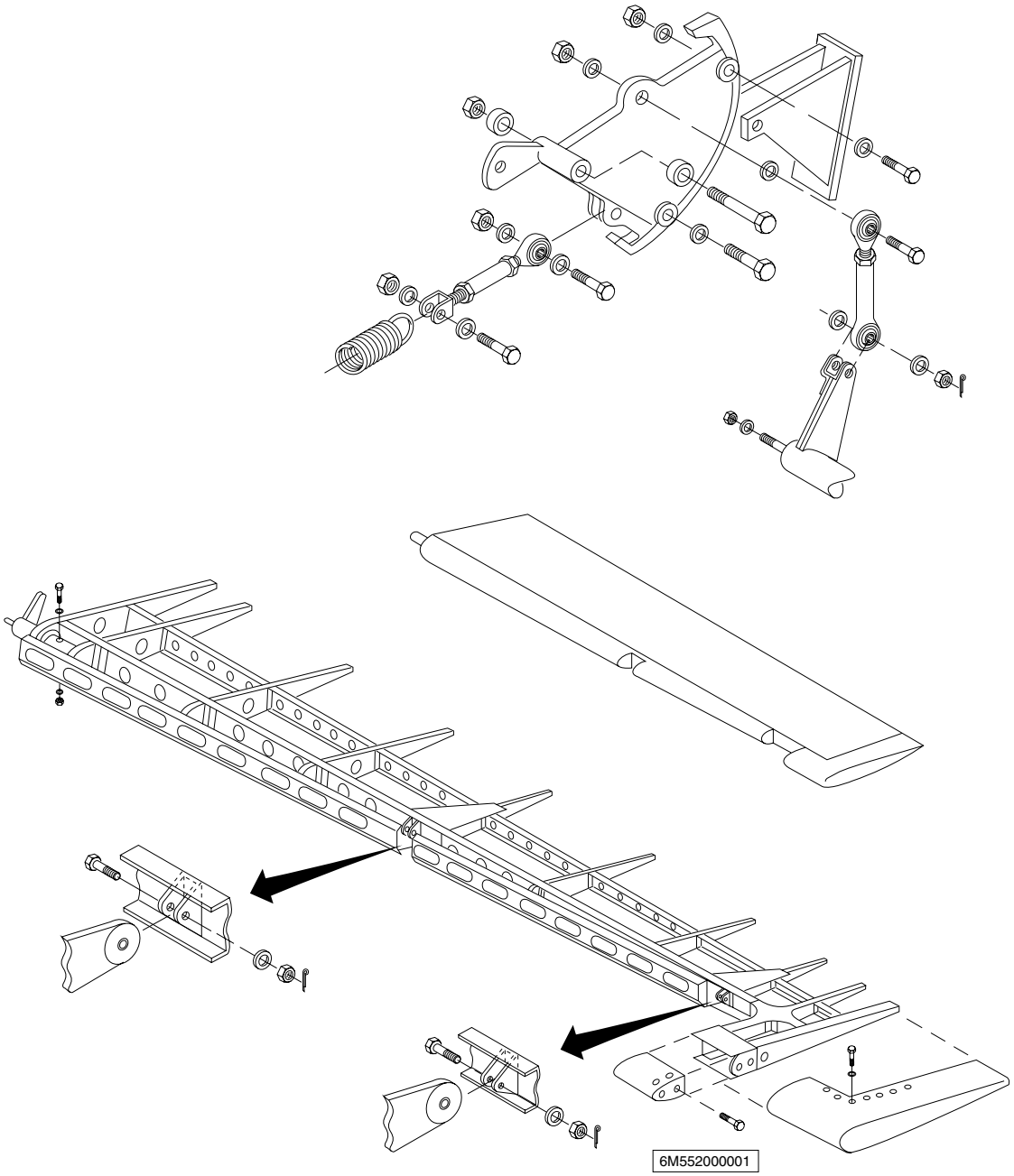
Horizontal Stabilizer
Figure 1

ELEVATOR – DESCRIPTION AND OPERATION

1. Elevator Components

A. Ribs – There are nine ribs on the elevators made of 2024-T4 0.032 inch aluminum alloy. They are attached to the spar with AN rivets.

B. Spar – The elevator spar is built of 2024-T4 0.063 inch aluminum alloy sheet. The spar is used to attach the ribs and the torque tube.



Elevator
Figure 1

C. Stringers – The stringers are made of 2024-T3 0.063 inch aluminum alloy and are used to separate the ribs and tie the skin onto the structure.

D. Skin – The skin used on the elevator is made from 2024-T3 aluminum alloy flat sheet. It is attached to the structure with rivets.

E. Elevator Tip Fairing – The elevator tip fairing is made of aluminum attached to the elevator with countersunk screws.

F. Elevator Torque Tube – A torque tube of 4130 chrome moly actuates the elevator and serves as the third hinge point. The torque tube is secured to the elevator with bolts.

G. Counterbalance – A one piece antimony lead counterbalance is attached to the forward outboard edge of the elevator and enclosed by the tip fairing.

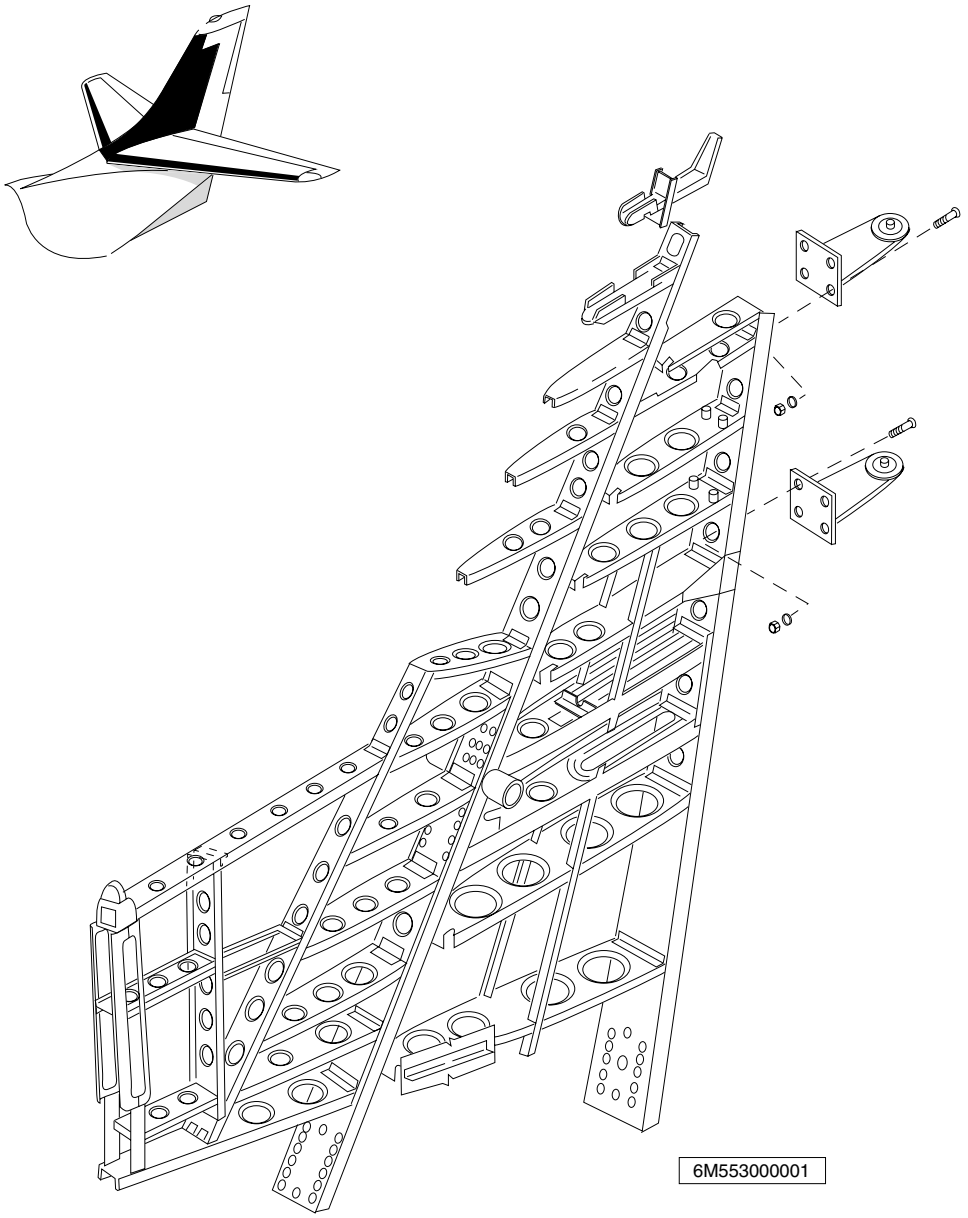
VERTICAL STABILIZER – DESCRIPTION AND OPERATION

1. Vertical Stabilizer Components

A. Tip – The removable stabilizer tip is manufactured of aluminum and contains an anti-collision light. The tip is attached with screws.

B. Spars – The spars for the stabilizer are built of 2024-T4. 0 063 inch aluminum alloy sheet. The spar is made by hydroforming and heat treated to 2024-T4. There are two such spars in the stabilizer.

C. Ribs – The stabilizer ribs are formed parts made of 2024-0 heat treated 2024-T



Vertical Stabilizer
Figure 1

D. Stringers – The stringers are made of extruded bulb angle. They are attached to the frame for skin attachment.

E. Skin – the skin is made of 2024-T3. 0 032 inch aluminum alloy. It is attached to the frame work with rivets.

F. Inspection Doors – There are a total of seven inspection and access doors on the stabilizer. The doors are installed on doublers riveted to the skin and attached to the skin with screws.

G. Hinge Assembly – Two hinge assemblies attached to the rear spar are used to support the rudder.

H. Dorsal Fin – On the Merlin IV SA226-AT and Metro SA226-TC aircraft, the dorsal fin runs from F.S. 483 to F.S. 579 at the vertical stabilizer front spar. It is constructed of laminated fiberglass and is attached to the fuselage and the vertical stabilizer with screws.

I. On the Merlin III SA226-T model aircraft, the dorsal fin extends from F.S. 267 to F.S.379 at the vertical stabilizer front spar.

1. Vertical Fin Trunnion Fitting Bushings

A. Removal

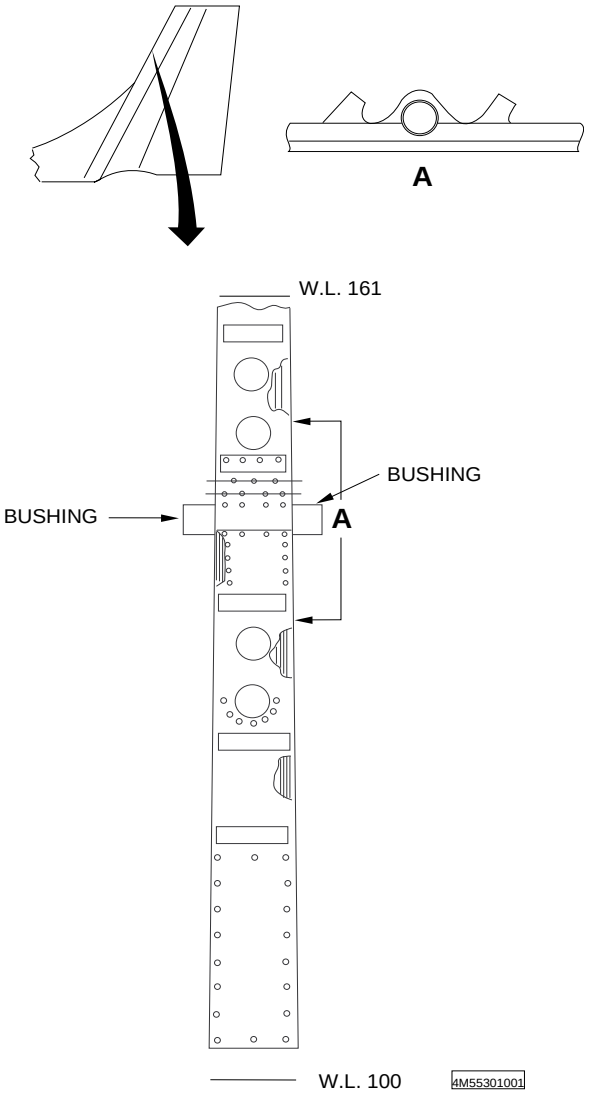
(1) Remove old bushing taking care not to score or damage the channel bore interior.

(2) Inspect channel bore interior for corrosion or wear.

B. Installation

(1) Clean the channel bore interior of any dust or debris that may prevent the proper seating of the bushing.

(2) Install bushing wet with epoxy primer MIL-PRF-23377.



Vertical Fin Trunnion Fitting Bushings
Figure 201

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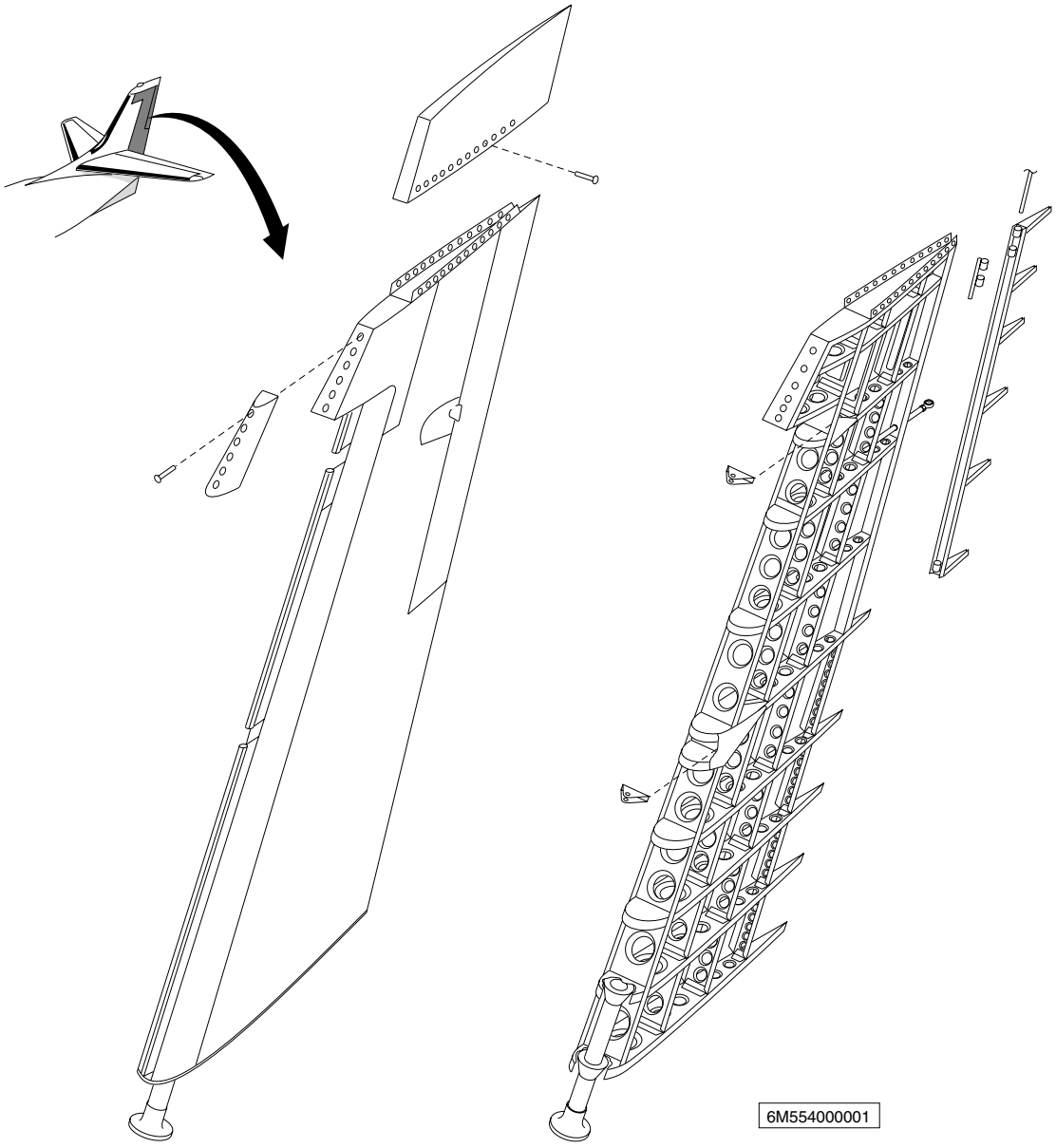
RUDDER – DESCRIPTION AND OPERATION

1. Rudder Components

A. Spar – The spar is one piece 89.00 inches (226.06 cm) long, made of 2024-T4 0.063 inch aluminum. The spar is used to attach the frames and skins.

B. Ribs – The ribs are constructed in two pieces of 2024-T4 0.032 inch aluminum alloy. They are attached to the front and rear spars with 2024-T3 0.032 inch aluminum clips. The skin is attached to the ribs.

C. Intercostal – There are two sets of intercostals made in sections to tie the ribs together. They are stamping of 2024-T4 flat aluminum material.



Rudder
Figure 1

D. Skin – The skin is made from flat sheet 2024-T3 0.025 inch aluminum and is attached to the frame with rivets.

E. Rudder Cap Assembly – The rudder cap is made of welded aluminum and attached to the rudder with screws.

F. Counterbalance – The lead counterbalance is attached to the top front of the rudder with six bolts.

G. Hinge – The hinge is made of extruded aluminum channel, and attached to the rudder with bolts and nut plates.

H. Trim Tab – The rudder trim tab is made from 2024-T5 0.016 inch aluminum skin and attached to the rudder with a piano hinge. It is actuated by a rod from the rudder trim actuator.

I. Torque Tube Assembly – The torque tube assembly is made of stainless steel, attached to the rudder with rivets and is mounted to the bellcrank with bolts. For details of the rudder and torque tube, see Chapter 27, Flight Controls.

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MAINTENANCE MANUAL

CHAPTER 56 – WINDOWS

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MAINTENANCE MANUAL

CHAPTER 56 – WINDOWS

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WINDOWS – DESCRIPTION AND OPERATION

1. General

A. Cockpit Compartment Windshields/Windows

(1) The cockpit compartment incorporates a three piece windshield and two side windows. The windshields consist of two electrically heated flat glass panes, located in front of the pilot and copilot. The single-curvature center windshield and the compound-curvature side windows are constructed of acrylic.

(2) On SA226 – T and SA226 – AT airplanes, dual acrylic panels are installed for the center windshield and each of the cockpit side windows. Bleed air flows between the panels for defogging on T225 and up and AT006 and up.

(3) On SA226 – TC airplanes, single acrylic panels are installed for the center windshield and each of the cockpit side windows as standard equipment. The dual panel construction and the bleed air defog are available for optional installation on TC205 and up.

B. Cabin Compartment Windows

(1) On SA226 – T airplanes, serial numbers T201 through T248, six dual panel windows are installed in the cabin passenger area. On serial numbers T249 and up, an additional window of the same size and type plus two smaller dual panel windows are installed in the aft portion of the fuselage.

(2) ON SA226 – AT airplanes, serial numbers AT001 through AT024, 14 dual panel round windows, of the dry air sandwich type, are installed in the cabin area. On serial numbers AT025 and up, 15 dual panel oblong windows are installed in the cabin area. In addition, window shades are installed as standard equipment.

(3) On SA226 – TC airplanes, serial numbers TC201 through TC212, 19 dual panel round dry air sandwich type windows are installed in the cabin area. On serial numbers TC213 and up, 19 dual panel oblong sandwich type windows are installed in the cabin area. In addition, window shades are installed on some SA226 – TC aircraft as an option.

NOTE: All cabin compartment windows are constructed of acrylic material.

2. Glass Windshield Description

A. General - Glass Windshield

The two cockpit heated windshields located directly forward of the pilot and copilot seats are toughened glass assemblies that incorporate a vinyl interlayer, a metallic retainer and insert, an electric heating grid, electric temperature sensors, anti-static protection, and an electrical terminal block. The windshield identification label is located on the lower inboard side of the windshield inner surface and may include the windshield manufacturer name and part number, the M7 Aerospace part number, the windshield serial number, and the date of manufacture. There are three different base numbers associated with the cockpit glass windshields. These base part numbers include 26-21126, 27-19442, and 27-19482. The information contained herein is applicable to all glass windshields unless otherwise noted.

B. Basic Construction - Glass Windshield

The glass windshields incorporate an inner glass ply, a vinyl interlayer, an outer glass ply, edge metallic insert and retainer, electric heating grid, electric temperature sensor(s), busbar, electric terminal block, and an anti-static coating on the outer surface.

A weather seal extends around the entire periphery of the windshield and is located between the outer edge of the outer glass ply and the metal retainer as shown in Figure 2.

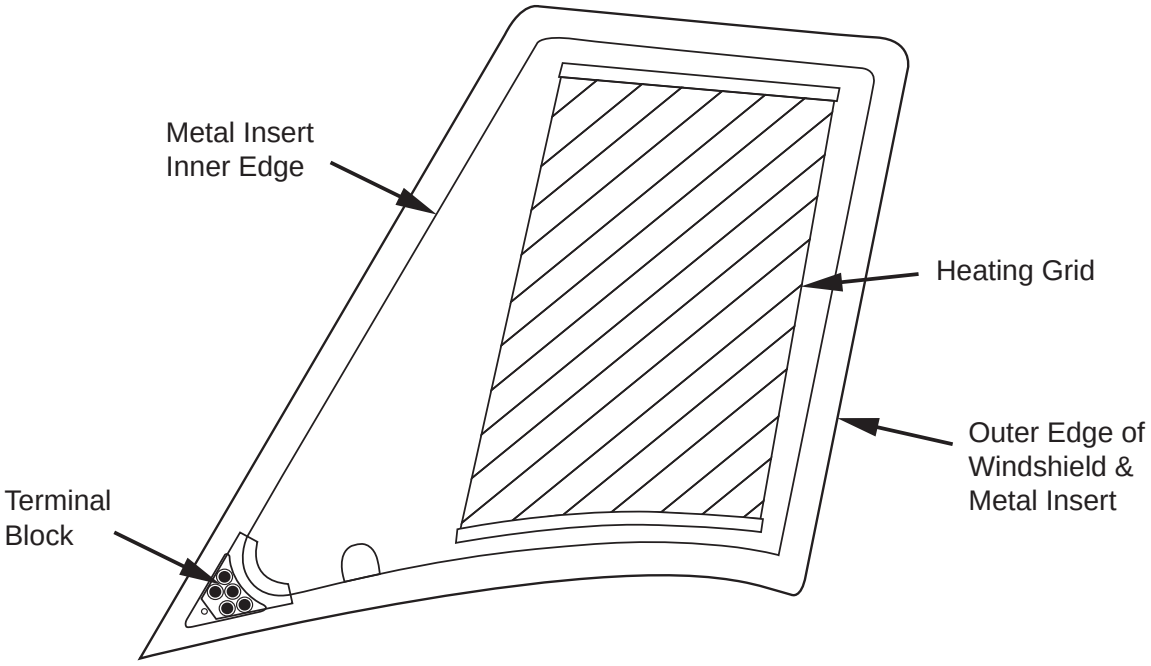


Figure 1
Glass Windshield Basic Arrangement

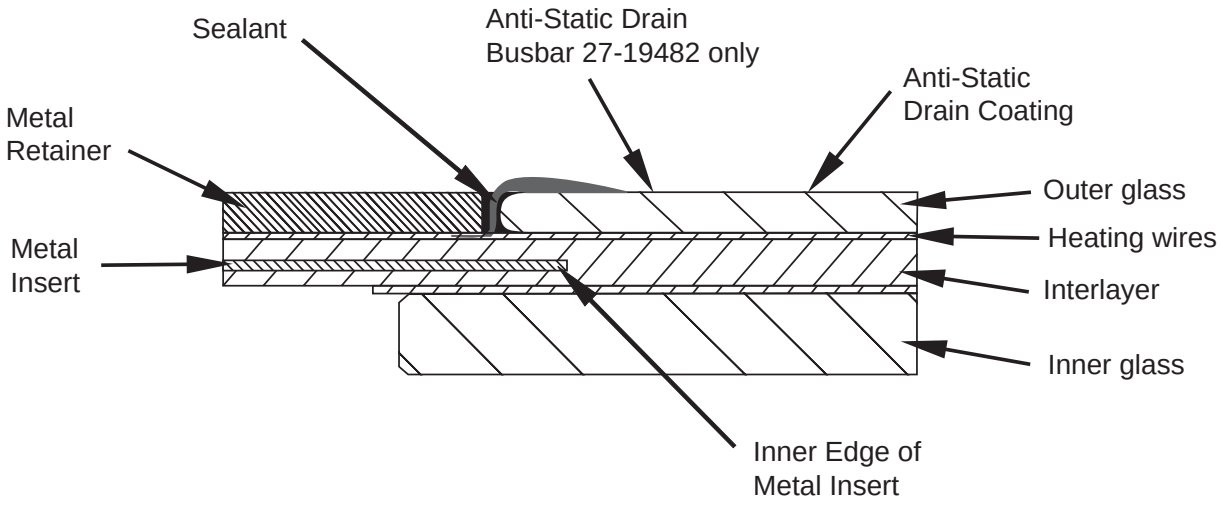


Figure 2
Glass Windshield Edge Cross Section

C. Electrical Components - Glass Windshield

The glass windshield incorporates an electric heating system to provide anti-icing capability. Additionally, electrical components are included to provide for the dissipation of static electric charges on the windshield. The elements of the electric heating system include a heating grid, temperature sensors, and an electric terminal block located on the windshield inner surface on the lower, inboard side. Provisions for the anti-static discharge system include an anti-static film and an anti-static tab. An anti-static busbar is included on the 27-19482 windshields only.

The electric heating grid is located between the interlayer and the outer glass as shown in Figure 2 and covers a portion of the windshield as shown in Figure 1. The heating busbar is located between the interlayer and the outer glass and extends along the lower, the inboard, and the upper edge of the windshield as shown in Figure 4. The temperature sensors and the electric lead wires to the temperature sensors are located between the interlayer and the outer glass ply. Some windshields incorporate a single temperature sensor and some incorporate two temperature sensors. The second temperature sensor is incorporated on some windshields in order to provide a spare sensor in case the initial sensor experiences a fault or failure.

The outer surface of the windshield incorporates a conductive coating for anti-static discharge purposes. An anti-static tab is installed on the outer surface of the windshield in the lower inboard portion as shown in Figure 5 to provide electrical connectivity between the conductive coating on the windshield outer surface and the electric terminal. Windshields with base part number 27-19482 also incorporate an anti-static discharge busbar bonded to the outer surface as shown in Figure 5.

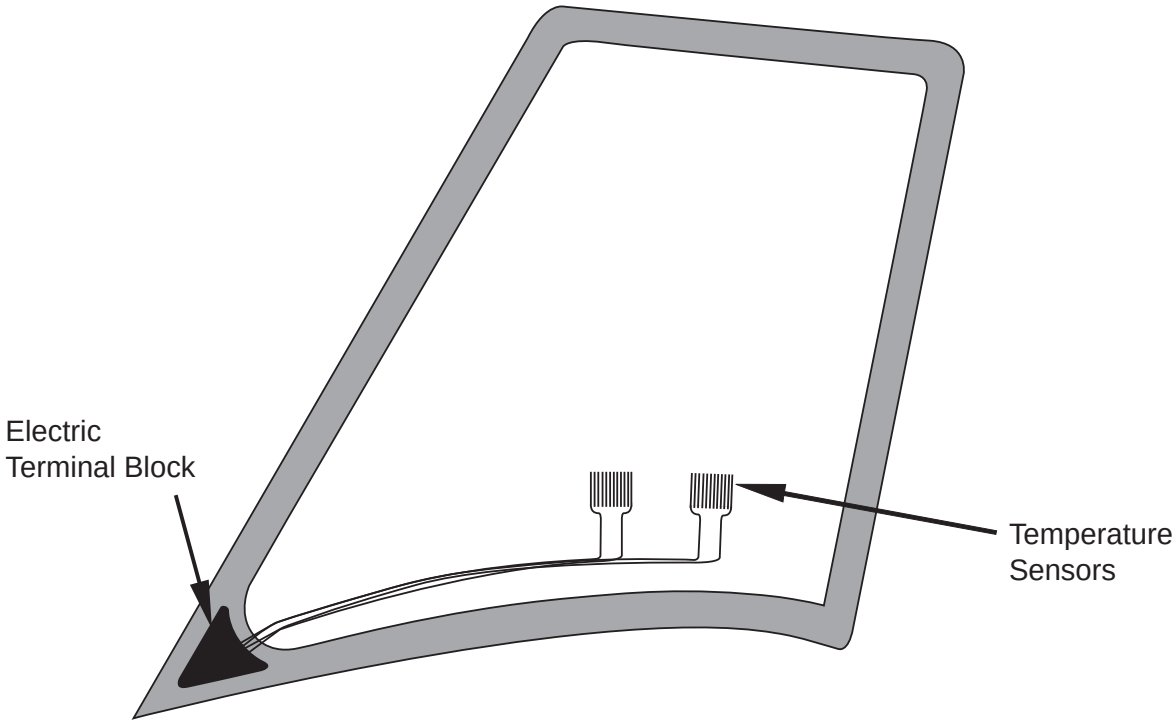


Figure 3
Temperature Sensors

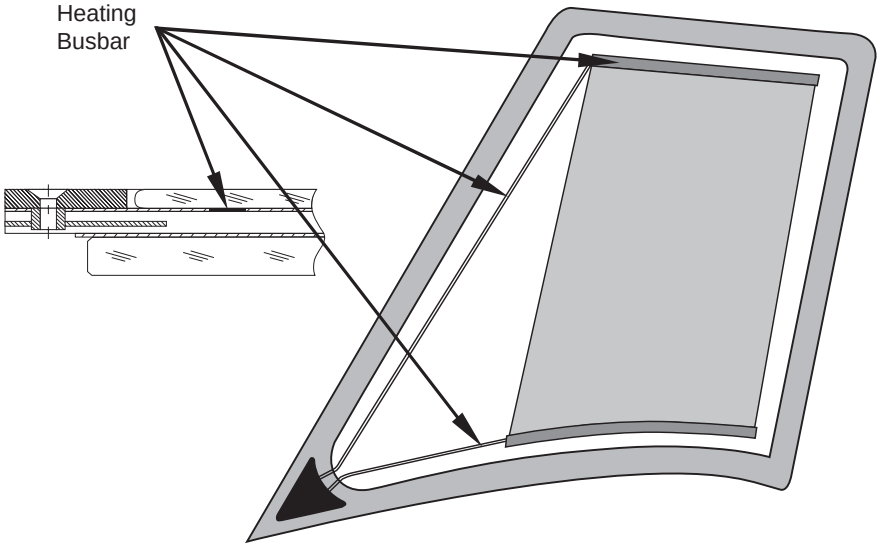


Figure 4
Windshield Heat Bus Bar

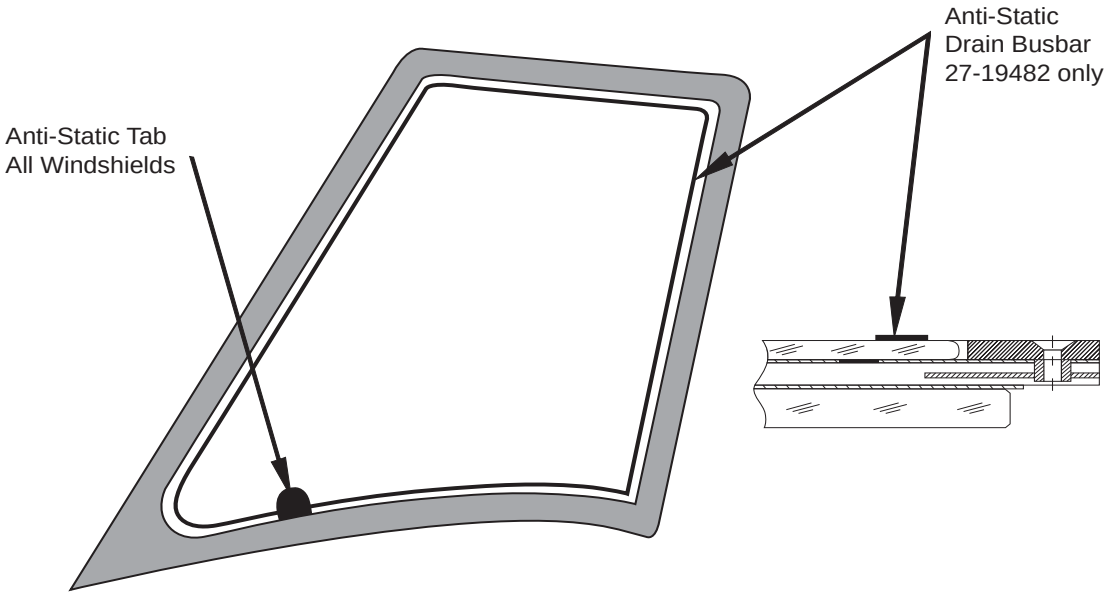


Figure 5
Windshield Anti-Static Drain Busbar
(27-19482 Only)

D. Critical Optical Zone - Glass Windshield

A Critcal Optical Zone is defined for the glass windshield as shown in Figure 6. Special criteria are applicable to the optical characteristics within the Critical Optical Zone. The zone extends 2.00 inches toward the center of the windshield as measured from the windshield "daylight opening" and 0.50 inches around the temperature sensors. The daylight opening is defined as the inner edge of the metal insert shown in Figure 2.

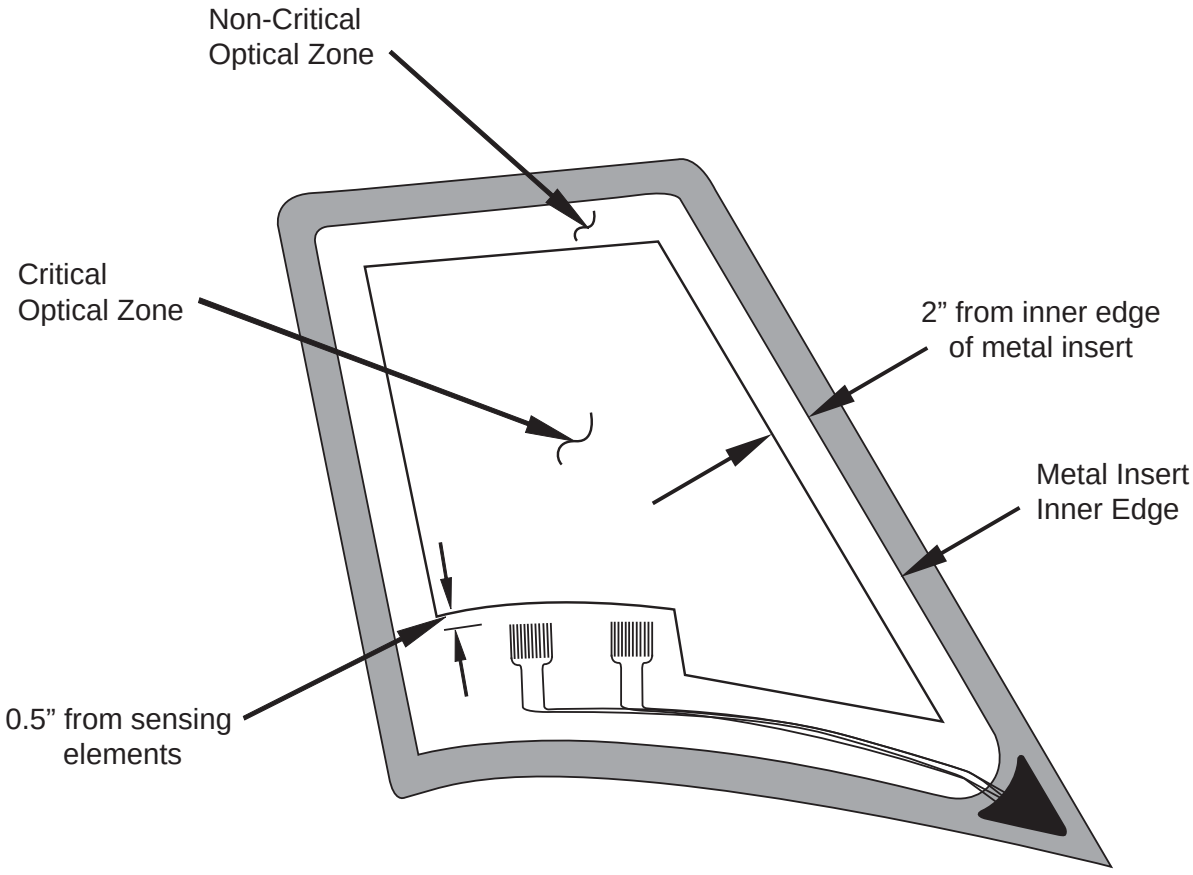
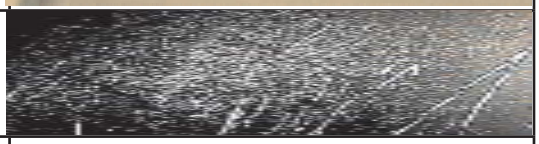
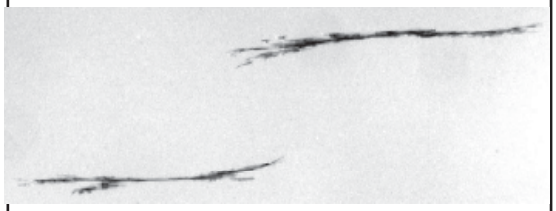


Figure 6
Windshield Critical Optical Zone

1. Inspection/Check – Windows

A. Definitions

Term	Definition	Notes
Blemish	Any imperfection in the transparent material not otherwise listed in this list of definitions.	
Bubbles	Clear brown or black spherical shapes located in the transparent material.	
Edge Chip	Area where a portion of the transparency material has detached from the edge of the transparency material.	
Crack	Narrow break, fissure or separation in the transparent material that extends through the entire thickness of the material.	
Craze **Crazing**	A network of fine breaks on the surface of the acrylic transparency.	
Critical Optical Zone	The viewing area of the windshield used during ground and flight operations.	See Figure 6, Chapter 56-00-00 - Description and Operation
Delamination	The disbonding or separation of one layer of material in a transparency assembly from another layer of transparency. For the heated windshield, delamination can occur by disbonding of either the inner or the outer glass panel from the interlayer material. Delamination appears as clear, flat and smooth bubbles with a circular or irregular edge.	
Dig	Gouging where small localized glass particles are removed from the surface of the glass material.	
Discoloration	Areas of black, blue, brown or white indications usually located around the edge of the heated windshield or in the heated area on the heated windshield.	

Distortion	Contoured lines in transparent material that cause changes in shape of objects when viewed through the transparency.	
Electrical Arcing / Windshield Heating Inoperative	Indicated by localized dark brown or black burn marks within the windshield around the heating wires, busbar or under the terminal block.	Windshield heating inoperative can be caused by electrical short circuit across a discontinuity in the busbar, heating wires or terminal block.
Glass Shear	In the heated windshield, particles of glass that detach from the glass surface adjacent to the interlayer material. Can appear similar to delamination but can be differentiated by appearance of rainbow colors, shear planes, and/or crushed contours that appear as white "frosting" when viewed with a flashlight.	
Haze	A foggy appearance when looking through the transparency.	
Interlayer Cracking	In the heated windshield, cracking that occurs within the vinyl interlayer material and appears as a black line or series of black lines. Interlayer cracking most typically occurs adjacent to the metal insert around the periphery of the windshield.	
Mark-Off	An almost non-existent depression on the surface possessing practically no depth. It is discernible only due to noticeable rim caused by mold surface defects as a result of forming operation.	
Moisture Ingress	Penetration of moisture or liquid between two layers of material in the transparency assembly. Moisture ingress indicates a degraded seal in the transparency assembly and is evidenced as a change from optically clear to cloudy or milky appearance. Moisture in close proximity to the heating grid on the heated windshield can cause discoloration or heating failures.	

Scratch	An abrasion on the surface of the material caused by contact with rough or sharp objects.	
Scratch - Type A	Unobtrusive abrasion on the surface of the transparency that is not easily seen or felt by dragging a fingernail across the scratch.	Generally considered to be less than 0.002 inches deep.
Scratch - Type B	Obtrusive abrasion on the surface of the transparency that is relatively easily seen or felt by dragging a fingernail across the scratch.	Generally considered to be 0.002 inches deep or greater.
Tong Marks	Small smooth depressions in the glass measuring approximately 0.080 inches (2 mm) in width. Tong marks are usually located within 0.20 inches (5 mm) from the edge of the glass and are due to the glass toughening process. Tong marks are located in the non-critical optical zone, will not grow in size and are acceptable.	

B. Inspection – Glass Windshields

NOTE: The glass windshield Critical Optical Zone is shown in Figure 201.

(1) Inspect the left and right cockpit heated windshields

(a) With the use of maintenance stands or ladders, obtain adequate access to perform detailed visual inspection of the entire left and right cockpit heated windshield from the outside of the aircraft.

(b) Ensure the surface of the glass is free from dirt, oil or other contaminants. Clean the windshield per 56-00-00, paragraph 2.A.

(c) With the use of a flashlight, perform a detailed visual inspection of the windshield inner glass ply, the interlayer and the outer glass ply (see Figure 2). Inspect for indications of delamination, interlayer cracking, moisture ingress, glass shear, digs, chips, cracks, scratches, electrical arcing or other defects. The viewing angle for the inspection should extend from a line of sight that is perpendicular to the surface of the glass to an angle extending approximately 45 degrees to either side of this plane.

(d) The limitations for discrepancies associated with the glass components are shown in Table 201. It is noted that the inner and outer glass plies can have different limitations. If the limitations of Table 201 are exceeded, replace the windshield prior to the next flight.

Discrepancy Limits			Outer Glass Ply	Inner Glass Ply	
			All Zones	Critical Optical Zone	Non-Critical Optical Zone
Blemish			Does not impair visibility	Diameter ≤ 0.062" No more than 2 at least 12" apart	Diameter ≤ 0.062" No more than 2 at least 12" apart
Edge Chip (No limit on maximum quantity)			Width ≤ 0.080" Depth ≤ 0.002"	**Not Allowed**	Width ≤ 0.080" Depth ≤ 0.002"
Crack			**Not Allowed**	**Not Allowed**	**Not Allowed**
Dig			Width ≤ 0.062" Depth ≤ 0.002"		
Glass Shear			Monitor (1) (2)	**Not Allowed**	**Not Allowed**
Scratch	Type A	Max Length	Does not impair visibility	6"	6"
		Total Length		20"	20"
	Type B	Max Length		2"	6"
		Total Length		2"	6"

(1) For indications of glass shear in the outer glass ply of the 27-19482 windshield, monitor the size and location of the glass shear indication. If outer glass ply glass shear indications increase in size or quantity, replace/repair windshield at earliest opportunity.

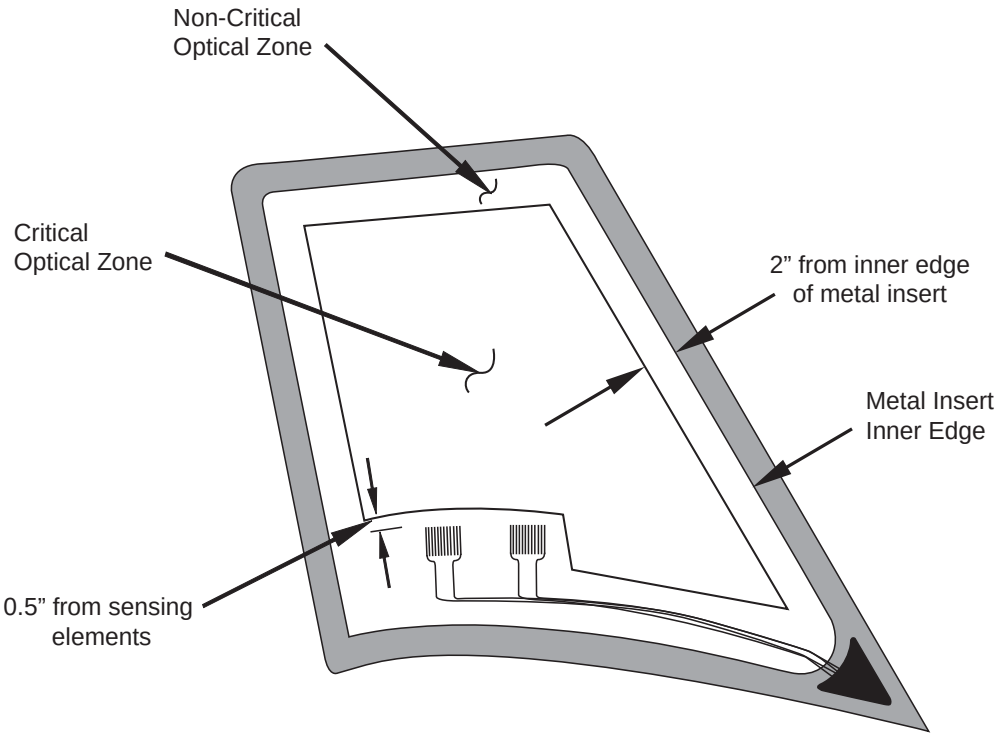
(2) Indications of glass shear in the 26-21126 or 27-19442 windshield require replacement prior to next flight.

Windshield Glass Limitations
Table 201

(e) Inspect the interlayer material for indications of bubbles. Bubbles in the interlayer are usually found in the windshield heated area and can be caused by faulty windshield heat or heat controller. Bubbles are allowed as long as visibility for airplane operation is not unacceptably affected. If air bubbles are in or adjacent to the heated area, perform functional test of Windshield Heat Control Unit and the Windshield Panel per Chapter 30-40-10 of this manual.

(f) Inspect the windshield for indications of delamination. Delamination is to be limited to the non-critical optical zone shown in Figure 201. If delamination is indicated, take care to differentiate between conditions of delamination and glass shear with the aid of a flashlight and by inspecting for rainbow colors. The presence of rainbow colors or distinct shear planes is an indication of glass shear. Delamination is allowed in the non-critical viewing zone as long as visibility for airplane operation is not unacceptably affected. Delamination **is not** permitted in the Critical Optical Zone.

(g) Inspect the windshield for indications of moisture ingress. Moisture ingress is allowed in the non-critical optical zone shown in Figure 201 as long as visibility for airplane operation is not unacceptably affected. Moisture ingress can be due to degraded weather seal. Repair weather seal as required at earliest opportunity. Continued moisture ingress can lead to degradation of the interlayer, electrical arcing and windshield heat system failure. Moisture ingress **is not** permitted in the Critical Optical Zone.



Windshield Critical Optical Zone
Figure 201

(h) Inspect the windshield for indications of discoloration. Discoloration adjacent to the heating busbar or windshield heating element can be caused by electric arcing as the result of moisture ingress. If present, perform operational test of the windshield panel per Chapter 30-40-10 of this manual. If moisture ingress is identified, repair weather seal at the earliest opportunity. Monitor changes in discoloration at each scheduled inspection.

(i) Inspect the interlayer material for indications of interlayer cracking particularly in the area adjacent to the metal insert (see Figure 202). An interlayer crack will appear as a a black line or a series of black lines when viewed from the edge of the crack and as a series of ridges and furrows when viewed perpendicular to the crack surface. If interlayer cracking is indicated, replace windshield prior to the next flight.

(j) Inspect the windshield weather seal around the outer periphery of the outer glass ply for signs of cracking, erosion and loose or missing areas of the seal. Repair the seal per section **3.**, Cockpit Glass Windshield Seal Repair, of this chapter.

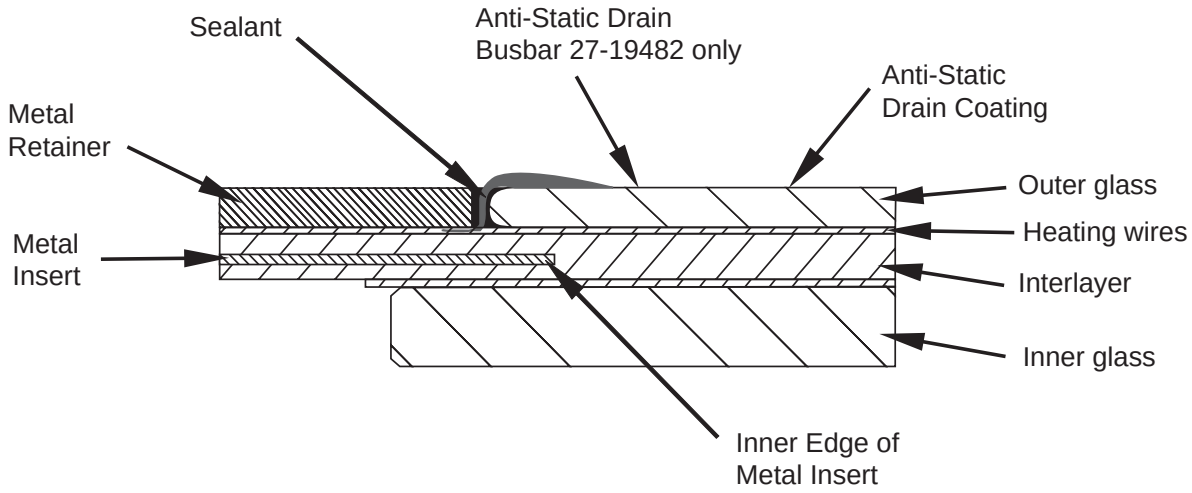
(k) Any discrepancy or combinations of discrepancies that result in unacceptable optical characteristics for the operation of the airplane require repair or replacement of the windshield.

(l) Inspect the glass windshield attachments for signs of loose or missing fasteners.

(m) For 27-19482 windshields, perform an electrical continuity test of the anti-static coating. Place one probe of the multimeter, set for continuity check, on an exposed part of the windshield retainer and the other probe anywhere on the anti-static

busbar located on the outer glass surface. If this checks indicates a lack of electrical continuity, the windshield must be replaced.

(n) Measure the resistance between the windshield metal retainer and the airframe structure. Resistance must not exceed 0.10 ohms.



Glass Windshield Edge Cross Section
Figure 202

C. Inspection - Cockpit Acrylic Windows

NOTE: The semi-critical area of the cockpit acrylic windows is defined as the perimeter of the window extending 1.00 inch (25.4 mm) towards the center of the window from the edge of the window retainer. The critical area of the cockpit acrylic windows is the area bounded by the semi-critical area.

(1) Mark-off

(a) Mark-off of such low intensity that vision quality is not impaired and is not visible when looking through the window is acceptable.

(2) Distortion

(a) A moderate amount of distortion is acceptable along the lower portion of the window if confined to 1.500 inches (3.81 cm) above the window retainer strip.

(b) Distortion may not be so severe that it restricts vision or diverts runway lines or section lines more than 45°.

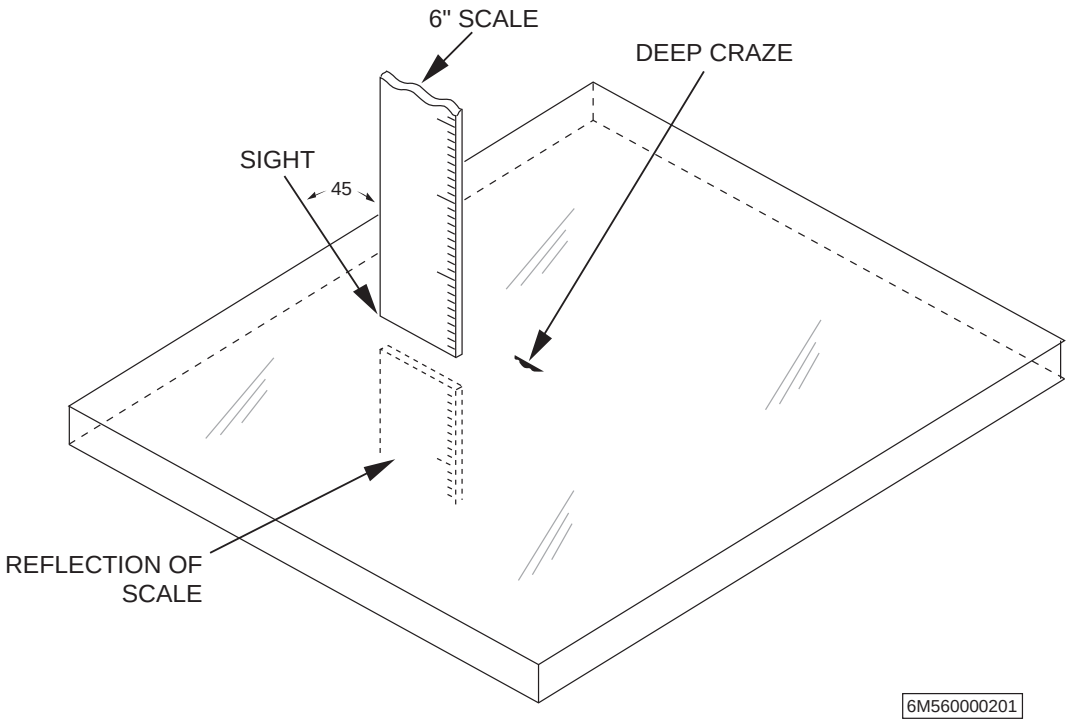
(c) Slight horizontal distortion is acceptable if; there are no more than two (2) lines, that do not occupy more than 25% of the window area, and are separated by a minimum of 6.00 inches (15.24 cm).

(3) Crazing (See Figure 203)

(a) Crazing in the window is critical.

(b) Determine depth of craze using a calibrated depth gauge or scale.

(c) Crazing in any portion of the windows require replacement. Crazing of depths up to 0.031 inch (0.787 mm) is cause for replacement at the earliest opportunity. Crazing of depth greater than 0.031 inch (0.787 mm) depth is cause for replacement prior to next flight.



Determining Depth of Craze
Figure 203

(4) Haze

(a) Haze or foggy appearance on the glass is not acceptable if the amount of haze/fog causes an obstruction of vision in the area used for operation of the aircraft.

(5) Scratches

NOTE: The semi-critical area of the cockpit acrylic windows is defined as the perimeter of the window extending 1.00 inch (25.4 mm) towards the center of the window from the edge of the window retainer. The critical area of the cockpit acrylic windows is the area bounded by the semi-critical area.

(a) In the semi-critical viewing area of the cockpit acrylic windows, scratches located in the lower portion of the window and less than 0.010 inch (0.25 mm) deep are acceptable if they do not extend more than 1.00 inch (25.4 mm) from the outside of the window retainer and are less 1.00 inch (25.4 mm) long.

(b) Scratches in acrylic windows may be reworked if:

1 No more than 0.020 inch (0.51 mm) deep.

2 0.375 inch (9.53 mm) panes are a minimum of 0.345 inch (8.76 mm) thick after rework.

3 0.250 inch (6.35 mm) panes are no less than 0.200 inch (5.08 mm) thick after rework.

4 0.188 inch (4.78 mm) panes are no less than 0.150 inch (3.81 mm) thick after rework.

5 No vision distortions in critical and semi-critical areas as a result of rework.

(c) Replace acrylic windows with scratches that cannot be reworked IAW these standards.

(6) Cracks

(a) Refer to **1.**, E. Crack Inspection – Acrylic Windows.

D. Inspection – Cabin Windows

NOTE: A non-critical area is a window not generally used for flight vision.

(1) Mark-Off

(a) A moderate amount of mark-off is acceptable provided visibility is not impaired.

(2) Distortion

(a) Distortion along the sides of the windshield within 0.0500 inch (1.27 cm) or less of the retainer is acceptable.

(b) Distortion along the upper portion of the windshield is acceptable if the distortion does not extend downward more than 2.00 inches (5.08 cm) from the top and does not impair vision.

(c) Distortion of the main body of the side window is acceptable providing the distorted area does not exceed 25% of the window area.

(3) Crazing

(a) Crazing in the window is critical.

(b) Crazing in any portion of the windows require replacement. Crazing of depths up to 0.031 inch (0.787 mm) is cause for replacement at the earliest opportunity. Crazing of depth greater than 0.031 inch (0.787 mm) depth is cause for replacement prior to next flight..

(c) Determine depth of craze using a calibrated depth gauge or scale.(See Figure 203)

(4) Haze

(a) Haze or foggy appearance on the glass is not acceptable if the amount of haze/fog causes an obstruction of vision in the area used for operation of the aircraft.

(5) Cracks

(a) Refer to **1.**, E. Crack Inspection – Acrylic Windows.

(6) Chips, Scratches, and Crazing Limitations.

(a) Scratches in acrylic windows may be reworked if:

1 They are less than 0.020 in (0.508 mm) deep.

2 0.250 in (6.350 mm) panes are a minimum of 0.200 in (5.080 mm) thick after rework.

3 0.188 in (4.78 mm) panes are a minimum of 0.150 in (3.81 mm) thick after rework.

4 No vision distortions in critical and semi-critical areas as result of rework.

(b) Replace acrylic windows with scratches that cannot be reworked IAW these standards.

E. Crack Inspection – Acrylic Windows

NOTE: Inspection of cabin window outer panes may be facilitated by the use of Air Aid 04-56001-1 and -2 Aircraft Window Inspection Kit. If this kit is not used, the outer window frames will have to be removed.

(1) Inspect outer window panes as follows:

(a) Inspect sealant for signs of wear, deterioration, and positive contact with mounting surfaces.

(b) Inspect window frames for cracks, loose rivets or screws, corrosion, and structural defects.

(2) Inspect cabin outer retainers as follows:

NOTE: Windows can be inspected with retainers in place, see **1.**, E., (3) Crack Inspection with Outer Retainer In Place.

(a) Slowly remove outer retainer screws.

(b) Gently separate the retainer from the outer pane.

NOTE: It may be necessary to cut through sealant around outer edges of window with a sharp bladed instrument. If cutting through sealant is not successful:

WARNING: WEAR EYE PROTECTION AND GLOVES, WHEN USING SAFETY WIRE TO CUT WINDSHIELD SEALANT.

1 Dig a small hole in the sealer and push through a piece of 0.020 inch thick stainless safety wire.

2 With one (1) person inside and one (1) outside, pull wire through sealer using a sawing action. Remove outer panel.

(c) Remove sealant residue from window frame and retainer.

(d) Inspect outer pane IAW 1., E., (4) Crack Limitations, this section. Check for cracks extending from retaining screw holes.

(e) If replacement of outer pane or retainer is not required, remove old sealant and install the retainer with new sealant.

(f) Stage torque retainer and outer panel screws as follows:

1 Begin at 12 o'clock position, Employ a diagonal sequential tightening schedule, i.e., 12 to 6, 3 to 9, 12:05 to 6:05, 3:05 to 9:05, etc.

2 Do not torque more than 4.50 inch-pounds (0.505 N·m) on initial torque.

3 Retorque all screws in the same sequence, 15 to 25 inch-pounds (1.69 to 2.82 N·m).

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION,

CAUTION: PREVENT METHYL PROPYL KETONE (MPK) FROM CONTACTING ACRYLIC WINDOW PANE. DAMAGE TO THE ACRYLIC WILL RESULT.

(g) Remove all excess sealant with methyl propyl ketone (MPK) or suitable substitute.

(h) Retouch screw heads as required for cosmetic purposes.

(3) Inspect cabin windows outer retainer without removal using Inspection Kit 04-56001-1 or -2.

NOTE: A 45 degree acrylic prism and glycerin may be used in lieu of the Inspection Kit when inspecting cockpit acrylic window panes.

(a) Clean the area to be inspected with soap and water ensuring surface is free of dirt, wax, and oil.

(b) Determine which immersion oil to use. Use type A in cold weather; type B in hot weather.

(c) Apply a small amount of immersion oil to surface of window adjacent to the mounting edge. On initial application, use a small amount of oil applied directly to one 90° face of the inspection prism in contact with glass or acrylic (see Figure 204).

(d) Press oiled face of the prism to acrylic. Slide prism around until a constant film of oil extends across prism face and window surface.

(e) Look into 90° face of prism (see Figure 204). The image of an unfractured fastener hole will appear as a frosty cylinder.

(f) If hole is countersunk, cylinder will appear to have a cone setting on one end as seen in Figure 204.

(g) The image of a cracked hole will appear as a frosty or reflective ear or projection, extending from one hole (see detail A-A of Figure 205).

(h) The image of a crack from one hole to another will appear as a frosty or reflective irregular surface (see detail B-B of Figure 205).

(4) Crack Limitation

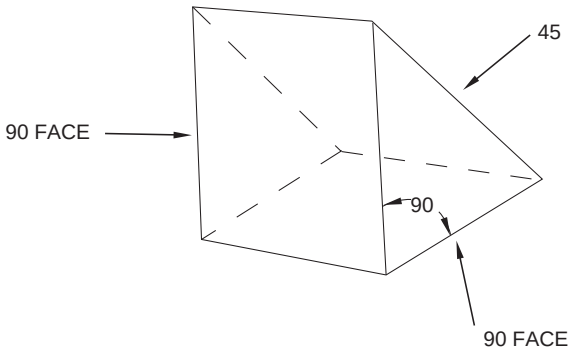
(a) The maximum acceptable crack length from any single screw is 0.300 inch (7.62 mm) (see Figure 205).

(b) The maximum combined length of multiple cracks into the space between two (2) adjacent screw holes is 0.300 inch (7.62 mm) (see Figure 205).

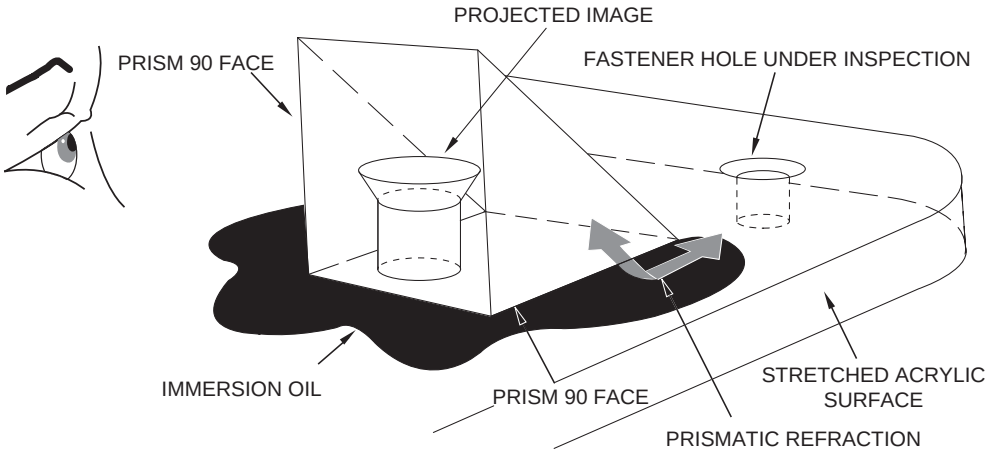
(c) If a crack is less than 0.300 inch (7.62 mm) long, inspect the window every twenty-five (25) flight hours.

(d) A maximum of three (3) outer window pane screw holes with cracks less than 0.299 inch (7.59 mm) are acceptable before window pane must be replaced.

(e) Replace outer window pane if a crack or combination of adjacent facing cracks exceed 0.300 inch (7.62 mm) long.



A. PRISM FACES



B. PRISM INSPECTION

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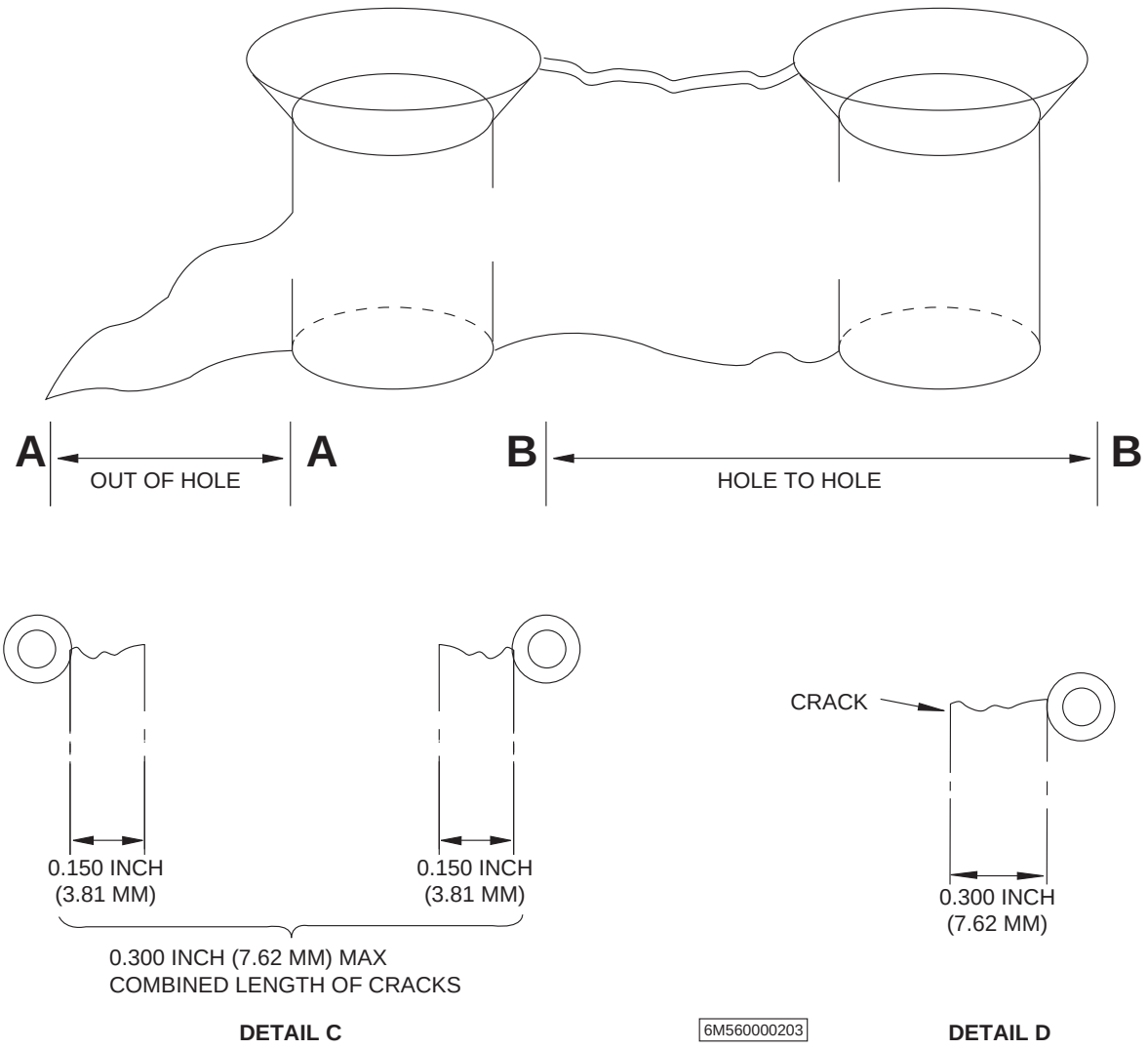
Visual Inspection
Figure 204



Crack Limitations
Figure 205

F. Polishing Equipment - Acrylic Windows

NOTE: The following equipment is acceptable for refinishing acrylic windows:

Mirco-Mesh Polishing Kit, obtained from;
Micro-Surface Finishing Products, Inc.,
Box 456, Wilton, Iowa 52778.

MIL-M-58091A (Aircraft Acrylic Plastic Window Refinishing Maintenance Kit).

Glass Polish Kit, P/N RS691560-00-450-3622,
Dept. of the Army TB-55-1560-276-241.

G. Polishing Procedure - Acrylic Windows

NOTE: The following polishing procedure is intended for application to acrylic windows only. This procedure is **NOT** to be applied to the cockpit glass winshield.

(1) Work an area considerably larger than damaged portion to blend with surrounding surface.

(2) Clean area thoroughly between each sanding phase.

(3) Slightly increase polish area with each sanding phase.

(4) Begin by sanding damaged area with 1500 polysand until original damage has been removed.

(5) Clean surface and lightly sand with 1800 polysand, working each square foot for approximately three (3) minutes.

(6) Lightly sand with 2400 polysand, working each square foot for approximately three (3) minutes and alternating horizontal and vertical strokes every twenty (20) seconds.

(7) Lightly sand with 3200 polysand, working each square foot for approximately two (2) minutes and alternating horizontal and vertical strokes every twenty (20) seconds.

(8) Lightly sand with 3600 polysand, working each square foot for approximately two (2) minutes and alternating horizontal and vertical strokes every twenty (20) seconds.

(9) Lightly sand with 4000 polysand, working each square foot for approximately two (2) minutes and alternating horizontal and vertical strokes every twenty (20) seconds.

(10) Lightly sand with 6000 polysand, working each square foot for approximately one and one half (1.5) minutes, alternating horizontal and vertical strokes every fifteen (15) seconds.

(11) Very lightly sand with 8000 polysand, working each square foot for approximately one and one half (1.5) minutes, alternating horizontal and vertical strokes every fifteen (15) seconds.

(12) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax; apply with fingertips using a circular motion.

(13) Buff clean with soft flannel cloth or cotton batting.

(14) Inspect polished area for imperfections.

(15) Repeat process as necessary.

(16) Ensure repair area does not exceed minimum material thickness requirements.

H. Precautions Prior to Painting

CAUTION: PROTECT WINDOWS DURING PAINTING. PAINT STRIPPERS, METAL BRIGHTENERS, AND SOLVENTS MAY DAMAGE EXPOSED SEALANTS, ANTI-STATIC COATINGS AND ACRYLICS.

(1) Clean windows IAW Cleaning Procedures, this Section.

(2) Apply 2.00 in (5.080 cm) wide aluminum tape. Start at bottom of window, applying each successive layer in shingle fashion, overlapping the previous layer by 0.500 in (12.70 mm).

(3) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax; apply with fingertips using a circular motion. Buff clean with soft flannel cloth or cotton batting.

(4) Inspect visually for imperfections. If any damage is visible, repeat necessary steps.

(5) Use a hard rubber roller to press tape smooth onto surface. Trim excess tape with a razor blade before smoothing.

(6) Inspect tape to ensure there are no fluid entry points.

(7) After painting, remove tape and clean all windows IAW Cleaning Procedures, this Section.

2. Cleaning/Painting – Windows

A. Cleaning – Glass

NOTE: The following procedure is recommended for glass windshields with surface coating for anti-static protection.

(1) Remove dirt and other substances from glass surface with clean water.

(2) Wash with mild soap and water or a 50/50 solution of isopropanol and water. Do not use abrasive material or strong acids. Wipe surface with a soft cloth or sponge.

(3) Rinse thoroughly and dry. DO NOT WAX GLASS.

(4) Clean wiper blades frequently to help prevent windshield scratching.

B. Cleaning – Acrylic

(1) Flush dirt from the acrylic surfaces with clean water.

(2) Wash with mild soap and water, 50/50 solution of isopropanol, and water or aliphatic naptha, type two.

(3) Do not use abrasive materials, strong acids, methanol, or methyl propyl ketone (MPK).

(4) Wipe surface with a soft cloth or sponge.

(5) Rinse thoroughly and dry.

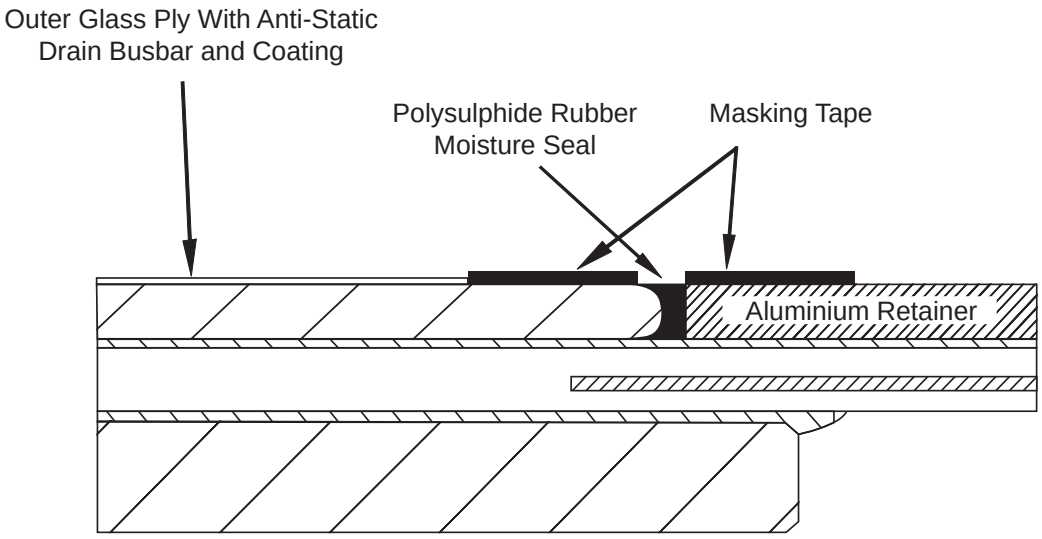
(6) After cleaning, acrylic surface may be polished IAW Polishing Procedure, this section.

WINDOWS – MAINTENANCE PRACTICES

3. Cockpit Glass Windshield Seal Repair

A. The 26-21126, 27-19442 and 27-19482 cockpit heated glass windshields incorporate a polysulfide seal against moisture all around the periphery of the windshield. The integrity of this seal must be maintained in order to prevent moisture ingress which can lead to discoloration, electrical arcing, interlayer degradation, windshield heating system failure, etc. If the integrity of the windshield is compromised as indicated by any of these faults or damage, or if the seal indicates signs of erosion or cracking, repair the seal as indicated in this section.

(1) Apply masking tape on the outer glass ply and the metal retainer immediately adjacent to the seal as shown in Figure 206.



Application of Protective Masking
Figure 206

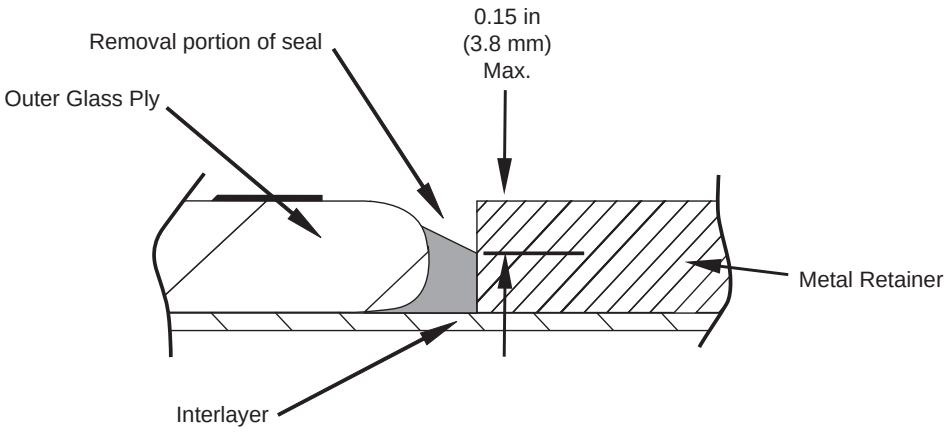
(2) Use a sharpened piece of plastic or sharpened piece of glass reinforced plastic (GRP) to remove the damaged portion of the polysulfide sealant between the outer glass ply and the metal retainer. Limit the depth of seal removal to no more than 0.15 in (3.8 mm) as damage to the interlayer and heating grid wires can result (see Figure 207).

CAUTION: **DO NOT USE METAL TOOL FOR REMOVAL OF THE POLYSULFIDE SEALANT AS THIS CAN DAMAGE THE WINDSHIELD THEREBY REQUIRING REPLACEMENT**

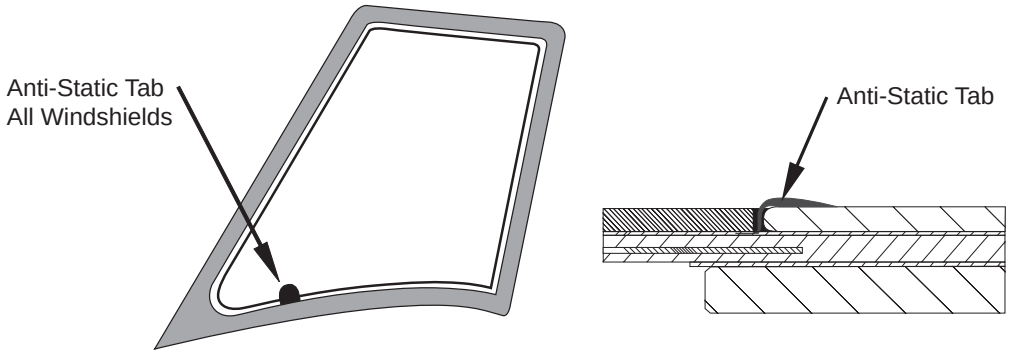
(3) For the removal of the damaged seal in the area of the anti-static tab, special precautions are required to prevent damage to the electrical conductor extending from under the metal retainer to the outer glass surface (see Figure 208).

(4) Remove the masking tape applied to the outer glass ply and the metal retainer in step 3.,A.,(1).

(5) Remove all loose dirt dust, pieces of sealant and other foreign material with a vacuum or a clean lint free cloth.



Removal of Seal
Figure 207



Anti-Static Tab
Figure 208

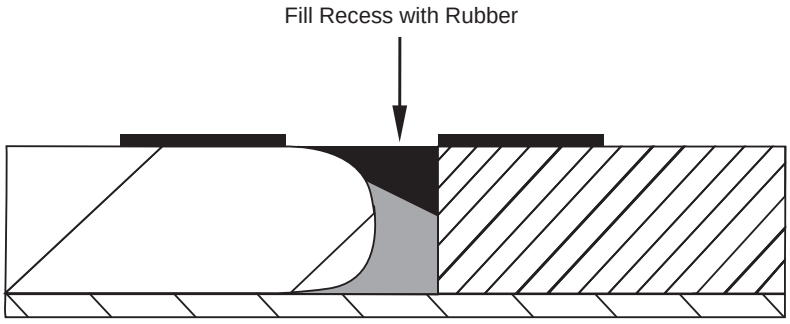
(6) Remove all dirt, oil, grease or other surface contamination with a clean lint free cloth moistened with Isopropyl Alcohol or Aliphatic Naptha (TT-N-95). Do not allow solvent to dry. Wipe immediately with a clean, dry lint free cloth.

(7) Apply masking tape on the outer glass ply and the metal retainer immediately adjacent to the seal as shown in Figure 206.

(8) Mix an adequate amount of PR-1425, Class B polysulfide sealant per mixing and handling instructions shown in the SA226/227 Structural Repair Manual (SRM) 51-30-03, step 1.C.

(9) Apply PR-1425, Class B polysulfide sealant to the clean, dry recess in the windshield seal. Smooth wet sealant with a flat, clean piece of plastic or glass reinforced plastic (see Figure 209).

(10) Remove the masking tape immediately after filling the recess and before the sealant starts to cure. Later removal of the tape can compromise the seal which will then require repair to the seal again.



Windshield Sealant Application
Figure 209

(11) Allow the sealant to cure (refer to manufacturer's technical data sheet for cure times).

(12) Once the sealant has fully cured, gently clean any residue left by the masking tape from the outer glass ply and the metal retainer.

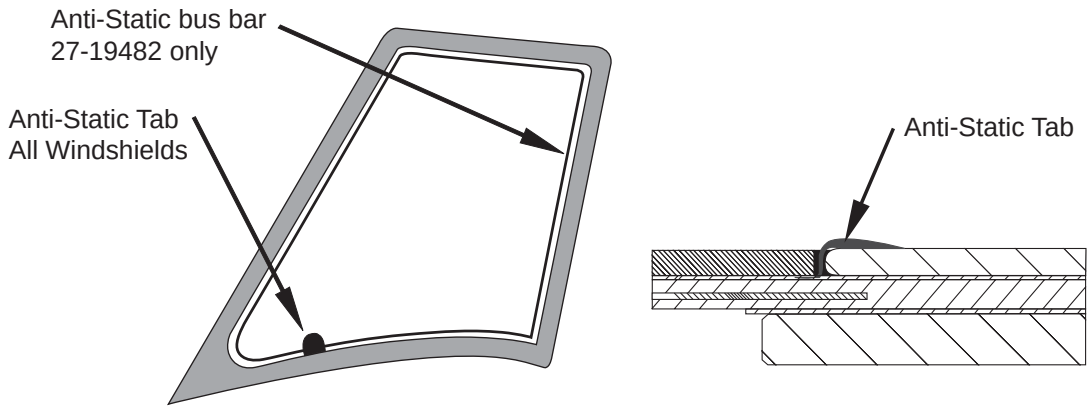
4. Anti-Static Tab Repair

A. The cockpit glass heated windshields incorporate an anti-static coating on the outer surface of the outer glass ply and an anti-static tab cover as shown in Figure 209. The purpose of the anti-static coating and the anti-static tab is to provide an electrical path for the dissipation of static electric charges that can develop immediately adjacent to the windshield. The integrity of this electrical load path is critical for the proper dissipation of these static electric charges. The anti-static tab incorporates a copper strap (copper foil tape and a copper braid) that is mechanically and electrically bonded to the inner surface of the metal retainer and to the outer surface of the glass. The 27-19482 windshields incorporate an electrical bus bar on the outer surface of the outer glass ply to which the anti-static tab is bonded. The 26-21126 and 27-19442 windshields bond the anti-static tab directly to the glass coating. A fiberglass cover with a polyvinyl butyral sublayer is bonded on top of the foil tab to provide protection. Inspect the anti-static cover for signs of wear, erosion, looseness, etc. If damage to the anti-static tab is indicated, repair as follows.

(1) Loose Tab Cover

(a) If possible, inspect the condition of the copper strap (braid and tape) that extends from the outer glass into the seal area. If the strap is missing or broken, the windshield must be replaced.

(b) Check the electrical continuity between the outer glass anti-static coating and the windshield retainer using a multimeter. For 27-19482 windshields, place one probe of the multimeter on an exposed part of the retainer (no paint, primer or sealant) and the other probe anywhere on the anti-static bus bar located on the outer glass surface. For the 26-21126 and 27-19442 windshields, the anti-static copper strap must be exposed to do the continuity check. If the strap is so exposed, place one probe of the meter on an exposed part of the retainer (no paint, primer or sealant) and the other probe anywhere on the anti-static strap where it is bonded on the outer glass surface. Use caution so as to not damage the copper strap. If this check indicates a lack of electrical continuity, the windshield must be replaced.



Anti-Static Tab
Figure 210

(c) Clean the glass surface, the retainer, the windshield area and the anti-static cover. Remove all dirt, oil, grease or other surface contamination with a clean lint free cloth moistened with Isopropyl Alcohol or Aliphatic Naptha (TT-N-95). Do not allow solvent to dry. Wipe immediately with a clean, dry lint free cloth.

(d) Position the tab cover such that it is centered on the copper strap and extends beyond the strap. Bond the cover to the glass surface using PR-1422 or PR-1425, Class B sealant.

(e) Apply a small bead of sealant (PR-1422 or PR-1425, Class B) around the entire periphery of the anti-static cover. The bead of sealant should not extend more than approximately 0.20 in (5.1 mm) past the edge of the anti-static tab cover.

(f) For 27-19482 windshields, recheck the electrical continuity between the outer glass coating and the windshield retainer as indicated in 4.,A.,(1),(b).

(2) Missing Tab Cover

If the fiberglass cover over the anti-static copper strap is missing, incorporate the following repair:

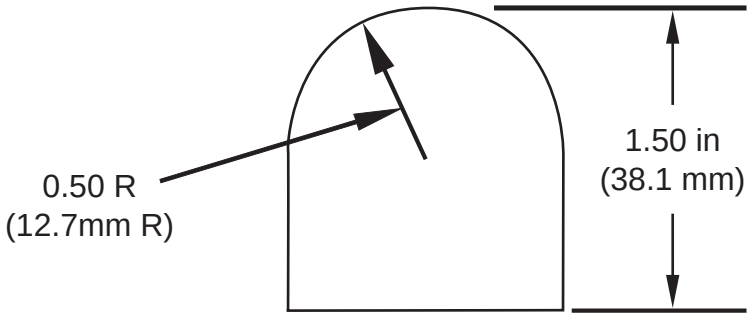
(a) Inspect the condition of the copper strap (braid and tape) that extends from the outer glass into the seal area. If the strap is missing or broken, the windshield must be replaced.

(b) If the copper strap is not attached to the outer glass, bond it to outer glass surface (or the anti-static busbar for the 27-19482 windshield) using electrically conductive adhesive such as Ellsworth SEC1233 or other adhesive that has an operating temperature range of -40°F to +160°F and provides acceptable electrical characteristics per step 4.A.(2)(c).

(c) Check the electrical continuity between the outer glass anti-static coating and the windshield retainer using a multimeter set for continuity check. For 27-19482 windshields, place one probe of the meter on an exposed part of the retainer (no paint, primer or sealant) and the other probe anywhere on the anti-static bus bar located on the outer glass surface. For the 26-21126 and 27-19442 windshields, the anti-static copper strap must be exposed to check the electrical continuity. Place one

probe of the meter on an exposed part of the retainer (no paint, primer or sealant) and the other probe anywhere on the anti-static strap where it is bonded on the outer glass surface. Use caution so as to not damage the copper strap. If this check indicates a lack of electrical continuity, the windshield must be replaced.

(d) Fabricate a new cover using polyvinyl butyral or polyurethane sheet with a minimum thickness of 0.040 in (1.03 mm). The size and shape of the repair cover are as shown in Figure 211.



Anti-Static Tab Repair Cover
Figure 211

(e) Clean the glass surface, the retainer, the windshield area and the anti-static cover. Remove all dirt, oil, grease or other surface contamination with a clean lint free cloth moistened with Isopropyl Alcohol or Aliphatic Naptha (TT-N-95). Do not allow solvent to dry. Wipe immediately with a clean, dry lint free cloth.

(f) Position the tab cover such that it is centered on the copper strap and extends beyond the strap. The tab cover should extend beyond the outer glass ply seal and onto the top of the windshield retainer. Bond the cover to the glass surface using PR-1422 or PR-1425, Class B sealant. Use caution so that the adhesive does not contact the copper strap.

(g) Apply a small bead of sealant (PR-1422 or PR-1425, Class B) over the entire tab repair cover and around the entire periphery of the anti-static cover. The sealant should not extend more than approximately 0.20 in (5.1 mm) past the edge of the anti-static tab cover.

(h) For 27-19482 windshields, recheck the electrical continuity between the outer glass coating and the windshield retainer as indicated in 4.,B.,(2).

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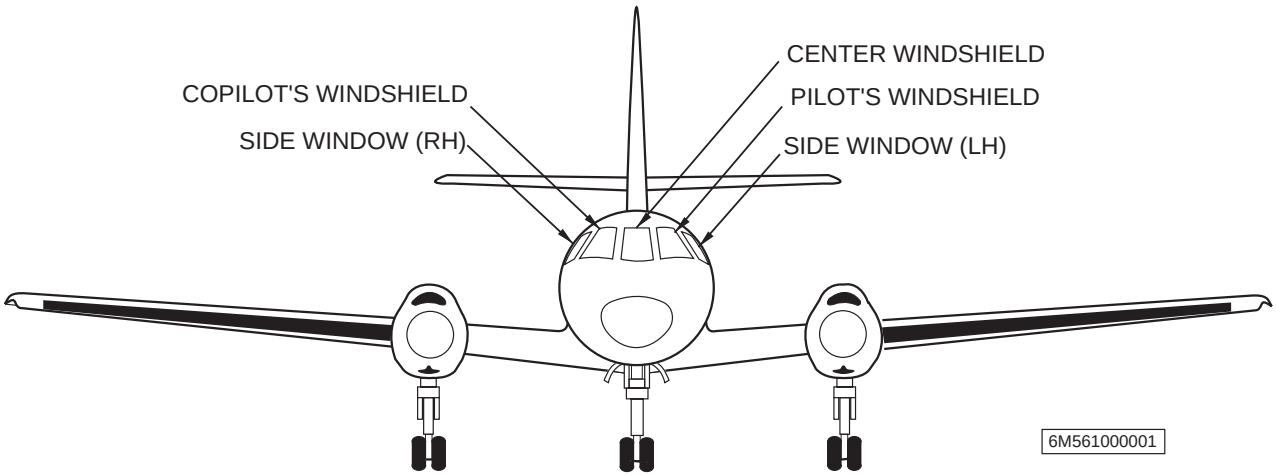
COCKPIT COMPARTMENT WINDSHIELDS/WINDOWS – DESCRIPTION AND OPERATION

1. General

A. The cockpit compartment windows consist of individual windshields for the pilot and copilot, a center windshield panel, and two side windows.

B. A window purge system, preventing moisture and/or condensation from forming between the side window outer and inner panels is installed on some aircraft.



Cockpit Windshields/Windows
Figure 1

2. Heated Windshield

A. Separate windshields are provided for the pilot and copilot. The windshields are constructed of two panes of tempered glass, laminated with a vinyl interlayer and surrounded by an aluminum frame.

B. The outside of the outer pane is Nesa coated (tin oxide film coating) and grounded to dissipate electro-static charges. A metal insert, bonded in the lower outboard corner of each windshield, provides connections for electrical wiring from the windshield deicing system.

C. Replacement windshields have dual temperature sensors.

(1) If one sensor should fail, disconnect connector at S1 and connect to S2 at lower inboard corner of windshield IAW SA226 Wiring Manual.

(2) If both sensors fail, windshield must be replaced.

3. Center Windshield

A. The center windshield consists of an inner and outer acrylic panel on SA226 – T and SA226 – AT aircraft, and a single acrylic panel on SA226 – TC aircraft. The dual panel center windshield is optional for the SA226 – TC aircraft.

4. Cockpit Side Window

A. The side window consists of an inner and outer acrylic panel on SA226 – T and SA226 – AT aircraft. The inner panel is installed to provide a quieter cabin. The outer panel is sealed and stressed to retain cabin pressure.

B. SA226 – TC aircraft has a single acrylic panel with a dual panel side window for an optional installation.

5. Window Defogging

A. Cockpit window defogging is provided by a desiccant system, or a window purge system.

(1) Desiccant System.

(a) A small copper tube is installed between the window panels with one end of the tube open to the air trapped between the panels. A plastic bag filled with desiccant material (silica gel) is attached to the opposite end of the tube. These bags are located behind the interior panel covering the respective window.

(2) Window Purge System

(a) SA226 – T and SA226 – AT aircraft are supplied with a window purge system. SA226 – TC aircraft has a window purge system as an optional installation.

(b) Bleed air is extracted from the door seal inflation plumbing and is filtered through water separator. After moisture has been extracted, a portion of the bleed air is routed to the cockpit window purge system. The accumulated water is dumped overboard, through fuselage drains, when the cabin is unpressurized.

6. Definition of Terms – Cockpit Side Window (Figure 2)

A. Aft Frame Retainer – Attached to the aft portion of the aircraft side window frame.

B. Lower Frame Retainer – Attached to the lower window sill of the aircraft window frame.

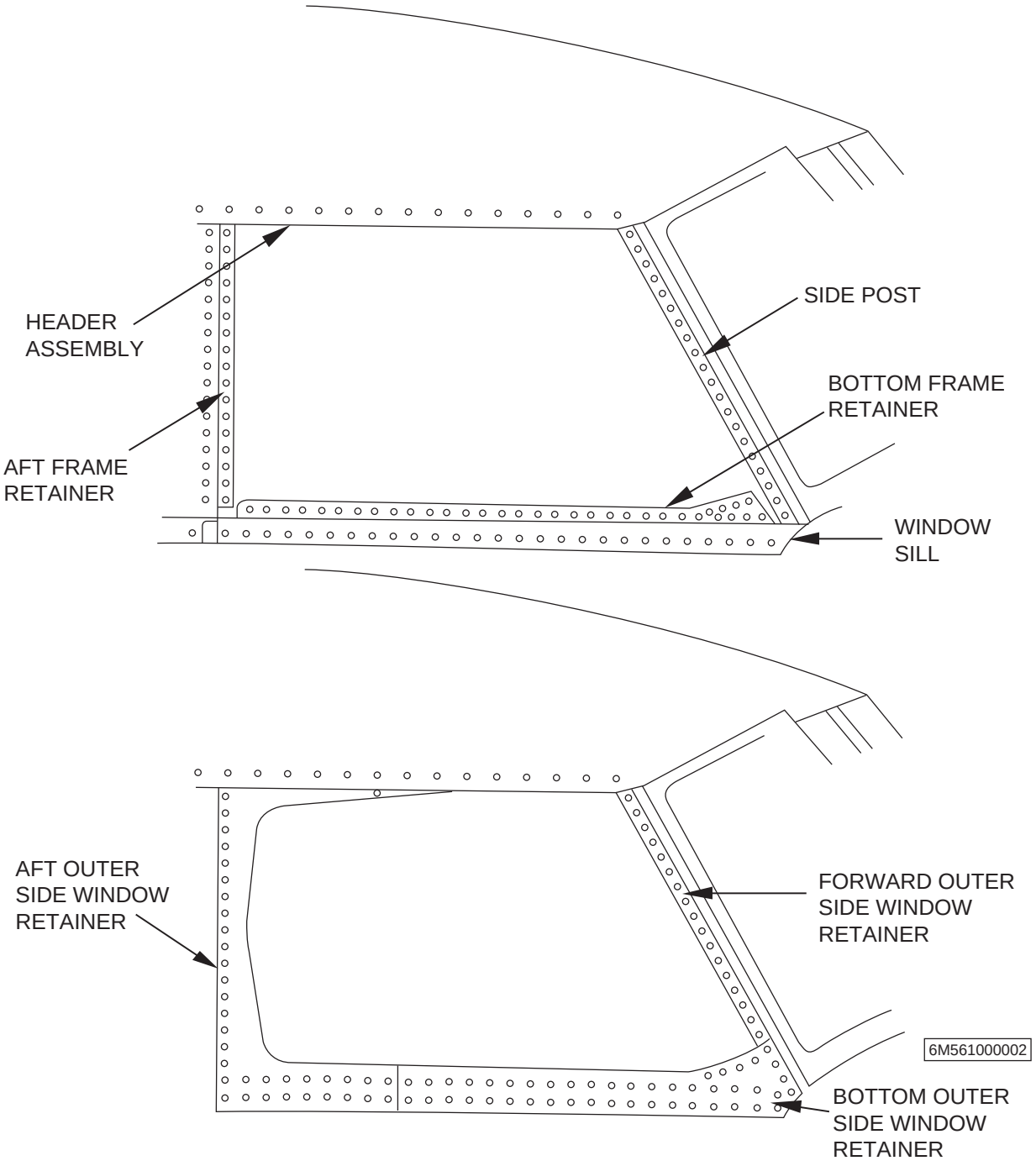
C. Side Post – That portion of the aircraft window frame assembly where the side window and heated glass side window join for installation.

D. Header Assembly – That portion of the aircraft window frame at the top, beneath the canopy skin, consisting of two angle brackets.

E. Aft Outer Side Window Retainer – The outer retainer of the side window covering half of the outer areas of the aft portion of the window.

F. Lower Outer Side Window Retainer – The outer retainer of the side window covering the lower forward portion of the window.

G. Forward Outer Side Window Retainer – The outer retainer of the side window covering the front side portion of the window.



Retainers
Figure 2

COCKPIT COMPARTMENT WINDSHIELDS/WINDOWS – MAINTENANCE PRACTICES

1. Removal/Installation – Heated Windshield

A. Removal – Heated Windshield

(1) Remove instrument panel glareshield.

(2) Remove interior panels and covers.

(3) Remove windshield mounting screws.

(4) Remove panel and cover clips.

NOTE: Mark location of clips for reference during windshield installation.

(5) Remove inner fasteners.

(6) Remove wiring from windshield.

(7) Remove windshield by pushing panel outward.

(8) If windshield sealant does not release after pushing panel outward, proceed as follows:

(a) Cut through sealant around outer edges of windshield with sharp-bladed instrument taking care not to damage the windshield or aircraft structure.

(b) Spray Ram Chemicals Aerosol Mold Release 225 (or equivalent) into crevices made from cutting around outer edges of windshield.

WARNING: WEAR EYE PROTECTION AND GLOVES, WHEN USING SAFETY WIRE TO CUT WINDSHIELD SEALANT.

(9) If windshield sealant will not release after cutting and spray application of mold release agent, proceed as follows:

(a) Dig a small hole in sealer and push through a length of 0.020 inch (0.51 mm) thick stainless steel safety wire.

(b) Wrap ends of wire around a screwdriver or similar tool.

(c) With one person inside and another outside, pull wire in a reciprocating motion through sealer until windshield is free. Remove windshield.

(10) Vacuum or brush clean all airplane structure surfaces to be sealed to remove sealant, dust, loose debris, and other surface contaminants. Wipe surfaces with a clean lint free cloth moistened with isopropyl alcohol or aliphatic naptha solvent to remove all grease, oil and other surface contaminants. Do not allow solvent to dry before wiping surfaces with a clean, dry, lint free cloth.

(11) If windshield is to be re-installed, clean all sealant, dust, and loose debris from the mating surfaces of the windshield. Wipe surfaces with a clean lint free cloth moistened with isopropyl alcohol or aliphatic naptha solvent to remove all grease, oil and other surface contaminants. Do not allow solvent to dry before wiping surfaces with a clean, dry, lint free cloth.

B. Installation – Heated Windshield

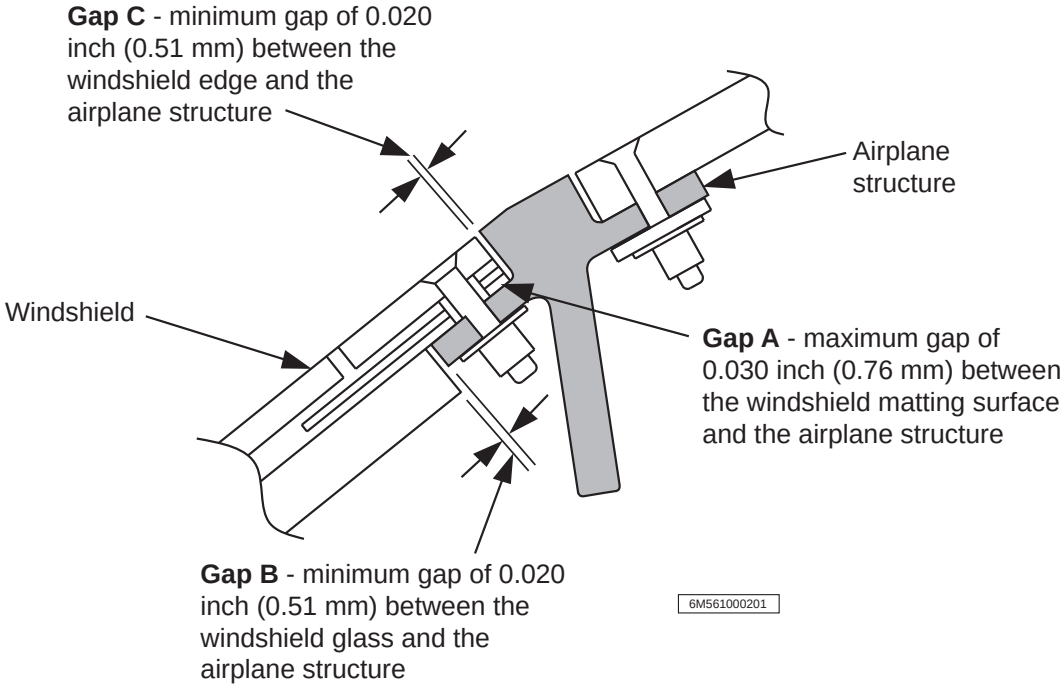
NOTE: The following procedures apply to all heated window installations, using either reusable existing heated windshields or new heated windshield panel with required drilling, trimming, counterboring, countersinking, and deburring completed IAW paragraphs **1.**, E. through H., this section.

(1) Inspect heated windshield panel for distortion and defects; mask windshield for protection.

(2) Inspect the airplane structure to which the windshield will be attached. Vacuum or brush clean all airplane structure surfaces to be sealed. Remove sealant, dust, loose debris, and other surface contaminants. Wipe surfaces with a clean, lint free cloth moistened with isopropyl alcohol or aliphatic naptha solvent to remove all grease, oil, and other surface contaminants. Do not allow solvent to dry before wiping surfaces with a clean, dry, lint free cloth.

(3) Wipe surfaces of windshield mating surfaces with a clean, lint free cloth moistened with isopropyl alcohol or aliphatic naptha solvent to remove all grease, oil, and other surface contaminants. Do not allow solvent to dry before wiping surfaces with a clean, dry, lint free cloth.

(4) Dry install the windshield onto the airplane structure to check flatness of windshield support structure. Maintain a maximum clearance of 0.030 in (0.76 mm) between the windshield assembly mating surface and the airplane structure, identified as Gap A in Figure 201. If Gap A exceeds 0.030 in (76 mm), aluminum shims may be bonded to the frame as required with EC1838 structural adhesive to meet the maximum allowable gap criteria.



Heated Windshield Critical Gaps
Figure 201

(5) From inside the airplane, inspect the gap between the edge of the windshield inner glass ply and the airplane structure identified as Gap B in Figure 201. Ensure a minimum value at Gap B of 0.020 in (0.51 mm) exists all around the windshield glass edge.

(6) From the outside of the airplane, inspect the gap between the edge of the windshield assembly and the airplane structure identified as Gap C in Figure 201. Ensure a minimum value at Gap C of 0.020 in (0.51 mm) exists all around the windshield edge.

(7) Remove the windshield from the airplane structure.

(8) Ensure that all windshield bushings are installed so that they are either flush or slightly recessed and held in place with sealant.

(9) Select four holes on the windshield to provide an electrical ground path to the airframe structure, one from the upper half and one from the lower half on each inboard and outboard post. Clean all sealant, coatings, etc. from the inner surface of selected holes on the windshield retainer. Apply electrically conductive chemical conversion coat conforming to MIL-DTL-81706 per MIL-DTL-5541, Type 1, Class 3.

(10) Identify corresponding holes selected in step (9) on the airplane structure. Clean all sealant, coatings, etc. from the inner surface of selected holes on the post and nutplate retainer. Apply electrically conductive chemical conversion coat conforming to MIL-DTL-81706 per MIL-DTL-5541, Type 1, Class 3.

(11) Identify these four locations on the inner and outer surface of the windshield.

(12) Apply a thin coat of Ram Chemicals Aerosol Mold Release No. 225 to the airplane structure faying surfaces.

(13) Mix an adequate amount of PR-1425, Class B (or approved equivalent sealant) per the mixing and handling instructions shown in the SA226/227 Structural Repair Manual (SRM) 51-30-03.

(14) Apply a thin coat of PR-1425, Class B (or approved equivalent sealant) to windshield frame faying surface.

NOTE: PR-1425, Class B sealant may also be used as an underlay material to fill a minor mismatch of heated windshield with surrounding windshield frame.

(15) Apply approximately 0.188 inch (4.775 mm) thickness of PR-1425, Class B sealant (or equivalent) to each corner of windshield frame.

(16) Position windshield into aircraft mounting structure.

(17) Apply a small amount of PR-1425, Class B sealant (or equivalent) around each screw shank prior to installing into screw holes except for the four locations identified in steps (9) through (11). DO NOT apply sealant to the screw shank at these locations.

(18) Measure the torque required to overcome the friction lock feature on the windshield installation fasteners. Install all 89 windshield installation screws with this measured torque plus 4.5 in-lbs (0.51 N-m). Allow sealant to cure to tack free state. Refer to PR-1425 product data sheet or see SA226/227 Structural Repair Manual (SRM) 51-30-03, for cure times.

NOTE: Do not exceed the specified torque values during initial tightening.

NOTE: Refer to Figure 201A for screw length requirements.

(19) Stage-torque screws 15 to 20 inch-pounds (1.69 to 2.26 N·m), using a cross sequence tightening schedule (e.g., 12 o'clock to 6 o'clock, 3 o'clock to 9 o'clock, 12:05 o'clock to 6:05 o'clock, etc.).

(20) Install interior panel and cover clips.

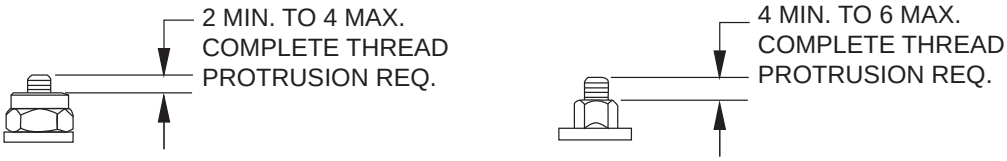
(21) Install all inner retainers.

(22) Tighten inner retainer screws hand tight, then loosen one-quarter (1/4) turn, using a cross sequence tightening schedule.

PERMISSIBLE TO VARY SCREW LENGTH UP OR DOWN ONE (1) SIZE TO ACCOMMODATE VARIANCES IN EITHER THE STRUCTURE OR WINDOW THICKNESS. WHEN INSTALLING HARDWARE ENSURE THE FOLLOWING CRITERIA ARE MET:

a. THE SCREW BEING INSTALLED IS OF SUFFICIENT LENGTH TO ENSURE NO PORTION OF THE THREADS REMAIN BELOW THE SURFACE OF THE STRUCTURE ONCE INSTALLATION IS FINAL.

b. AFTER FINAL TORQUE OF THE FASTENER, THREAD PROTRUSION BEYOND THE NUT SHALL MEET REQUIREMENTS SHOWN IN ILLUSTRATIONS BELOW.



STANDARD HEIGHT
- FIBER LOCKING NUT (MS20365 SERIES, 3/8 WRENCH SIZE) OR,
- FIBER LOCKING NUTPLATE

REDUCED HEIGHT
- STEEL LOCKING NUT AND (MS21042 SERIES, 3/16 OR 1/4 WRENCH SIZE) OR,
- STEEL LOCKING NUTPLATE

NOTE: FOR INSTALLATION UTILIZING A NUT AND WASHER COMBO, A MINIMUM 1 EA 0.016 INCH (0.406 MM) THICK WASHER IS REQUIRED BENEATH THE NUT. IT IS PERMISSIBLE TO USE UP TO TWO (2) WASHERS IN ANY COMBINATION OF THICK OR THIN BENEATH THE NUT TO ACHIEVE THE REQUIRED THREAD PROTRUSION.

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Windshields / Windows - Screw Length Requirements
Figure 201A

(23) Install wiring to electro-thermal windshield per the SA226 Wiring Manual, Chapter 30.

(24) Install heated windshield panels and covers.

(25) Fill spaces between windshield and aircraft primary structure with sealant.

(26) After sealant has cured, remove all masking tape and clean windshield. Install glareshield.

(27) Perform pressurization check IAW Adjustment/Test - Pressurization, Chapter 21, of this manual.

(28) Perform operational test, high and low, of windshield heat system IAW Chapter 30-40-10, Adjustment/Test - Windshield De-ice, of this manual.

C. Required Equipment

(1) Hand drill, high-speed, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed.

(3) High-speed steel drill bit, 0.1935 inch.

(4) Counterbore bit, 0.313 inch with 0.031 inch radius and 0.1935 inch pilot.

(5) Countersink bit, 0.1935 inch.

(6) Torque wrench.

(7) Steel rule.

(8) Rasp.

(9) Sheet metal Cleco fasteners or draw mono-locks.

(10) China marker.

(11) Spacers, aluminum.

(12) Sealant, PR-1425, Class B, or the following approved equivalent:

(a) PR-1422, Class B.

D. Preparation

(1) Inspect new windshield for general condition. Mask the glass portions of the windshield for protection.

E. Drilling and Trimming – Heated Windshield

(1) Inspect the aircraft structure to which the windshield will be attached. Clean all contaminants including sealant and sealant residue from the airplane structure.

(2) Dry install the windshield onto the airplane structure.

NOTE: Approximately 0.040 inch (1.02 mm) space should be allowed for sealing between the outer edges of the new windshield frame and the aircraft primary structure.

(3) Trim the frame of the new windshield assembly as required with rasp to allow windshield to be positioned on the airplane structure while maintaining a minimum gap of 0.020 inch (0.51 mm) between the outer frame of the windshield assembly and the airplane structure (Gap C, Figure 201). The minimum amount of windshield edge trimming required should be done so that fastener hole minimum edge distance criteria can be maintained. From inside the airplane, ensure a minimum gap of 0.020 inch exists between the edge of the windshield inner glass ply and the airplane structure (Gap B, Figure 201). **DO NOT CONTACT WINDSHIELD GLASS WITH RASP.**

(4) With new windshield held firmly in place, and using the aircraft primary structure retainer as reference, drill three (3) 0.190 / 0.194 in. (4.83 / 4.93 mm) holes, one (1) hole at top and one (1) at each side of windshield frame. The minimum edge distance on the hole (distance from the center of the hole to the edge of the windshield) is to be 0.33 inch (8.38 mm).

NOTE: Drill holes from cockpit side of frame.

(5) Remove windshield and place it on a flat work area with inner side down. Place old windshield, inner side down, on top of new windshield.

(6) Align old windshield with new windshield, securing windshields together with bolts and nuts through holes in new windshield frame. Nuts and bolts should only be tightened snug. Excessive torque on bolt/nut will damage the windshield.

(7) Use old windshield as a template guide, and drill remaining 0.190 / 0.194 in. (4.83 / 4.93 mm) holes through new windshield frame while maintaining a minimum edge distance of 0.33 inch (8.38 mm).

NOTE: Drill holes on a drill press to help ensure proper alignment.

F. Counterboring – Heated Windshield

(1) Place new windshield on drill press with inner side facing up. Adjust drill press to counterbore a maximum depth of 0.192 - 0.196 inch (4.88 - 4.98 mm) into windshield assembly (see Figure 201b).

(2) Counterbore all holes with 0.313 inch counterbore bit (0.322/0.326 inch diameter counterbore (8.18/8.28 mm)) with maximum 0.031 inch (0.79 mm) fillet radius and a 0.1935 inch (4.915 mm) pilot.

G. Countersinking – Heated Windshield (Figure 202)

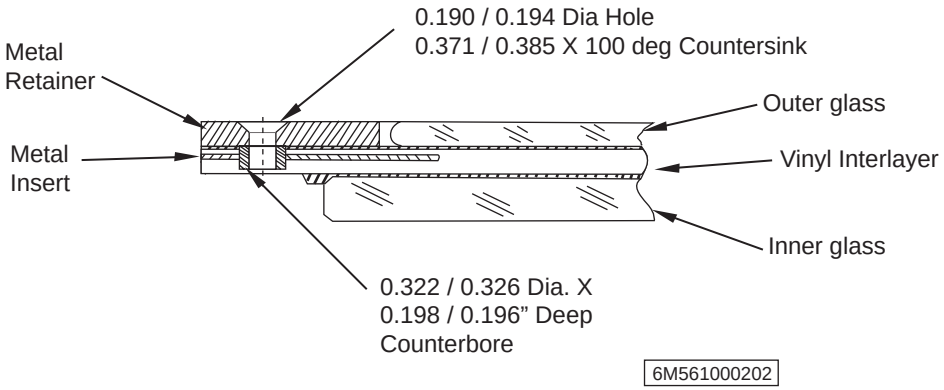
(1) Place new windshield on drill press with outer side facing up.

(2) Countersink all holes 0.371/0.385 inch (9.42/9.78 mm) diameter by 100 degrees.

NOTE: Countersink must have an included angle of 100° and produce the degree of flushness required with retaining screws installed.

H. Deburring

(1) Remove chips and burrs on both sides of the windshield frame.



Countersinking Heated Windshield
Figure 202

I. Installing Spacers

(1) Apply a small amount of PR-1425 sealant around the outside circumference of each spacer and position the spacers in the counterbored holes. Ensure spacers are installed either flush or slightly recessed.

J. Cleaning

(1) Remove masking tape.

(2) Remove dirt and other substances from glass surface with clean water.

(3) Wash with mild soap and water or a 50/50 solution of isopropanol and water. Do not use any abrasive materials or any strong acids or bases. When wiping the surface, use a soft clean cloth or sponge and use a straight rubbing motion.

(4) Rinse thoroughly and dry; do not apply wax.

2. Removal/Installation – Center Windshield (Outer Panel)

A. Removal – Center Windshield (Outer Panel)

(1) Remove instrument panel glareshield.

(2) Remove interior paneling as required.

(3) Remove center windshield retainer screws and retainer.

(4) Remove panel and cover clips.

NOTE: Mark location of clips for future reference during windshield installation.

(5) Remove sealant residue from window frame and retainer.

NOTE: It may be necessary to cut through the sealant around outer edges of windshield with a sharp bladed instrument in order to help loosen the sealant.

WARNING: WEAR EYE PROTECTION AND GLOVES, WHEN USING SAFETY WIRE TO CUT WINDSHIELD SEALANT.

(6) If windshield sealant will not release after cutting, proceed as follows:

(a) Dig a small hole in sealant and push through a length of 0.020 inch thick stainless steel safety wire.

(b) Wrap ends of wire around a screwdriver or similar tool.

(c) With one person inside and another outside, pull wire in a reciprocating motion through sealant until windshield is free. Remove windshield.

(7) Remove inner windshield panel, if applicable, IAW para **3.,** A., Removal – Center Windshield (Inner Panel).

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(8) Clean window frame with methyl propyl ketone (MPK). Remove all sealant residue.

(9) Remove window purge tubing, if applicable, and stow until required for installation.

B. Required Equipment

(1) Power plane, Rockwell model 167A or equivalent.

(2) Router, Rockwell model 1002 with model 1001 router base or equivalent.

(3) Hand drill motor, high speed, 15,000-25,000 RPM.

(4) Drill press, automatic constant feed.

(5) Band saw, with 14 teeth per inch blade.

(6) High-speed steel drill bits, 0.166 inch, 0.113 inch, and 0.098 inch for drilling acrylic.

(7) Torque wrench.

(8) Countersink bit, 0.156 inch.

(9) Counterbore bit 0.250 inch.

(10) Steel rule.

(11) Rasp.

(12) Sheet metal Cleco fasteners or draw mono-locks.

(13) China marker.

(14) Poly-Flow tubing, 44-P-1/4 (0.250 x 0.312).

(15) Sealant, MIL-S-8802.

C. Preparation – Old Windshield Removed Intact

(1) Inspect new windshield for distortion and general condition. Mask for protection.

D. Making Template

NOTE: A template must be custom made to fit the inside contour of the old panel so that existing hole locations may be adequately transferred to the inside contour of the new panel.

(1) Place 0.032 inch thick sheet metal on inside of old windshield panel for forming of template.

(2) Form template to fit contour of outer windshield panel and secure with clamps.

(3) Scribe a trim line around edge of old windshield panel onto template.

(4) Drill new holes in template with a 0.166 inch high-speed steel drill bit, using old windshield as a guide.

(5) Remove template and trim excess material following scribe line established in step **2.**, D., (3).

E. Drilling

(1) Place template on inside of new outer windshield panel.

(2) Clamp template to outer windshield panel.

(3) Using template as a guide, hold drill perpendicular to panel surface and drill holes in new windshield from inner side of new panel with a 0.166 inch high-speed steel drill bit, using the following sequence.

(a) Slowly drill one (1) hole at 3 o'clock position and secure with Cleco.

(b) Drill one (1) hole at 9 o'clock position and secure with Cleco.

(c) Drill one (1) hole at 12 o'clock position and secure with Cleco.

(d) Drill one (1) hole at 6 o'clock position and secure with Cleco.

(e) Drill remaining holes applying no feed pressure on drill motor. Allow drill to lightly feed itself through panel.

(4) Enlarge holes to 0.250 inch (6.350 mm) using counterbore with appropriate pilot. Again start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) of material thickness.

(5) Turn window panel over and finish counterboring holes.

(6) Use sandpaper #400 or countersink to remove any ridges around hole edges.

(7) Using trimmed template as a guide, scribe a line around edge of template onto new panel.

F. Trimming

(1) Remove template and trim excess material following scribe line. Use band saw with 14 teeth per inch blade.

NOTE: Where only minor trimming is required, a power plane or rasp is recommended.

(2) Smooth trimmed edges by sanding.

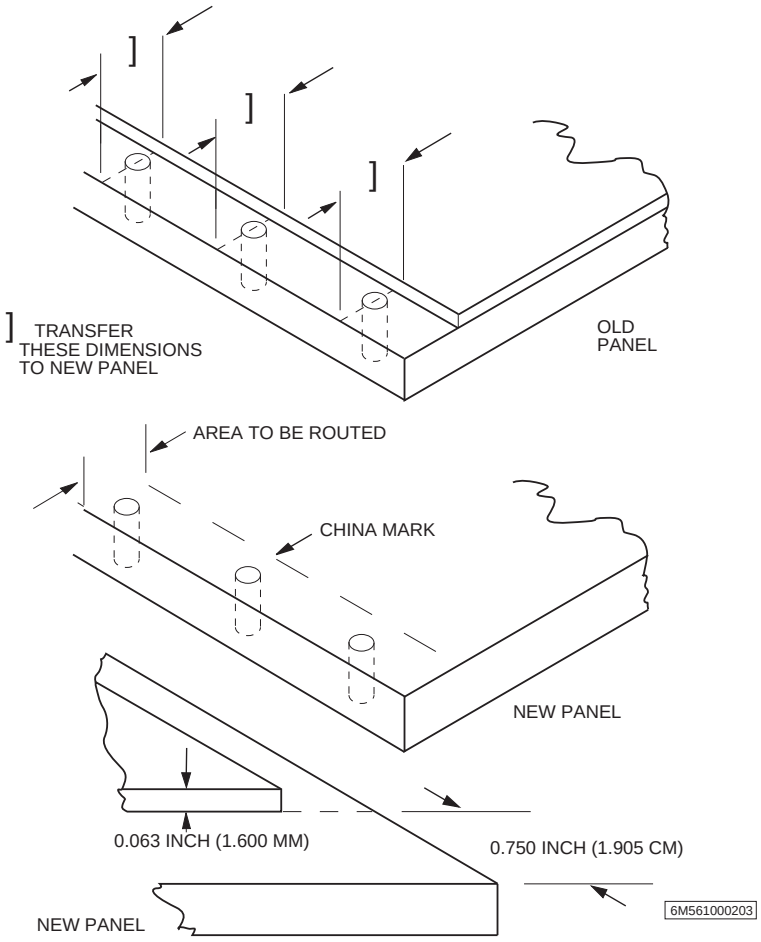
G. Routing – Upper Edge

(1) Determine width of old panel routed area as follows:

(a) Measure from upper edge, across each hole, to edge of routed area.

(b) Transfer width dimensions onto new panel with china marker.

(c) After transferring dimensions, draw a continuous line across new panel (see Figure 203).



Routing
Figure 203

(2) Adjust router to cut 0.125 inches (3.175 mm) deep.

CAUTION: DO NOT EXCEED ROUTING DEPTH OF 0.125 IN (3.175 MM) ON UPPER EDGE.

(3) Position new panel on a flat surface. Rout out marked area by starting at one end, working in one direction to the opposite end.

(4) Check depth of routed area.

H. Routing Sides and Lower Edge

CAUTION: DO NOT EXCEED ROUTING DEPTH OF 0.063 IN (1.600 MM) OR A WIDTH OF 0.750 IN (1.90 CM) ON SIDES.

(1) Set router depth to 0.063 inch (1.600 mm).

(2) Start routing from top of one side, working in one direction toward bottom edge in one continuous motion.

(3) Ensure depth and width of routed area on first of two sides is correct (see Figure 203).

(4) Repeat steps **2.**, H., (1) and (2) for opposite side.

(5) Repeat steps **2.**, H., (1) and (2) for lower edge.

I. Deburring

(1) Remove chips and burrs from holes on inner and outer side of center windshield panel.

J. Installing Bushing

(1) Install bushings in all 0.250 in (6.350 mm) holes using Imperial Eastman Poly-Flow tubing, 44-P-1/4 (0.250 x 0.312), for bushing material. Trim bushings flush with each side of holes.

K. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask holes on each side to prevent dried polish from falling between panels after installation and to keep sealant contaminant free.

(3) For hairline scratches, use polysand procedure IAW Polishing Procedure, this Section.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax with fingertips in a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect for imperfections. If any damage is noted, repeat necessary steps.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

L. Preparation – Old Windshield Not Intact

NOTE: Follow these procedures if the windshield panel is damaged to the extent that it cannot be used as a template for drilling and trimming.

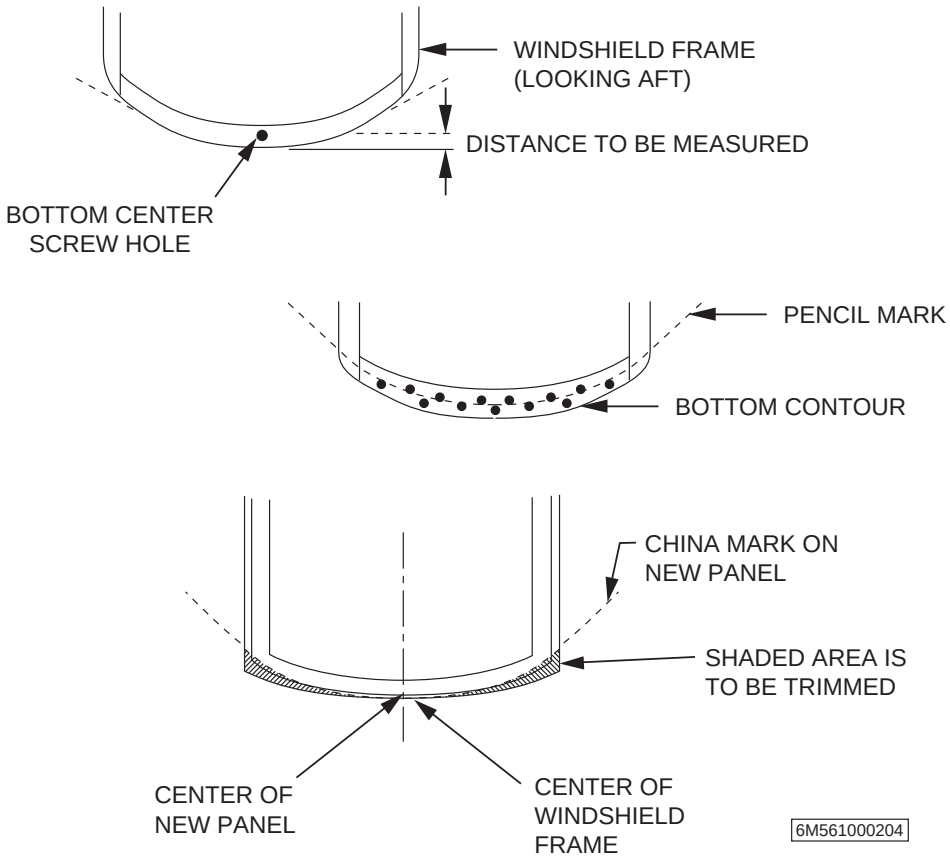
(1) Inspect new windshield for nicks, scratches, and general condition. Mask for protection.

M. Trimming – Lower Edge (Figure 204)

(1) Measure distance (Dimension A) between bottom center of aircraft frame and midpoint of the center screw hole.

(2) Use a caliper or similar tool to maintain Dimension A while drawing a line on windshield frame, following lower edge contour.

(3) Position new windshield in frame with bottom center aligned with line established in step **2.,** M., (2).



Trimming Bottom
Figure 204

(4) Center windshield in aircraft frame. Allow approximately 0.063 inch (1.600 mm) space on both sides of windshield.

(5) Determine amount of lower edge trimming required.

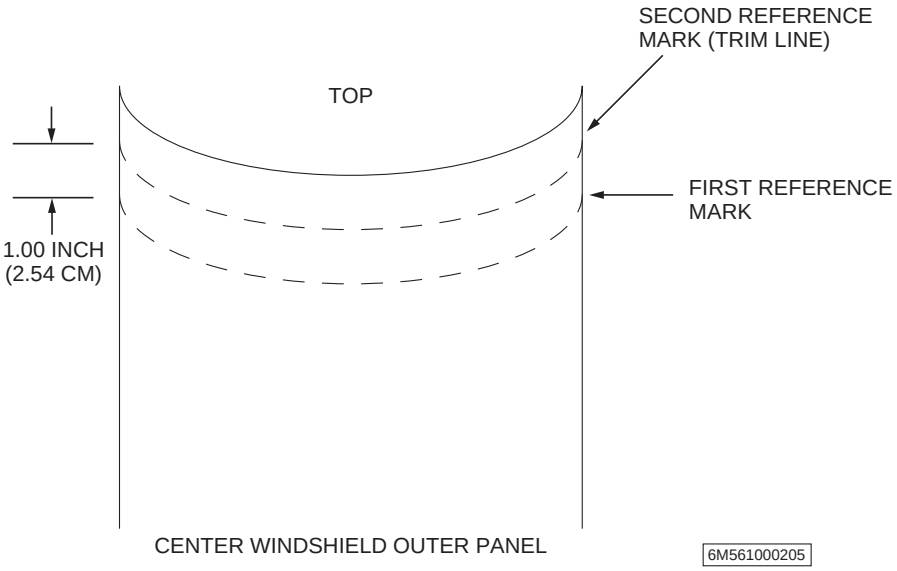
(6) Following line established in step **2.,** M., (2), scribe windshield with a china marker on lower portion of windshield frame. Area below china marker line is trim area.

(7) Trim windshield along china marker line with 14 teeth per inch blade band saw.

(8) Position windshield in aircraft frame and check contour.

(9) Repeat marking and trimming as necessary until lower edge of windshield fits contour of lower edge of aircraft frame.

(10) Smooth edges by sanding.



Trimming and Routing
Figure 205

N. Trimming – Sides

(1) Allow a minimum gap of 0.063 inch (1.600 mm) between windshield and aircraft frame on each side.

(2) Trim sides of windshield as required with a power plane or rasp. Use a rasp for minor trimming.

O. Trimming – Upper Edge

(1) Position windshield in frame, allowing upper edge to rest outside canopy.

(2) From inside cockpit area, make a reference mark with china marker. Mark top, inside of windshield to correspond with top of canopy frame.

(3) Make a second reference mark 1.00 inch (2.54 cm) above first reference mark (see Figure 205). Area between first and second reference marks is the area to be routed out.

NOTE: Routing is necessary to permit new panel to fit between the top of the aircraft frame and the canopy. The area above the top (second) reference mark is the area to be trimmed.

(4) Trim excess material from area above second reference mark with 14 teeth per inch blade band saw.

P. Routing – Top

(1) Adjust router to cut 0.125 inch (3.175 mm) deep.

(2) Remove windshield and place on a suitable work bench or flat surface.

(3) Rout area between trimmed edge and reference mark by starting at one end and work in one direction to opposite end of panel.

(4) Check depth of routed area. Initial routing depth should not exceed 0.125 inch (3.175 mm).

(5) Place panel in aircraft frame. Check for fit, contour, and space.

(6) If further routing is required, remove windshield and perform the following steps:

(a) Mark area requiring additional routing using outside top of canopy frame as a reference.

(b) Make additional mark on outside of windshield panel with china marker.

(c) Remove panel and rout as required.

(d) Smooth edges by sanding.

(e) Position windshield in frame and check for fit.

(7) Smooth all edges of new windshield by sanding.

Q. Drilling

NOTE: Screw holes should be drilled (using high speed drill) prior to routing sides and bottom to facilitate viewing during drilling.

(1) Position outer panel in windshield frame. Align to fit, contour and space. Hold outer panel firmly in place.

(2) Drill first hole at the bottom of windshield frame.

(3) Viewing through outside of windshield panel, drill a 0.166 inch hole to align with hole in bottom of aircraft windshield frame.

CAUTION: USE A DRILL STOP TO AVOID DAMAGING WINDSHIELD FRAME NUTPLATES. DRILL HOLES PERPENDICULAR TO EXISTING CONTOURS TO FACILITATE SCREW HEAD FLUSHNESS.

(4) Drilling perpendicular to panel surface, drill holes in new windshield with a 0.166 inch high-speed steel drill bit, using the following sequence.

(a) Slowly drill one (1) hole at 3 o'clock position and secure with Cleco.

(b) Drill one (1) hole at 9 o'clock position and secure with Cleco.

(c) Drill one (1) hole at 12 o'clock position and secure with Cleco.

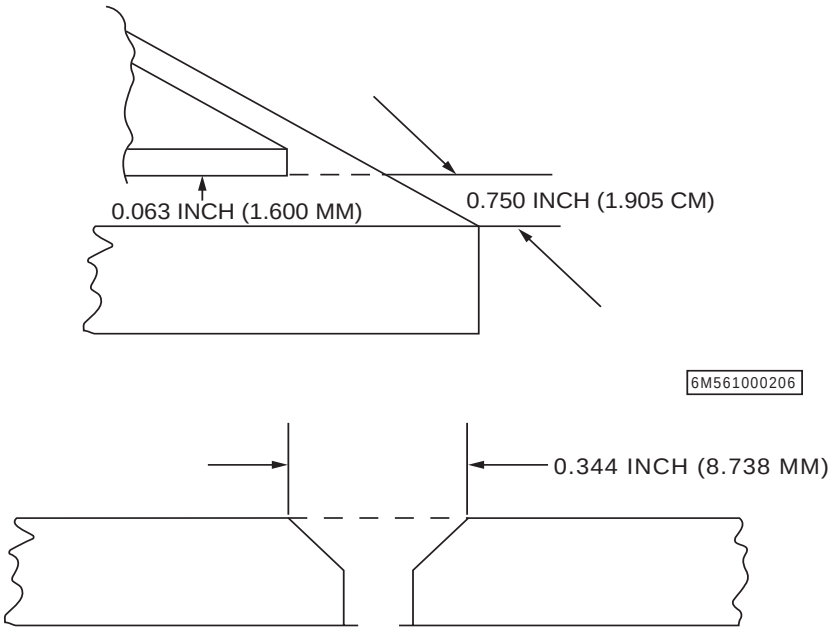
(d) Drill one (1) hole at 6 o'clock position and secure with Cleco.

(e) Drill remaining holes applying no feed pressure on drill motor. Allow drill to lightly feed itself through panel.

(5) Enlarge holes to 0.250 inch (6.350 mm) using counterbore with appropriate pilot. Again start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) of material thickness.

(6) Turn window panel over and finish counterboring holes.

(7) Use sandpaper #400 or countersink to remove any ridges around hole edges.



Routing and Counterboring
Figure 206

R. Routing – Sides and Bottom

CAUTION: DO NOT EXCEED ROUTING DEPTH OF 0.063 IN (1.600 MM) OR WIDTH OF 0.750 INCH (1.905 CM).

(1) Set router to cut a maximum depth of 0.063 in (1.600 mm) and a maximum width of 0.750 inch (1.905 cm).

(2) Start work from top of one side of panel. Work in one direction in long, continuous motion.

(3) Ensure depth and width of routed area on first of two sides is correct (see Figure 203).

(4) Repeat steps **2.**, R., (2) and (3) for opposite side.

(5) Repeat steps **2.**, R., (2) and (3) for lower edge.

S. Deburring

(1) Remove chips and burrs from holes on inner and outside of center windshield panel.

T. Installing Bushings

(1) Install bushings in all 0.250 inch (6.350 mm) holes, using Imperial Eastman Poly-Flow tubing, 44-P-1/4 (0.250 x 0.312), for bushing material. Trim bushings flush with each side of the hole.

U. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent dried polish from falling between panels.

(3) For hairline scratches, use polysand procedure IAW Polishing Procedure, this Section.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax with fingertips in a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect for imperfections.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

V. Installation – Center Windshield (Outer Panel)

NOTE: Before installing the new center windshield outer panel, inspect for distortion and general condition.

(1) Install window purge tubing, if applicable.

(2) Install inner windshield, if applicable, IAW **3.**, T., this chapter.

(3) Apply thin coat of sealant to center windshield frame of aircraft.

(4) Place windshield in frame and apply thin coat of sealant around outer edge of panel on routed area.

(5) Install panel and cover clips.

(6) Place nutplate strips/nuts in place on inner side of windshield frame.

(7) Place outer retainer in place around windshield panel.

(8) Align screw holes in outer retainer with windshield and nut/nutplate holes.

(9) Secure with Cleco fasteners, draw mono-locks, or screws.

(10) Apply small amount of sealant around each screw prior to installing.

(11) Install screws and tighten just enough to let sealant extrude from around screws. Let sealant cure for twenty four (24) hours.

NOTE: Do not exceed the specified torque values during initial tightening.

NOTE: Refer to Figure 201A for screw length requirements.

(12) After sealant has cured, stage torque screws to a maximum of 15 inch-pounds (1.69

N·m) in a diagonal sequential tightening schedule such as; 12 o'clock to 6 o'clock, 3 to 9, 12:05 to 6:05, 3:05 to 9:05, etc.

(13) Repeating stage torque sequence, torque screws 15 to 25 inch-pounds (1.69 to 2.82 N·m).

(14) Apply sealant to outside fill spaces between retainer, outer panel, and aircraft frame.

(15) After sealant cures, remove all masking, and clean windshield.

(16) Install interior panel and cover.

(17) Install instrument panel glareshield.

(18) Perform pressurization check IAW Adjustment/Test - Pressurization, Chapter 21, of this manual.

3. Removal/Installation – Center Windshield (Inner Panel)

A. Removal – Center Windshield (Inner Panel) Intact

(1) Remove center windshield outer panel IAW para **2.**, A., Removal – Center Windshield (Outer Panel), this section.

(2) Remove instrument panel glareshield.

(3) Remove window purge tubing, if applicable, and stow until required for installation.

(4) Remove all screws, washers, and nuts from bottom of old inner panel securing it to bottom retainer.

(5) Remove side and top retainers.

(6) Remove old center windshield inner panel.

NOTE: It may be necessary to cut through the sealant around outer edges of windshield with a sharp bladed instrument in order to help loosen the sealant.

WARNING: WEAR EYE PROTECTION AND GLOVES, WHEN USING SAFETY WIRE TO CUT WINDSHIELD SEALANT.

(7) If windshield sealant will not release after cutting, proceed as follows:

(a) Dig a small hole in sealant and push through a length of 0.020 inch thick stainless steel safety wire.

(b) Wrap ends of wire around a screwdriver or similar tool.

(c) With one person inside and another outside, pull wire in a reciprocating motion through sealant until windshield is free. Remove windshield.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(8) Clean windshield frame, retainers, and fillers with methyl propyl ketone (MPK). Remove all sealant residue.

B. Required Equipment

(1) Refer to para **2.**, B., Required Equipment, Center Windshield (Outer Panel), this section for appropriate listing.

C. Preparation – Old Windshield Removed Intact

(1) Inspect new windshield for distortion and general condition. Mask for protection.

D. Making Template

NOTE: A template must be custom made to fit the inside contour of the old panel so that existing hole locations may be adequately transferred to the inside contour of the new panel.

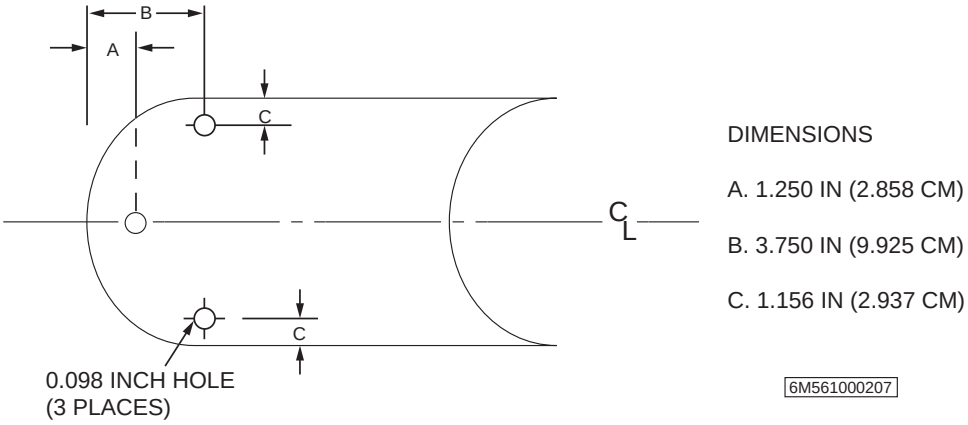
(1) Place 0.032 inch thick sheet metal template material on inside of old inner windshield panel for forming of template.

(2) Form template to fit contour of inner windshield panel and secure with clamps.

(3) Scribe a trim line around edge of old windshield panel onto template.

(4) Drill new holes in template with a 0.166 inch drill bit, using old windshield as a guide.

(5) Remove template and trim excess material following scribe line established in step **3.**, D., (3).



Purge System Drilling
Figure 207

E. Drilling

(1) Position template on inside of new inner windshield panel.

(2) Clamp template to inner windshield panel.

(3) Using template as a guide and drilling perpendicular to panel surface, drill holes in new windshield with a 0.166 inch high-speed steel drill bit, using the following sequence.

(a) Slowly drill one (1) hole at 3 o'clock position and secure with Cleco.

(b) Drill one (1) hole at 9 o'clock position and secure with Cleco.

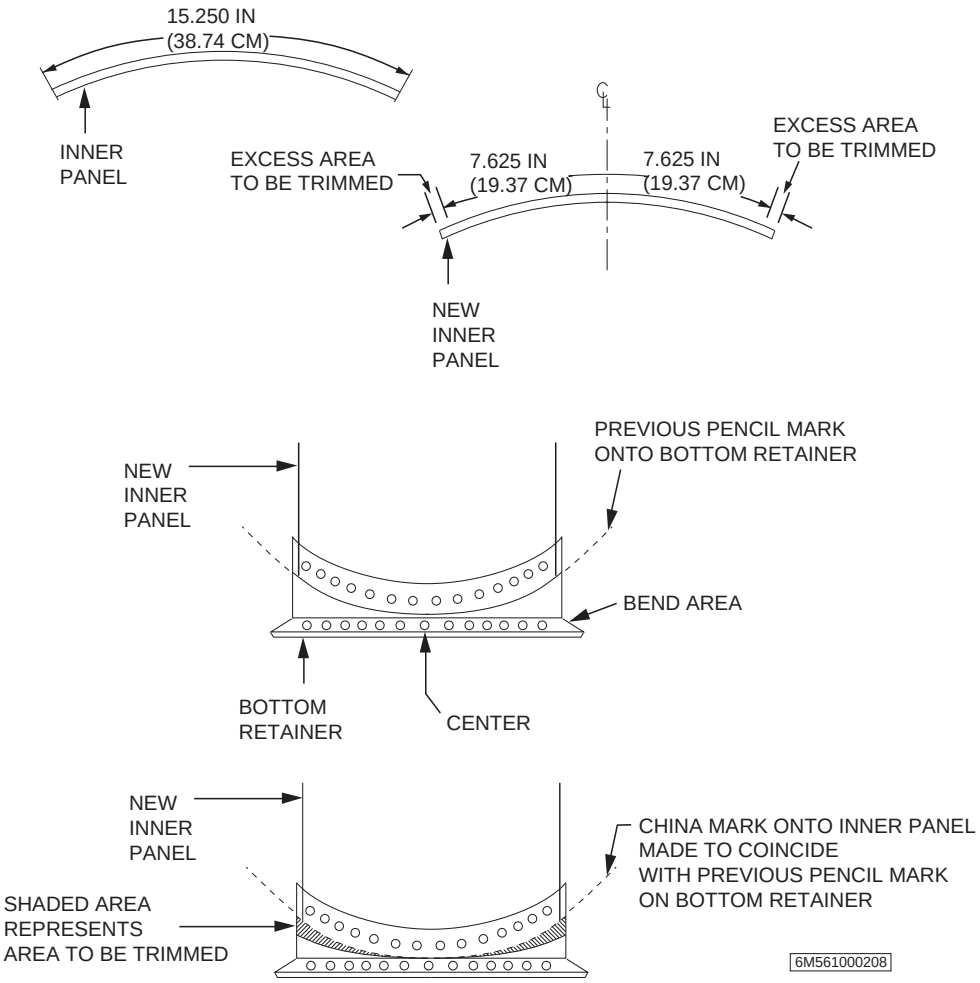
(c) Drill one (1) hole at 12 o'clock position and secure with Cleco.

(d) Drill one (1) hole at 6 o'clock position and secure with Cleco.

(e) Drill remaining holes applying no feed pressure on drill motor. Allow drill to lightly feed itself through panel.

(4) Enlarge holes to 0.250 inch (6.350 mm) using counterbore with appropriate pilot. Again start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) of material thickness.

(5) Turn window panel over and finish counterboring holes.



Preparation
Figure 208

(6) Use sandpaper #400 or countersink to remove any ridges around hole edges.

(7) Scribe a line around edge of template onto new panel, using trimmed template as a guide.

(8) On aircraft utilizing window purge system, drill three (3) 0.098 inch holes through the inner panel (see Figure 207).

F. Trimming

(1) Remove template and trim excess material following scribe line. Use band saw with 14 teeth per inch blade.

NOTE: Where only minor trimming is required, a power plane or rasp is recommended.

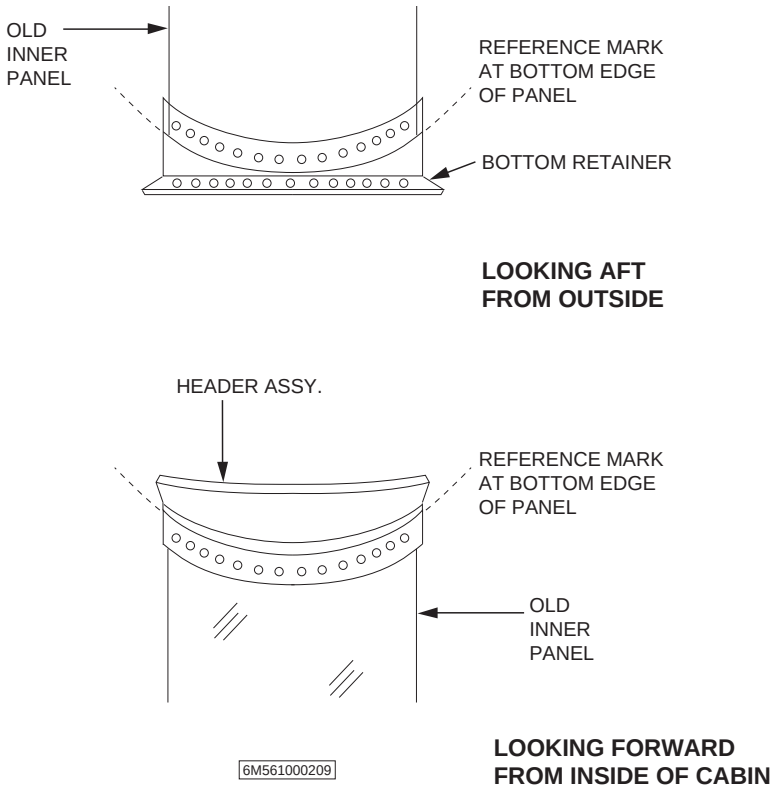
(2) Smooth trimmed edges by sanding.

G. Deburring

(1) Remove chips and burrs from holes on inner and outer side of center windshield panel.

H. Installing Bushings

(1) Install bushings in all 0.250 in (6.350 mm) holes, using Imperial Eastman Poly-Flow tubing, 44-P-1/4 (0.250 x 0.312), for bushing material. Trim bushings flush with each side of holes.



Marking Upper and Lower Edge
Figure 209

COCKPIT COMPARTMENT WINDSHIELDS/WINDOWS – MAINTENANCE PRACTICES

I. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask holes on each side to prevent dried polish from falling between panels after installation and to keep sealant contaminant free.

(3) For hairline scratches, use polysand procedures IAW, steps **1.**, G., (5) through (16), Section 56-00-00 this chapter.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax with fingertips in a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect for imperfections. If any damage is noted, repeat necessary steps.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

J. Removal – Center Windshield (Inner Panel) Not Intact

(1) Follow procedures IAW para **3.**, A., steps (1) through (8), this section.

(2) Make reference mark with pencil along bottom edge of old inner panel onto surface of bottom retainer to be used to facilitate later trimming on new inner panel.

(3) Make reference mark with pencil along bottom edge of old inner panel onto surface of aircraft windshield header assembly to facilitate later trimming on new inner panel.

K. Required Equipment

(1) Refer to para **2.**, B., Required Equipment, Center Windshield (Outer Panel), for appropriate listing.

L. Preparation – Old Windshield Not Intact

(1) Inspect new windshield for distortion and general condition. Mask for protection.

M. Trimming – Lower Edge

(1) Position new inner panel windshield frame so that the center of the lower edge of the panel rests along the "bend area" of the lower retainer (see Figure 208).

(2) Allow minimum gap of 0.063 inch (1.600 mm) between inner panel and aircraft frame.

(3) Make reference mark with china marker on new inner panel to coincide with previous mark made on bottom retainer. The area remaining below the china mark is area to be trimmed.

(4) Trim inner panel along china marker line with a 14 teeth per inch blade band saw.

(5) Smooth trimmed edges by sanding.

N. Trimming – Sides

(1) The initial width of the new inner panel should be 15.250 inches (38.74 cm) measured across the outer contour of the new inner panel (see Figure 208).

(2) Measure a distance of 7.625 inches (19.37 cm) from centerline to each side of new panel and make reference mark with china marker. Area each side of china marker line is trim area (see Figure 208).

(3) Trim panel along china marker line with a 14 teeth per inch blade band saw.

(4) Allow minimum gap of 0.063 inch (1.600 mm) between windshield and aircraft frame on each side.

(5) Smooth edges by sanding.

O. Trimming – Upper Edge

(1) Position inner panel in frame. Align bottom of inner panel within pencil mark on outer side of bottom retainer.

(2) With inner panel securely held in place and equal distance on each side, make reference mark with china marker on inside of inner panel, coinciding with pencil reference mark along top of aircraft windshield header assembly (see Figure 209).

(3) Trim top of inner panel along china marker line with a 14 teeth per inch blade band saw.

(4) Smooth edges by sanding.

P. Drilling

(1) After trimming, position inner panel in windshield frame. Align to fit, contour, and space. Hold inner panel firmly in place.

(2) Drill first hole at the bottom of windshield frame.

(3) Viewing through outside of windshield panel, drill a 0.166 inch hole to align with hole in bottom of aircraft windshield frame.

CAUTION: USE A DRILL STOP TO AVOID DAMAGING WINDSHIELD FRAME NUTPLATES. DRILL HOLES PERPENDICULAR TO EXISTING CONTOURS TO FACILITATE SCREW HEAD FLUSHNESS.

(4) Drill perpendicular to panel surface and drill holes in new windshield with a 0.166 inch high-speed steel drill bit, using the following sequence.

(a) Slowly drill one (1) hole at 3 o'clock position and secure with Cleco.

(b) Drill one (1) hole at 9 o'clock position and secure with Cleco.

(c) Drill one (1) hole at 12 o'clock position and secure with Cleco.

(d) Drill one (1) hole at 6 o'clock position and secure with Cleco.

(e) Drill remaining holes applying no feed pressure on drill motor. Allow drill to lightly feed itself through panel.

(5) Enlarge holes to 0.250 inch (6.350 mm) using counterbore with appropriate pilot. Again start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) of material thickness.

(6) Turn window panel over and finish counterboring holes.

(7) Use sandpaper #400 or countersink to remove any ridges around hole edges.

(8) On aircraft utilizing window purge system, drill three (3) 0.098 inch holes through the inner panel (see Figure 207).

Q. Deburring

(1) Remove chips and burrs from holes on inner and outer side of center windshield panel.

R. Installing Bushings

(1) Install bushings in all 0.250 inch (6.350 mm) holes using Imperial Eastman Poly-Flow tubing, 44-P-1/4 (0.250 x 0.312) for bushing material. Trim bushings flush with each side of the hole.

S. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent dried polish from falling between panels.

(3) For hairline scratches, use polysand procedure IAW steps **1.**, G., (5) through (16), Section 56-00-00 this chapter.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax with fingertips in a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect for imperfections.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

T. Installation – Center Windshield (Inner Panel)

NOTE: Before installing the new center windshield inner panel, inspect for distortion and general condition.

(1) Apply thin coat of sealant to center windshield frame of aircraft. Apply to sides, outside area of bottom retainer, and to inside area of canopy header assembly.

(2) Install top and side fillers to windshield frame.

(3) Apply thin coat of sealant to inner surfaces of top and side fillers. Install new inner panel.

(4) Apply thin coat of sealant around sides and top of new panel for retainer installation. Install side and top retainers.

(5) Install screws, washers, and nuts to bottom of new panel securing it to bottom retainer.

(6) Install window purge tubing, if applicable.

(7) Install instrument panel glareshield.

(8) Install outer windshield panel IAW para **2.**, V., Installation – Center Windshield (Outer Panel), this section.

4. Removal/Installation – Side Window (Outer Panel)

A. Removal – Side Window (Outer Panel)

(1) Remove instrument panel glareshield.

(2) Remove front window post cover.

(3) Remove optional acoustical window, if applicable.

NOTE: The optional acoustical window is force-fitted into place and secured within a rubber retainer. This window should be removed from the front of the window after removing the FWD window post cover.

(4) Remove window frame from side window.

(5) Remove retaining screws.

(6) Remove paneling clips.

NOTE: Mark location of clips for reference during installation.

(7) Remove all outer and inner retainers.

(8) If windshield sealant does not release after removing retainers, proceed as follows:

(a) Cut through sealant around outer edges of window with sharp-bladed instrument.

(b) Spray Ram Chemicals Aerosol Mold Release 225 (or equivalent) into crevices made from cutting around outer edges of window.

WARNING: WEAR EYE PROTECTION AND GLOVES, WHEN USING SAFETY WIRE TO CUT WINDOW SEALANT.

(9) If sealant does not release after cutting around outer edges of window and application of mold release agent, proceed as follows:

(a) Dig a small hole in sealant and push through a length of 0.020 inch thick stainless steel safety wire.

(b) Wrap ends of wire around a screwdriver or similar tool.

(c) With one (1) person inside and another outside, pull wire in a reciprocating motion through sealant until outer side window is free. Remove window.

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(10) Clean window frame with methyl propyl ketone (MPK). Remove all sealant residue.

B. Required Equipment

(1) Hand drill, high-speed motor, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed.

(3) Band saw, with 14 teeth per inch blade.

(4) High-speed steel drill bits, 0.166 inch, 0.113 inch, and 0.098 inch, for drilling acrylic.

(5) Torque wrench.

(6) Countersink bit, 0.156 inch.

(7) Counterbore bit, 0.250 inch.

(8) Steel rule.

(9) Rasp.

(10) Sheet metal Cleco fasteners or draw mono-locks.

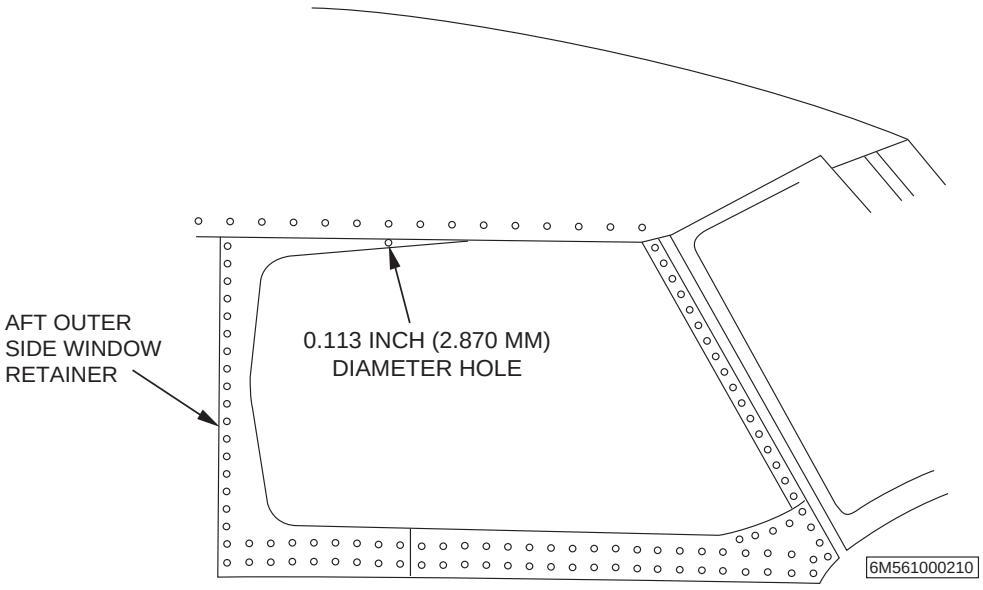
(11) China marker.

(12) Nomar Pressure Sensitive Paper (3M Company) or Permacel 01 Pressure Sensitive Paper (Permacel Tape Corp.).

(13) Poly-Flow tubing 44-P-1/4 (0.250 x 0.312).

(14) Sealant, MIL-S-8802 or the following approved equivalent:

(a) Sealant, MIL-S-81733.



Drilling
Figure 210

(15) Scratch-removing compound – Wilco No. 35, Louis P. Hershey Lustar, or other conforming to MIL-C-18767.

(16) Permatex polish No. 403D or Vista automobile wax.

C. Preparation – Old Window Intact

(1) Inspect new window for distortion and general condition; mask for protection.

D. Making Template

(1) Use masking paper as a template to transfer outside dimensions of old panel onto new panel.

NOTE: When pressure sensitive paper is not available, use pressure sensitive paper applied to new panel for protection and storage.

(2) Place pressure sensitive paper on outside area of old panel. Trim paper to match outer dimensions of old panel.

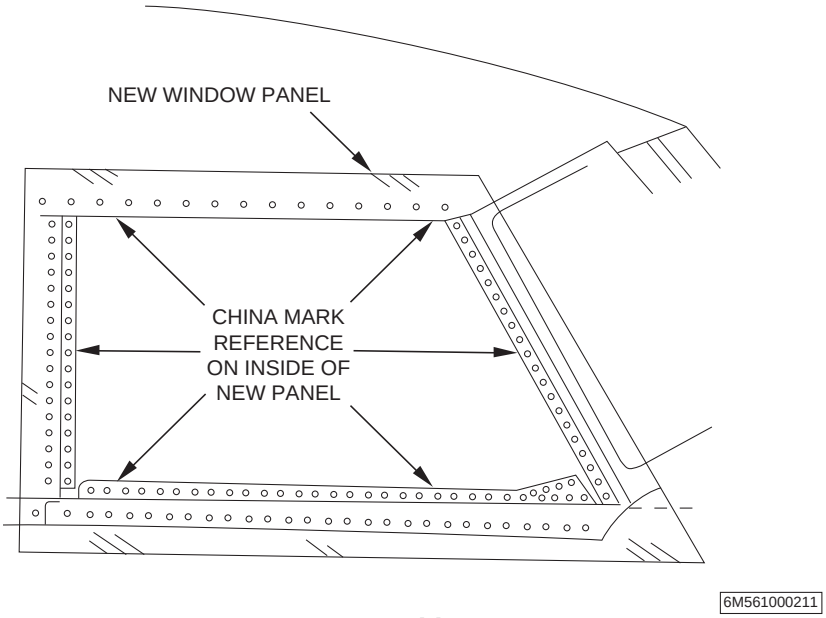
(3) Remove paper and place on outside face of new panel. Center paper so that trim area is equal on all sides.

E. Trimming

(1) Trim along edge of new panel (faced with protective paper) with a 14 teeth per inch blade band saw.

NOTE: Use a power plane or rasp for minor trimming.

(2) Smooth trimmed edges by sanding.



Marking
Figure 211

F. Drilling (See Figure 210)

NOTE: Remove a 1.00 inch (2.54 cm) strip of paper from edges of panel to allow for drilling.

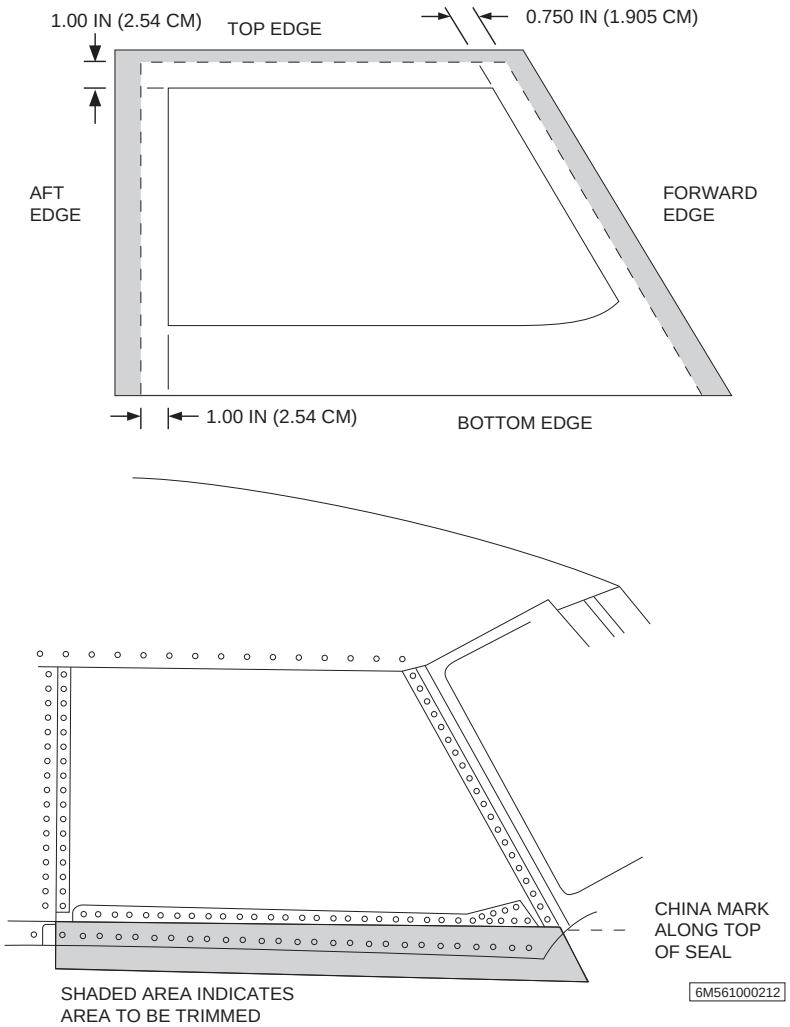
(1) Place new panel in aircraft side window frame.

(2) Position panel to achieve proper fit, contour, and space.

(3) Position lower and aft retainers in place and secure at several equally spaced points with Cleco fasteners or screws.

NOTE: Forward outer retainer cannot be positioned until holes are drilled into new panel.

CAUTION: USE A DRILL STOP TO RESTRICT DRILLING MORE THAN WINDOW PANEL THICKNESS TO PREVENT ELONGATED HOLES IN WINDOW FRAME SIDE POST.



Trimming
Figure 212

(4) Drill forward outer retainer holes to align with holes in window frame side post with a 0.166 inch drill bit, using the following sequence.

(a) Slowly drill one (1) hole at 3 o'clock position and secure with Cleco.

(b) Drill one (1) hole at 9 o'clock position and secure with Cleco.

(c) Drill one (1) hole at 6 o'clock position and secure with Cleco.

(5) Using outer retainers as templates, drill starter holes for all remaining holes except one (1) hole at top of window, with 0.166 inch high-speed steel drill bit.

NOTE: "Starter Holes" are holes that DO NOT penetrate entire thickness of material.

(6) Drill one (1) 0.113 inch (2.870 mm) diameter hole at top of window and secure with Cleco.

(7) Remove retainers and panel to a suitable work area. Complete drilling of holes on drill press.

(8) Enlarge holes to 0.250 inch (6.350 mm) (except 0.113 inch [2.870 mm] diameter hole) using counterbore with appropriate pilot. Start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) of material thickness.

(9) Turn window panel over and finish counterboring holes.

(10) Use sandpaper #400 or countersink to remove any ridges around hole edges.

G. Deburring

(1) Remove chips and burrs from holes.

H. Installing Bushings

(1) Install bushings in all 0.250 inch (6.35 mm) diameter holes, using Imperial Eastman Poly-Flow tubing, 44-P-1/4 (0.250 x 0.312), for bushing material. Trim bushings flush with each side of hole.

I. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent dried polish from falling between panels.

(3) For hairline scratches, use polysand procedures IAW steps **1.**, G., (5) through (16), Section 56-00-00 this chapter.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax. Apply with fingertips in a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect for imperfections.

(6) Remove masking from holes and mask remaining areas of panel with pressure- sensitive masking paper.

J. Preparation – Old Window Not Intact

(1) Inspect new window panel for distortion and general condition. Mask for protection.

K. Marking and Trimming (see Figures 211 and 212)

(1) Position new window panel in frame. Align panel to provide equal distance on each side.

(2) From inside cockpit, make a reference mark on new panel conforming to dimensions of window retainers.

(3) Remove panel to work area.

(4) Draw a reference mark 1.00 inch (2.54 cm) aligned with mark at top and aft side of new panel.

(5) Make a reference mark 0.750 inch (1.90 cm) aligned with mark at FWD side of panel (see Figure 212).

(6) Trim panel with 14 teeth per inch blade band saw (see Figure 212).

(7) Smooth all trimmed edges by sanding.

L. Trimming – Bottom (see Figure 212)

(1) Place new window in position with upper edge in window frame header assembly.

NOTE: Further trimming of upper corners may be required to properly fit panel into header assembly. Use a rasp for final trimming.

(2) Align forward and aft sides of panel equally in aircraft frame.

(3) While holding new panel firmly in place, make a reference mark on outside of panel to align with top of window sill.

(4) Remove panel to work area and trim excess with 14 teeth per inch blade band saw (see Figure 212).

(5) Smooth all trimmed edges by sanding.

M. Drilling

(1) Remove 1.00 inch (2.54 cm) strip of paper from panel along both edges to allow for drilling.

NOTE: Use a high speed drill (15,000-25,000 RPM) to drill holes and eliminate chipping. Use outer window retainers as reference templates for drilling holes in panel.

(2) Position and align panel in side window frame.

(3) Position lower and aft outer retainers in place and secure at several equally space points with Cleco fasteners or screws.

CAUTION: USE A DRILL STOP TO RESTRICT DRILLING MORE THAN WINDOW PANEL THICKNESS TO PREVENT ELONGATED HOLES IN WINDOW FRAME SIDE POST.

(4) Drill forward outer retainer holes to align with holes in window frame side post with a 0.166 inch drill bit, using the following sequence.

(a) Slowly drill one (1) hole at 3 o'clock position and secure with Cleco.

(b) Drill one (1) hole at 9 o'clock position and secure with Cleco.

(c) Drill one (1) hole at 6 o'clock position and secure with Cleco.

(5) Using outer retainers as templates, drill starter holes for all remaining holes except one (1) hole at top of window, with 0.166 inch high-speed steel drill bit.

NOTE: "Starter Holes" are holes that DO NOT penetrate entire thickness of material.

(6) Drill one (1) 0.113 inch (2.870 mm) diameter hole at top of window and secure with Cleco.

NOTE: This hole is needed to secure aft outer side retainer to panel IAW para **4.**, S., Installation – Side Window (Outer Panel), this section.

(7) Remove retainers and panel to a suitable work area. Complete drilling of holes on drill press.

(8) Enlarge holes to 0.250 inch (6.350 mm) (except 0.113 inch [2.870 mm] diameter hole) using counterbore with appropriate pilot. Start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) of material thickness.

(9) Turn window panel over and finish counterboring holes.

(10) Use sandpaper #400 or countersink to remove any ridges around hole edges.

N. Deburring

(1) Remove chips and burrs from holes.

O. Installing Bushings

(1) Install bushings in all 0.250 inch (6.35 mm) holes, using Imperial Eastman Poly-Flow tubing, 44-P-1/4 (0.250 x 0.312), for bushing material. Trim bushings flush with each side of hole.

P. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent dried polish from falling between panels.

(3) For hairline scratches, use polysand procedures IAW steps **1.**, G., (5) through (16), Section 56-00-00 this chapter.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax with fingertips in a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect for imperfections. If any damage is visible, repeat necessary steps.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

Q. Installation – Side Window (Outer Panel)

NOTE: The following procedures apply to all outer side window panel installations, using either reusable existing panels or new panels predrilled with applicable holes IAW para **4.**, F., Drilling, this section.

(1) Inspect panel for general condition and defects in material. Mask panel after inspection.

(2) Apply thin coat of MIL-S-8802 sealant (or equivalent) to window sill, lower frame retainer, side post, aft frame retainer, and between the two (2) angle brackets in header assembly in aircraft side window frame.

NOTE: MIL-S-8802 sealant may also be used as an underlay material to fill a minor mismatch of side window (outer panel) with surrounding side window frame.

MIL-S-81733 sealants are approved equivalents for MIL-S-8802 sealant.

(3) Position new panel in frame.

(4) Apply thin coat of MIL-S-8802 sealant (or equivalent) to applicable surfaces of aft outer, lower outer, and forward outer side window retainers.

(5) Apply a small amount of MIL-S-8802 sealant (or equivalent) around each screw prior to installing outer retainers and outer window panel.

(6) Install outer retainers, secure with screws just snug, and allow sealant to set.

(7) Install small screw, washer, and nut, securing aft outer retainer to outer window panel just snug, and allow sealant to set.

(8) Install inner side window panel with gaskets and nutplate strips.

(9) Install screws just snug, and allow sealant to set.

(10) Stage-torque screws 15 to 20 inch-pounds (1.69 to 2.26 N·m) starting at 12 o'clock position in a cross sequence installation pattern, e.g., 12 o'clock to 6 o'clock, 3 o'clock to 9 o'clock, etc.

NOTE: Do not exceed the specified torque values during initial tightening.

NOTE: Refer to Figure 201A for screw length requirements.

(11) Apply MIL-S-8802 sealant (or equivalent) to fill spaces between retainers, outer window panel, and aircraft frame. After sealant cures, remove masking and clean side windows.

(12) Install outer window panel and cover clips.

(13) Install window purge tubing.

(14) Install optional acoustical windows, if applicable.

(15) Perform pressurization check IAW Adjustment/Test – Pressurization, Chapter 21, this manual.

(16) Install instrument panel glareshield.

5. Removal/Installation Side Window (Inner Panel)

A. Removal – Side Window (Inner Panel)

(1) Remove instrument panel glareshield.

(2) Remove front window post cover.

(3) Remove optional acoustical window, if applicable.

(4) Remove window frame from side window.

(5) Remove all retaining screws.

(6) Remove paneling clips.

NOTE: Mark location of clips for reference during installation.

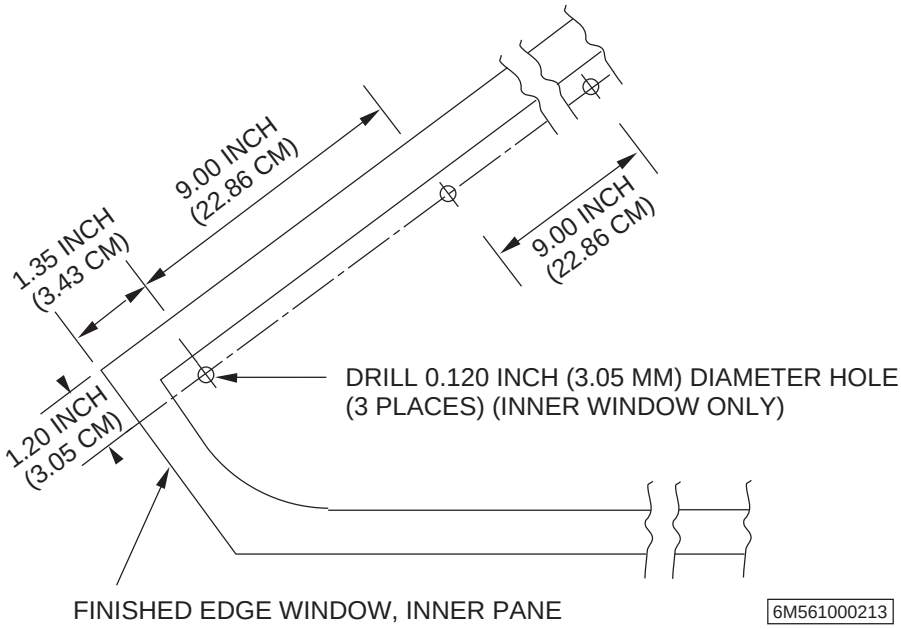
(7) Remove all inside fasteners.

NOTE: It is not necessary to remove the side window outer panel in order to remove the inner panel. Only the screws and fasteners require removal.

(8) Remove inner panel.

NOTE: It may be necessary to cut through the sealant around the outer edges of the window with a sharp-bladed instrument to help loosen sealant.

(9) Remove top, aft, and bottom window fillers.



Drilling
Figure 213

WARNING: METHYL PROPYL KETONE (MPK) IS EXTREMELY FLAMMABLE AND SHOULD BE USED ONLY USED ONLY IN WELL VENTILATED AREAS. OBSERVE SAFETY PRECAUTIONS. MPK CONSTITUTES A MODERATE HEALTH HAZARD THROUGH INHALATION, SKIN ABSORPTION, EYE IRRITATION, OR INGESTION.

(10) Clean window frame of all sealant residue using methyl propyl ketone (MPK).

B. Required Equipment

(1) Hand drill, high-speed motor, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed.

(3) Band saw, with 14 teeth per inch blade.

(4) High-speed steel drill bits, 0.166 inch, 0.113 inch, and 0.120 inch, for drilling acrylic.

(5) Torque wrench.

(6) Countersink bit, 0.156 inch.

(7) Counterbore bit, 0.250 inch.

(8) Steel rule.

(9) Rasp.

(10) Sheet metal Cleco fasteners or draw mono-locks.

(11) China marker.

(12) No. 343 Nomar Pressure Sensitive Paper (3M Company) or Permacel 01 Pressure Sensitive Paper (Permacel Tape Corp.).

(13) Poly-Flow tubing 44-P-1/4 (0.250 x 0.312).

(14) Sealant, MIL-S-8802, or the following approved equivalent:

(a) Sealant, MIL-S-81733.

(15) Scratch-removing compound – Wilco No. 35, Louis P. Hershey Lustar, or other conforming to MIL-C-18767.

(16) Permatex polish No. 403D or Vista automobile wax.

(17) Dupont Imron No. 5027UT Red, or Axalta Imron AF700 BC, 5027ES Monza Red paint.

C. Preparation

(1) Inspect new window for distortion and general condition. Mask for protection.

D. Making Template

(1) Use masking paper as a template to transfer outside dimensions of old panel onto new panel.

NOTE: When pressure-sensitive paper is not available, use pressure-sensitive paper applied to new panel for protection and storage.

(2) Position pressure-sensitive paper on outside area of old panel. Trim paper to match outer dimensions of old panel.

(3) Remove paper and place on outside of new panel. Center paper so that trim area is equal on all sides.

E. Trimming

(1) Trim panel by hand. Do not heat up edges. Smooth and polish all rough edges and marks.

NOTE: Use a power plane or rasp for minor trimming.

(2) Smooth and polish trimmed edges.

(3) Paint (spray or brush) entire outer edge (thickness) of window with Dupont Imron No. 5027UT Red, or Axalta Imron AF700 BC, 5027ES Monza Red paint.

F. Drilling

(1) Remove a 1.00 inch (2.54 cm) strip of pressure-sensitive paper covering new panel along edges on both sides to allow for drilling.

(2) Position and align new inner panel in aircraft side window frame.

(3) Drill holes in panel to align with holes in window frame with a 0.166 inch drill bit, using the following sequence.

(a) Slowly drill one (1) hole at 3 o'clock position and secure with Cleco.

(b) Drill one (1) hole at 9 o'clock position and secure with Cleco.

(c) Drill one (1) hole at 6 o'clock position and secure with Cleco.

CAUTION: USE A DRILL STOP TO RESTRICT DRILLING BEYOND WINDOW PANEL THICKNESS TO PREVENT DAMAGING HOLES IN WINDOW FRAME SIDE POST.

(4) With inner panel in place and temporarily secured, use outer panel screw holes as guides to locate and start drill remaining holes except one (1) hole at top of window, with 0.166 inch high-speed steel drill bit.

(5) Drill one (1) 0.113 inch (2.870 mm) diameter hole at top of window and secure with Cleco.

(6) Remove retainers and panel to a suitable work area. Complete drilling of holes on drill press.

(7) Enlarge holes to 0.250 inch (6.350 mm) (except 0.113 inch [2.870 mm] diameter hole) using counterbore with appropriate pilot. Start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) of material thickness.

(8) Turn window panel over and finish counterboring holes.

(9) Use sandpaper #400 or countersink to remove any ridges around hole edges.

(10) Drill three (3) purge system holes in side window inner panel, using a 0.120 inch high-speed steel drill bit and high-speed hand drill (see Figure 213).

G. Deburring

(1) Remove chips and burrs from holes.

H. Installing Bushings

(1) Install bushings in all 0.250 inch (6.35 mm) holes, using Imperial Eastman Poly-Flow tubing, 44-P-1/4 (0.250 x 0.312). Trim bushings flush with each side of hole.

I. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent dried polish from falling between panels.

(3) For hairline scratches, use polysand procedures IAW Polishing Procedure, this chapter.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax. Apply with fingertips in circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect for imperfections.

(6) Remove masking from holes and mask remaining areas of panel with pressure-sensitive masking paper.

J. Installation – Side Window (Inner Panel)

NOTE: The following procedures apply to all inner side window panel installations, using either reusable existing panels or new panels predrilled with applicable holes IAW para **5.**, F., Drilling, this section.

(1) Inspect panel for general condition and defects in material. Mask panel after inspection.

(2) Apply thin coat of MIL-S-8802 sealant (or approved equivalent sealant) to both sides of top, aft, and bottom window panel fillers, and install each in place.

NOTE: MIL-S-8802 sealant may also be used as an underlay material to fill a minor mismatch of side window (inner panel) with surrounding side window frame.

MIL-S-81733 sealants are approved equivalents for MIL-S-8802 sealant.

(3) Position inner panel in place.

(4) Apply thin coat of MIL-S-8802 sealant (or equivalent) to inner surface of nutplate strip retainers.

(5) Install nutplate strip retainers.

(6) Install inner side window panel and cover clips.

(7) Apply small amount of sealant MIL-S-8802 sealant (or equivalent) around each screw prior to installing.

(8) Install screws just snug and allow MIL-S-8802 sealant (or equivalent) to set.

NOTE: Do not exceed the specified torque values during initial tightening.

NOTE: Refer to Figure 201A for screw length requirements.

(9) Stage-torque screws 15 to 20 inch-pounds (1.69 to 2.26 N·m), starting at 12 o'clock position in a cross sequence installation pattern, e.g., 12 o'clock to 6 o'clock, 3 o'clock to 9 o'clock, etc.

NOTE: Do not exceed the suggested minimum torque values during initial tightening.

(10) Apply MIL-S-8802 sealant (or equivalent) to fill spaces between retainers, outer window panel, and aircraft frame. Allow sealant to cure.

(11) After MIL-S-8802 sealant (or equivalent) cures, remove masking and clean windows.

(12) Install window purge tubing.

(13) Install inner side window covers.

(14) Perform pressurization check IAW Adjustment/Test – Pressurization, Chapter 21, this manual.

(15) Install optional acoustical window panels, if applicable.

(16) Install instrument panel glareshield.

K. Installation – Side Window (Inner Panel) Old Panel Not Intact

NOTE: The following procedures apply to installation of cockpit side windows when the old panel is not intact for use as a template or new panel is not predrilled.

(1) Inspect new inner panel for general condition and defects in material.

(2) Place inner panel into position and trim panel to obtain clearance around the edges.

NOTE: Trim aft edge as necessary.

Trim panel by hand. Do not heat up edges. Smooth and polish all rough edges and marks.

(3) Paint (spray or brush) entire outer edge (thickness) of window with Dupont Imron No. 5027UT Red, or Axalta Imron AF700 BC, 5027ES Monza Red paint.

(4) Drill screw holes in inner panel;

(a) Cover panel with Normar Pressure Sensitive Paper (3M) or Permacel 01 Pressure Sensitive Paper (Permacel) adhesive backed paper to prevent surface damage.

(5) Hold panel on window frame and drill holes in new windshield with a 0.166 inch drill bit, using the following sequence.

(a) Slowly drill one (1) hole at 3 o'clock position and secure with cleco.

(b) Drill one (1) hole at 9 o'clock position and secure with cleco.

(c) Drill one (1) hole at 12 o'clock position and secure with cleco.

(d) Drill one (1) hole at 6 o'clock position and secure with cleco.

(e) Drill start remaining holes.

(6) Remove panel and position on work bench with wood backup.

(7) Drill remaining holes applying no feed pressure on drill motor. Allow drill to lightly feed itself through panel.

(8) Enlarge holes to 0.250 inch (6.350 mm) using counterbore with appropriate pilot. Again start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) of material thickness.

(9) Turn window panel over and finish counterboring holes.

(10) Use sandpaper #400 or countersink to remove any ridges around hole edges.

(11) Drill three (3) window vent holes 0.120 inch (3.05 mm) diameter through inner panel, see Figure 213. Clean and deburr holes.

(12) Clean and polish new inner panel.

(13) Clean faying surfaces of inner retainer with MIL-S-38736, PR148 (PRC) or Turco #657.

(14) Install bushings in all 0.250 inch (6.35 mm) diameter holes of inner panel and trim flush with each side of hole.

(15) Cover installation area with PR1422-B2 sealant.

(16) Apply a thin coat of sealant to both inner and outer fay surfaces of the window frame.

(17) Install fillers.

(18) Apply a thin coat of sealant to inner fay surface of fillers.

NOTE: Ensure sealant does not to get in open ends of purge tubes.

(19) Install new inner panel.

(20) Apply a thin coat of sealant to the outer fay surface of inner retainers and install retainers.

(21) Apply a small amount of sealant around each screw.

(22) Install screws.

NOTE: Do not exceed the specified torque values during initial tightening.

NOTE: Refer to Figure 201A for screw length requirements.

(23) Stage torque screws as follows;

(a) Start at 12 o'clock and use a diagonal sequential tightening schedule i.e., 12 to 6, 3 to 9, 12:05 to 6:05, 3:05 to 9:05, etc. Torque screws to 4.5 inch-pounds (0.51 N·m).

(b) Repeat same sequence, tightening screws to 15 to 25 inch-pounds (1.69 to 2.82 N·m).

(24) Install push rivets in inner panel vent holes.

(25) Perform pressurization check IAW Adjustment/Test – Pressurization, Chapter 21, this manual.

(26) Perform inspection of screw holes for cracks IAW 56-00-00 **1.**, E, (3) and (4).

(27) Install inner side window covers.

(28) Install optional acoustical window panels, if applicable.

(29) At next scheduled, Letter Check or Phase Inspection, perform inspection of screw holes for cracks IAW 56-00-00 **1.**, E, (4).

L. Desiccant Filter Check – Dual Side Window Panels

(1) Remove window cover.

(2) Remove side panel under window.

(3) Check color of desiccant.

NOTE: If desiccant is pink replace desiccant. If desiccant is blue, check is complete, replace access panel.

(4) Place suitable container under filter assembly.

(5) Remove two (2) lines from "T" fitting on filter assembly.

(6) Remove tie strap.

(7) Remove filter assembly.

(8) Remove screw cap from bottom of filter.

(9) Remove old desiccant and replace with new or refreshed desiccant.

(10) Install screw cap to bottom of filter.

(11) Install two (2) lines on "T" fitting on filter assembly.

(12) Install side panel under window.

(13) Install window cover.

M. Nitrogen Moisture Purge – Dual Side Window Panels

(1) Remove window cover.

(2) Remove side panel under window.

(3) Place suitable container under filter assembly.

(4) Remove two (2) lines on "T" fitting on filter assembly (forward line is drain line).

CAUTION: DO NOT USE PRESSURE TO REMOVE PUDDLED MOISTURE ACCUMULATION. PRESSURE WILL BUILD RAPIDLY BETWEEN PANELS AND FRACTURE INNER PANE.

(5) Attach regulated dry nitrogen source to aft line.

NOTE: If moisture has puddled, allow it to drain or attach syringe or similar device to drain line and withdraw water.

(6) Remove three (3) push rivets at forward edge of inner panel.

CAUTION: APPLY NO MORE THAN 10 PSI (0.703 KG/CM^2) OF PRESSURE.

(7) Apply no more than 10 psi (0.703 kg/cm^2 of pressure from dry nitrogen source for one (1) hour after all visible moisture is removed.

(8) Disconnect nitrogen source.

(9) Install three (3) previously removed push rivets.

(10) Install new or refreshed desiccant in filter.

(11) Install two previously removed lines on "T" fitting on filter assembly.

(12) Install side panel under window.

(13) Install window cover.

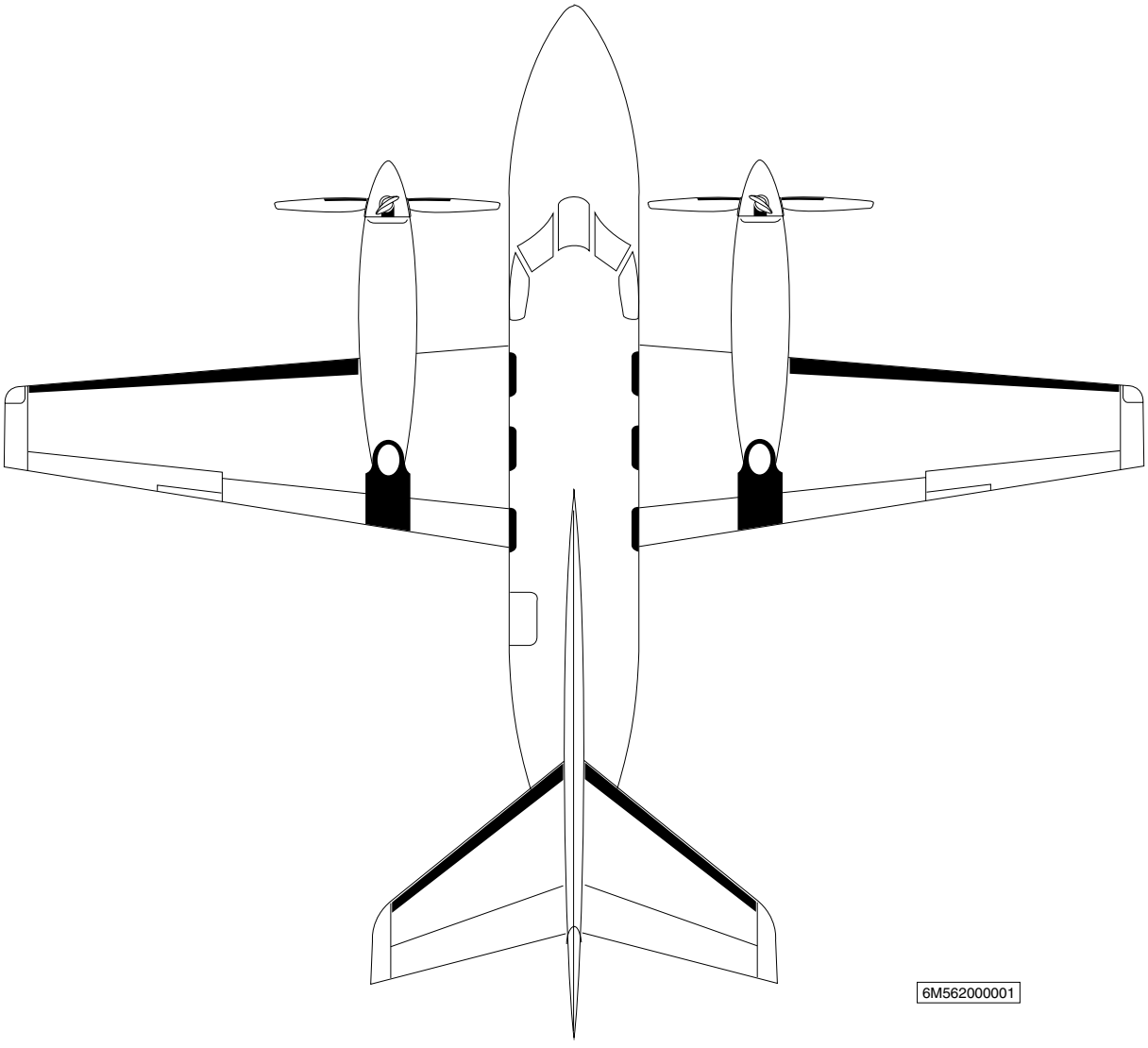
CABIN COMPARTMENT WINDOWS – DESCRIPTION AND OPERATION

1. General

A. All cabin windows are uniform in construction consisting of an inner and outer acrylic panel. The acrylic window panels are sealed around the outer edges. The panels are designed to sustain the load imposed by cabin pressurization and external airloads.

2. Cabin Windows SA226 – T

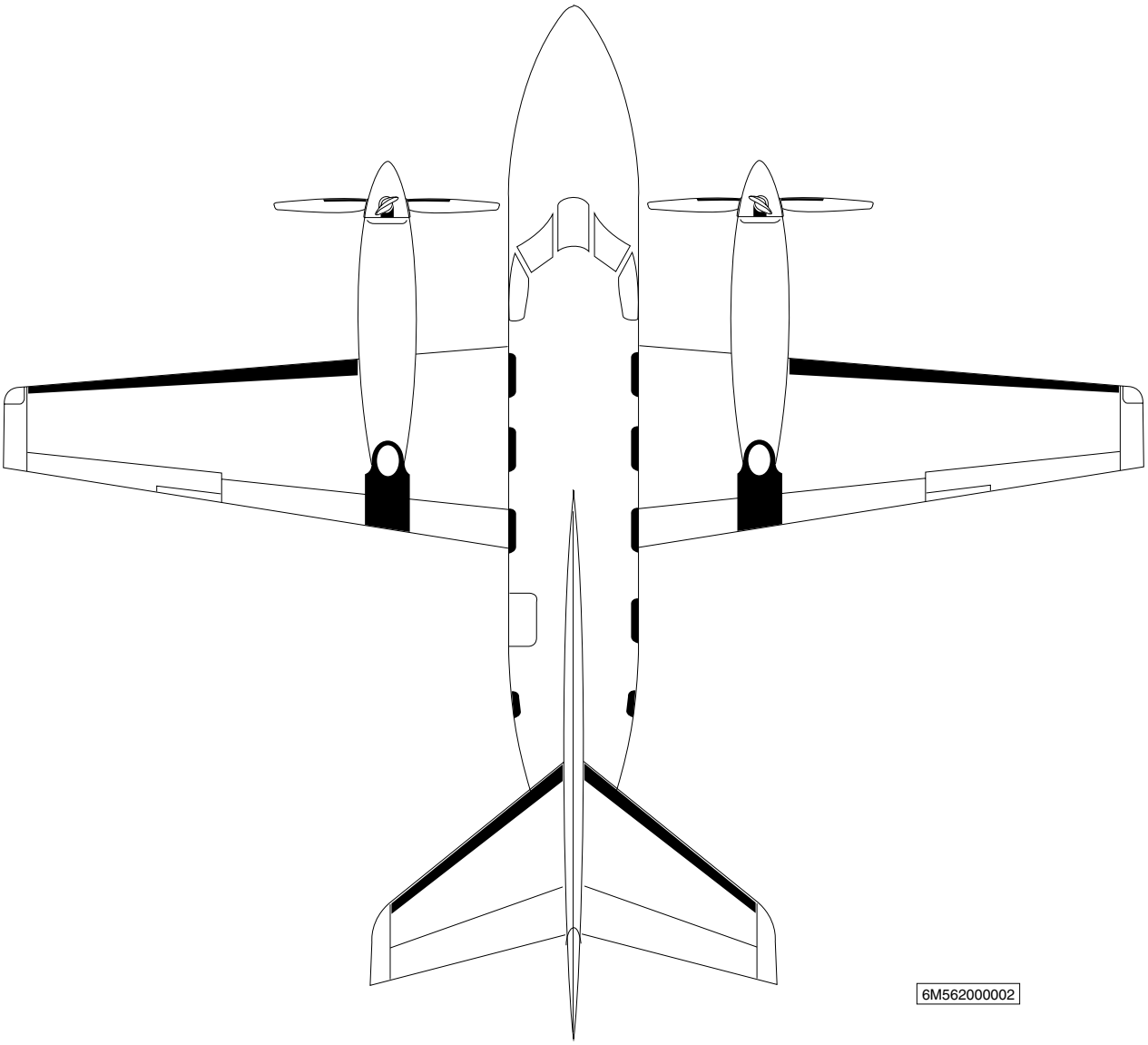
A. The cabin compartment of early SA226 – T aircraft (SN 201-248) contains six windows, three on the left side of the fuselage and three on the right side (see Figure 1). The dual panel windows are of the dry air sandwich type. Either the inner or outer panel will sustain full cabin differential pressure in the event of failure of the other pane. A desiccant bag (moisture collector) is attached at the bottom of each window to prevent fogging and to remove moisture.



SA226 – T Cabin Windows
(SN 201-248)
Figure 1

CABIN COMPARTMENT WINDOWS – DESCRIPTION AND OPERATION

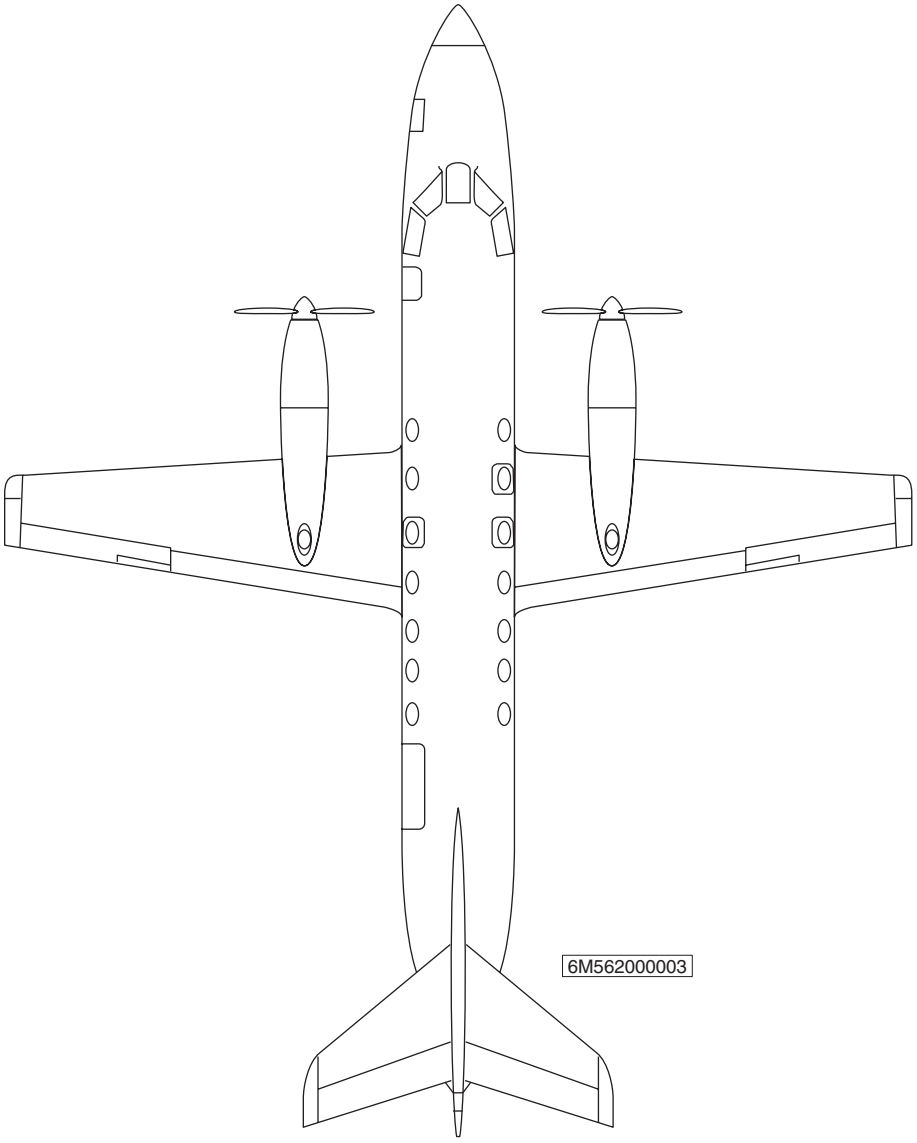
B. The cabin compartment of later SA226 – T aircraft (SN 249-999) contains nine windows, four on the left side of the fuselage and five on the right side (see Figure 2). The two windows at the aft end of the cabin compartment are smaller than the other seven windows. All windows are dual panel dry air sandwich type. A desiccant bag is attached to the bottom of each of the six forward windows to prevent fogging and to remove moisture. The three aft windows have drilled inner panels to allow circulation of cabin air; this air circulation between panels eliminates moisture and prevents fogging.



SA226 – T Cabin Windows
(SN 249-999)
Figure 2

3. Cabin Windows SA226 – AT, TC

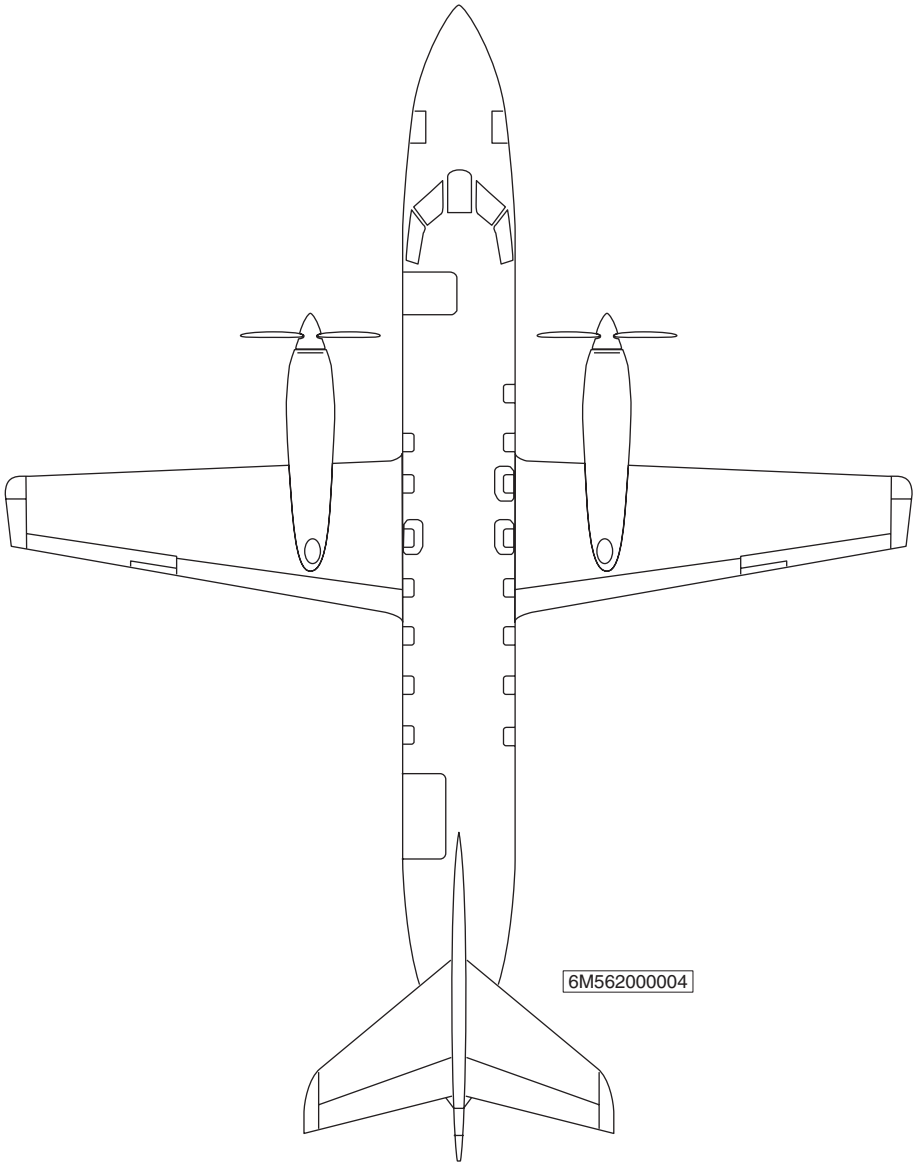
A. The cabin compartment of SA226 – AT aircraft, SN 001-024, incorporates 14 acrylic dual panel round windows of the dry air sandwich type. Seven windows are installed on the right hand side, and seven on the left hand side.



SA226 – AT Cabin Windows
(SN 001-024)
Figure 3

CABIN COMPARTMENT WINDOWS – DESCRIPTION AND OPERATION

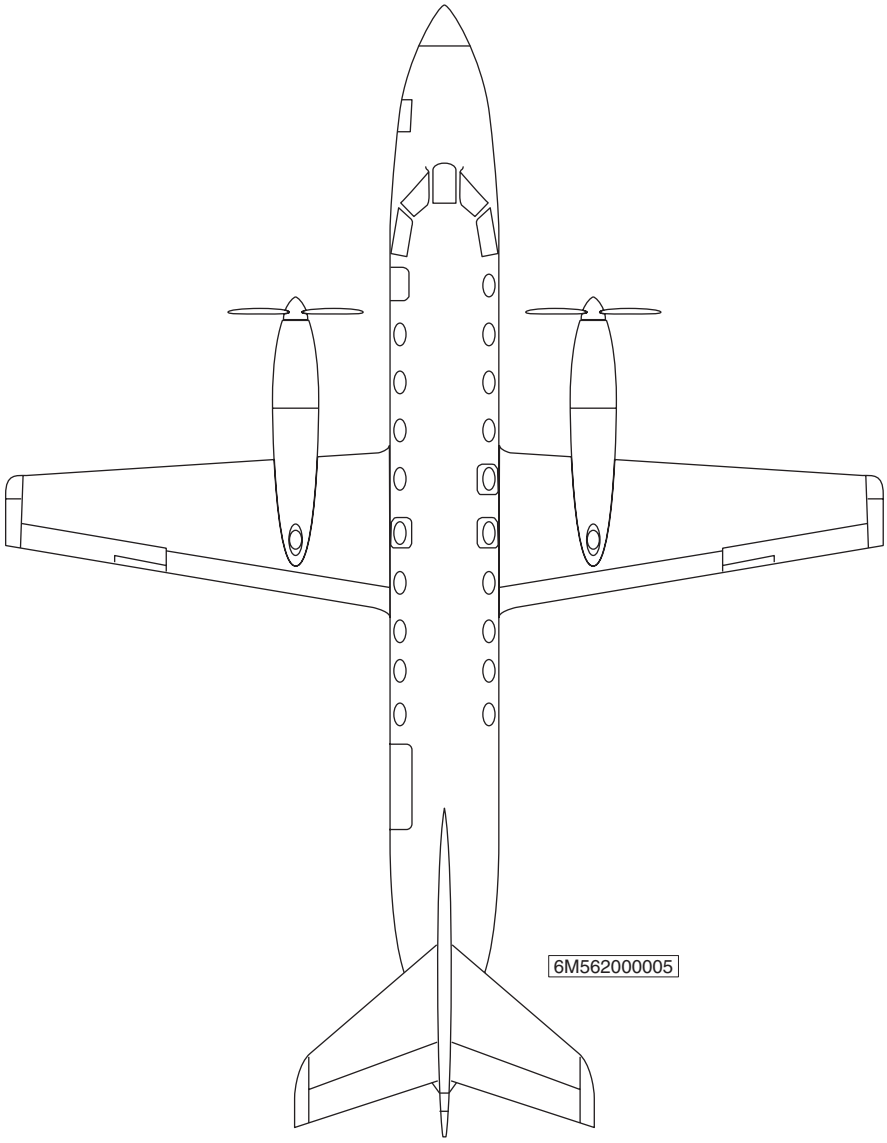
B. The cabin compartment of SA226-AT aircraft, SN 025-999 incorporates 15 acrylic dual panel oblong windows of the dry air sandwich type. Seven windows are installed on the left hand side, and eight windows on the right hand side. A desiccant bag is attached to the window assembly on each of the three escape hatches to prevent fogging and to remove moisture from between the panes. On SN 048 and up, four 0.125 in (3.175 mm) holes are drilled into all cabin inner windows other than the escape hatches allowing cabin air to circulate between the panels to eliminate moisture and prevent fogging. All cabin windows are installed with shades as standard equipment. An additional acoustical acrylic window panel may be installed on each cabin window as an optional item.



SA226 – AT Cabin Windows
(SN 025-999)
Figure 4

CABIN COMPARTMENT WINDOWS – DESCRIPTION AND OPERATION

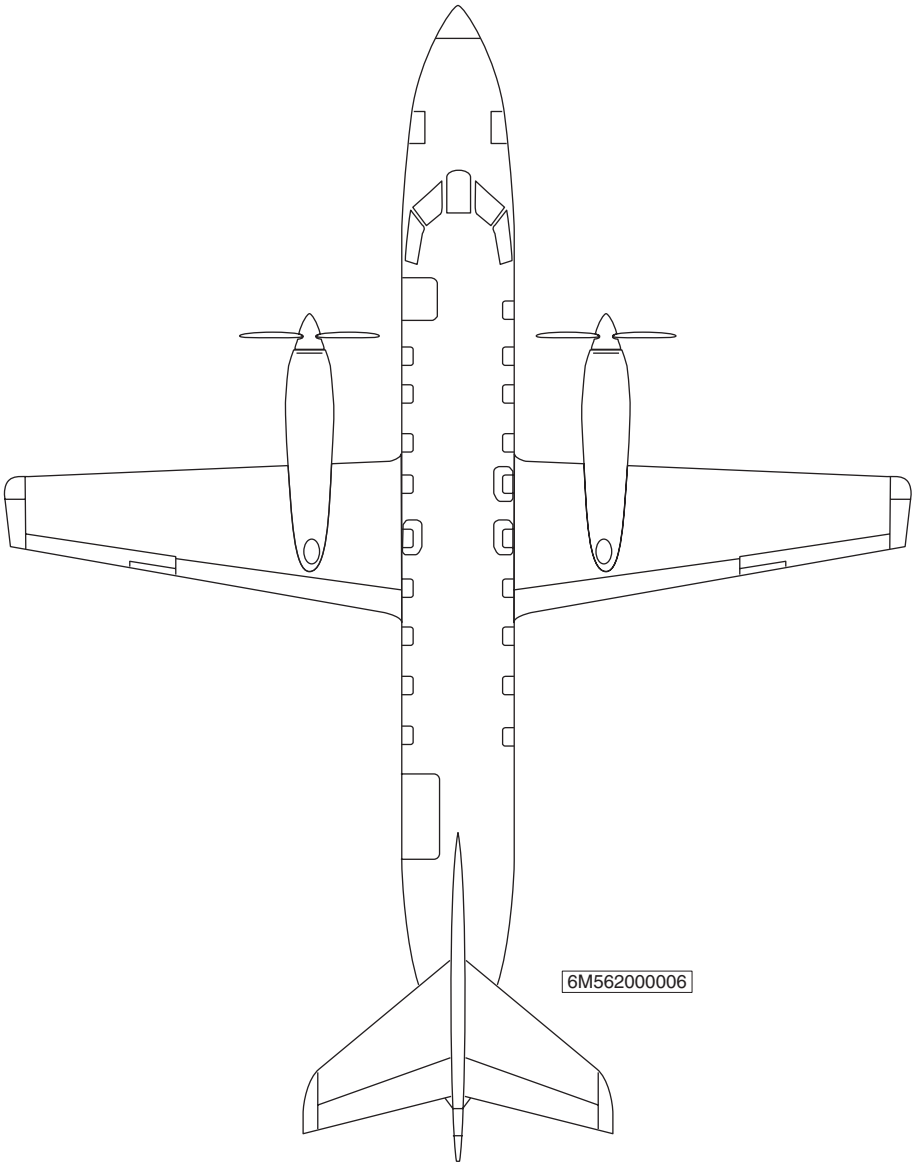
C. The cabin compartment of SA226 – TC aircraft, SN 201-212, incorporates 19 acrylic dual panel round windows of the dry air sandwich type. Nine windows are installed on the left hand side, and ten on the right hand side.



SA226 – TC Cabin Windows
(SN 201-212)
Figure 5

CABIN COMPARTMENT WINDOWS – DESCRIPTION AND OPERATION

D. The cabin compartment of SA226-TC aircraft, SN 213-999, incorporates 19 acrylic dual panel oblong windows of the dry air sandwich type. Nine windows are installed on the left hand side and ten on the right hand side. A desiccant bag is attached to the window assembly on each of the three escape hatches to prevent fogging and to remove moisture from between the panels. On SN 217 and up, four 0.125 in (3.175 mm) holes are drilled into all cabin inner windows other than the escape hatches allowing cabin air to circulate between the panels to eliminate moisture and prevent fogging. All cabin windows are installed without shades. However, installation of shades is an optional item. An additional acoustical acrylic window panel may be installed on each cabin window as an optional item.



SA226 – TC Cabin Windows
(SN 213-999)
Figure 6

SA226 – T CABIN WINDOWS – MAINTENANCE PRACTICES

1. Removal/Installation – SA226 – T Cabin Windows

A. Removal – SA226 – T Cabin Windows (Outer Panel)

(1) Remove interior Royalite paneling from around window panel to be removed.

(2) On applicable windows, remove desiccant bag tubing and temporarily stow aside until required during installation procedure.

(3) Remove Royalite panel clips.

NOTE: Prior to removal of clips, mark location for future reference during installation.

(4) Remove all retaining screws, nuts, and washers from each outer panel retainer.

(5) Remove inner panel. Clean, inspect for damage, and store until ready for reinstallation.

(6) Remove outer window retainer.

(7) Remove outer window panel.

NOTE: It may be necessary to cut the sealant around the outer edges of the panel with a sharp bladed instrument to help loosen the sealant.

WARNING: WEAR EYE PROTECTION AND GLOVES WHEN USING SAFETY WIRE TO CUT WINDOW SEALANT.

(a) Dig a small hole in the sealer and push through a piece of 0.020 stainless safety wire.

(b) Wrap ends of wire around screwdriver or similar tool.

(c) With one person inside and one outside, pull wire in a reciprocating motion through sealer until window is free. Remove outer panel.

(8) Remove the window seal and discard.

(9) Clean window frame of all sealant residue with MIL-C-38736 cleaning solvent compound.

B. Required Equipment

(1) Hand drill motor, high speed, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed (15,000 RPM).

(3) Drill bit, 0.166 inch (135° high speed), or equivalent.

(4) Drill bit, 0.173, 0.172, and 0.1695 inch.

(5) Torque Wrench.

(6) Countersink bit, 0.156 inch.

(7) Counterbore bit 0.196 inch.

(8) Counterbore bit 0.250 inch.

(9) Rasp.

(10) Cleco fasteners or wedge-lock fasteners.

(11) Sealant, MIL-S-8802 or equivalent.

(12) Solvents, MIL-S-38736.

(13) Adhesive.

(14) Alodine #1200 (MIL-C-5541), Conversion Coating.

(15) Plastic Polish.

(16) Auto Wax Vista (Local Purchase).

(17) Zinc Chromate Primer TT-P-1756 (Local Purchase).

NOTE: To determine if a window needs to be replaced, refer to 56-00-00, Inspection/ Check Windows.

C. New Window Preparation

(1) Inspect new window for distortion and general condition. Mask for protection.

NOTE: The new outer cabin window is received with both of its sides previously routed. Therefore, no routing procedure need be accomplished.

D. Trimming

(1) Place new window in position in aircraft structure.

(2) Determine amount of trimming required for proper fit. Allow approximately 0.030 in (0.762 mm) space between edges of window panel and aircraft structure to prevent fretting.

(3) Trim window panel where required with a rasp. Do not heat up edges.

(4) Smooth and polish all rough edges and marks.

(5) Bevel inside edge of outer pane to 0.05 inch radius.

E. Drilling

CAUTION: DRILL A CLEAN HOLE FREE OF CHIPS, VENTS, AND RESIDUAL STRESSES. ALLOW DRILL TO DO WORK WITH LITTLE OR NO FORCE FROM OPERATOR. NEVER DRILL ACRYLIC IN CONJUNCTION WITH METAL. METAL CHIPS WILL SCRATCH ACRYLIC.

(1) Drill inner retainer screw holes in window pane.

(2) Deburr holes and break sharp edges in inner retainer.

(3) Cover pane with Normar Pressure Sensitive Paper (3M Company) or Permacel 01 Pressure Sensitive Paper (Permacel Tape Corp.) adhesive back paper to prevent surface damage.

(4) Observe the following when drilling the inner window retainer:

(a) Use 0.166 inch drill bit (135° high speed), or equivalent.

(b) Save old pane and practice drilling holes to get the feel of the drill motor, bit, counterbore, and general operation. Try several different procedures and select the one that's best for you.

(c) Hold drill perpendicular to pane surface.

(d) Use a counterbore to enlarge holes to final size.

(e) Carefully remove ridges around holes with fine sandpaper.

(5) Hold pane in retainer. Slowly drill one (1) hole at 3 o'clock position and secure with cleco.

(6) Drill one (1) hole at 9 o'clock position and secure with cleco.

(7) Drill one (1) hole at 12 o'clock position and secure with cleco.

(8) Drill one (1) hole at 6 o'clock position and secure with cleco.

(9) Start drill remaining holes. After all holes are marked, move pane to work bench.

(10) Position pane on work bench. Back up window panel with a wood block or similar material.

(11) Drill all holes applying no feed pressure. Allow drill to lightly feed itself through panel.

(12) Enlarge holes to 0.250 inch using counterbore with appropriate pilot. Again start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) material thickness.

(13) Turn window pane over and finish counterboring holes.

(14) Use sandpaper #400 or countersink to remove any ridges around hole edges.

(15) Align new outer retainer with old outer retainer. Locate hole pattern and transfer to new retainer using old retainer as a template.

NOTE: The outer retainer may be locally manufactured. The new retainer is of thicker material with chamfered edges and the holes for the screw receptacles are countersunk.

F. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent dried polish from falling inside between the panels after installation and to keep sealant contaminant free.

(3) For hairline scratches, use polysand procedure IAW steps **1.**, G., (5) through (16), Section 56-00-00, this Chapter.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax; apply with fingertips using a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect visually for imperfections. If any damage is visible, repeat necessary steps.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

G. Installation – SA226 – T Cabin Window (Outer Panel)

(1) Inspect new cabin window for distortion and general condition.

(2) Clean faying surfaces of inner retainer with MIL-S-38736.

(3) Prepare a sufficient quantity of class B-2 sealant IAW MIL-S-8802 to cover installation area.

(4) Apply thin coat of class B-2 sealant to inner retainer.

(5) Install new window seal with adhesive.

CAUTION: BE CAREFUL NOT TO SCRATCH WINDOW PANE.

(6) Install new bushings in 0.0250 inch holes in inner and outer panes. Trim bushings flush to match pane thickness. Make bushings from Imperial Eastman Poly-Flow tubing No. 44-PP.

(7) Install inner window pane.

(8) Position outer window pane in place in the aircraft structure.

(9) Apply thin coat of sealant to inner side of outer retainer, Install retainer in position.

(10) Apply small amount of sealant around each screw. Install screws, washers, and nuts.

NOTE: Check length of screws installed. Ensure protrusion of at least two (2) full threads beyond nut.

(11) Torque screws 4.5 inch-pounds (0.505 N·m) starting at 12 o'clock in a diagonal sequence such as; 12 to 6, 3 to 9, etc.

(12 Using same sequence torque screws 15 to 25 inch-pounds (1.69 to 2.82 N·m).

(13) Apply sealant to outside fill space between outer window retainer and aircraft structure.

(14) On applicable windows, install desiccant bag tubing until flush with inner perimeter of retainer and bond with EC2216 (3M) or equivalent epoxy base adhesive. Insert end of tube inside desiccant bag and seal airtight with #472 (3M) tape. Attach bag to aircraft frame with #472 (3M) tape.

(15) Remove masking from outer window pane and clean window.

(16) Perform pressurization check IAW Adjustment/Test – Pressurization, Chapter 21, this manual.

(17) Install Royalite panel and cover clips (applicable aircraft).

(18) Install interior Royalite paneling.

H. Removal – SA226 – T Cabin Windows (Inner Panel)

(1) Remove interior Royalite paneling from around window panel.

(2) Mark location of panel clips for reference during installation. Remove clips.

(3) Remove inner retainer nuts.

(4) Remove inner retainer screws.

(5) Remove inner window panel.

NOTE: It may be necessary to cut through the sealant around the outer edges of the panel with a sharp bladed instrument in order to help loosen the sealant.

(6) Clean inner retainer of all sealant residue with MIL-C-38736 cleaning compound.

I. Required Equipment

(1) Hand drill motor, high speed, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed.

(3) Drill bit, 0.166 inch and 0.098 inch, for drilling acrylic.

(4) Torque wrench.

(5) Countersink bit, 0.156 inch.

(6) Counterbore bit, 0.196 inch.

(7) Counterbore bit, 0.250 inch.

(8) Rasp.

(9) Cleco fasteners or draw-mono-locks.

(10) Sealant, MIL-S-8802.

J. New Window Preparation

(1) Inspect new inner window panel for distortion and general condition. Mask for protection.

(2) The new inner window is received previously routed. Therefore, no routing procedure is required.

K. Trimming

(1) Locate and cut out slot in inner retainer for desiccant bag tube (if applicable).

(2) Place new inner window in position within the aircraft structure.

(3) Determine amount of trimming required for proper fit. Allow 0.030 in (0.762 mm) space between edge of window panel and aircraft structure to prevent fretting.

(4) Trim new window panel where required with a rasp. Do not heat up edges.

(5) Smooth and polish rough edges and marks.

L. Drilling

CAUTION: DRILL A CLEAN HOLE, FREE OF CHIPS, VENTS, AND RESIDUAL STRESSES. NEVER DRILL ACRYLIC IN CONJUNCTION WITH METAL. METAL CHIPS WILL SCRATCH ACRYLIC.

(1) Drill retainer screw holes.

(2) Deburr holes and break sharp edges in inner retainer.

(3) Cover pane with Nomar Pressure Sensitive Paper (3M Company) or Permacel 01 Pressure Sensitive Paper (Permacel Tape Corp.) adhesive back paper to prevent surface damage.

(4) Observe the following when drilling inner window retainer.

(a) Use 0.166 inch drill bit (135° high speed, New York Twist), or equivalent.

NOTE: Save old pane and practice drilling to get the feel of the drill motor, bit, counterbore, and general operation. Try several different procedures and select the one that's best for you.

(b) Hold drill perpendicular to pane surface.

(c) Use a counterbore to enlarge holes to final size.

(d) Carefully remove ridges around holes with fine sandpaper.

(5) Hold pane in retainer. Slowly drill one (1) hole at 3 o'clock position and secure with Cleco, or draw-mono-lock.

(6) Drill one (1) hole at 9 o'clock position and secure with Cleco or draw-mono-lock.

(7) Drill one (1) hole at 12 o'clock position and secure with Cleco, or draw-mono-lock.

(8) Drill one (1) hole at 6 o'clock position and secure with Cleco, or draw-mono-lock.

(9) Start drill remaining holes. After all holes are marked, move pane to work bench.

(10) Position window pane on work bench. Back up window pane with a wooden block or similar material.

(11) Drill all holes applying no feed pressure. Allow drill to lightly feed itself through panel.

(12) Enlarge holes to 0.250 inch using counterbore with appropriate pilot. Again start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately one third (1/3) material thickness.

(13) Turn window pane over and finish counterboring holes.

(14) Use sandpaper #400 or countersink to remove any ridges around hole edges.

(15) Align new inner retainer with old inner retainer. Locate hole pattern and transfer to new retainer using old retainer as a template.

M. Drilling Purge Holes

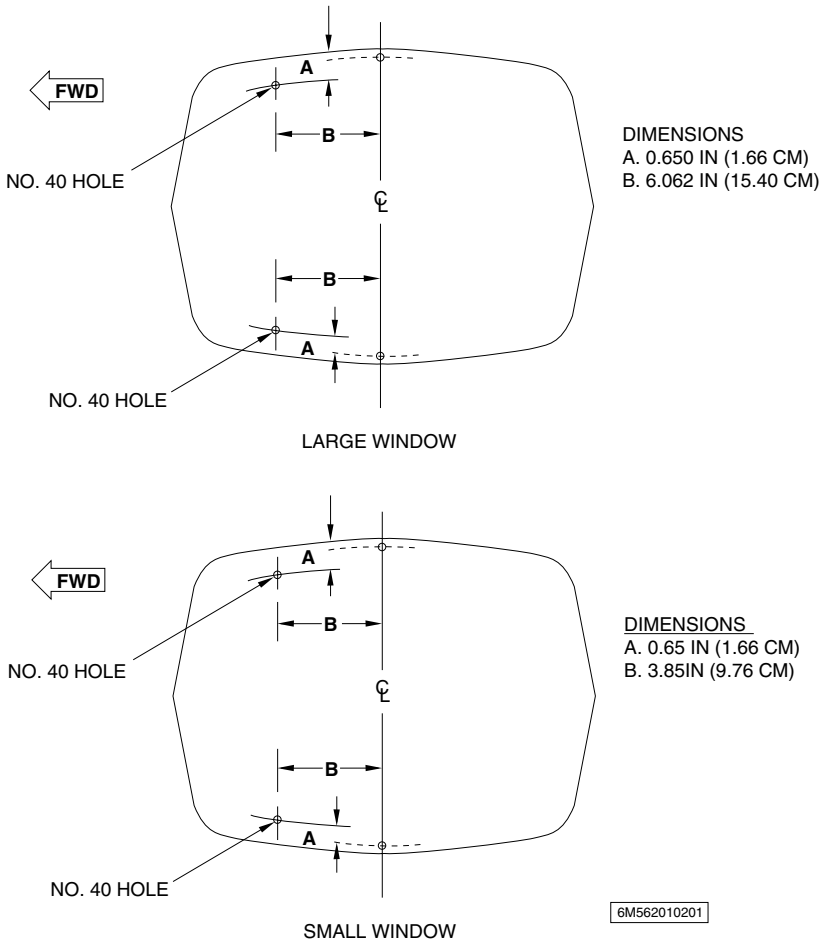
(1) On SN 249 and up, drill two holes through the inner panel on the large aft window, and two small left and right aft windows for cabin air circulation.

(2) On the large right aft window, determine the centerline of the window. The holes are to be drilled 6.062 in (15.40 cm) forward of the centerline and 0.65 in (1.66 cm) from the screw hole centerline toward the center of the window (see Figure 201).

(3) Using a 0.098 inch drill bit, drill two (2) holes through the inner panel.

(4) On the two smaller left and right aft windows, the holes are to be drilled 3.85 in (9.76 cm) forward of the window centerline and 0.65 in (1.66 cm) from the screw hole centerline toward the center of the window (see Figure 201).

(5) Using a 0.098 inch drill bit, drill two holes through the inner panel.



Drilling Purge Holes
Figure 201

N. Deburring

(1) Use a 0.156 inch countersink bit to remove burrs from all holes on inner and outer side of window panel.

O. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent polish from falling inside the panels after installation and to keep sealant contaminant free.

(3) For hairline scratches, use polysand procedures IAW Polishing Procedures this chapter.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax; apply with fingertips using a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect visually for imperfections. If any damage is visible, repeat necessary steps.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

P. Installation – SA226 – T Cabin Windows (Inner Panel)

(1) Inspect new window pane for distortion and general condition.

(2) Clean faying surfaces of inner retainer with MIL-S-38736.

(3) Install new window seal with 08001 adhesive (3M). Leave a 0.120 in (3.05 mm) gap between ends of seal to match location of cutout in inner retainer for desiccant bag tube.

(4) Temporarily install a piece of tubing, rubber seal, or gasket material in desiccant bag tube cutout to keep sealant out of holes. Leave end of piece long enough to go through inner pane and retainer.

(5) Prepare a sufficient quantity of Class B-2 sealant IAW MIL-S-8802 to cover installation area.

(6) Apply thin coat of Class B-2 sealant to the inner retainer, and install filler.

(7) Remove masking from the outer side of the new inner window pane.

(8) Apply a thin coat of sealant to inner surface of filler, and install new inner window pane.

(9) Apply a thin coat of sealant to outer surface of inner retainer, and install retainer.

(10) Apply a thin coat of sealant around each screw prior to installing into screw holes. Install all screws, washers, and nuts.

NOTE: Check length of screws installed. Ensure protrusion of at least two (2) full threads beyond nut.

(11) Install all Royalite panel and cover clips.

(12) Install all inner retainer nuts.

(13) Torque all screws 4.5 inch-pounds (0.505 N·m) starting at 12 o'clock in a diagonal sequence such as; 12 to 6, 3 to 9, etc.

(14) Using same sequence as above, torque screws 15 to 25 inch-pounds (1.69 to 2.82 N·m).

(15 Remove temporary piece of material from desiccant bag tube hole.

(16) Install desiccant bag tube until end is flush with inner pane. Bond tube to inner pane with EC2216 (3M) or equivalent epoxy base adhesive.

NOTE: Ensure desiccant bag is dry before installation.

(17) Route tube straight down. Insert end of tube inside a 6 mil plastic bag filled with 0.25 pint of type IV Desiccant. Seal bag airtight to tube with 3472 (3M) tape, and tape bag to aircraft.

(18) Remove masking tape from inner side of window pane and clean window.

(19) Perform pressurization check IAW Adjustment/Test – Pressurization, Chapter 21, this manual.

(20) Install interior paneling.

SA226 – AT, TC CABIN WINDOWS – MAINTENANCE PRACTICES

1. Removal/Installation – SA226 – AT, TC Cabin Windows

A. Removal – SA226 – AT, TC Cabin Windows (Outer Panel)

NOTE: The maintenance procedures in section 56-20-20 apply to SA226 – AT 001-024, and SA226 – TC 201-212.

(1) Remove interior Royalite paneling from around window panel to be removed.

(2) On applicable windows, remove desiccant bag tubing and temporarily stow aside until required during installation procedure.

(3) Remove all inner retainer nuts.

(4) Remove Royalite panel clips.

NOTE: Prior to removal of clips, mark location for future reference during installation procedure.

(5) Remove all retaining screws.

NOTE: Removal of the inner window is not required in order to remove the outer window.

(6) Remove the outer window retainer.

(7) Remove outer window panel.

NOTE: It may be necessary to cut through the sealant around the outer edges of the panel with a sharp bladed instrument in order to help loosen the sealant.

WARNING: WEAR EYE PROTECTION AND GLOVES WHEN USING SAFETY WIRE TO CUT WINDOW SEALANT.

(a) Dig a small hole in the sealer and push through a piece of 0.020 stainless safety wire.

(b) Wrap ends around screwdriver or similar tool.

(c) With one person inside and one person outside, pull wire in a reciprocating motion through sealer until window is free. Remove outer panel.

(8) Clean window frame of all sealant residue with MIL-C-38736 cleaning solvent compound.

B. Required Equipment

(1) Hand drill motor, high speed, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed (15,000 RPM).

(3) Drill bit, 0.166 inch (135° high speed), or equivalent.

(4) Torque wrench.

(5) Countersink bit, 0.156 inch.

(6) Rasp file.

(7) Sheet metal “cleco” fasteners or draw-mono-locks.

(8) Sealant, MIL-S-8802 PR 1422 or equivalent.

(9) Solvents, MIL-S-38736.

(10 Plastic Polish.

(11) Auto Wax Vista (Local Purchase).

NOTE: To determine if a window needs to be replaced, refer to 56-00-00, Inspection/ Check Windows.

C. New Window Preparation

NOTE: Inspect new outer window panel for worthiness. Refer to 56-00-00, Inspection/Check-Windows. Mask for protection.

D. Trimming

(1) Place the new outer window into position in aircraft structure.

(2) Determine amount of trimming required for proper fit. Approximately 0.030 in (0.762 mm) space should be allowed between the edges of window panel and aircraft structure to prevent fretting.

(3) Trim window panel where required with a rasp. Do not heat up edges.

(4) Smooth and polish all rough edges and marks.

(5) Bevel inside edge of outer pane to 0.05 inch radius.

E. Drilling

CAUTION: DRILL A CLEAN HOLE FREE OF CHIPS, VENTS, AND RESIDUAL STRESSES. ALLOW DRILL TO DO WORK WITH LITTLE OR NO FORCE FROM OPERATOR. NEVER DRILL ACRYLIC IN CONJUNCTION WITH METAL. METAL CHIPS WILL SCRATCH ACRYLIC.

(1) Cover pane with Nomar Pressure Sensitive Paper (3M Company) or Permacel 01 Pressure Sensitive Paper (Permacel Tape Corp.) adhesive back paper to prevent surface damage.

(2) Align and hold pane in retainer. Slowly drill one (1) hole at 3 o'clock position and secure with cleco or draw-mono-lock.

NOTE: Use 0.166 inch drill bit (125° high speed), or equivalent.

(3) Drill one hole at 9 o'clock position and secure cleco or draw-mono-lock.

(4) Drill one hole at 12 o'clock position and secure cleco or draw-mono-lock.

(5) Drill one hole at 6 o'clock position and secure cleco or draw-mono-lock.

(6) Start drill remaining holes. After all holes are marked, move pane to work bench.

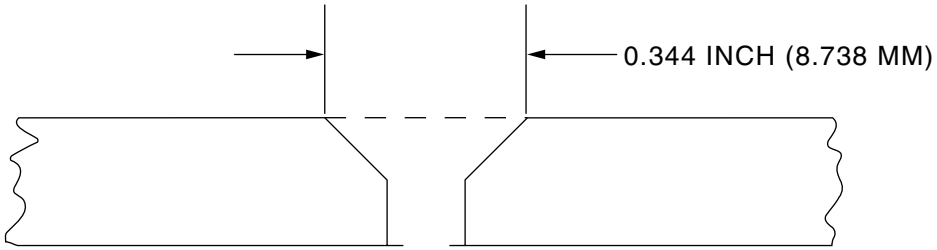
(7) Position pane on work bench. Back-up window with a wood block or similar material.

(8) Drill all holes applying no feed pressure. Allow drill to lightly feed itself through pane.

F. Countersinking

(1) Place the new window pane onto a drill press with the outerside facing up. Countersink all holes using a 0.156 inch countersink bit.

NOTE: The countersink shall have an included angle of 100° and shall be of the proper diameter to produce the degree of flushness required after the screws are installed. The countersink is correct when the diameter of the countersink measures 0.34 (11/32) inches across the diameter of each hole (see Figure 201).



Countersinking
Figure 201

G. Deburring

NOTE: Use a 0.156 inch countersink bit to remove burrs from all holes on inner and outer side of window panel.

H. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent polish from falling inside between the panels after installation and to keep sealant contaminant free.

(3) For hairline scratches, use polysand procedure steps **1.**, G., (5) through (16) Section 56-00-00.

(4) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax; apply with fingertips using a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect visually for imperfections. If any damage is visible, repeat necessary steps.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

SA226 – AT, TC CABIN WINDOWS – MAINTENANCE PRACTICES

I. Installation – SA226 – AT, TC Cabin Windows (Outer Panel)

NOTE: Inspect window for distortion and general condition.

(1) Apply a thin coat of sealant to the outer window spacer (aircraft primary structure).

(2) Remove masking from the inner side of the new outer panel.

(3) Position outer window panel in place in the aircraft structure.

(4) Apply a thin coat of sealant to the inner side of the outer retainer. Install outer retainer.

(5) Apply small amount of sealant around each screw. Install all screws.

NOTE: Check length of screws installed, Ensure protrusion of at least two (2) full threads beyond nut.

(6) Install all Royalite panel and cover clips.

(7) Install all inner retainer nuts.

(8) Torque screws 4.5 inch-pounds (0.505 N·m) starting at 12 o'clock in a diagonal sequence such as; 12 to 6, 3 to 9 etc.

(9) Using same sequence torque screws 15 to 25 inch-pounds (1.69 to 2.82 N·m).

(10) Apply sealant to outside fill space between the outer window retainer and the aircraft frame.

(11) On applicable windows, install desiccant bag tubing until flush with inner perimeter of retainer and bond with EC2216 (3M) or equivalent epoxy base adhesive. Insert end of tube inside desiccant bag and seal airtight with #472 (3M) tape. Attach bag to aircraft frame with #472 (3M) tape.

(12) Remove masking from outer side of window panel and clean window (see Section 56-00-00, Maintenance Practices).

(13) Install interior Royalite paneling.

J. Removal – SA226 – AT, TC Cabin Windows (Inner Panel)

(1) Remove interior Royalite paneling from around window panel to be removed.

(2) Remove all inner retainer nuts.

(3) Remove Royalite panel clips.

NOTE: Prior to removal of clips, mark location for future reference during installation procedure.

(4) Remove all retaining screws.

NOTE: Removal of the outer window is not required in order to remove the inner window.

(5) Remove the inner window retainer.

(6) Remove inner window panel.

NOTE: It may be necessary to cut through the sealant around the outer edges of the panel with a sharp bladed instrument in order to help loosen the sealant.

WARNING: WEAR EYE PROTECTION AND GLOVES WHEN USING SAFETY WIRE TO CUT WINDOW SEALANT.

(a) Dig a small hole in the sealer and push through a piece of 0.020 stainless safety wire.

(b) Wrap ends around screwdriver or similar tool.

(c) With one person inside and one person outside, pull wire in a reciprocating motion through sealer until window is free. Remove outer panel.

(7) Clean window frame of all sealant residue with MIL-C-38736 cleaning solvent compound.

K. Required Equipment

(1) Hand drill motor, high speed, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed (15,000 RPM).

(3) Drill bit, 0.166 inch (135° high speed), or equivalent.

(4) Torque wrench.

(5) Countersink bit, 0.156 inch.

(6) Rasp file.

(7) Sheet metal “cleco” fasteners or draw-mono-locks.

(8) Sealant, MIL-S-8802 PR 1422 or equivalent.

(9) Solvents, MIL-S-38736.

(10 Plastic Polish.

(11) Auto Wax Vista (Local Purchase).

NOTE: To determine if a window needs to be replaced, refer to 56-00-00, Inspection/ Check Windows.

L. New Window Preparation

(1) Inspect new outer window panel for worthiness. Refer to 56-00-00, Inspection/Check-Windows. Mask for protection.

(2) The new inner window is received previously routed. Therefore, no routing procedure is required.

M. Trimming

(1) Locate and cut out slot in inner retainer for desiccant bag tube (if applicable).

(2) Place the new outer window into position in aircraft structure.

(3) Determine amount of trimming required for proper fit. Approximately 0.030 in (0.762 mm) space should be allowed between the edges of window panel and aircraft structure to prevent fretting.

(4) Trim window panel where required with a rasp. Do not heat up edges.

(5) Smooth and polish all rough edges and marks.

N. Drilling

CAUTION: DRILL A CLEAN HOLE FREE OF CHIPS, VENTS, AND RESIDUAL STRESSES. ALLOW DRILL TO DO WORK WITH LITTLE OR NO FORCE FROM OPERATOR. NEVER DRILL ACRYLIC IN CONJUNCTION WITH METAL. METAL CHIPS WILL SCRATCH ACRYLIC.

(1) Cover pane with Nomar Pressure Sensitive Paper (3M Company) or Permacel 01 Pressure Sensitive Paper (Permacel Tape Corp.) adhesive back paper to prevent surface damage.

(2) Align and hold pane in retainer. Slowly drill one (1) hole at 3 o'clock position and secure with cleco or draw-mono-lock.

NOTE: Use 0.166 inch drill bit (125° high speed), or equivalent.

(3) Drill one hole at 9 o'clock position and secure cleco or draw-mono-lock.

(4) Drill one hole at 12 o'clock position and secure cleco or draw-mono-lock.

(5) Drill one hole at 6 o'clock position and secure cleco or draw-mono-lock.

(6) Start drill remaining holes. After all holes are marked, move pane to work bench.

(7) Position pane on work bench. Back-up window with a wood block or similar material.

(8) Drill all holes applying no feed pressure. Allow drill to lightly feed itself through pane.

O. Deburring

NOTE: Use a 0.156 inch countersink bit to remove burrs from all holes on inner and outer side of window panel.

P. Cleaning and Polishing

(a) Remove masking paper.

(b) Mask all holes on each side to prevent polish from falling inside between the panels after installation and to keep sealant contaminant free.

(c) For hairline scratches, use polysand procedure steps **1.**, G., (5) through (16) Section 56-00-00.

(d) Apply a thin coat of Permatex plastic polish No. 403D or Vista automobile wax; apply with fingertips using a circular motion. Buff clean with soft flannel cloth or cotton batting.

(e) Inspect visually for imperfections. If any damage is visible, repeat necessary steps.

(f) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

O. Installation – SA226 – AT, TC Cabin Windows (Inner Panel)

NOTE: Inspect window for distortion and general condition.

(1) Apply a thin coat of sealant to the inner retainer.

(2) Remove masking from the outer side of the new inner panel.

(3) Position outer window panel in place in the aircraft structure.

(4) Apply small amount of sealant around each screw. Install all screws.

NOTE: Check length of screws installed, Ensure protrusion of at least two (2) full threads beyond nut.

(5) Install all Royalite panel and cover clips.

(6) Install all inner retainer nuts.

(7) Torque screws 4.5 inch-pounds (0.505 N·m) starting at 12 o'clock in a diagonal sequence such as; 12 to 6, 3 to 9 etc.

(8) Using same sequence torque screws 15 to 25 inch-pounds (1.69 to 2.82 N·m).

(9) Apply sealant to outside fill space between the outer window retainer and the aircraft frame.

(10) Remove masking from inner side of window panel and clean window (see Section 56-00-00, Maintenance Practices).

(11) Install interior Royalite paneling.

SA226 – AT, TC CABIN WINDOWS – MAINTENANCE PRACTICES

1. Removal/Installation – SA226 – AT, TC Cabin Windows

A. Removal – SA226 – AT, TC Cabin Windows (Outer Panel)

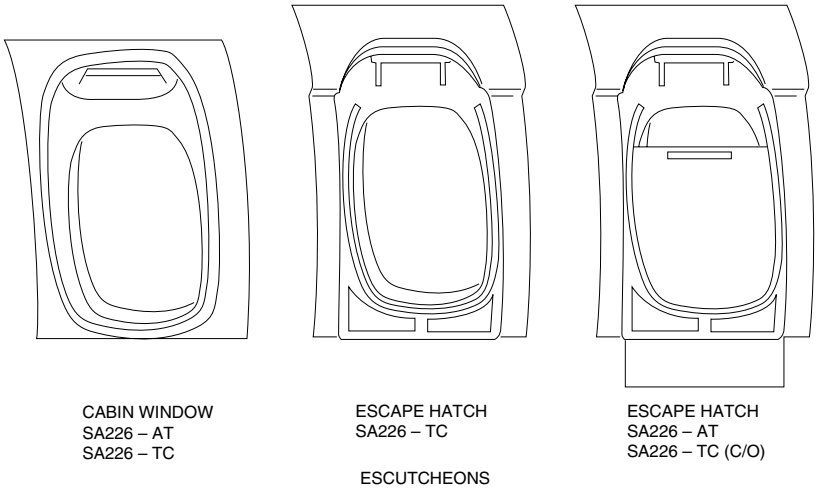
NOTE: The maintenance procedures in section 56-20-30 apply to SA226 – AT 025-999, and SA226 – TC 213-999.

(1) Remove interior Royalite paneling from around window panel to be removed.

(2) Mark location of panel clips for reference during installation. Remove clips.

(3) On applicable windows, remove desiccant bag tubing and temporarily stow aside until required during installation procedure.

(4) Remove the interior escutcheon and inner window assembly (on applicable aircraft), inspect for damage and store until ready for installation.



Escutcheons/Inner Retainers
Figure 201

(5) On applicable aircraft remove screws from each inner nutplate retainer.

(6) Remove the outer window retainer.

NOTE: It may be necessary to cut through the sealant around the outer edges of the window with a sharp bladed instrument to help loosen the sealant.

WARNING: WEAR EYE PROTECTION AND GLOVES, WHEN USING SAFETY WIRE TO CUT WINDOW SEALANT.

(7) If sealant will not release after cutting, proceed as follows:

(a) Dig a small hole in the sealer and push through a piece of 0.020 stainless steel safety wire.

(b) Wrap ends of wire around a screwdriver or similar tool.

(c) With one (1) person inside and another outside, pull wire in a reciprocating motion through sealer until window is free. Remove window.

(8) Clean window frame with MIL–C–38736 cleaning solvent compound.

B. New Window Preparation – Required Equipment

(1) Hand drill motor, high speed, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed (15,000 RPM).

(3) Special drill bits, 0.166 inch (135° high speed, New York Twist) or equivalent.

(4) Alternate drill bits, 0.173 inch, 0.172 inch, and 0.1695 inch.

(5) Torque Wrench.

(6) Countersink bit, 0.156 inch.

(7) Counterbore bit, 0.250 inch.

(8) Rasp.

(9) Cleco fasteners or wedge-lock fasteners.

(10) Sealant MIL-S-8802 or equivalent.

(11) Solvents, MIL-S-38736.

(12) Adhesive.

(13) Alodine, MIL-C-5541.

(14) Plastic Polish.

(15) Auto Wax Vista (Local Purchase).

(16) Zinc Chromate Primer TT-P-1756 (Local Purchase).

C. New Window Preparation

(1) Inspect new window for distortion and general condition. Mask for protection.

D. Trimming

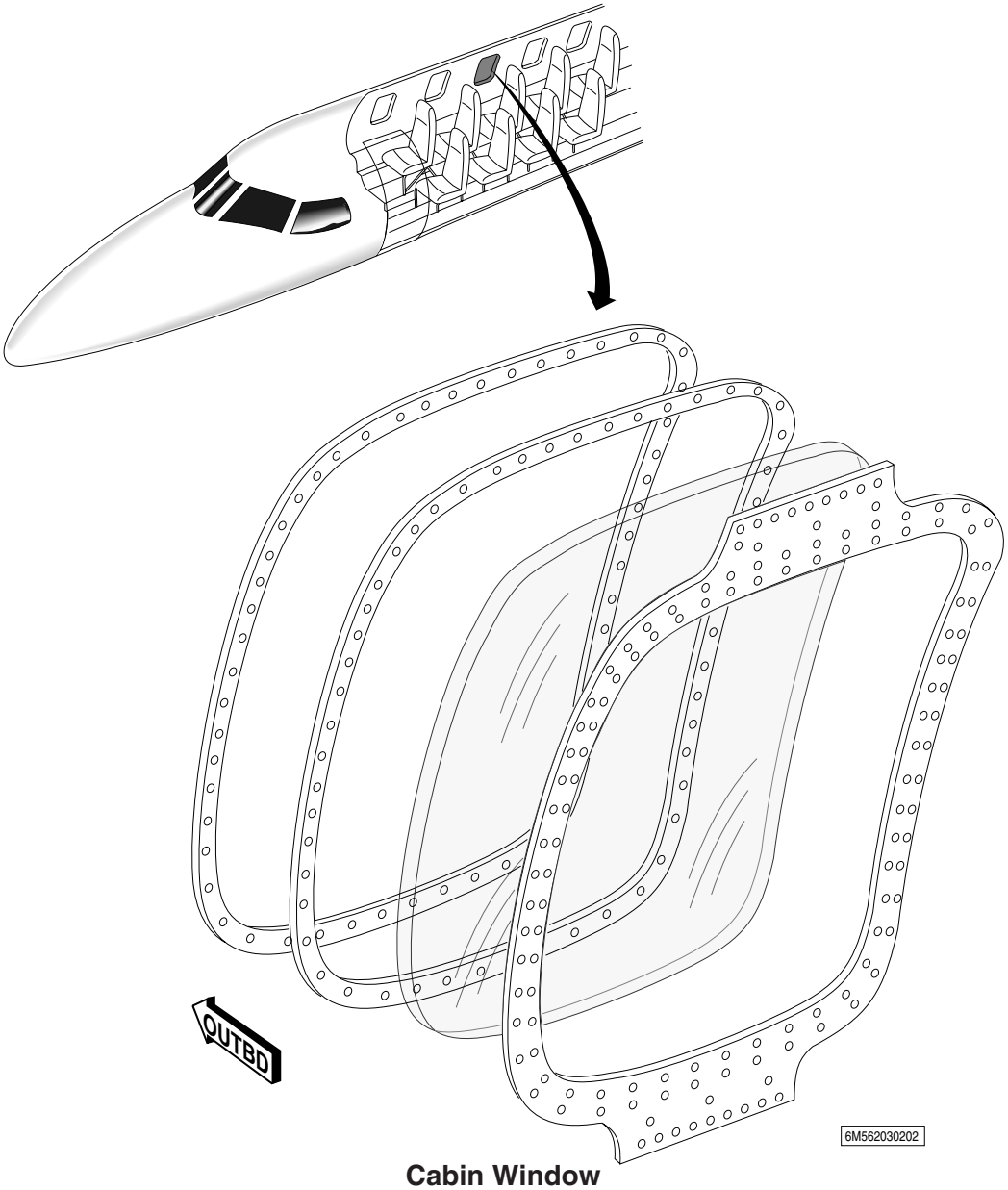
(1) Place the new outer window in position within aircraft structure.

(2) Determine amount of trimming required for proper fit. Allow approximately 0.030 in (0.762 mm) space between edges of window panel and aircraft structure to prevent fretting.

(3) Trim window panel where required with a rasp. Do not heat up edges.

(4) Smooth and polish all rough edges and marks.

(5) Bevel inside edge of outer pane to 0.05 inch radius.



Cabin Window
Figure 202

E. Drilling

CAUTION: DRILL A CLEAN HOLE FREE OF CHIPS, VENTS, AND RESIDUAL STRESSES. ALLOW DRILL TO DO WORK WITH LITTLE OR NO FORCE FROM OPERATOR. NEVER DRILL ACRYLIC IN CONJUNCTION WITH METAL. METAL CHIPS WILL SCRATCH ACRYLIC.

(1) Drill retainer screw holes in window pane.

(2) Deburr holes and break sharp edges in inner retainer.

(3) Cover pane with Normar Pressure Sensitive Paper (3M Company) or Permacel 01 Pressure Sensitive Paper (Permacel Tape Corp.) adhesive back paper to prevent surface damage.

(4) Observe the following when drilling inner window retainer:

(a) Use 0.166 inch drill bit (135° high speed, New York Twist), or equivalent.

(b) Save old pane and practice drilling holes to get the feel of the drill motor, bit, counterbore, and general operation. Try several different procedures and select the one that's best for you.

(c) Hold drill perpendicular to pane surface.

(d) Use a counterbore to enlarge holes to final size.

(e) Carefully remove ridges around holes with fine sandpaper.

(5) Hold pane in retainer. Slowly drill one (1) hole at 3 o'clock position and secure with cleco.

(6) Drill one (1) hole at 9 o'clock position and secure with cleco.

(7) Drill one (1) hole at 12 o'clock position and secure with cleco.

(8) Drill one (1) hole at 6 o'clock and secure with cleco.

(9) Start drill remaining holes. After all holes are marked, move pane to work bench.

(10) Position pane on work bench with wood back up.

(11) Drill all holes applying no feed pressure. Allow drill to lightly feed itself through pane.

(12) Enlarge holes to 0.250 inch using counterbore with appropriate pilot. Again start drill motor with no feed pressure allowing counterbore to lightly pull itself through to approximately 1/3 material thickness.

(13) Turn window pane over and finish counterboring holes.

(14) Use sandpaper #400 or countersink to remove any ridges around hole edges.

(15) Align new outer retainer with old outer retainer. Locate hole pattern and establish onto new retainer using old retainer as a template.

NOTE: The outer retainer may be locally manufactured. The new retainer is of thicker material with chamfered edges and the holes for the screw receptacles are countersunk.

F. Cleaning and Polishing

(1) Remove masking paper.

(2) Mask all holes on each side to prevent polish from falling between the panels.

(3) For hairline scratches, use polysand procedure IAW Polishing Procedures, this chapter.

(4) Apply a thin coat of Permatex plastic polish, or Vista automobile wax. Apply with fingertips using a circular motion. Buff clean with soft flannel cloth or cotton batting.

(5) Inspect visually for imperfections. If any damage is visible, repeat necessary steps.

(6) Remove masking from holes and mask remaining areas of panel with pressure sensitive masking paper.

G. Installation – SA226 – AT, TC Cabin Windows (Outer Panel)

NOTE: Inspect new cabin window for distortion and defects in material.

(1) Clean faying surfaces of inner retainer with MIL-S-38736.

(2) Prepare a sufficient quantity of class B-2 sealant IAW MIL-S-8802 to cover the installation area.

(3) Apply thin coat of Class B-2 sealant to inner retainer.

CAUTION: BE CAREFUL NOT TO SCRATCH WINDOW PANE.

(4) Install new bushings in 0.250 in (6.350 mm) holes in inner and outer panes. Trim bushings flush to match pane thickness. Make bushings from Imperial Eastman Poly-Flow tubing No. 44-PP.

(5) Position window pane in place in the aircraft structure.

(6) Apply thin coat of sealant to the inner side of the outer retainer. Install retainer in position.

(7) Apply small amount of sealant around each screw. Install all screws, washers, and nuts.

NOTE: Check length of screws installed. Ensure protrusion of at least two (2) full threads beyond nut.

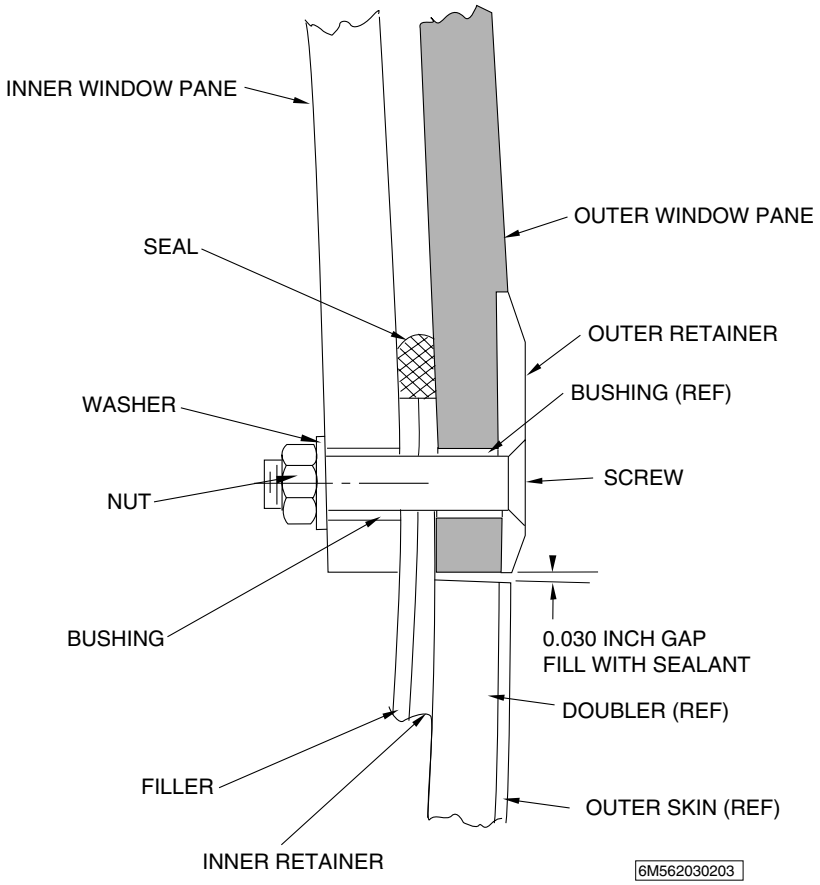
(8) Torque all screws 4.5 inch-pounds (0.505 N·m) starting at 12 o'clock in a diagonal sequence such as; 12 to 6, 3 to 9, etc.

(9) Using same sequence, torque screws 15 to 25 inch-pounds (1.69 to 2.82 N·m).

(10) Apply sealant to outside fill space between outer window retainer and aircraft structure.

(11) On applicable windows, install desiccant bag tubing until flush with inner perimeter of retainer and bond with EC2216 (3M) or equivalent epoxy base adhesive. Insert end of tube inside desiccant bag and seal airtight with #472 (3M) tape. Attach bag to aircraft frame with #472 tape.

(12) Remove masking from outer window pane and clean window.



Dual Window Pane
Figure 203

(13) Perform pressurization check IAW Adjustment/Test – Pressurization, Chapter 21, this manual.

(14) Install Royalite panel and cover clips (applicable aircraft).

(15) Install window escutcheon on (applicable aircraft).

(16) Install interior Royalite paneling.

H. Removal – SA226 – AT, TC Cabin Windows (Inner Panel)

(1) Remove interior paneling from around window panel to be removed.

(2) Mark location of panel clips for reference during installation. Remove clips.

(3) Remove interior escutcheon.

(4) Remove inner window retainer assembly.

(5) On applicable aircraft remove screws from each inner nutplate retainer.

(6) Remove inner retainer from retainer assembly. by drilling rivets using a 0.125 drill bit.

(7) Remove the inner window panel.

(8) Clean window frame with MIL-C-38736 cleaning compound.

I. New Window Preparation – Required Equipment

(1) Hand drill motor, high speed, 15,000-25,000 RPM.

(2) Drill press, automatic constant feed.

(3) Drill bit, 0.125 inch.

(4) Riveting tools.

(5) Countersink bit, 0.156 inch.

(6) Sealant, MIL-S-8802.

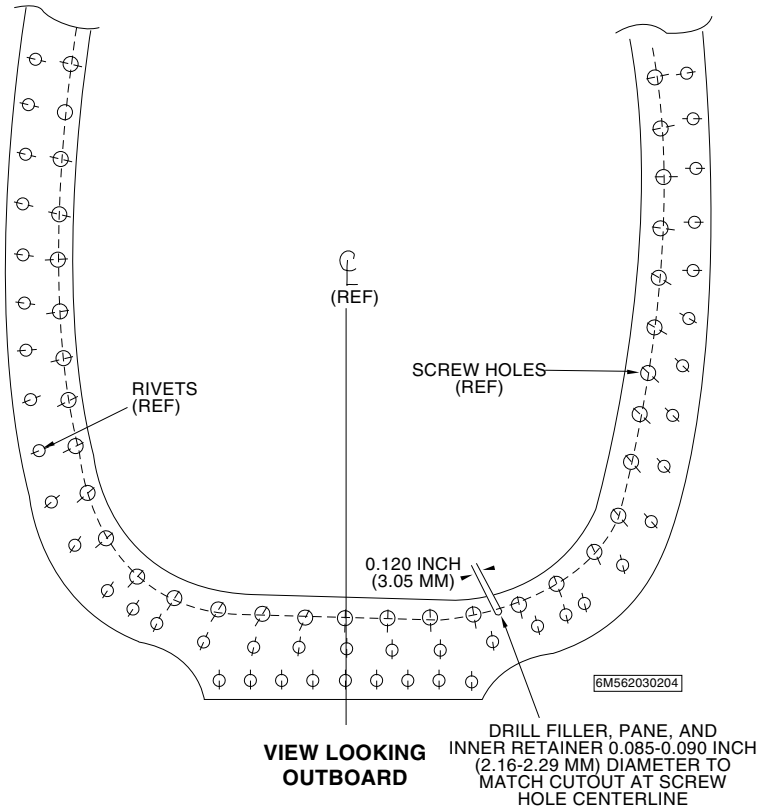
J. New Window Preparation

(1) Inspect new inner window panel for distortion and general condition. Mask for protection.

(2) The new inner cabin window does not require drilling, trimming, or routing except for desiccant bag tube slot.

K. Trimming

(1) Locate and cut out slot in inner retainer for desiccant bag tube (applicable windows).



Desiccant Tube Cutout
Figure 204

L. Drilling Purge Holes

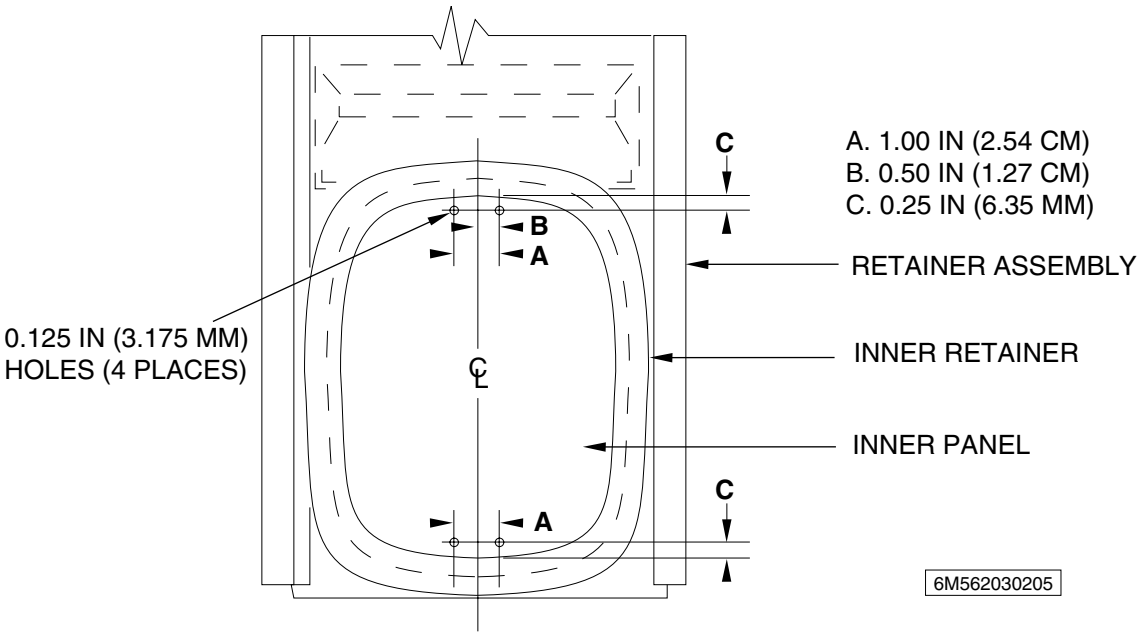
NOTE: On AT048 and up, and TC217 and up, four (4) 0.125 inch holes must be drilled in all inner panels except those installed in escape hatches.

(1) Place new window panel in inner retainer assembly, and place inner retainer on new window panel.

(2) Measure and mark positions for the four purge holes (see Figure 205).

(3) Drill four (4) 0.125 inch holes at the positions marked.

(4) Use countersink to deburr hole edges on inner and outer side of pane.



Drilling Purge Holes
Figure 205

M. Deburring

(1) Use sandpaper #400 or countersink to remove any ridges around hole edges.

N. Installation – SA226 – AT, TC Cabin Windows (Inner Panel)

(1) Clean faying surfaces of inner retainer with MIL-S-38736.

(2) Install new window seal with 08001 adhesive (3M). Leave 0.120 in (3.05 mm) gap between ends of seal to match location of cutout in inner retainer for desiccant tube cutout (applicable windows).

CAUTION: BE CAREFUL NOT TO SCRATCH WINDOW PANE.

(3) Install bushings in all 0.250 in (6.350 mm) holes in inner pane. Trim bushings flush to match pane thickness. Make bushings from Imperial Eastman Poly-Flow tubing No. 44-PP.

(4) To keep sealant out of desiccant bag tube holes, temporarily install a piece of tubing, rubber seal, or gasket material in the cutout. Leave end of piece long enough to go through filler, inner pane, and retainer (applicable windows).

(5) Prepare a sufficient quantity of sealant to cover the installation area.

(6) Apply thin coat of sealant to both inner and outer fay surfaces of the inner retainer.

(7) Install filler.

(8) Apply thin coat of sealant to inner fay surface of filler.

(9) Install inner pane.

(10) Apply thin coat of sealant to outer fay surface of inner retainer.

(11) Apply thin coat of sealant to inner fay surface of outer retainer. Install retainer.

(12) Apply small amount of sealant around each screw. Install screws and nuts (see Figure 203).

NOTE: Check length of screws installed. Ensure protrusion of at least two (2) full threads beyond nut.

(13) Torque all screws 4.5 inch-pounds (0.505 N·m) starting at 12 o'clock in a diagonal sequence such as; 12 to 6, 3 to 9, etc.

(14) Using same sequence, torque screws 15 to 25 inch-pounds (1.69 to 2.82 N·m).

(15) Apply sealant to outside fill space between outer window retainer and aircraft structure.

(16) Remove temporary piece of material from desiccant bag tube slot (applicable windows).

(17) Install desiccant bag tubing until flush with inner perimeter of retainer and bond with EC2216 (3M) or equivalent epoxy base adhesive. Insert end of tube inside desiccant bag and seal airtight with #472 (3M) tape. Attach bag to airframe with #472 tape.

(18) Remove masking and clean window.

(19) Perform pressurization check IAW Adjustment/Test – Pressurization, Chapter 21, this manual.

(20) Install Royalite panel and cover clips.

(21) Install window escutcheon.

(21) Install interior Royalite paneling.

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MAINTENANCE MANUAL

CHAPTER 57 – WINGS

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57-40-10	MP	201	OCT 01/13	57-50-00	DO	2	APR 01/04
57-40-10	MP	202	OCT 01/13	57-50-00	DO	3	APR 01/04

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MAINTENANCE MANUAL

CHAPTER 57 – WINGS

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MAINTENANCE MANUAL

CHAPTER 57 – WINGS

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WINGS – DESCRIPTION AND OPERATION

1. General

A. The SA226-T, AT, and TC aircraft have a full cantilever one piece wing with a span of 46.25 feet. The wing consists of two outer panels which contain integral fuel cells and a center section which contains various system accessories.

B. The front and rear spars are continuous from tip to tip. Wing attachment fittings on the spars transmit structural loads to box formers and stringers in the fuselage.

C. The removable components are the wing tips and the fairings. The fairings contour the center section to the fuselage.

D. The flaps operate on hinge assemblies riveted to the rear spar. The ailerons are attached to the wing by three hinge brackets bolted to the rear spar.

1. Inspection/Check – Wings

A. Inspection – Wings

(1) Inspect fiberglass for signs of delamination, cracks, erosion, and elongated attach holes.

(2) Inspect sand damaged area and splice in fiberglass cloth and resin. (Sand to proper contour and refinish).

B. Plexiglass – Wing Tip Position Light

(1) Check plexiglass for distortion and cracks.

(2) Remove and replace if distorted. Polish out scratches with glass polish.

C. Aluminum Alloy

(1) Inspect for cracks, dents, scratches, buckles, corrosion, and separation of doublers.

(2) Inspect attachment bolts and screws for looseness, cracks, and corrosion.

(3) Inspect air scoops for cracks, erosion, and delamination.

(4) Inspect wing tips for cracks, dents, and signs of damage.

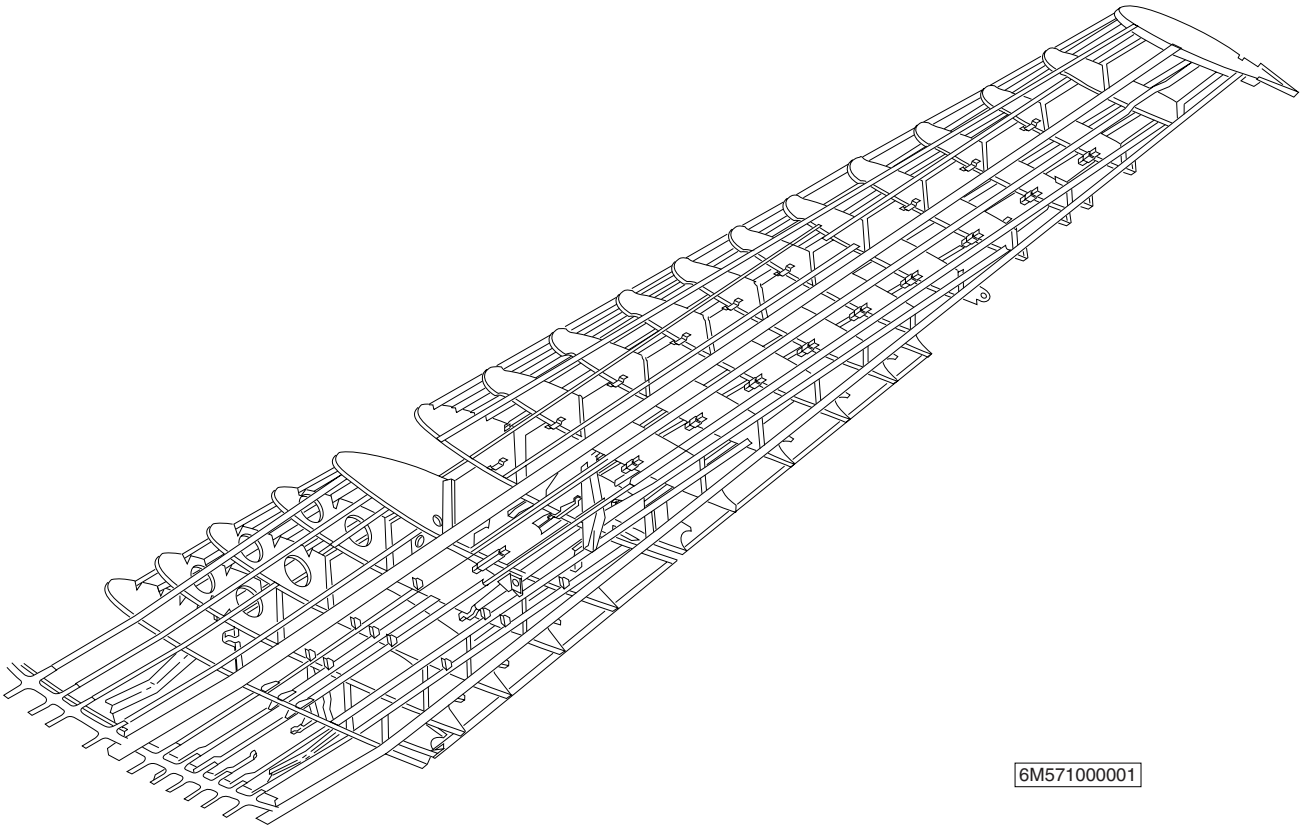
MAIN FRAME – DESCRIPTION AND OPERATION

1. General

A. The main frame of the wing consists of two spars, frames to form the airfoil, and stringers to add support to the wing skin, see Figure 1.

B. Spars

(1) The main and rear spars are the built-up I beam type and constructed of 2024-T3 aluminum, reinforced with T16AL-4V Titanium Alloy. The spars are continuous from tip to tip and pass under the fuselage pressure vessel. Steel reinforced fuselage attach points are located on each side of the fuselage under removable fairings.



Main Frame
Figure 1

C. Ribs

(1) The wing ribs are hydroformed in three sections and attached to the spars by one inch 90 degree angles. The ribs are made of 2024-0 aluminum alloy and heat treated. The interior ribs serve as baffles for the integral fuel tanks and at wing stations 132, 174, and 218 incorporate flapper valves to prevent the surge of fuel to the wing tip in a wing low attitude.

D. Stringers

(1) The stringers are made from 2014-T3511 extruded bulb angle. The two stringers on top and bottom of the leading edge are used to form the offset for the deicing boot. These two stringers are made from 2024-T3, .82 inches thick. The stringers are attached to the ribs by a .040 formed angle. The rivet holes are dimpled to accept the skin.

AUXILIARY STRUCTURES – DESCRIPTION AND OPERATION

1. General

A. Wing Tip

(1) The wing tip is a removable aluminum structure that houses the two position lights, the flux gate, and provides access to the lines for the fuel cell vent system. They are attached to the wing with No. 8 aluminum screws.

B. Trailing Edge

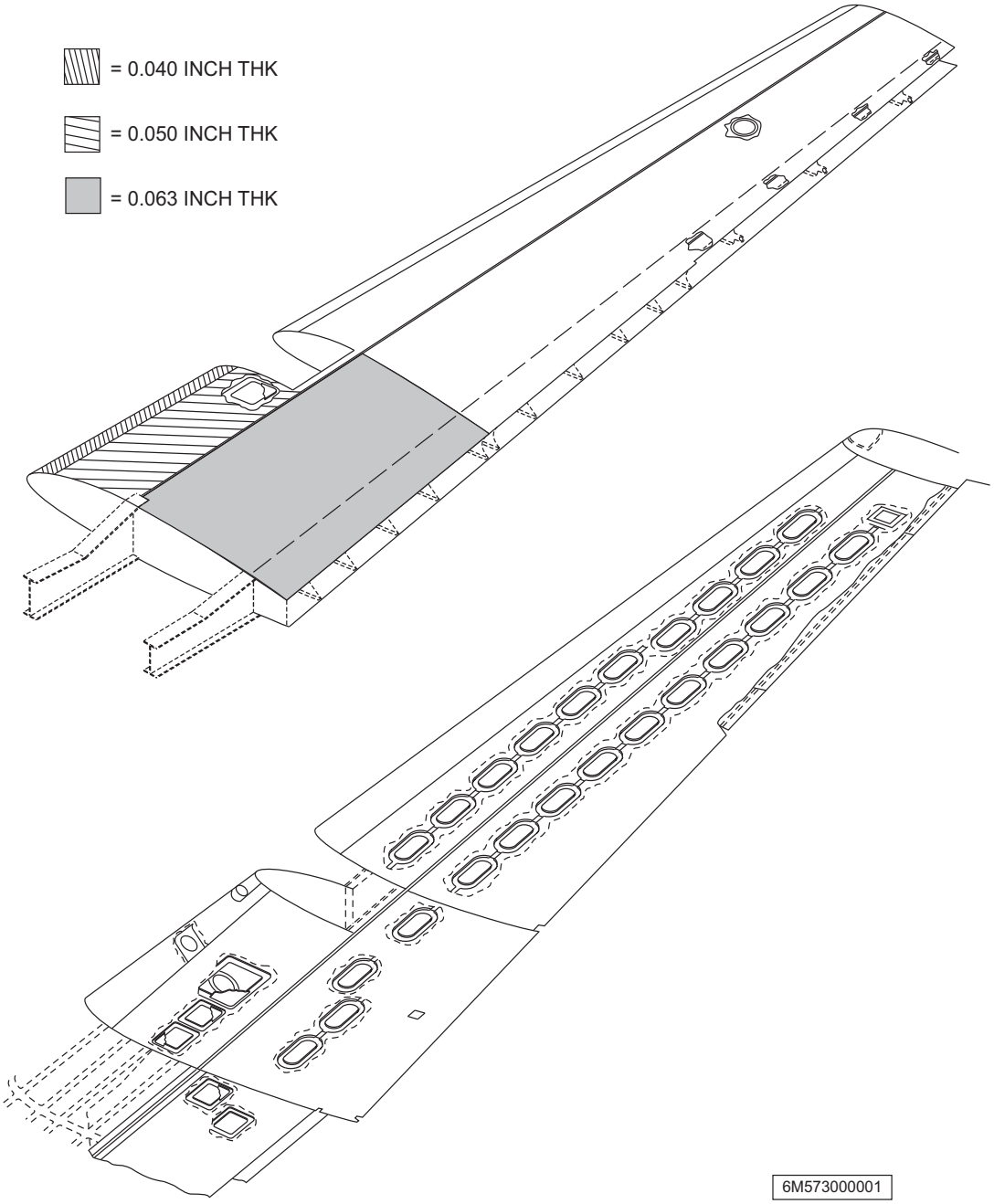
(1) The trailing edge houses the retaining structure for flight control surfaces, and the actuating mechanism for flaps and ailerons. The left wing housing the aileron trim tab control cables and actuator.

PLATES AND SKINS – DESCRIPTION AND OPERATION

1. General

A. Skin

(1) The skin used for the wing is chemically etched aluminum alloy. The thickness varies from 0.032 inches to 0.063 inches. Outboard of the nacelles the wing is 0.032 inches thick on top and bottom. The top inboard wing thickness varies. See Figure 1 for areas and thickness.



Plates and Skins
Figure 1

B. Access Panels (Figure 1)

(1) Forty eight access panels to the fuel cells are on the underside of the wing. The panels are installed by MS24694S3 screws into a doubler which retains sealed nutplates.

(2) There are eight access panels on the center section of the wing. The two forward on the front spar are used to cover air conditioning, deicing, and pressurization components. They are installed with AN509-8R8 screws. The four between the spars are for access to the fuel crossover tube and antennas and landing lights.

(3) The two large doors of the rear spar are used to cover and house the flap interconnect and aileron bellcrank. This door is attached to the airframe with AN509-8R8 screws.

(4) The doublers are attached to the wing with 1/8 rivets AN426-4. Between the doubler and the panel a 3/32 inch phenolic lightning arrester is installed.

C. Fairings

(1) The fairings used to contour the wing to the fuselage union are made of laminated fiberglass. The fairings extend from the front of the wing to the trailing edge. Components of various aircraft systems are located beneath the fairings. They are attached to the fuselage and wing with nut plates and screws.

PLATES AND SKINS – MAINTENANCE PRACTICES

A. Removal – Wing Access Panel

(1) Remove screws securing access panel to wing.

(2) Remove access panel.

B. Installation – Wing Access Panel

(1) Using a clean cloth moistened with methyl propyl ketone (MPK) clean access panel, phenolic seal, and wing skin recess.

(2) Wipe dry with second clean cloth.

(3) Check access panel and phenolic seal for cracks and general condition.

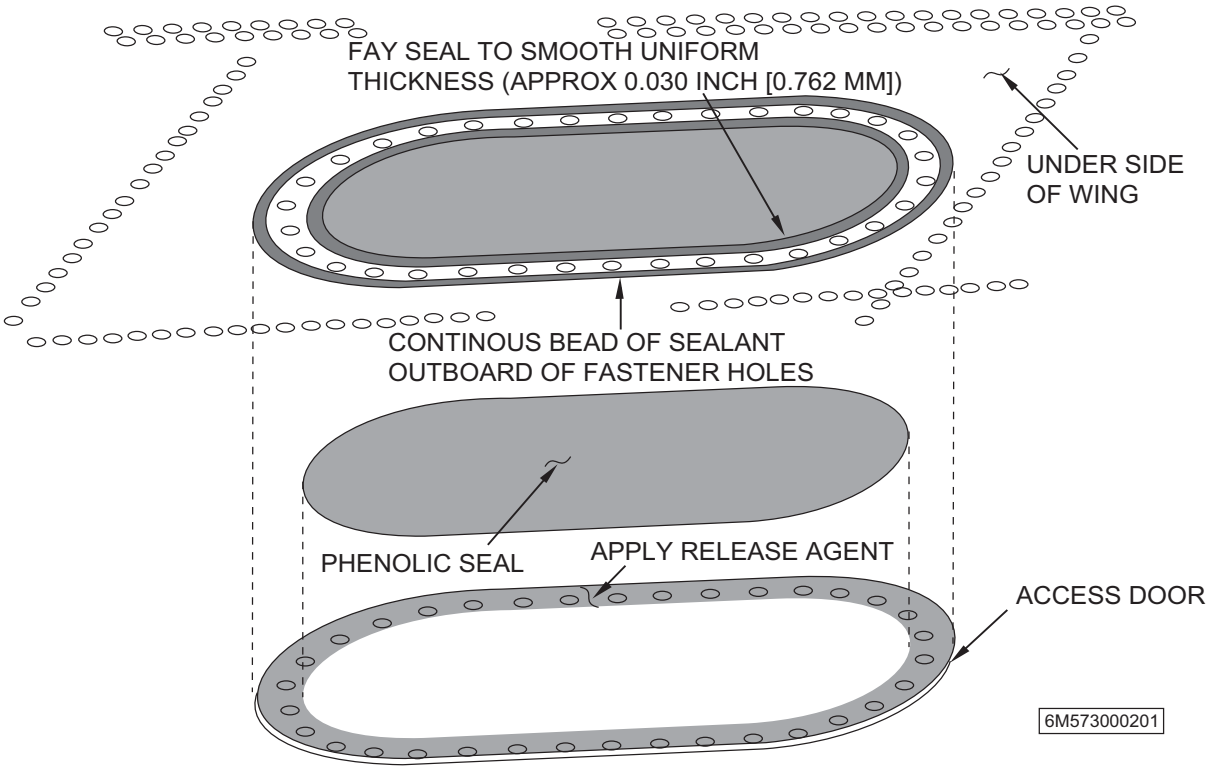
(4) Apply MIL-S-8802 Class B sealant to wing skin recess (Figure 201).

(5) Install phenolic seal onto wing skin recess. Ensure small excess of sealant is squeezed out in a continuous bead around periphery of seal.

(6) Apply release agent (Paraffin, or Mold Release 225) to access panel (Figure 201).

(7) Secure access panel to wing recess with screws. Ensure small excess of sealant is squeezed out in a continuous bead around periphery of joint.

(8) Remove excess sealant from exterior skin surface using cloth moistened with MPK.



Sealing Access Panels
Figure 201

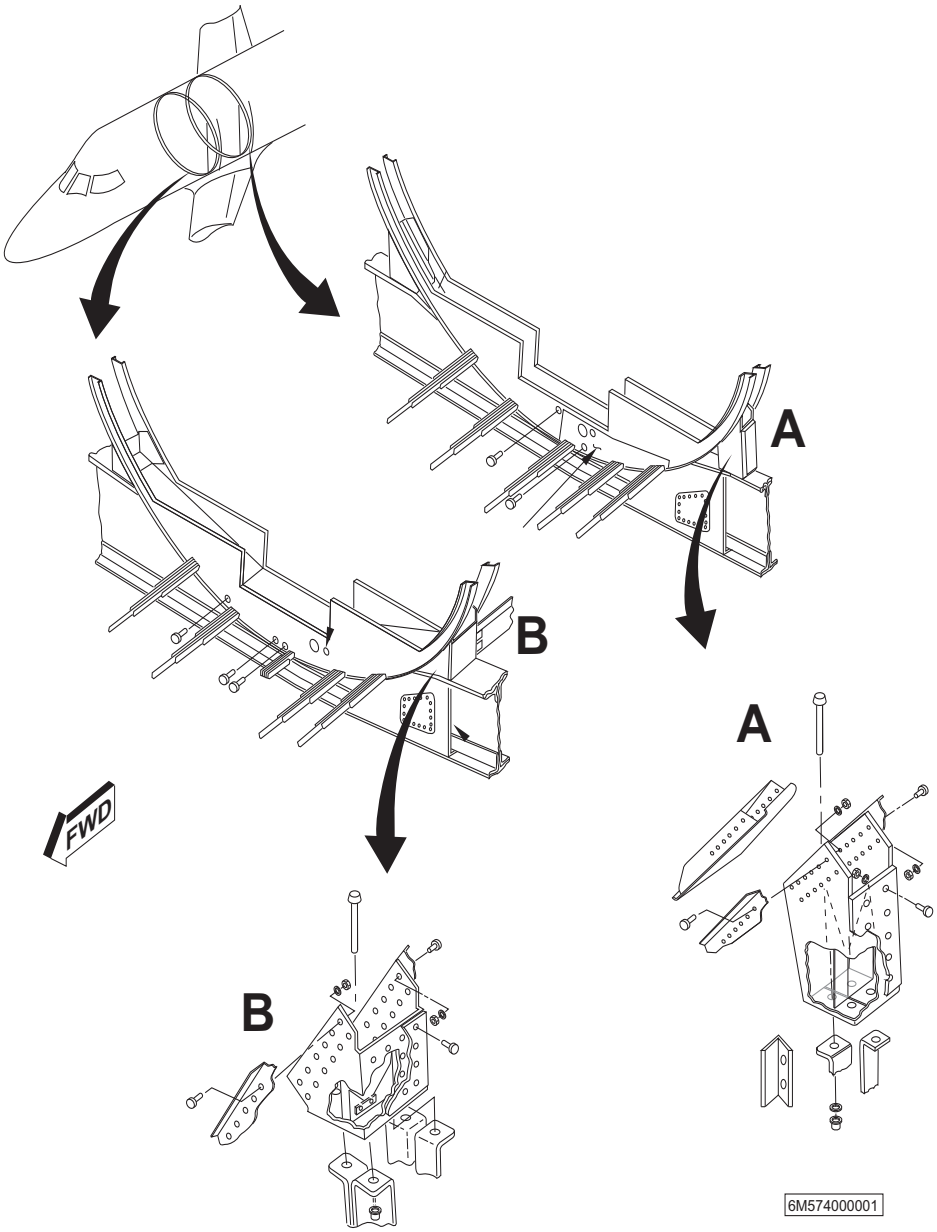
MAINTENANCE MANUAL

ATTACH FITTINGS – DESCRIPTION AND OPERATION

1. General

A. Wing To Fuselage (Figure 1)

(1) The wing is attached to the fuselage at four points on the main and rear spar. The attach fittings are made from routed-out aluminum billets. The fittings are attached to the wing with four 5/16" MS20005 bolts (main spar) and four 1/4" MS20004 bolts (aft spar). The fittings are in turn attached to the beltframes with NAS1103 bolts. There are additional bolts through the spar center section which attach the bellframes to the spar center section pressure plates. These bolts are secured in nut plates attached to the inside of the pressure plates.



Attach Fittings
Figure 1

ATTACH FITTINGS – MAINTENANCE PRACTICES

1. Removal/Installation - Wing-to-Fuselage Attach Fittings

A. Removal and Installation - Wing-to-Fuselage Attach Fittings

(1) Main Spar Fittings

(a) Remove electrical power from aircraft. Turn OFF main batteries and ensure ground power unit is not connected to the aircraft. Tag out circuit breakers and battery switches to prevent injury or damage to the aircraft.

(b) Remove interior furnishings, cabin seats, carpet, floorboards, and side panels as required to gain access to wing carry-thru area from inside the fuselage.

(c) Defuel aircraft and sump fuel in accordance with SA226 Maintenance Manual 12-30-15.

(d) Jack and shore aircraft in accordance with SA226 Maintenance Manual 07-10-00 and 07-20-00.

(e) Remove wing belly skins and wing-to-fuselage fairings fore and aft of the main spar. Note size and type of fasteners for reinstallation.

(f) Remove accessories, plumbing, and equipment as required for access to the main spar pressure plate outboard (approx. BL 22) access panels.

(g) Remove front and aft side main pressure plate access panels.

(h) Remove under-wing boost pump access panel. Remove boost pumps and plumbing as required for access to aft outboard fitting attach fastener

(i) Remove under wing access panel, forward of spar and immediately outboard of WS 27.

(j) Remove air conditioning duct valve, anti-ice valve, and water separator inside under-wing access panel. Bag and tag ends of tubes and parts for reinstallation.

(k) Remove inboard and outboard tension bolts through spar cap from main spar forward and aft fitting halves. The fasteners will be replaced with new hardware.

NOTE: Fastener heads may be ground off to facilitate removal. Take care not to damage fitting. Before punching fasteners through into the spar and wing interor, check to ensure clearance with other fasteners through spar and WS 27 rib webs below. Remove other fasteners as necessary to avoid damage.

(l) Remove fitting hardware through fuselage frame webs and retain for size and type.

(m) Remove 27-22118 Mount Assembly. Permissible to trim 27-22025 Doubler on inner frame cap in accordance with Figure 203 if necessary to remove Mount Assy. Retain shims between frame and fittings noting orientation and location for later installation.

(n) High frequency Eddy Current (HFEC) inspect the inside diameter of the open bolt holes (4) in the titanium upper spar cap per SA226/227 Structural Repair Manual (SRM) 51-30-07. Contact M7 Engineering if a crack is found.

(o) Build a tooling fixture to capture locations and hole alignment of holes in spar cap.

(p) Duplicate holes from fixture to new fitting halves. Hole size 0.312/0.316" diameter. Deburr. Drill holes per SA226/227 Structural Repair Manual (SRM) 51-40-07.

(q) Assemble the 27-22121 and 27-22123 fitting halves loosely to form the 27-22118 Mount Assembly (Figure 201). Do not pick up holes in fittings from splice plate until fittings are located following the next step.

(r) Install mount assembly in accordance with the following installation notes:

1 Ensure all four spar bolt holes are aligned through all components in stack-up.

2 Permissible to remove 0.060" material locally from underside of base of mount assy to match irregularities of mating spar caps. Fittings must maintain minimum base thickness of 0.290". Alodine and prime any reworked surfaces and bare edges per SA226/227 Structural Repair Manual (SRM) 51-30-02.

Ensure that more than one-half of forward and aft edge of mount assy is in contact with spar caps. Max allowable remaining gap is 0.020". Fill remaining gaps with Devcon-F per step 1.A.(1)(r)3.

3 Apply Devcon-F (putty type) between mating surfaces of mount assy and spar to fill any existing gaps less than 0.020" after mount assy is made to contact spar cap along more than half of both fwd and aft edges. Prior to application of Devcon-F, clean and treat mating surfaces with Devcon release agent.

4 Replace bolts and wet install per SA226/227 Structural Repair Manual (SRM) 51-30-03 through spar cap:

Inboard Bolts

MS20005-25	Bolt,
MS20002C5	Washer,
52LH4367-054	Nut,
4367-05	Washer

Outboard Bolts

MS20005-23	Bolt,
MS20002C5	Washer,
52LH4367-054	Nut,
4367-05	Washer

NOTE: Grip length of fasteners may vary depending on structure and sealant build up at each fastener location. Measure each location to ensure proper length.

5 Torque MS20005 bolts through spar cap to 190 - 220 in-lbs in accordance with SA226/227 Structural Repair Manual (SRM) 51-40-07 using a cross sequence tightening schedule (e.g., 12 o'clock to 6 o'clock, 3 o'clock to 9 o'clock, 12:05 o'clock to 6:05 o'clock, etc.). Overcoat all nuts in fuel tank with sealant per SA226/227 Structural Repair Manual (SRM) 51-30-03.

(t) Pick up existing fastener holes through frame webs. Hole size 0.190/0.194" diameter. Reinstall any shims between frame web and fittings that were existing in same location/orientation as previous. Shims shall be fay surface sealed per SA226/227 Structural Repair Manual (SRM) 51-30-03. Contact M7 Aerospace if new shims are required (see Note 4, Figures 201 and 202). Install new existing type (NAS1103) fasteners or install new NAS6203-( ) Bolt, NAS1149F0332P Washer (or any other NAS1149 cad-plated), and MS21042L3 Nut. Install fasteners per SA226/227 Structural Repair Manual (SRM) 51-40-07.

(u) Pick up existing fastener holes through 27-31264 Support Clips. Maintain fastener type and size. Install fasteners per SA226/227 Structural Repair Manual (SRM) 51-40-07.

(w) Pick up existing holes through the 27-22124 splice plate(s). Maintain 2D edge distance on new fittings. Install fasteners per SA226/227 Structural Repair Manual (SRM) 51-40-07. Fay surface seal the splice plates per SA226/227 Structural Repair Manual (SRM) 51-30-03. See Figure 201 for fastener type and hole size.

(x) M7 Aerospace recommends to install new hardware for all locations that hardware was removed for this modification.

(2) Aft Spar Fittings

(a) Remove electrical power from aircraft. Turn OFF main batteries and ensure ground power unit is not connected to the aircraft. Tag out circuit breakers and battery switches to prevent injury or damage to the aircraft.

(b) Remove interior furnishings, cabin seats, carpet, floorboards, and side panels as required to gain access to wing carry-thru area from inside the fuselage.

(c) Defuel aircraft and sump fuel in accordance with SA226 Maintenance Manual 12-30-15.

(d) Jack and shore aircraft in accordance with SA226 Maintenance Manual 07-10-00 and 07-20-00.

(e) Remove wing belly skins and wing-to-fuselage fairings fore and aft of the aft spar as required for access. Note size and type of fasteners for reinstallation.

(f) Remove accessories, plumbing, and equipment as required for access to the aft spar pressure plate outboard (approx. BL 22) access panels.

(g) Remove front and aft side aft spar pressure plate access panels.

(h) Remove under-wing boost pump access panel. Remove boost pumps and plumbing as required for access to forward outboard fitting attach fastener.

(i) Remove 27-71003 or 27-19097 Tube Assy (aileron control system).

(j) Remove flap walking beam. Remove flap.

(k) Remove or secure inboard flap hinge linkage parts to allow clear access to nut side of aft, outboard fitting bolt.

(l) Remove inboard and outboard tension bolts through spar cap from main spar forward and aft fitting halves. The fasteners will be replaced with new hardware.

NOTE: Fastener heads may be ground off to facilitate removal. Take care not to damage fitting. Before punching fasteners through into the spar and wing interor, check to ensure clearance with other fasteners through spar and WS 27 rib webs below. Remove other fasteners as necessary to avoid damage.

(m) Remove fitting hardware through fuselage frame webs and retain for size and type.

(n) Remove 27-22119 Mount Assy. Permissible to trim 27-22025 Doubler on inner frame cap in accordance with Figure 203 if necessary to remove Mount Assy. Retain shims between frame and fittings noting orientation and location for later installation.

(o) Build a tooling fixture to capture locations and hole alignment of holes in spar cap.

(p) Duplicate holes from fixture to new fitting halves. Hole size 0.250/0.254" diameter. Deburr. Drill holes per SA226/227 Structural Repair Manual (SRM) 51-40-07.

(q) Assemble the 27-22135 and 27-22136 fitting halves loosely to form the 27-22119 Mount Assembly (Figure 202). Do not pick up holes in fittings from 27-22137 splice until fittings are located following the next step.

(r) Install mount assembly in accordance with the following installation notes:

1 Ensure all four spar bolt holes are aligned through all components in stack-up.

2 Permissible to remove 0.060" material locally from underside of base of mount assy to match irregularities of mating spar caps. Fittings must maintain minimum base thickness of 0.290". Alodine and prime any reworked surfaces and bare edges per SA226/227 Structural Repair Manual (SRM) 51-30-02. Ensure that more than one-half of forward and aft edge of mount assy is in contact with spar caps. Max allowable remaining gap is 0.020". Fill remaining gaps with Devcon-F per step 1.A.(2)(r)3.

3 Apply Devcon-F (putty type) between mating surfaces of mount assy and spar to fill any existing gaps less than 0.020" after mount assy is made to contact spar cap along more than half of both fwd and aft edges. Prior to application of Devcon-F, clean and treat mating surfaces with Devcon release agent.

4 Replace 4 bolts and wet install per SA226/227 Structural Repair Manual (SRM) 51-30-03 through spar cap:

MS20004-22	Bolt, Aft, Outbd & Inbd
MS20004-24	Bolt, Fwd, Outbd
MS20004-26	Bolt, Fwd, Inbd
MS20002C4	Washer,
52LH4367-048	Nut,
RM2935-04	Washer

NOTE: Grip length of fasteners may vary depending on structure and sealant build up at each fastener location. Measure each location to ensure proper length.

5 Torque MS20004 bolts through spar cap to 100 - 140 in-lbs in accordance with SA226/227 Structural Repair Manual (SRM) 51-40-07 using a cross sequence tightening schedule (e.g., 12 o'clock to 6 o'clock, 3 o'clock to 9 o'clock, 12:05 o'clock to 6:05 o'clock, etc.). Overcoat all nuts in fuel tank with sealant per SA226/227 Structural Repair Manual (SRM) 51-30-03.

(s) Pick up existing fastener holes through frame webs. Hole size 0.190/0.194" diameter.

Reinstall any shims between frame web and fittings that were existing in same location/orientation as previous. Shims shall be fay surface sealed per SA226/227 Structural Repair Manual (SRM) 51-30-03. Contact M7 Aerospace if new shims are required (see Note 4, Figures 201 and 202). Install new existing type (NAS1103) fasteners or install new NAS6203-( ) Bolt, NAS1149F0332P Washer (or any other NAS1149 cad-plated), and MS21042L3 Nut. Install fasteners per SA226/227 Structural Repair Manual (SRM) 51-40-07.

(t) Pick up existing fastener holes through 27-30000-087/-088 Bracket on fwd side. Maintain fastener type and size. Install fasteners per SA226/227 Structural Repair Manual (SRM) 51-40-07.

(u) Pick up existing holes through the 27-22137 splice plate(s). Maintain 2D edge distance on new fittings. Install fasteners per SA226/227 Structural Repair Manual (SRM) 51-40-07. Fay surface seal the splice plates per SA226/227 Structural Repair Manual (SRM) 51-30-03. See Figure 202 for fastener type and hole size.

(v) M7 Aerospace recommends to install new hardware for all locations that hardware was removed for this modification.

B. Close Up

(1) Reinstall any fasteners through the spar web or WS 27 rib web that were removed to facilitate removal of fitting tension bolts through spar cap. Install same type and size in accordance with SA226/227 Structural Repair Manual (SRM) 51-40-07.

(2) Ensure area is fillet sealed and overcoat fasteners in wet fuel tank per SA226/227 Structural Repair Manual (SRM) 51-30-03.

(3) Restore collector tank, reseal fasteners, and close collector tank access panel.

(4) Reinstall spar access panels and center section equipment and systems per applicable sections of the SA226 Maintenance Manual.

(5) Reinstall air conditioning duct valve, anti-ice valve, and water separator and reinstall under-wing access panel.

(6) Remove shoring.

(7) Refuel aircraft and check for leaks. Refer to SA226 Maintenance Manual, Chapter 28-10-10.

(8) Reinstall air conditioning duct valve, anti-ice valve, and water separator and reinstall under-wing access panel.

(9) Reinstall wing-to-fuselage fairing and interior parts and components per applicable sections of the SA226 Maintenance Manual.

(10) If aft spar fittings were replaced, continue through step 1.B.(15).

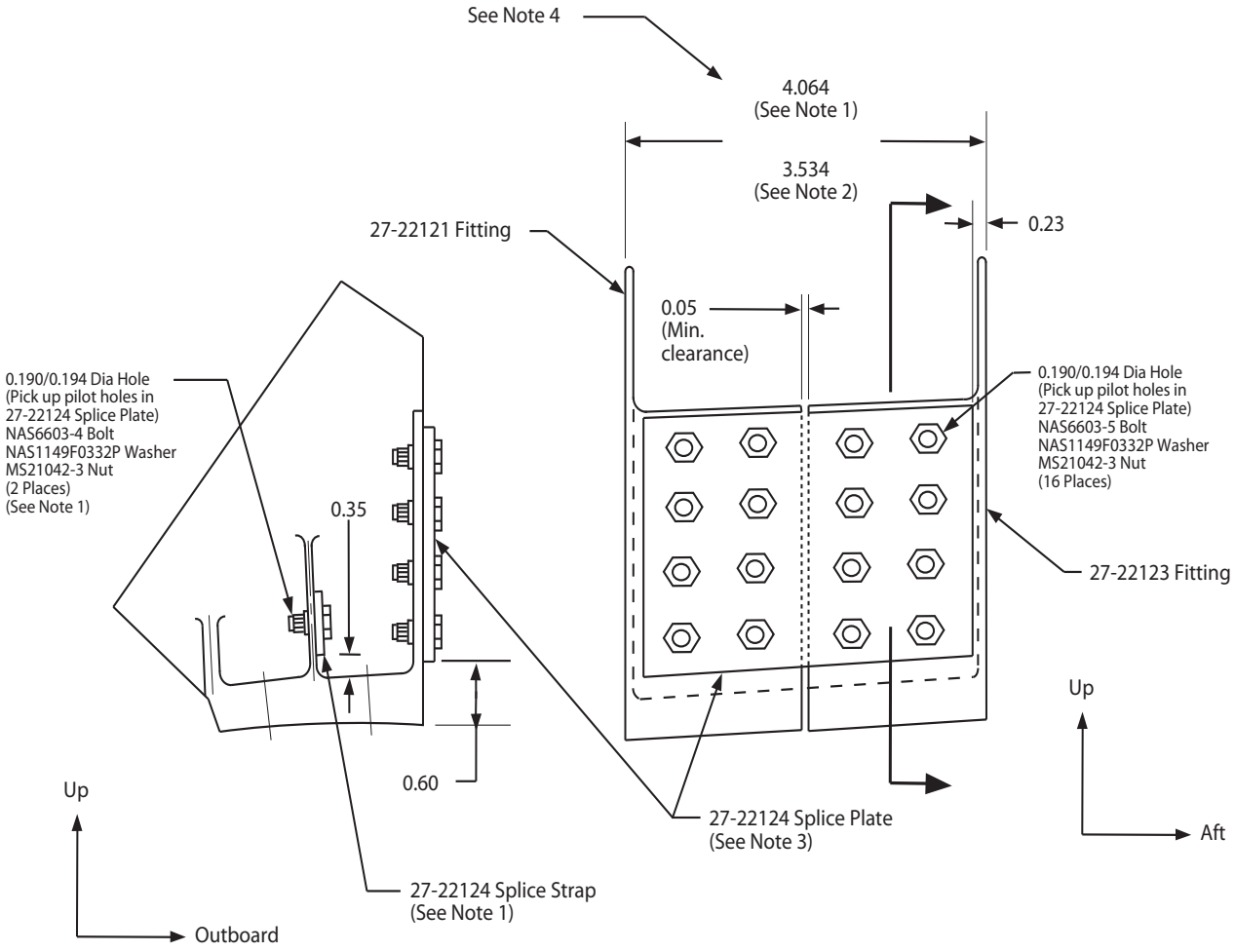
(13) Reinstall flap and flap walking beam.

(14) Reinstall 27-71003 or 27-19097 Tube Assy (aileron control system).

(15) Remove tags from battery switches and circuit breakers pulled in step 1.A.(1)(a) or 1.A.(2)(a). Connect GPU to aircraft.

(16) Perform aileron control system operational checks in accordance with SA226 Maintenance Manual 27-10-10.

(17) Perform flap control system operational checks in accordance with SA226 Maintenance Manual 27-50-10.



Note 1: Effective for SA226-T(B) 394 and up, TC 398 and up.

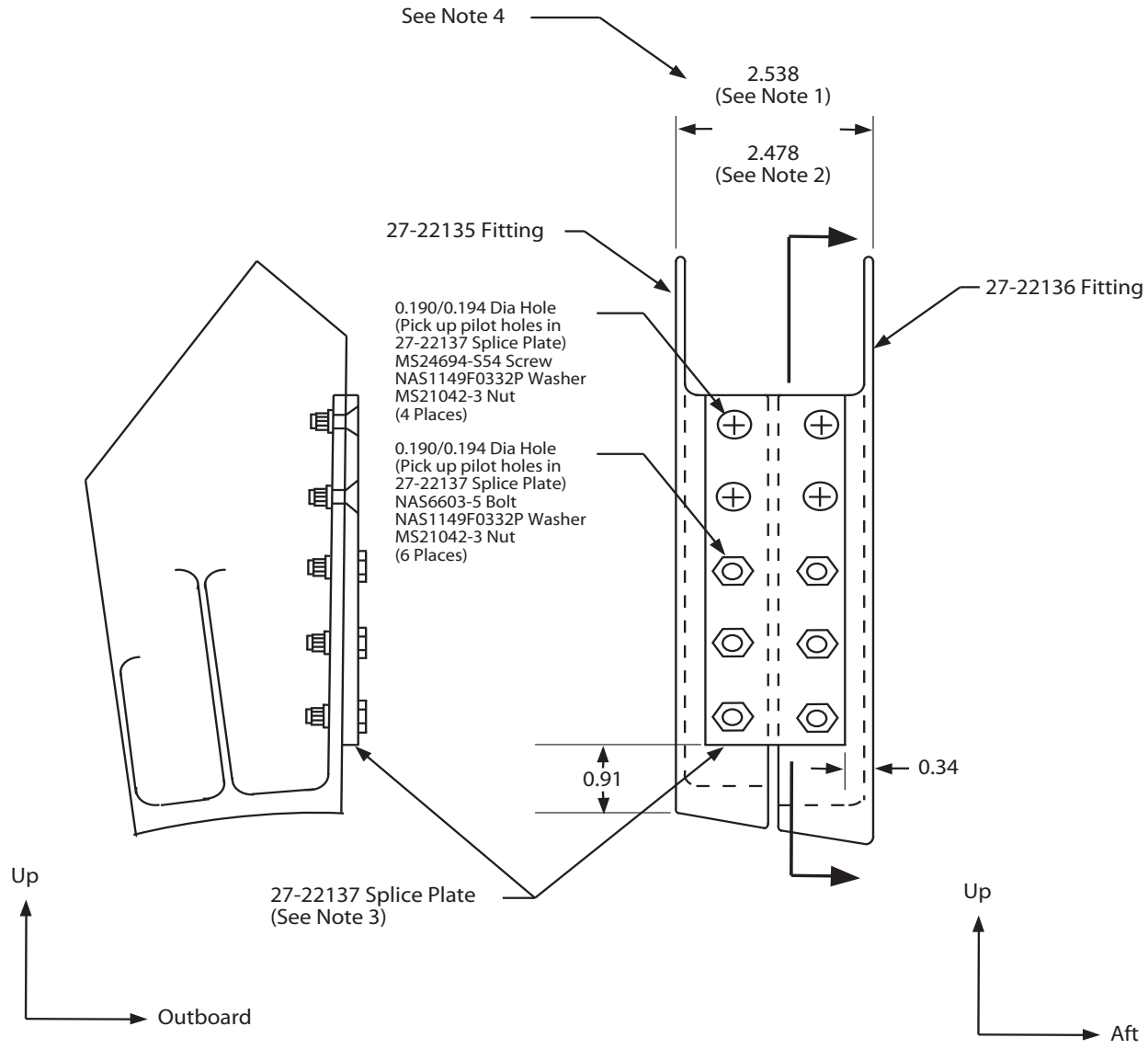
Note 2: Effective for all SA226 serial numbers not covered by Note 1.

Note 3: Fay surface seal in accordance with SA226/227 Structural Repair Manual 51-30-03.

Note 4: Dimensions shown are design parameters. Adjust to match aircraft conditions if necessary. Maintain 2D edge distance on all fastener locations.

Note 5: All dimensions are in inches.

Main Spar
(LH Shown, RH Opposite)
Figure 201



Note 1: Effective for SA226-T(B) 394 and up, TC 398 and up.

Note 2: Effective for all SA226 serial numbers not covered by Note 1.

Note 3: Fay surface seal in accordance with SA226/227 Structural Repair Manual 51-30-03.

Note 4: Dimensions shown are design parameters. Adjust to match aircraft conditions if necessary. Maintain 2D edge distance on all fastener locations.

Note 5: All dimensions are in inches.

Aft Spar
(LH Shown, RH Opposite)
Figure 202

Replacement Wing-to-Fuselage Attach Fitting Part Numbers					
Aircraft Model	Serial Number	Main Spar		Aft Spar	
		27-22121 Fwd Half	27-22123 Aft Half	27-22135 Fwd Half	27-22136 Aft Half
SA226-T SA226-T(B) SA226-TC SA226-AT	201 - 291 276 - 324 201 - 320 001 - 074	-013/-014	-013/-014	-007/-008	-007/-008
SA226-T(B) SA226-TC	325 - 391 321 - 396	-015/-016	-015/-016	-007/-008	-007/-008
SA226-T(B) SA226-TC	394 - 417 398 - 419	-019/-020	-019/-020	-007/-008	-007/-008

Table 201

Fitting-to-Spar Hardware and Alternates		
Location	Existing	Alternate [1]
Main Spar Inboard	MS20005-25 Bolt	MS21250-05025 Bolt [2]
	MS20002C5 Washer	
	52LH4367-054 Nut	NAS1727-5 Nut & Washer Combo [3]
	4367-05 Washer	
Main Spar Outboard	MS20005-23 Bolt	MS21250-05023 Bolt [2]
	MS20002C5 Washer	
	52LH4367-054 Nut	NAS1727-5 Nut & Washer Combo [3]
	4367-05 Washer	
Aft Spar, Aft Side	MS20004-22 Bolt	MS21250-04022 Bolt [2]
	MS20002C4 Washer	
	52LH4367-048 Nut	NAS1727-4 Nut & Washer Combo [3]
	RM2935-04 Washer	
Aft Spar, Fwd Side	MS20004-24 Bolt (Outbd)	MS21250-04024 Bolt [2]
	MS20004-26 Bolt (Inbd)	MS21250-04026 Bolt [2]
	MS20002C4 Washer	
	52LH4367-048 Nut	NAS1727-4 Nut & Washer Combo [3]
	RM2935-04 Washer	
Fitting-to-Frame Hardware and Alternates		
Location	Existing	Alternate
All Locations	LH7559-3 Nut	MS21042L3 Nut
All Locations	NAS1303 Nut NAS1304 Nut NAS1103 Nut	NAS6603 Nut NAS6604 Nut NAS6203 Nut

For Table NOTES, see page 210.

Table 202

Notes:

1. The Table permits the use of alternate hardware to the existing blueprint hardware. For example, an NAS1727-5 nut and washer combo can be used instead of a 52LH4367-054 nut and 4367-05 washer. Nut and washer combination cannot be used seperately. The NAS1727-5 nut and washer can be used with both MS20005-XX bolt or MS21250-05XXX bolt. Similarly, the 52LH4367-054 nut and 4367-05 washer may be used with the MS21250-05XXX bolt. Alternate parts vary in size from existing parts and may not fit depending on fastener location.

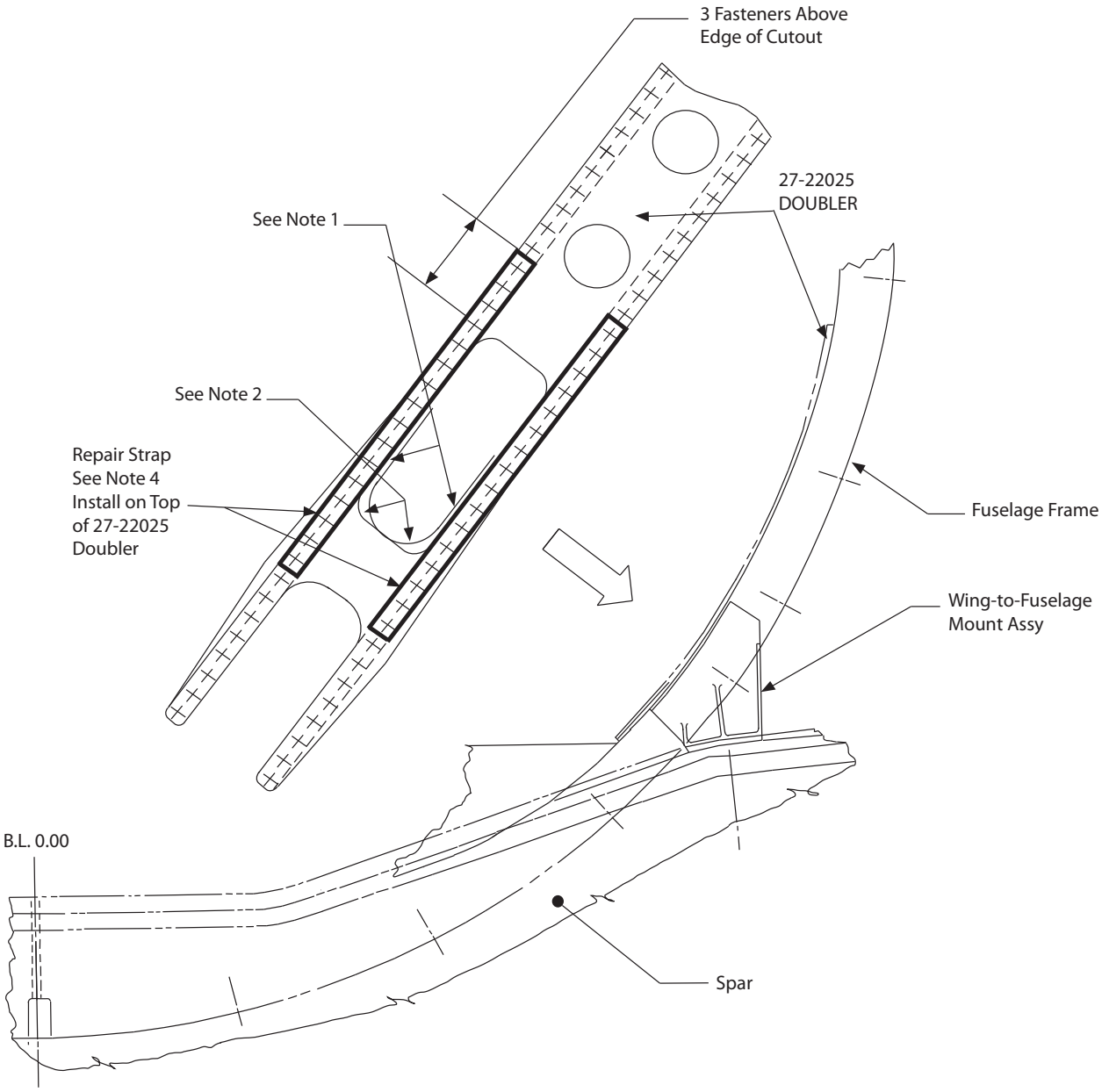
2. The four fasteners at one location must be of the same type. The four fasteners must be substituted together.

3. The NAS1727-4/-5 nuts a physically larger than the original nut/washer combination. This may result in requiring a longer grip length bolt to ensure correct safety on the bolt. See NOTE 5 for use of radius block if required.

4. Grip lengths of fasteners may vary depending on local conditions. Measure at each location to ensure proper length.

5. If the fastener is too close to the fitting gusset to permit proper installation, a radius block of sufficient thickness to lift the head or nut above the gusset may be employed. Create radius block from 2024-T3

Table 202 Notes



See NOTES on page 212.

Figure 203

NOTES:

1. Permissible to trim edge of doubler cutout flush with frame web.
2. Permissible to trim corner to minimum 0.50 radius.
3. The main spar is shown but similar modifications are allowed to the 27-22025 doubler at the aft spar location.
4. For SA226-T/-T(B)/-TC S/N below 258 and SA226-AT S/N below 067; manuafacture repair strap from 0.063 Thk, 2024-T3, AMS-QQ-A-250/5, 12.15 inches long X 0.76 inches wide and install as shown.
5. Conversion coat and prime trimmed edges and repair parts per SA226/227 Structural Repair Manual (SRM) 51-30-02.
6. Fay surface seal strap to frames as shown per SA226/227 Structural Repair Manual (SRM) 51-30-03 using same fastener type and size per SA226/227 Structural Repair Manual (SRM) 51-40-01.
7. All dimensions are in inches.

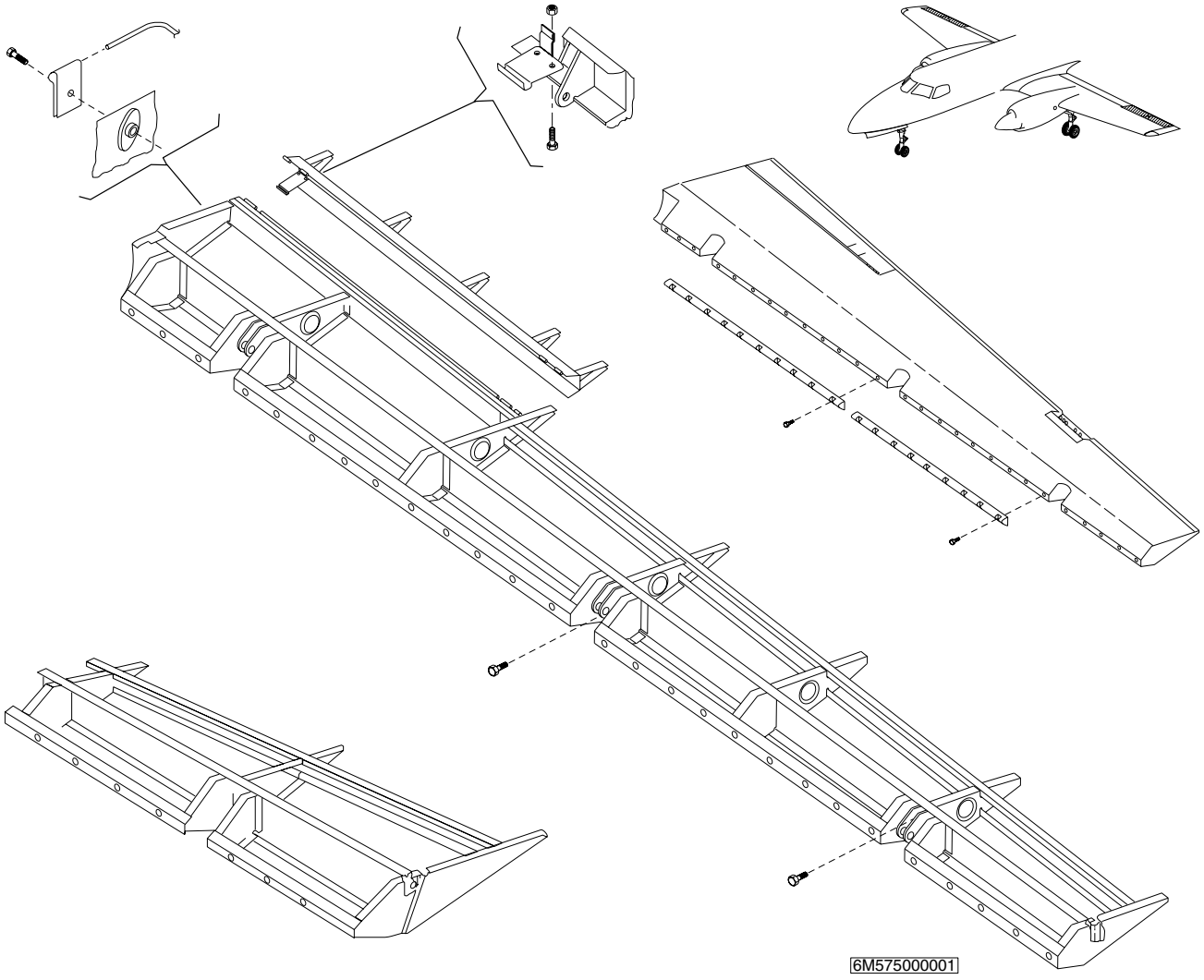
Figure 203 Notes

FLIGHT SURFACES – DESCRIPTION AND OPERATION

1. General

A. Ailerons (Figure 1)

(1) The ailerons are composed of formed ribs made of 2024-T4 aluminum alloy, two formed stringers made of 2024-T3 aluminum alloy, a formed and built-in spar made of 2024-T3 aluminum alloy. The skin is .020 2024-T3 aluminum alloy. The skin is riveted to the framework with standard AN426 rivets. A lead weight forward of the hinge line is used to balance the aileron.



Flight Surfaces
Figure 1

B. Aileron Trim Tab

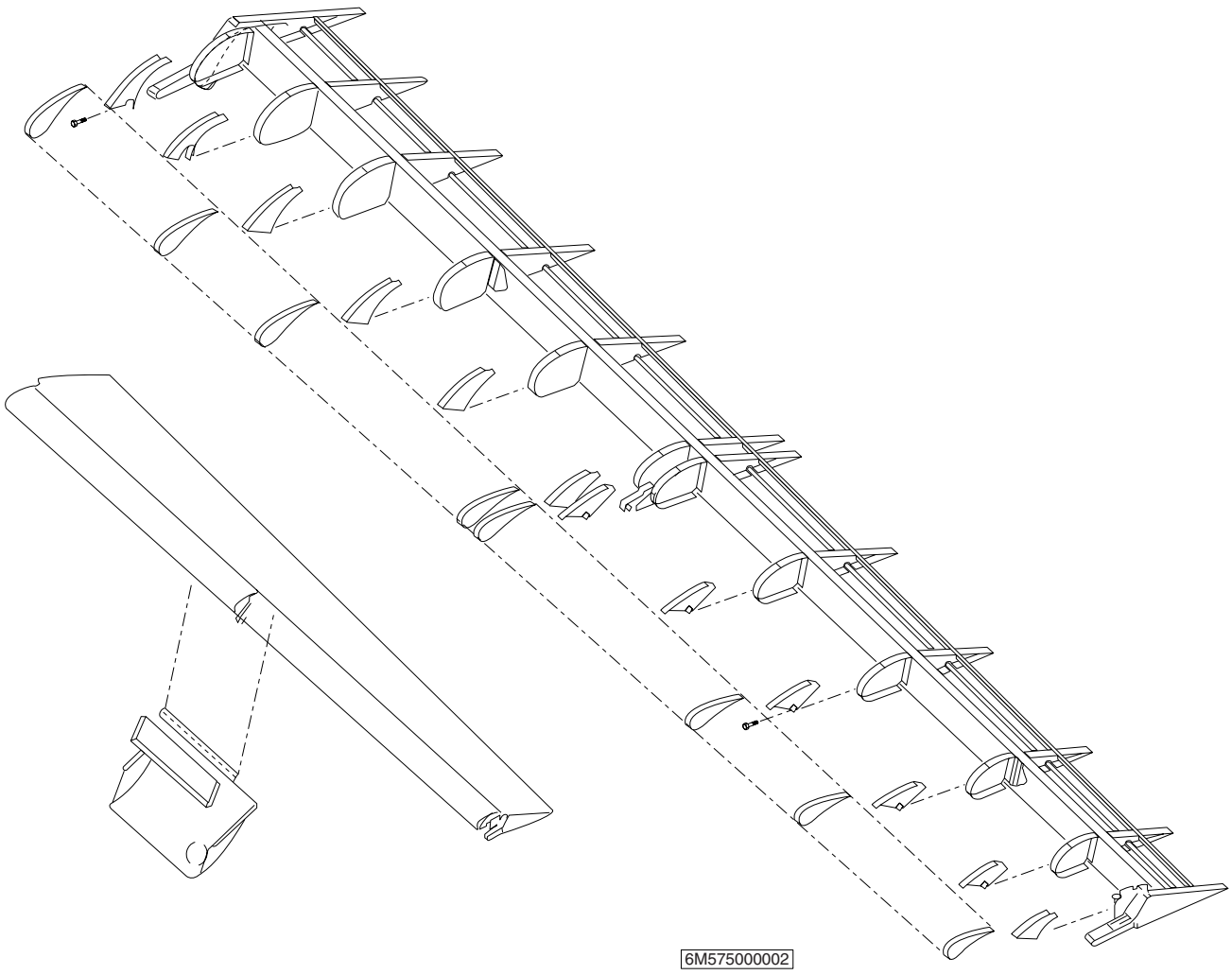
NOTE: Aileron trim tab is on the left side only.

(1) The aileron trim tab is made from formed ribs of 2024-0 aluminum alloy. The spar is a formed angle made of flat sheet. The skin is made from .016 2024-T3 aluminum alloy. The aileron hinges on a MS20257 piano hinge.

C. Flaps

(1) The wing flaps are interconnected by a mechanical link under the center section of the airplane. The flaps are actuated by two hydraulic cylinders. If one cylinder fails, the other will lower the flap through the interconnect. The flaps are made from 2024-T3 aluminum alloy frames The flaps are made in two parts divided at the flap actuator. They are made of .016 2024T3 skin. The fore flaps are attached to the flap with brackets riveted to the fore flap. The brackets are attached to the flap with No. 8 screws.

NOTE: For maintenance practices of the ailerons, trims, and flaps, see Chapter 27, Flight Controls.



Spars and Ribs
Figure 2

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MAINTENANCE MANUAL

SERVICE INFORMATION – PROPELLERS

1. Introduction

A. The following information is supplied as a supplement to this chapter. Subject matter is included which is not applicable to an ATA 100 format, but is still of interest to Merlin and Metro owner/operators. The information in this section is not part of the basic manual, and may be removed or retained at the discretion of the owner/operator.

B. Revisions and additions to this section will be made as additional information is available. Input from individual owner/operators is encouraged, and may be submitted to:

M7 Aerospace, LP
Post Office Box 790490
San Antonio, Texas 78279-0490
ATTN: Technical Publications Department

C. 61-01. Blade Angle Measurement

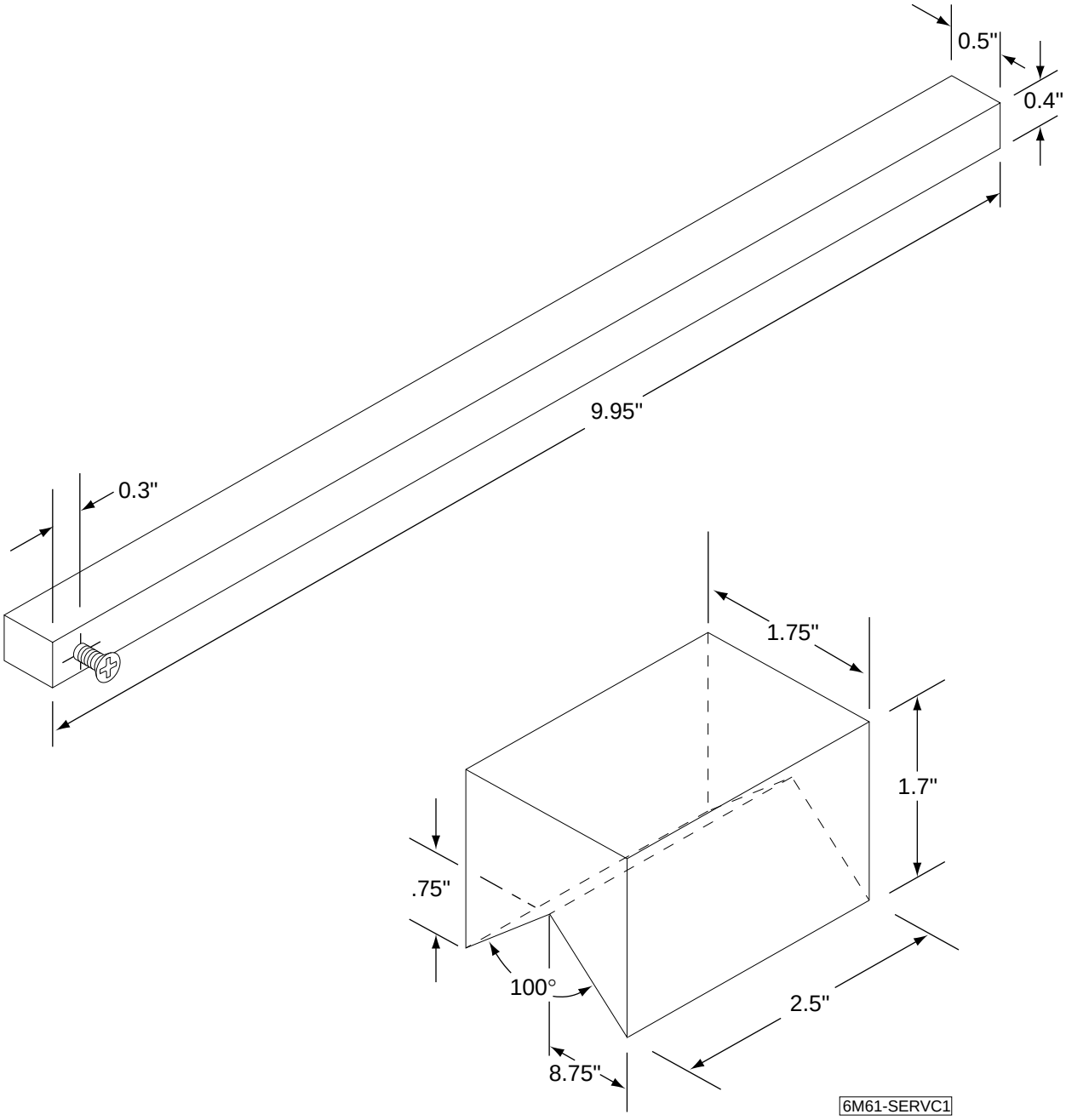
(1) Instructions for local manufacture of the two required blade angle measuring aids are shown in Figure 1. Use of these devices will ensure that blade angle measurements are accurate.

D. 61-02. Propeller Balancing

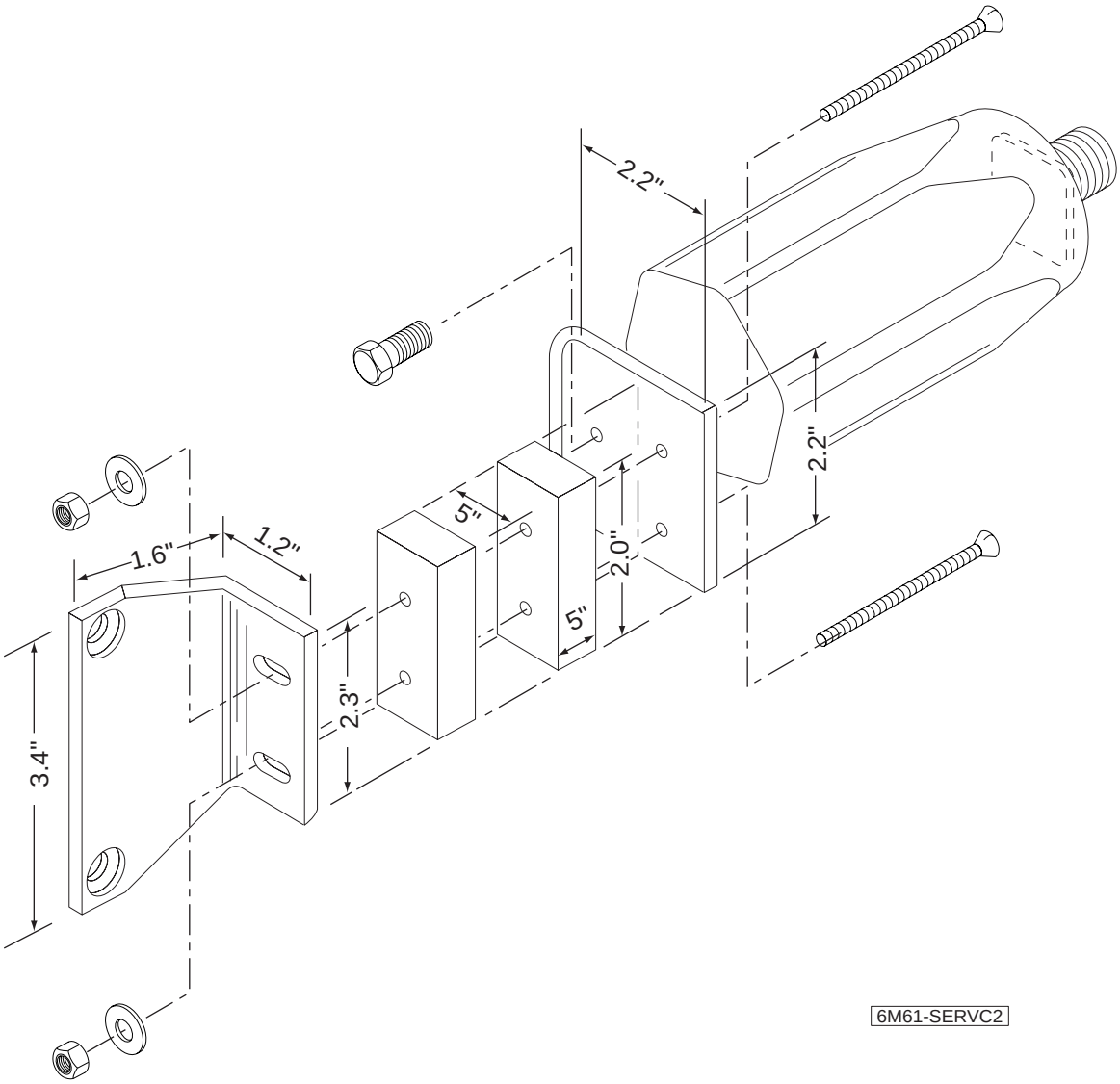
(1) M7 Aerospace has found that dynamically balancing the propellers results in greater passenger and crew comfort, less cracking of cowing and attachments, and improved service life of engine installed components. AiResearch Service Information Letter P 331-52 addresses this subject and recommends the propellers be balanced at least every 1,000 flight hours.

E. 61-03. Propeller Balancing Aids

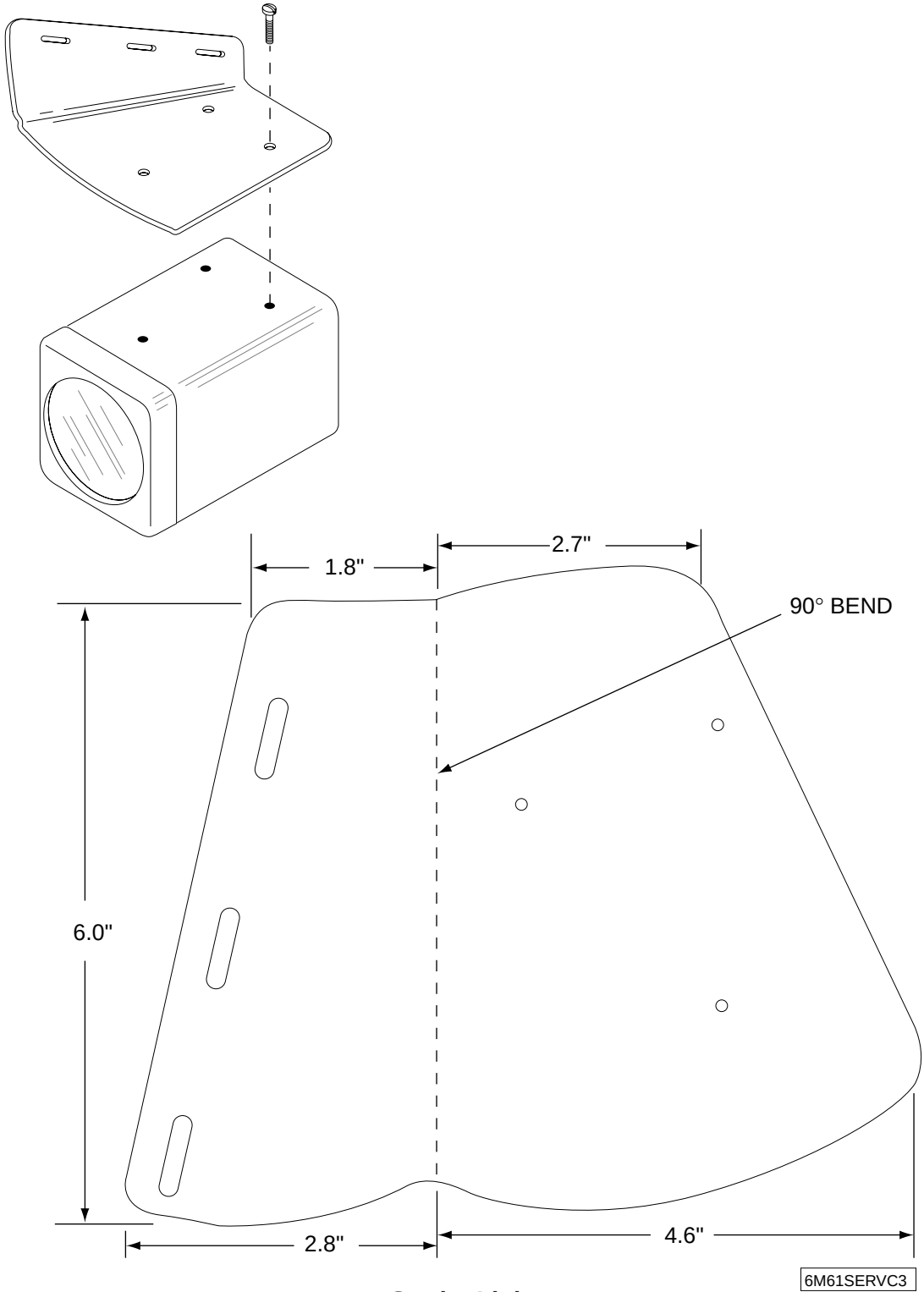
(1) Instructions for local manufacture of propeller balancing aids are shown in Figures 2 and 3. The vibration pick up model 544 brackets are made from 1/4 inch aluminum and the spacers are phenolic. The strobe light model 586 mount bracket is made from 1/16 inch aluminum alloy, and is attached to the strobe light at a 23 degree angle.



Blade Angle Measuring Aids
Figure 1



Vibration Pick-Up
Mount Bracket
Figure 2



Strobe Light Mount Bracket
Figure 3

PROPELLER SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The propeller used on the SA226-AT (Merlin IV) and the SA226-TC (Metro II) is a three blade, constant speed, full feathering, reversible unit with a diameter of 102 inches. It is flange mounted to the engine. Blade angle settings are accomplished at the 30 inch station as that location has proven to be the most accurate. See Table 1 for the applicable blade angle positions and values.

B. The propeller used on the SA226-T (Merlin III) is identical to that used on the SA226-AT/TC except that its diameter is 98 inches and the flight idle blade angle is different, see Table 1.

C. The propeller used on the SA226-T(B) (Merlin IIIB) is a four blade, constant speed, full feathering reversible unit with a diameter of 106 inches. It is flange mounted to the engine. Blade angle settings are accomplished at the 30 inch station. See Table 1 for applicable blade angle positions and values.

D. The propeller hub and blade types on the SA226-T, T(B), AT, TC airplanes are:

A/C TYPE	SA226-T	SA226-T(B)	SA226-AT	SA226TC
Hub	HC-B3TN	HC-B4TN-5EL	HC-B3TN-5	HC-B3TN-5
Hub		HC-B4TN-5HL		
Hub		HC-B4TN-5KL		
Blade	T10282HBD-4R	LT10282AB+2.5	T10282HB	T10282HB
	T10282DB-4R		T10282B	T10282B

2. Propeller Model Designations

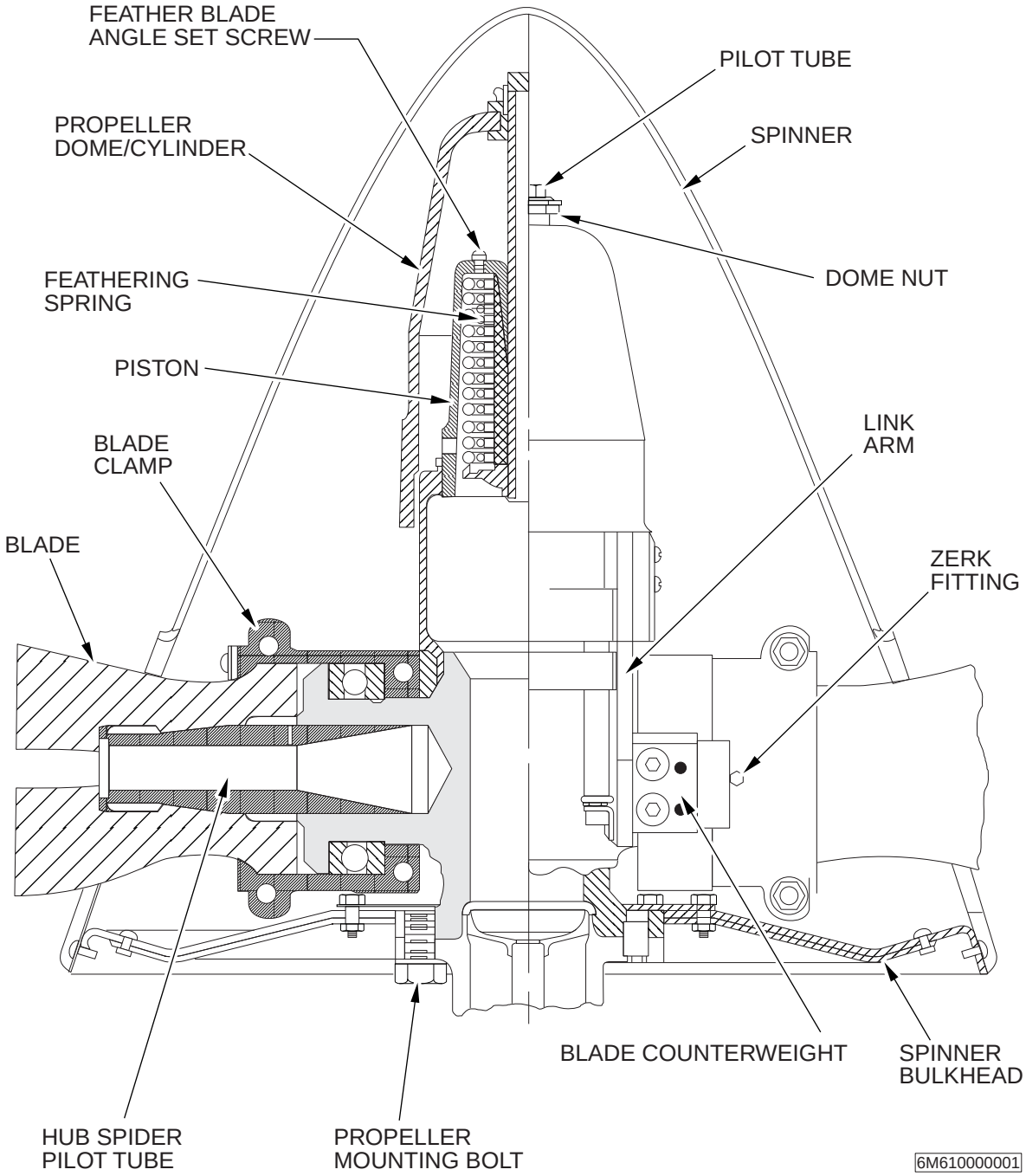
A. Symbols for propeller hub.

(1) HC – Hartzel controllable pitch.

(2) B – Basic design designation.

(3) 3 – Number of blades.

(4) T – Blade shank size.



Propeller Model
HC-B3TN-5
Figure 1

PROPELLER SYSTEM – DESCRIPTION AND OPERATION

THE APPLICABLE BLADE ANGLE POSITION AND VALUES FOR THE PROPELLER USED ON SA226 MODEL AIRCRAFT ARE AS FOLLOWS

T201-275, 277-291	FLIGHT IDLE	REVERSE	FEATHER	START LOCK	
	+13.5° ±0.2°	-6.0° ±0.8°	+89.7°+ 0.0° -0.5°	+2.0 ±1.0 with ±0.2° Variation Between Blades	
T(B) 276, 292-331	FLIGHT IDLE	REVERSE	FEATHER	START LOCK	REVERSE LOCK OUT
	+13.5° ±0.2°	*-2.0° ±0.8°	+89.5° +0.0° -0.5°	+2.0°±1.0° with ±0.2 Variation Between Blades	+2.0° to +2.5°
T(B)332-999	FLIGHT IDLE	REVERSE	FEATHER	START LOCK	REVERSE LOCK OUT
	+13.5° ±0.2°	-6.0° ±0.8°	+89.5°+0.0° -0.5°	+2.0° ±1.0° with ±0.2° Variation Between Blades	+2.0 to +2.5°
AT001-999	FLIGHT IDLE	REVERSE	FEATHER	START LOCK	
	+13.0° ±0.2°	-6.0° ±0.8°	+89.7°+0.0° -0.5°	+2.0° ±1.0° with ±0.2° Variation Between Blades	
TC201-999	FLIGHT IDLE	REVERSE	FEATHER	START LOCK	
	+13.0° ±0.2°	-6.0° ±0.8°	+89.7° +0.0° -0.5°	+2.0° ±1.0° with ±0.2° Variation Between Blades	

Table 1

The flight idle blade angle is set with the beta tube through the propeller pitch control. The reverse blade angle is determined by the cam on the pitch control. The feather, start lock, and reverse lock out are mechanically adjustable blade angle position.

*UNLESS AIRESEARCH SERVICE BULLETIN TPE 331-72-0238 IS COMPLIED WITH IN WHICH CASE THE LIMIT IS -6.0°.

(5) N – Shaft mounting.

(6) 5 – Specific design features.

B. Symbols for propeller blade.

(1) T – Needle bearing is installed in blade shank.

(2) 102 – Basic diameter before change.

(3) Design basic number.

(4) H – Blade material is 7076-T61.

(5) B – Deicing boots.

C. The constant speed, full feathering, reversible propellers are designed by Hartzell for use with the TPE 331 series turbo-prop engines. Oil pressure, working against internal spring and blade counterweight force, causes the blade angle to decrease. When oil is returned from the propeller to the engine case, the counterweight and internal spring forces cause an increase in the blade angle to the full feathered position.

D. The propellers are controlled in flight by the propeller governor referenced as the Propeller Governor Mode. The propellers are controlled on the ground by the propeller pitch control referenced as Beta Mode. Positioning the power lever ahead of, or behind, the flight idle gate determines the mode of operation.

E. A light for each engine, located in the annunciator panel, illuminates when the respective engine is operating in the Beta Mode.

3. Negative Torque System (NTS)

A. The negative torque system incorporated in the engine will automatically reduce propeller drag in the event of engine power loss.

4. Feathering

A. Feathering is accomplished by manually pulling the engine Stop and Feather control on the pedestal. Actuation of this control closes a fuel shutoff valve and opens the feathering valve.

5. Unfeathering

A. An electrically operated unfeathering pump unfeathers the propeller either in flight or on the ground. The electrical circuitry is arranged in such a way that the unfeathering pump is used in place of the electric starter to initiate engine rotation during an air start.

6. Propeller Synchronizing

A. A propeller synchronizer system is installed on the SA226-T, AT, TC for use during climb and cruise. The system has a limited range that will prevent the slave engine from losing more than a fixed amount of RPM if the master engine is feathered with the synchronizer on.

7. Propeller Synchrophasing

A. A propeller synchrophaser system is installed on the SA226-T(B) for use during climb, cruise, and descent. The system has a limited range that will prevent the slave engine from losing more than a fixed amount of RPM if the master engine is feathered with the synchrophaser on.

PROPELLER SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The propeller is hydraulically operated by engine oil pressure opposing internal feathering springs and blade counterweight force. The engine oil pressure is boosted to 400 psi and then routed through the governor pilot valve, feathering valve, pitch control unit, and the beta tube to the propeller piston and cylinder assembly.

B. The combined forces of the internal feathering spring installed in the cylinder assembly and counterweights attached to the blade clamps are used to increase the pitch of the blades. Oil pressure directed to the cylinder and piston assembly decreases the pitch. Feathering the blades is accomplished by releasing the oil pressure and permitting the feathering spring and counterweights to increase the pitch to the feathered position. Unfeathering the blades requires the electric unfeathering pump to direct oil pressure to the cylinder to decrease the pitch and permit the engine to windmill for air starting. To reverse the propeller, a follower sleeve in the pitch control unit is positioned by movement of the power lever which exposes ports in the beta tube. This directs governor oil pressure to the cylinder and decreases the pitch to reverse blade angle.

C. Centrifugal/mechanical stops (called stop locks) are installed to prevent the propeller blades from feathering during normal engine shutdown. As long as the propeller is rotating at speeds over 600 RPM, centrifugal force prevents the stops from extending from their housings and restricting propeller pitch changes. At lower RPM's when the engine is shutdown, spring force will overcome the centrifugal force and the stops will extend from their housings, engage the blades, and prevent the pitch from increasing further due to the force of the feathering spring.

2. Propeller Components

A. Blades

(1) The blades are solid and are constructed of machined aluminum alloy. The blade root is hollow to accommodate the pilot tube which protrudes from the hub spider.

B. Blade Pilot Tube

(1) The blade pilot tube centers the blade into the hub spider and provides a spindle for the blade to swivel around.

C. Blade Clamps

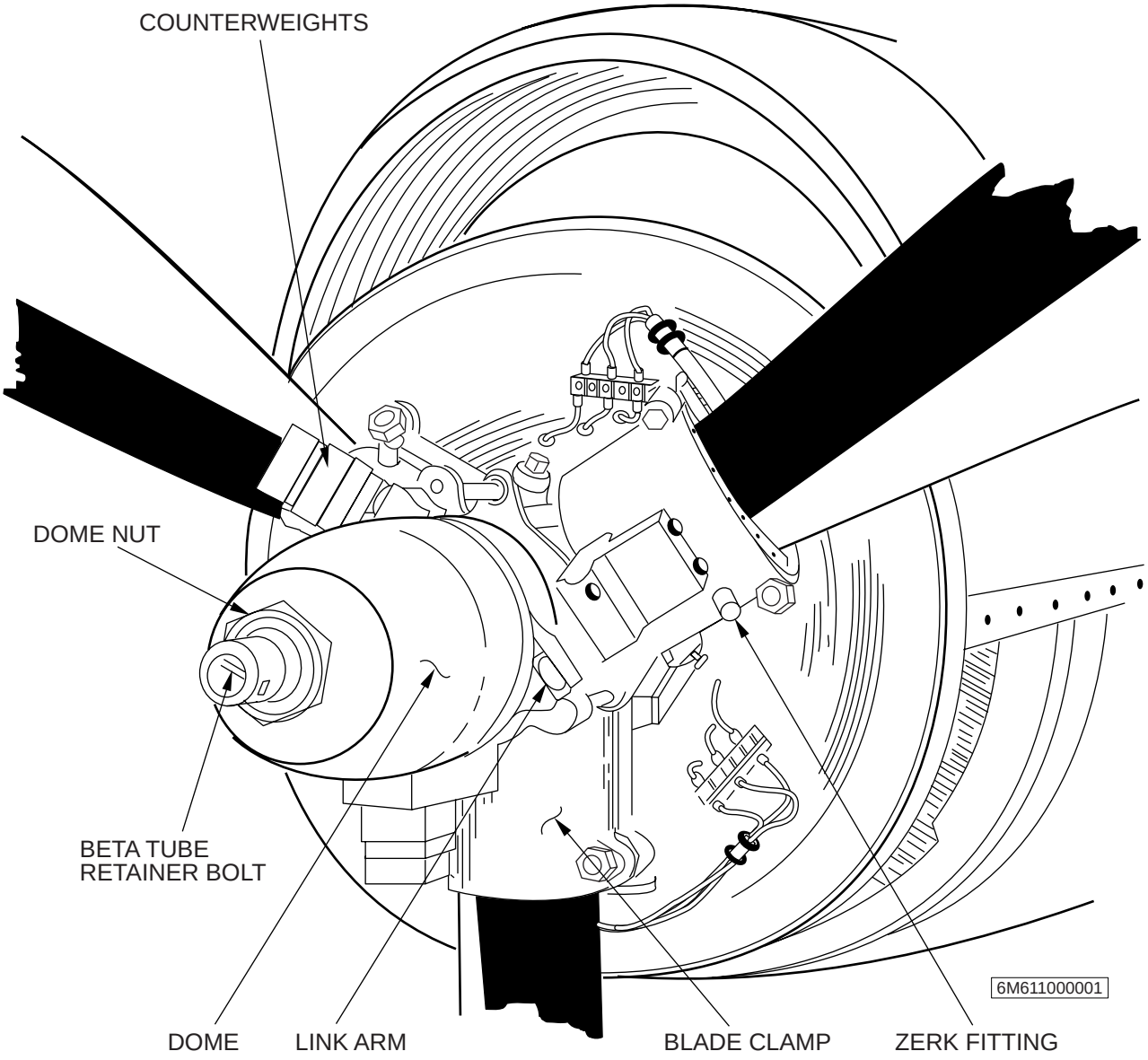
(1) The blade clamps hold the blades in the hub spider assembly.

D. Piston and Cylinder Assembly

(1) The piston and cylinder assembly (through the feathering springs, counterweights, and oil pressure transferred through the pilot tube) provides the force necessary to accomplish pitch changes.

E. Link Arms

(1) The link arms connect the piston to the propeller blade clamps and convert horizontal piston force into a swiveling movement of the propeller blades around their axis.



Propeller Installation Figure 1

F. Counterweights

(1) The counterweights are attached to the blade clamps to assist the feathering springs (of the piston and cylinder assembly) utilizing centrifugal force to increase the pitch angle of the blades.

G. Spinner Mounting Plate and Bulkhead Assembly

(1) The spinner bulkhead is installed on the propeller spider hub flange and provides for installation of the spinner mounting plate, deice platter, and the auto blade stops (start locks). The mounting plate (which is marked engine side and propeller side) is installed with the propeller side butted against the aft side of the spinner bulkhead. The mounting plate has an alignment arrow which must be aligned with a similar alignment arrow on the propeller spider hub flange during installation. The deice platter is installed on the engine side of the mounting plate and incorporates the slip ring disks which provides a means of transferring electrical power to the propeller deicing boots.

H. Hub Spider Assembly

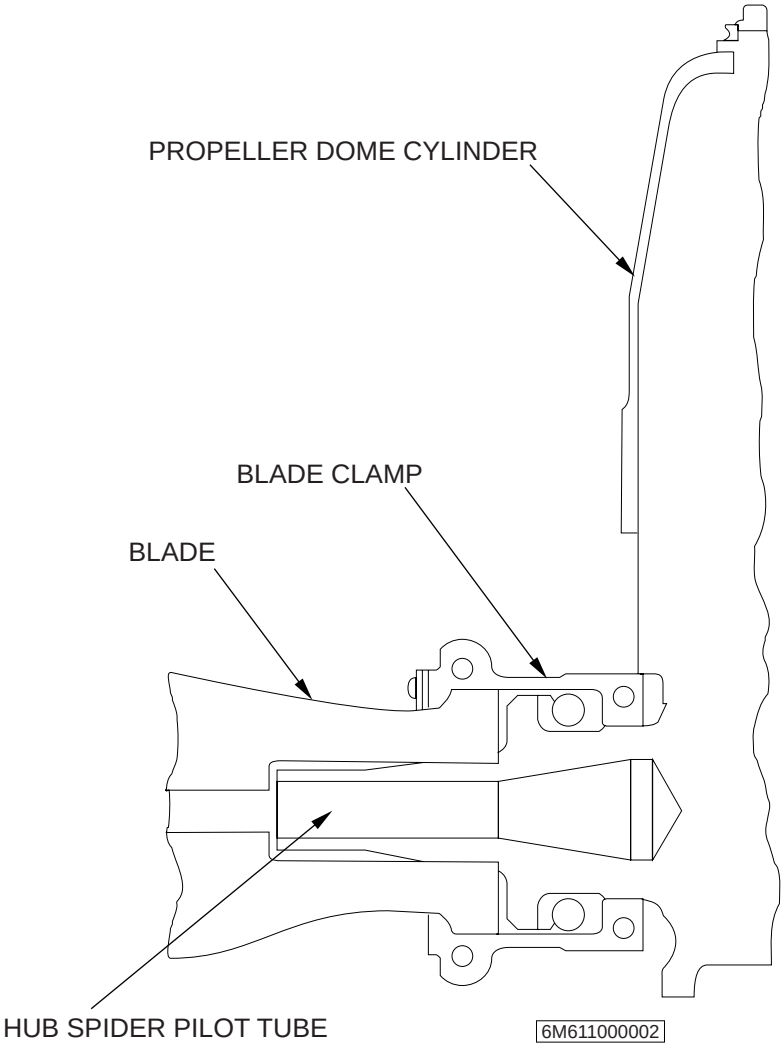
(1) The hub spider assembly consists of blade clamps. piston and cylinder assembly containing feathering springs, oil transfer (pilot) tube, link arms, and counterweights. The hub spider assembly is flange mounted to the engine drive shaft.

I. Deice Platter

(1) The deice platter is an aluminum disk mounted between the engine flange and the spinner bulkhead. On it are mounted the slip ring disks on the aft (engine) side and deice leads are mounted on the front (propeller) side.

J. Deice Leads and Terminal Clamp

(1) Deice leads from the slip ring disks are fastened to the blade deice boots at the terminal clamp, completing the necessary circuit for deicing operation. The terminal clamp is a molded terminal assembly mounted on the blade clamp. It serves as a mount for propeller balance weights as well as a junction for deicer leads.



Hub Spider Assembly
Figure 2

K. Brush Block/Brush Module

(1) The brush block/brush module selectively distributes the electrical power, (necessary for deicing the propeller blades), to the slip ring disks located on the aft (engine) side of the deice platter. It is mounted on the engine gear box and connected to the propeller deice timer unit.

L. Deice Boot

(1) The deice boot is constructed of fabric reinforced abrasion resistant rubber. It is impregnated with a wire mesh that distributes heat to the inner or outer half as selected by the automatic deice timer. The outer surface that is exposed to the elements has a glossy appearance and finish, While the side that touches the propeller has a dull appearance and finish.

M. Beta Tube

(1) The beta tube is an oil transfer device which provides oil, under pressure, from the propeller governor or pitch control, depending on the operating mode (beta or prop-governing), to control the blade angle.

N. Pilot Tube

(1) The pilot tube houses the forward end of the beta tube and secures the dome assembly to the hub spider. The beta tube is threaded into the forward end of the pilot tube and is secured with the beta tube retainer bolt.

O. Zerk Fittings

(1) The zerk (or grease) fittings are one-way valves that are located on the blade clamps. These are the means of filling the blade clamp interior with lubricating grease (under pressure). They allow grease to go in, but then seal themselves and prevent its escape.

P. Dome

(1) The propeller hub dome houses the piston, cylinder, and oil transfer (pilot) tube and acts as a seal against the hub.

Q. Dome Nut

(1) The dome nut secures the dome to the hub.

R. Spinner

(1) The propeller spinner protects the hub components from damage by foreign objects. It is designed to provide a streamlining effect to reduce drag.

S. Slip Ring Disks

(1) The slip ring disks are commutator rings made of copper and are located on the aft (engine) side of the deice platter. They transfer electrical power from the engine gear box mounted brush block module to the spinning propeller. They are insulated from each other and connect to deice leads on the front (propeller) side of the deice platter.

T. Start Locks

(1) The start locks are a centrifugal/mechanical device that physically lock the propeller blades. This is necessary to prevent them from being spring driven into feather during engine shutdown when oil pressure diminishes to zero. Attempting to start the engine with the blades in feather would be a very heavy load on the starter and could result in engine damage. The start locks eliminate this possibility.

U. Lock Plate

(1) The lock plate is the counterpart to the start lock and is mounted on the hub spider blade clamp. It is the unit that the start lock is positioned against.

3. Blade Tolerances

A. End Play in the blades – total permissible fore ad aft play is .1 inch.

B. Blade track – tolerance of 1/7 inch is permitted.

C. List of approved grease, by specification or name:

(1) MIL-G-81322

(a) Shell Oil Company – Aeroshell Grease #22

(b) Mobil Oil Company – Mobilgrease #28

(2) MIL-G-3545

(a) Shell Oil Company – Aeroshell Grease #5

(3) *MIL-G-23827

(a) Shell Oil Company – Aeroshell Grease #7

(b) Exxon Oil Company – Exxon 5114 EP Grease

(4) MIL Specification

(a) Hartzell Propeller, Inc. – Hartzell DG Grease

*These are synthetic greases and may cause swelling of the O-ring seals. Use these only if necessary.

PROPELLERS – MAINTENANCE PRACTICES

1. Servicing – Propellers

A. Every 100 hours, grease blade clamps through zerk fittings. Care should be taken to avoid blowing out clamp gaskets. Remove one zerk fitting and pump grease into the remaining fitting until grease appears through the hole where the zerk was removed. Replace zerk fitting.

2. Removal/Installation – Propellers

A. Removal – Propeller

(1) Remove the chin cowl.

(2) Feather the propeller prior to removal either by not placing the blades on the start locks during shutdown, or manually, as follows:

(a) Actuate the engine stop and feather control or manually rotate the blades with a paddle.

(b) While the blades are off the start locks, depress and hold start lock pins in the retracted position so that the blades can move into feather.

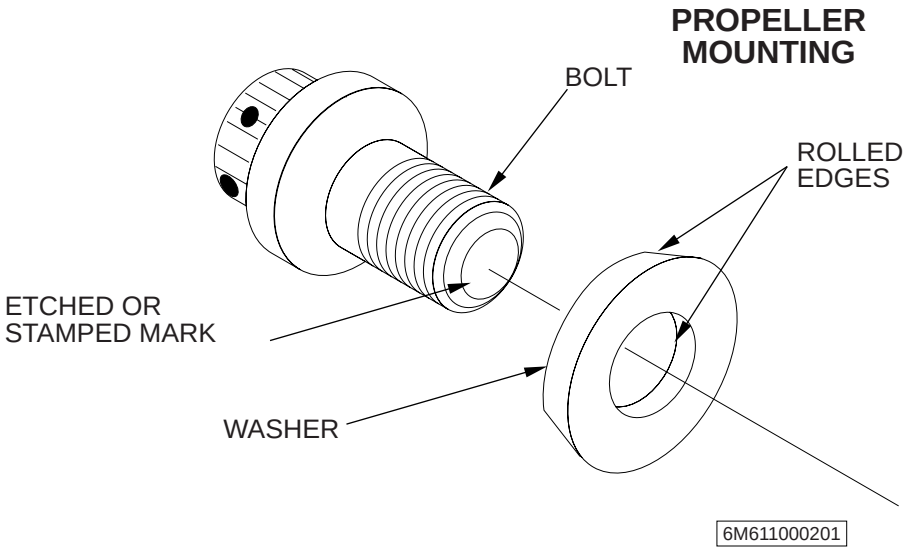
CAUTION: TAKE CARE WHEN USING A PADDLE TO PREVENT ABRUPT FEATHERING AFTER THE BLADES ARE RELEASED FROM THE START LOCKS.

(3) Remove the spinner.

NOTE: If propeller was not feathered during shutdown, it will be necessary to remove the spinner before manually feathering.

(a) Place an alignment mark on the spinner and on an adjacent blade so that it can be reinstalled just as removed.

(b) Remove the screws from around the spinner circumference (21 screws on T, AT, TC; 20 screws on T(B).



**Propeller
Mounting Bolt
Figure 201**

(4) Feather the Propeller

CAUTION: USE EXTREME CARE IN THE REMOVAL AND STORAGE OF THE BETA TUBE.

(5) Remove the beta tube.

(a) Remove the beta tube retainer bolt.

(b) Unscrew and remove the beta tube.

(6) While supporting the propeller, remove the propeller bolts.

(7) Remove the propeller assembly.

B. Installation – Propeller

(1) To prevent damage to the brushes in the deicer brush block, tape a piece of cardboard over the brushes.

(2) Clean the propeller and engine flanges, removing any possible nicks which might prevent mating and also remove all oil and grease from propeller bolt hole threads.

(3) Apply engine oil sparingly to PRP 909-8 O-ring and install it on the engine propeller shaft.

(4) Inspect beta O-ring condition and installation.

(5) Inspect the propeller to be certain it is the correct one.

(6) Align the dowel pins on the engine flange and install the propeller, spinner bulkhead, mounting plate, spacer, and deicer platter on to the engine shaft.

(7) Install eight B3339 bolts identified by one or more marks etched or stamped on the threaded end face of the bolt (the face opposite the head surface), with eight A-2048-2 stamped washers through the engine shaft flange into the propeller flange.

(8) Install the washers on the bolts with the rolled or chamfered edges against the bolt head. Apply MIL-T-5544 petrolated graphite to bolt threads and washer surface.

NOTE: Washers and bolts may be reused between overhauls, should propeller be removed and reinstalled. They must be replaced on installation after overhaul.

(9) Snug the bolts following the sequence for step 1 (see Figure 202). Turn the propeller in the direction of the rotation only.

(10) After all eight bolts are snug and propeller is well seated in position, torque bolts in sequence for step 2 shown in Figure 202. Use only adapter tool No. SAT1188 with an 18.5 inch torque wrench set to 25 foot-pounds. (Use snap-on-torque wrenches QJR 3200B).

NOTE: Use of other torque wrench-adapter combinations may not apply the correct torque when the values listed above are used. In all cases, the final torque applied to each bolt must be 100-105 foot-pounds.

(11) After all eight bolts have been torqued (see Figure 202, Step 2), reset the torque wrench to 51 foot-pounds and torque the bolts in sequence (Figure 202, Step 3). Apply a smooth even force to wrench handle, no jerking.

(12) After all eight bolts are torqued, reset torque wrench to 63/67 foot-pounds and torque the bolts again (Figure 202, Step 4).

(13) Secure the bolts head with .041 gauge safety wire.

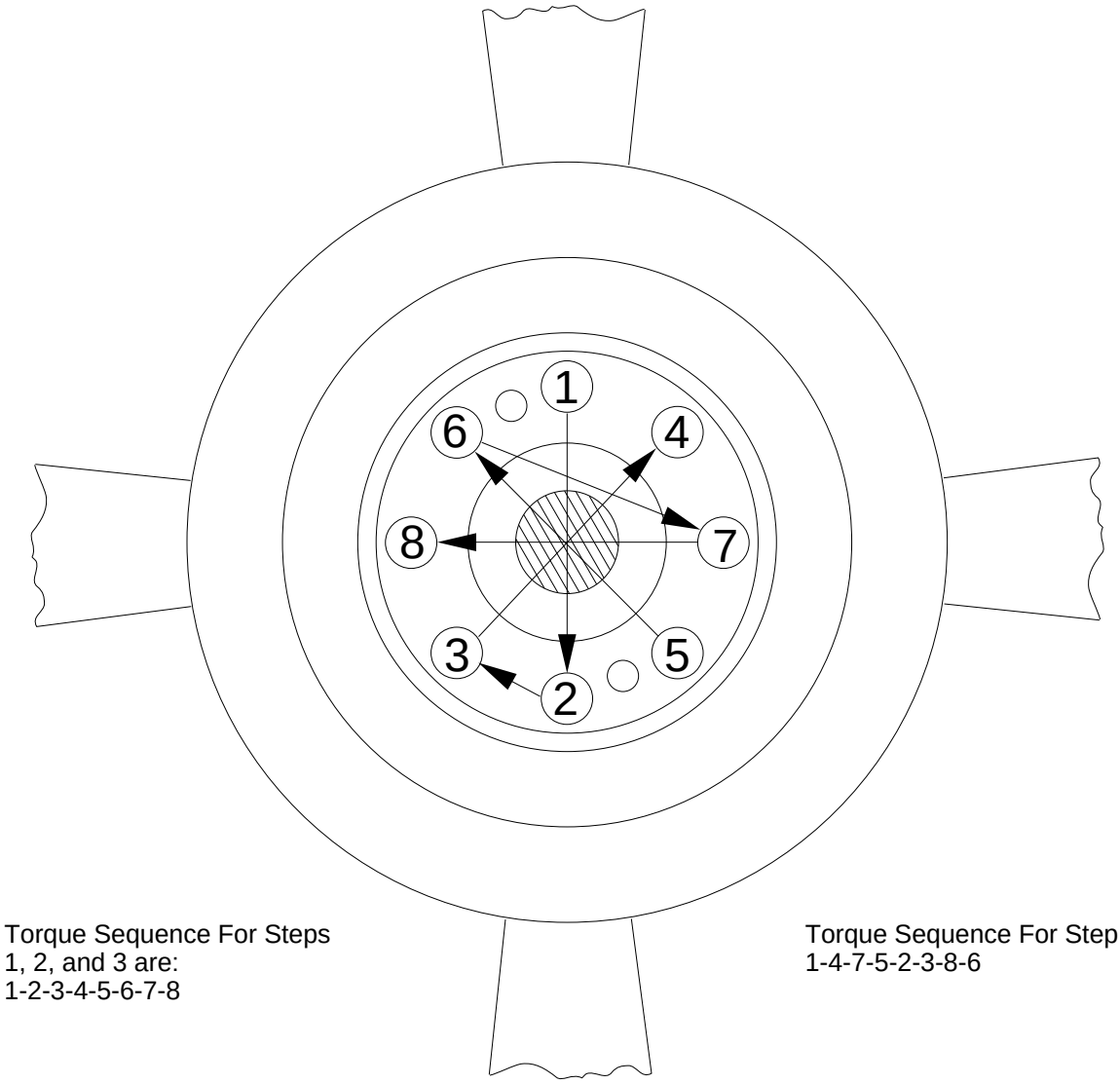
NOTE: If bolts have .55 safety wire hole, .032 safety wire is permissible. If safety wire hole is .098, .041 safety wire must be used.

(14) Remove the tape and cardboard from the brush block and the commutator rings is 1/32" at the forward end and 3/32" at the aft end relative the rotation of the commutator ring.

(a) Check that the gap between the brush block and the commutator ring is 1/32" at the forward end and 3/32" at the aft end relative the rotation of the commutator ring.

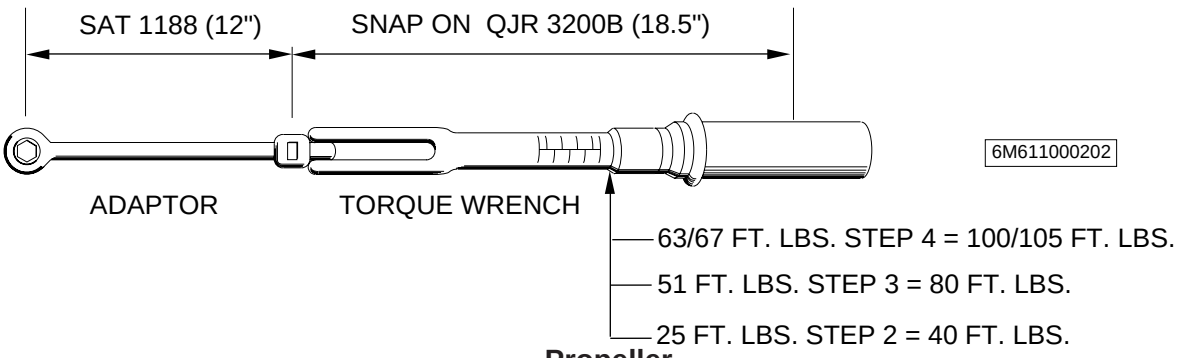
(b) As Illustrated (Figure 203), a 40 bit drill placed immediately above and a 30 bit drill placed immediately below the brush (placement relative to direction of ring rotation) will assure the correct gaps.

(c) Check that the brushes are aligned with the commutators so that the entire face of each brush is in contact with the commutator rings throughout the 360° rotation (see 61-10-30, Figure 201).

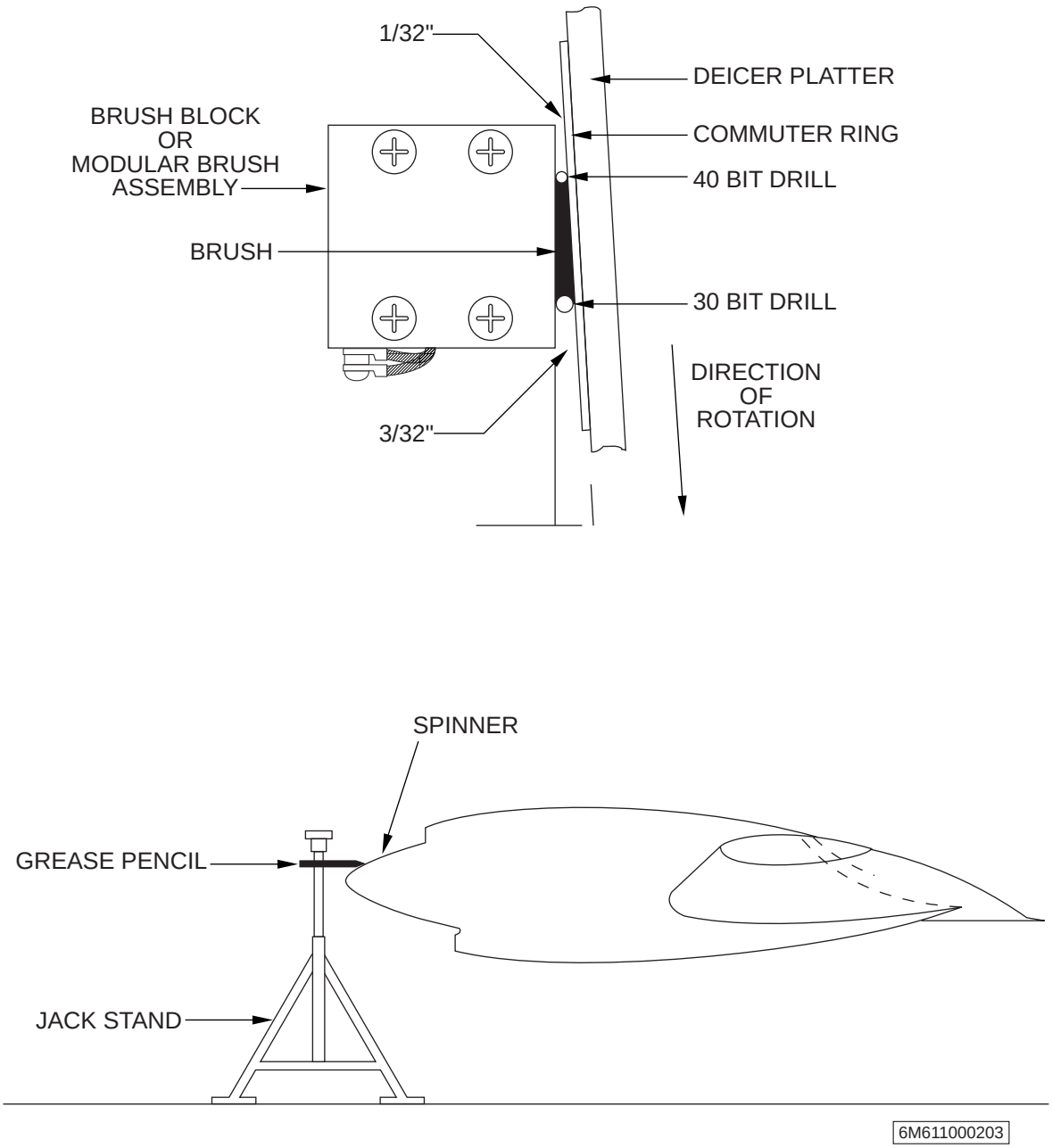


Torque Sequence For Steps 1, 2, and 3 are:
1-2-3-4-5-6-7-8

Torque Sequence For Step 4 is:
1-4-7-5-2-3-8-6



Propeller Torque Sequence
Figure 202



Brush Block and
Spinner Dome Realignment
Figure 203

(15) Install the beta tube and thread 16 turns.

NOTE: Before installation steps can continue, blade angle must be checked and adjusted as necessary. Refer to the Adjustment/Measurement section for the necessary steps. After blade angles have been checked and adjusted as necessary, continue.

(16) Install beta tube retainer bolt.

(17) Check propeller for safety and clearances between fixed and moving parts.

(18) Align the spinner mark with the mark made on the propeller blade and reinstall it with one screw between each blade.

(19) Check spinner trueness (Figure 203).

(a) Place a jack or other stationary object adjacent to the spinner.

(b) Secure a grease pencil or its equivalent to the jack at the height of the spinner and position it so it will mark the spinner approximately three inches in from the tip. Adjust the pencil elevation so that it only marks the spinner if it is out of true.

(c) Turn the propeller by hand.

(d) Note any marks made on the spinner and realign it as necessary until the spinner doesn't wobble.

(e) Replace and tighten the screws around the spinner circumference.

(20) Install the chin cowl.

C. Removal – Propeller Blade

(1) Remove the spinner.

(2) Feather the propeller.

(3) Mark the matching lines on the blade and the clamp.

(4) Record the blade and the clamp serial numbers.

(5) Disconnect the link arm. The screw is on the aft end of the link pin.

(6) Disconnect the three deicer wires from the blade clamp terminals.

(7) Remove the front and rear bolts from the blade clamp.

(8) Remove the clamp halves and gaskets.

NOTE: When the clamps are removed, the ball bearings inside may fall out. These should be caught in a suitable container.

(9) Remove the blade from the hub.

D. Installation – Propeller blade

(1) Fill the blade pilot tube with grease. Make certain that air is not trapped under the grease as this will change the propeller balance after engine run up.

(2) Install each blade on the same hub pilot tube from which it was removed. Apply a hardening gasket compound (Permatex) around the blade shank in the shoulder radius.

(3) Install the linkscrew into the clamp and tighten 45 to 50 foot-pounds. Secure with safety wire.

(4) Install the zerk fittings in the clamp. Assemble the link and sleeve onto the linkscrew and secure with a cotter pin.

(5) Install the stop plate with its bolts onto the base of the clamps in the same position as they were before they were removed. Secure with safety wire (not applicable on HC-B3TN-5M propellers).

(6) Install the counterweight slugs, if there were any, onto the counterweights. Secure with safety wire.

(7) Install the correct matching clamp and counterweight subassembly with new clamp gaskets.

NOTE: This process is best accomplished by peeling back the masking tape from about 180° of the bearing on the top side. Install the half of the clamp that has the counterweight. Peel the tape from the lower half of the exposed bearing and install the other clamp half. Slip the clamp gaskets between the clamps at the parting line. It is essential to locate the inner bearing race parting line at right angles with the blade clamp parting line.

ITEM/LOCATION	PART OR BOLT SIZE	TORQUE VALUES* FOOT-POUNDS	INCH-POUNDS
Mounting Bolts	9/16-18	100-105	1200-1260
Outer Blade Clamp Bolt	7/16-20	60-65	720-780
inner blade clamp Socket Screw	3/8-20	40	480
Counterweight Socket Screw	7/16-20	65	780
A-304 Linkscrew	1/2-20	35-45	420-540
Guide Rod End Socket Screw	5/15-24	10	120
Piston End Nut-Feathering	A-880-1	120	1440
PARTS TO BE CHECKED FOR SLIP			
Blade mounted in Blade Clamp	Blade	167	2004

*All torque values are based on non-lubricated (dry) threads. Tolerance on torque values is ± 10% unless otherwise stated.

Torque Values
Figure 204

(8) Fasten the clamp halves together with the socket head screws and bolts with nuts. Tighten the screws to 40 foot-pounds and the torque the nuts to 65 foot-pounds.

(9) Check the blades for slippage in the clamps with a blade bar, torquing each blade to 167 foot-pounds using blade bar and weights.

E. Spinner Bulkhead Assembly

(1) Support the propeller assembly, flange side up.

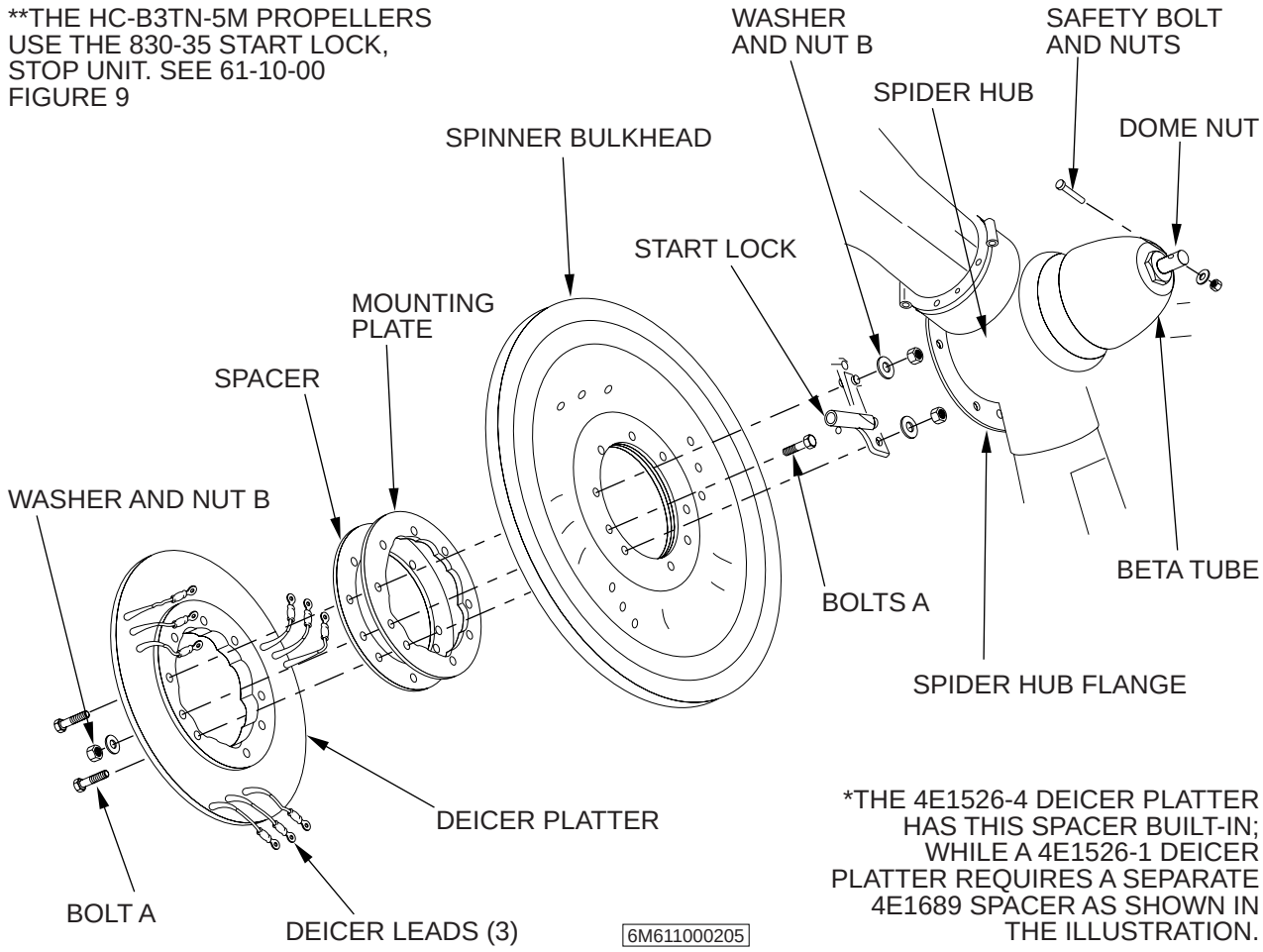
(2) Assemble the spinner bulkhead, mounting plate, spacer, and slip ring disk and secure them together with three bolts, washers, and nuts (bolts A and nuts and washers B in Figure 204). Feed the deicer leads through the spinner bulkhead and attach them to the terminal strips.

(3) Install the spinner bulkhead assembly on the spider hub flange.

(4) Install the start locks on the spinner bulkhead.

NOTE: The HC-B3TN-5M propeller start lock is installed on the dome guide rod collar.

(5) Install the blade deicer leads onto the spinner bulkhead terminal strips.



Spinner Bulkhead Assembly
Figure 205

3. Inspection/Check – Propellers

A. Inspection/Check – Propellers

(1) Remove the spinner. Clean and inspect for cracks, dents, or other damage.

(2) Feather the propeller.

(3) Inspect each blade for nicks, cracks, pitting, corrosion, or any other type of damage.

(4) Check each blade for fore and aft play; one-tenth of an inch of play is allowed. If this tolerance is exceeded, the propeller should be replaced. This check should be chordwise of the blade; otherwise, the blade will deflect and exaggerate the true play.

(5) Check each blade to determine whether or not they turn freely on the hub spider pilot tube. This can be accomplished by rocking the counterweights back and forth through the slight freedom allowed by the pitch change mechanism. If they appear tight, the pitch changing mechanism must be removed so that each blade can be checked individually. The propeller should be replaced if the blades are tight.

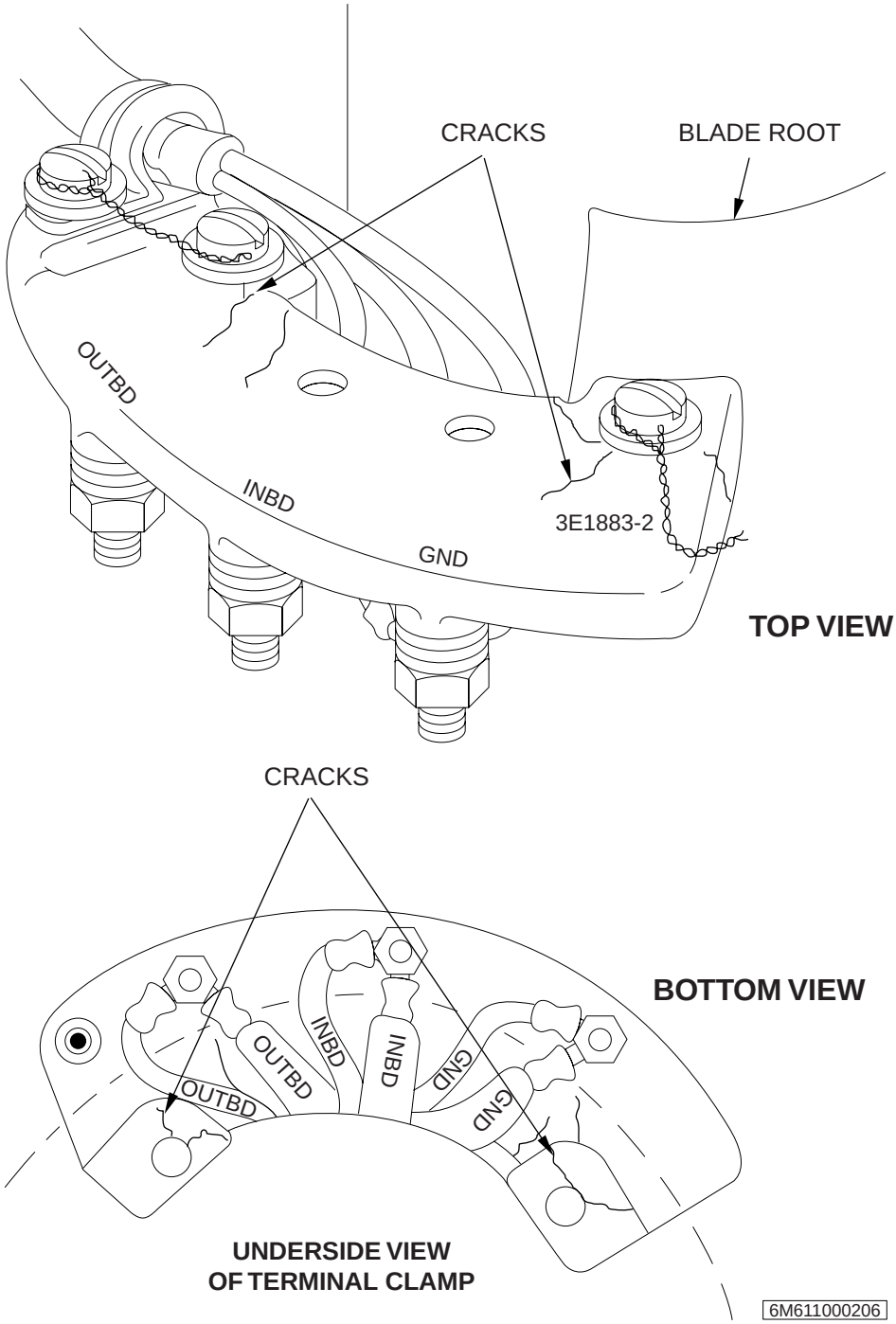
(6) Inspect the dome and each blade clamp for leakage.

(7) Inspect each start lock and spring retainer pin for security. Also check each spring to see if vibration has caused it to turn on its retainer pin in an effort to work its way out.

(8) Inspect each deicer boot for condition and any signs of damage.

(9) Check the cycling operation of each deicer boot.

(10) Inspect the propeller bulkhead for security, cracks, or other damage.



Deicer Terminal Clamp Assembly
Crack Inspection
Figure 206

(11) Inspect the slip ring disks, brush block, brushes, and electrical system connections for security and condition.

(12) Inspect each mounting bolt for condition, security, and correct torque value.

(13) Inspect terminal clamp for cracks. Cracks are usually in or around the mounting hole area and propagate along the mounting pad area (see Figure 206). Remove terminal clamp for thorough inspection. Cut safety wire and remove two mounting screws, if there is any sign of cracking or separation, terminal assembly must be replaced. If terminal assembly is sound, re-install as follows:

(a) Clean top of propeller blade clamp mounting surface to remove foreign material.

(b) Position deicer leads and wire harness leads as shown in Figure 206 and position terminal assembly on propeller blade clamp. Avoid crossing leads.

(c) Install attaching screws and washers. Check that leads are not pinched between terminal assembly and blade clamp. Tighten screws. Torque 15-20 inch-pounds and safety wire.

CAUTION: DO NOT PLACE WASHERS OR PROPELLER BALANCE WEIGHTS BETWEEN THE PAD OF THE TERMINAL CLAMP ASSEMBLY AND PROPELLER BLADE CLAMP.

(d) If propeller balance weights were present on the original terminal clamp assembly, reinstall on the new assembly in the same manner in which they were removed. Ensure that screws securing balance weights do not protrude through the terminal assembly and contact the leads or hub clamp.

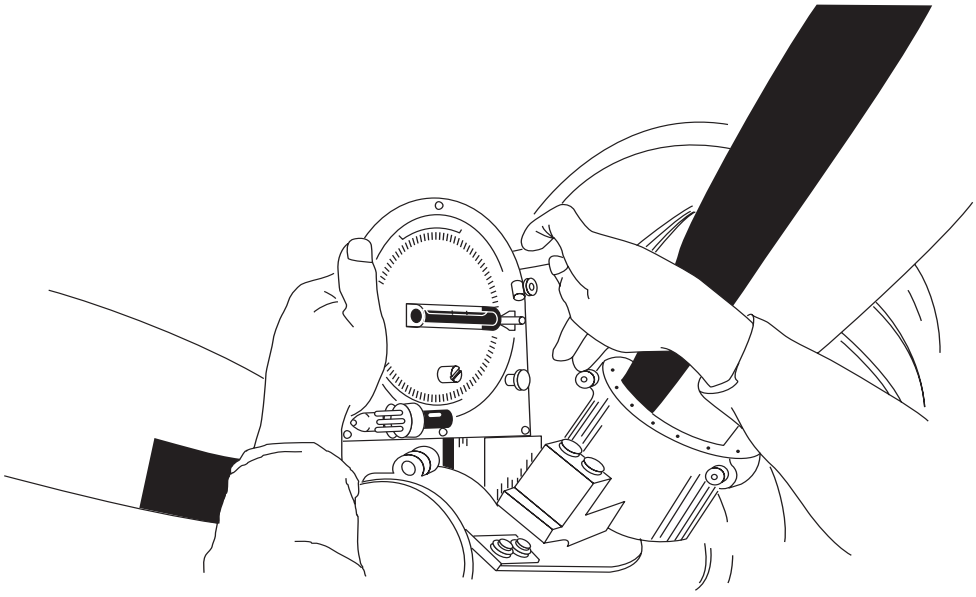
NOTE: Improper installation of the deicer terminals may result in overstressing and lead to cracking of the terminal clamp. If fractured terminal clamp is not replaced, it may separate from the blade clamp and exit the spinner dome, allowing the possibility of striking the aircraft fuselage.

(14) Perform any required maintenance as necessary.

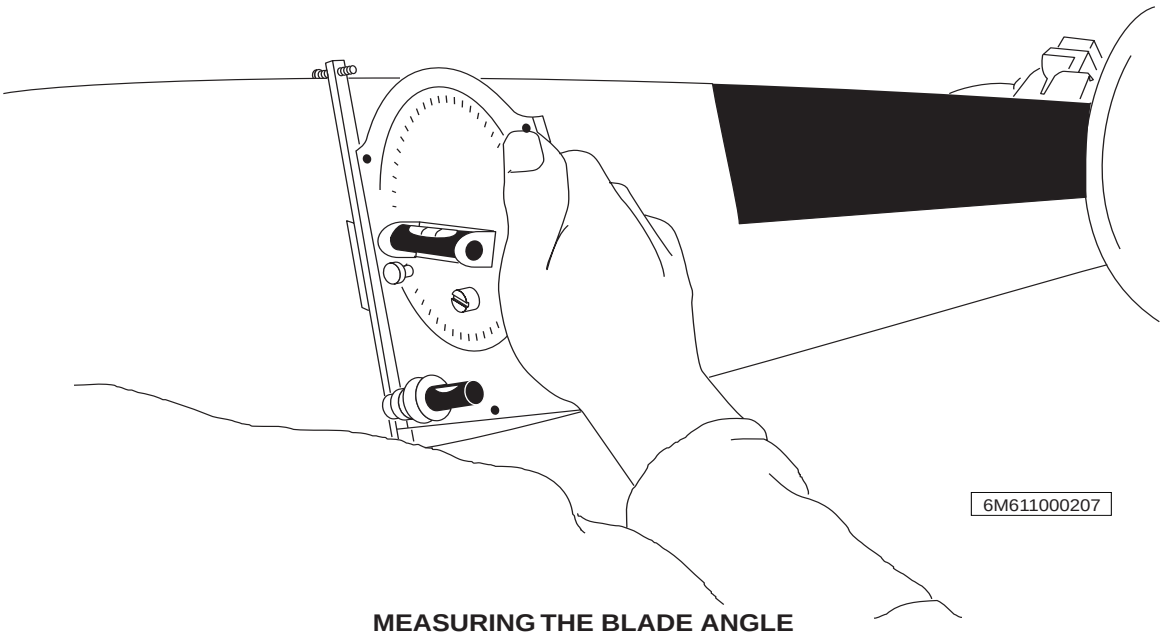
(15) Clean the propeller assembly with solvent; wipe with a clean cloth and allow to air dry.

(16) Install the spinner and true it according to previous instructions.

(17) Polish the spinner with a clean soft cloth.



ZEROING THE PROTRACTOR



MEASURING THE BLADE ANGLE

Blade Angle Measurement Figure 207

4. Adjustment/Test – Propellers

A. Adjustment/Test – Blade Angle Adjustment

(1) All blade angle measurements should be made using a propeller protractor. The protractor must be zeroed on the propeller piston while the blades are on the start locks. Fairchild has determined that the most accurate zeroing is accomplished with an adaptor block positioned on the propeller piston. Dimensions for the local manufacturer of this block are provided in the Service Information section of this chapter.

(2) The actual measurement of each blade's angle must be taken at the 30 inch station, usually identified by a yellow mark on the aft (thrust) side of the propeller, for the greatest accuracy. Because of the blade's curvature, a square steel bar must be used as a rest for the protractor. Dimensions for the local manufacture of this bar are provided in the Service Information section of this chapter.

(3) When performing checks in which the unfeather pump is used to move the blades (flight idle, reverse) observe the pump duty cycle of 1/2 minute with hot oil or one minute with cold oil. If the unfeather pump changes pitch while running (indicating the oil tank is empty) release the unfeather switch at once. Motor the engine by using the starter test switch or turn the propeller by hand as necessary to return the oil from the engine to the tank.

B. Adjustment/Test – Start Lock Blade Angle Adjustment

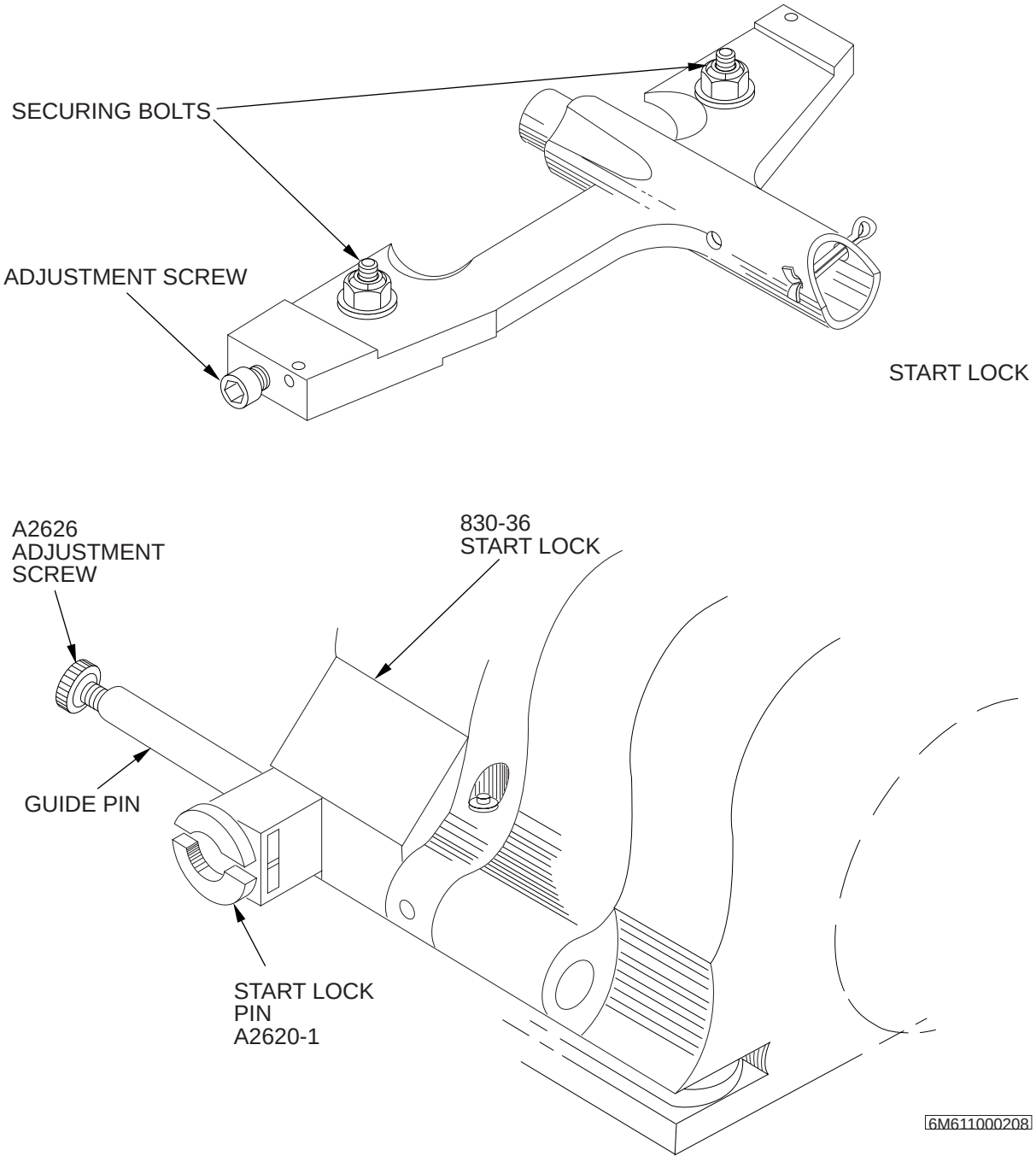
(1) Remove the spinner and zero the protractor.

(2) Measure each blade angle. It should be as shown in Table 1, Section 61-00-00.

(3) If the blade angle is incorrect, adjust the start locks as follows:

(a) Loosen the two bolts securing the start lock for the respective blade.

(b) Turn the adjustment screw as necessary to adjust the blade angle.



Start Lock and Start Lock Pin
for HC-B3TN-5M Propeller
Figure 208

(c) Secure the adjustment screw with .032 gauge safety wire.

(d) Secure the start lock bolts and torque to 48 inch-pounds.

(4) Install the spinner after all the blades are correct.

(5) True the spinner as previously described.

C. Adjustment/Test–HC-B3TN-5M Propeller Start Lock Check and Adjustment

NOTE: The HC-B3TN-5M propeller incorporates a new start lock system.

(1) Remove the spinner and zero the protractor.

(2) Measure each blade angle.

(3) If the blade angle is incorrect, adjust the start locks as follows:

(a) Power lever to reverse, unfeather pump on.

(b) Pressurize the propeller enough to lift piston guide rods and screws off of the stop pins.

(c) Pull the start lock pins out and lock with 1/4 inch wedges between the pins and housing (see Figure 208).

(d) Turn unfeather pump off and release enough pressure to allow guide pins to exit start lock housing.

(e) Use an allen wrench to adjust the guide pin screw.

NOTE: One quarter turn of the guide pin screw equals approximately 0.3° blade angle (in to increase and out to decrease blade angle).

(f) Remove the lock wedges from between the pins and housings.

(g) Turn unfeather pump on; move power lever to reverse until start locks engage.

(h) Unfeather pump off; move power lever forward.

(i) Measure blade angle. If further adjustment is required, repeat the above procedure until the correct blade angle is obtained.

(j) The A2626 adjustment screws are safetied with green Loctite (AA Weld Seal 89-41) prior to assembly. Reapply Loctite if the screw is adjusted.

D. Adjustment/Test – Reverse Lockout Blade Angle Adjustment (Merlin IIIB)

NOTE: The reverse lockout blade angle must be checked at the engine operational oil temperature (indicated in the green arc).

(1) Remove the spinner.

(2) Zero the protractor.

(3) Release the start lock and while holding each lock pin in the retracted position, insert a cotter pin through the housing to keep the pin retracted.

(4) Hang a ten pound weight on the counterweight at each blade to be checked. This will simulate an air load and remove slack from the linkage assembly.

(5) Place the power lever against the reverse lockout stop in the cockpit pedestal. Operate the unfeather pump.

WARNING: DO NOT MOVE THE SWITCH TO THE START POSITION. SERIOUS INJURY TO THE PERSON CHECKING THE BLADE ANGLES COULD RESULT.

(6) Measure blade angle at the 30 inch station.

(7) If the blade angles are incorrect, adjust as follows:

(a) Reposition the reverse lockout gate in the cockpit pedestal.

(b) Check blade angle as before.

(8) Remove the weight from the counterweight.

(9) Remove the cotter pins from the extension end of the start locks.

(10) Install spinner and true it according to previous instructions.

E. Adjustment/Test – Flight Idle (Hot Oil) Adjustment

NOTE: The flight idle blade angle must be checked at the engine operational oil temperature (indicated in the green arc).

(1) Remove the spinner.

(2) Zero the protractor.

(3) Release the start locks.

(4) Place the power lever in the flight idle position.

(5) Hang a ten pound weight on the counterweight of each blade to be checked.

(6) Move the starter/unfeather test switch to the unfeather position.

WARNING: DO NOT MOVE THE SWITCH TO THE STARTER TEST POSITION. SERIOUS INJURY TO THE PERSON CHECKING THE BLADE ANGLES COULD RESULT.

(7) Using the blade bar and protractor, measure the propeller blade angle at the 30 inch station. The correct blade angels are shown in Table 1, Section 61-00-00.

(8) If the blade angles are incorrect, adjust as follows:

(a) Remove the beta tube retainer bolt.

(b) Rotate the beta tube clockwise to decrease blade angle or counterclockwise to increase.

(c) Install the beta tube retainer bolt and torque to 40 inch-pounds.

(9) Remove the weight from the counterweight.

(10) Install the spinner and true it according to previous instructions.

F. Inspection/Check – Reverse Blade Angle Check

NOTE: The reverse blade angle stops are internal propeller stops and cannot be adjusted in the field.

(1) Remove the spinner.

(2) Zero the protractor.

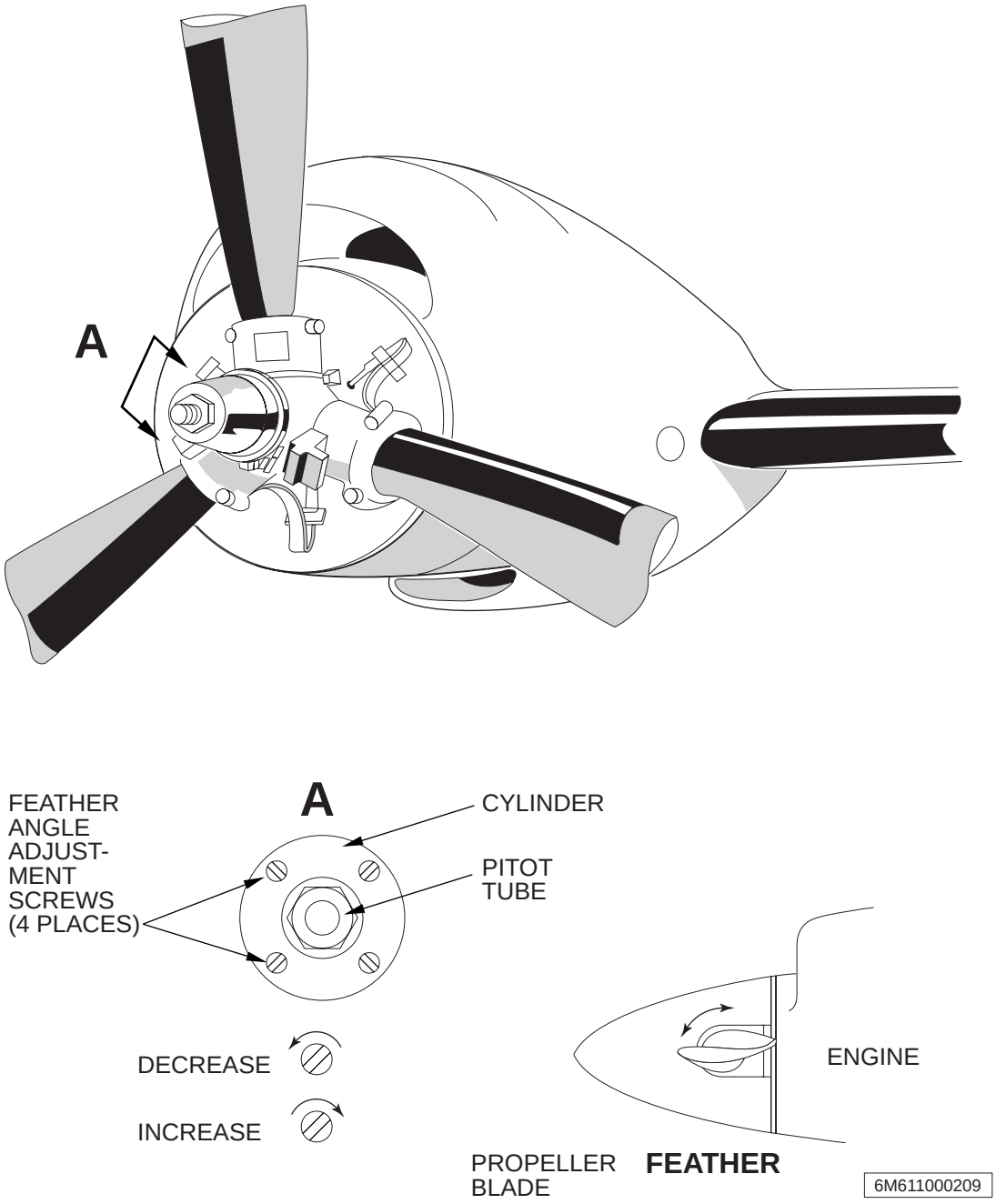
(3) Hold the power levers in full reverse and move the starter/unfeather test switch to the unfeather position.

WARNING: DO NOT MOVE THE SWITCH TO THE STARTER TEST POSITION. SERIOUS INJURY TO THE PERSON CHECKING THE BLADE ANGLES COULD RESULT.

(4) Using the blade bar and protractor, measure the blade angles at the 30 inch station.

(5) If the angles are incorrect, check the propeller pitch control pointer. If the pointer is not within 3° of zero, the entire engine requires rigging adjustment.

(6) Install the spinner and true it according to previous instructions.



Blade Angle Adjustment
Figure 209

G. Adjustment – Feather Blade Angle

CAUTION: Backward rotation of a feathered propeller in flight indicates that the blades have travelled beyond the feathered blade angle limits, and that the feathered angle stops require adjustment. Prolonged backward rotation will damage starter-generator brushes and might damage some seals.

NOTE: Perfect feathered blade angles will cause no rotation when the airplane is flown at airspeeds between 95 and 180 KIAS in coordinated flight.

Limited forward rotation in coordinated flight is permissible. The amount of forward rotation permitted varies with airspeed, as follows:

MAXIMUM PERMISSIBLE FORWARD ROTATION

AIRSPEED	SA226-T,-AT, TC	SA226-T(B)
95 KIAS to 150 KIAS	6 RPM (18 Blades per minute)	6 RPM (24 Blades per minute)
150 KIAS to 180 KIAS	10 RPM (30 Blades per minute)	10 RPM (40 Blades per minute)

(1) Remove the spinner. Exercise caution not to damage the propeller deice boots.

(2) Feather the propeller.

(3) Zero the protractor. Check the blade angle at the 30 inch station using the blade bar and protractor.

(4) Remove the beta tube retainer bolt.

(5) Remove the dome nut with a 1 13/16" wrench while holding the pilot tube with a 1" wrench.

(6) Remove the safety wire from link pin screws.

(7) Remove the link arm pin screws.

(8) Remove link arm pins.

(9) Swing the link arms away from the hub.

(10) Place an alignment mark (grease pencil, etc.) on the dome and the adjacent propeller blade.

(11) Remove the dome.

(12) Located at the forward end of the propeller cylinder are four 10/32 inch screws next to the pilot tube. They are safety wired to one another.

(13) Remove the safety wire from the adjustment screws.

(14) Turn the screws in to increase the blade angle and out to decrease. One full turn will change the blade angle 1.5°.

NOTE: The adjustment screws function as a mechanical stop by limiting the aft travel of the dome.

(15) After the screws have been adjusted, secure them with .021 gauge safety wire.

(16) Replace the dome on the propeller cylinder by lining it up with the alignment mark made on the propeller blade before removal.

(17) Place each link arm into its slot in the dome.

(18) Insert each link arm pin through the dome and into the link arm.

(19) Insert and torque each link arm pin screw to 40 - 50 inch-pounds, but do not secure with safety wire at this time.

(20) Replace and torque dome nut to 90 - 120 foot-pounds, while holding the pilot tube with a 1" wrench. Do not replace beta tube retainer bolt now.

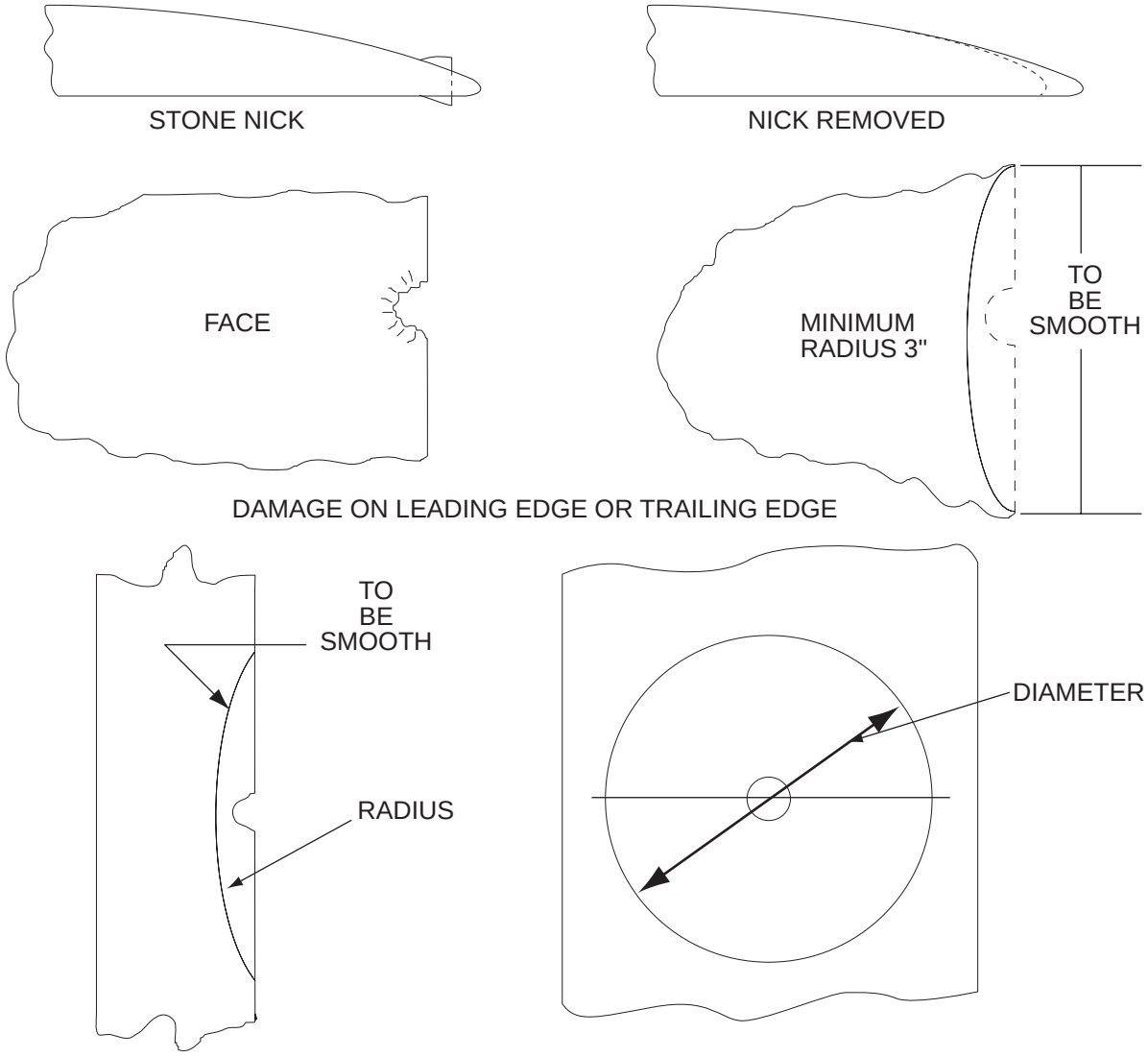
(21) With each blade at the feather angle, place the protractor on the blade bar at the 30 inch station and check the blade angle. The correct blade angles are shown in Table 1, Section 61-00-00. These are the normal blade angles. The final determination of acceptable adjustment is in the in-flight propeller action after the engine has been shutdown. If the in-flight limits prescribed in this section are not complied with, the blade angle must be reset.

(22) If blade angles are incorrect, repeat steps 4. thru 21. until correct angles are obtained. If (or when) the blade angle is correct, continue below.

(23) Secure each link arm pin screw with safety wire.

(24) Insert the beta tube retainer bolt in the hole of the pilot tube and torque to 40 inch-pounds.

(25) Replace the spinner and true it according to previous instructions.



RADIUS	MINIMUM WIDTH		MINIMUM THICKNESS	
	T/AT/TC	T(B)	T/AT/TC	T(B)
8	3.830	3.990	2.913	2.585
12	5.165	5.300	1.541	1.195
18	6.435	6.235	0.942	0.634
24	7.090	6.645	0.657	0.461
30	7.345	6.660	0.508	0.398
36	7.400	6.230	0.367	0.335
42	7.130	5.155	0.253	0.241
48	6.250	—	0.153	—

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Propeller Repairs
Figure 210

H. Repairing Nicks

(1) Equipment required.

(a) Fine cut round and flat files

(b) Emery tape or cloth.

(b) 10 X magnifying glass

(c) Crocus cloth

(d) Dye penetrant

NOTE: Nicks in the leading edge of blades can develop into cracks. It is important that all nicks be removed as soon as possible after they occur. This is especially important if they occur in the area zero to eighteen inches from the tip, since this area has the greatest vibratory blade stress.

I. Blade Rework

(1) To determine the amount of rework needed, use the following formula:

(a) Leading or trailing edge – depth of the damage times 10 for the repair diameter or times 5 for the radius.

(b) Thrust or camber side – depth of the damage times 20 for the repair diameter or times 10 for the radius.

(c) Use the table on Figure 210 to determine repair criteria.

(2) The radius value is inches as measured from the dome center. The minimum width and thickness is the minimum that the propeller can be reduced to in accomplishing the repair at the corresponding radius. If these tolerances would be exceeded to accomplish the repair according to the above formula, the blade must be replaced.

J. Repair Procedures:

NOTE: Blades with large nicks, gouges, etc. that may affect the structure, balance, or operation of the propeller should be referred to a qualified propeller repair station for repair or replacement.

(1) Local repairs may be made using files, small air, or electrical powered equipment with suitable grinding and polishing attachments. All repairs must be accomplished parallel to the blade axis.

(a) Begin with a round file removing damaged material to the bottom of the damaged area.

(b) Remove material from this point out on both sides; provide a smooth faired depression; maintain the original airfoil concept.

(c) Using an emery cloth, smoothly fair the area. Remove all traces of initial filing and rework.

(d) Using crocus cloth, polish the area.

(e) Inspect the reworked area with a 10X magnifying glass and dye penetrant. Assure no cracks remain.

NOTE: Camber section repairs that form a continuous line across the blade are not acceptable.

Eddy current, ultrasonics, and flourescent penetrants are also acceptable methods of crack detection.

(f) Chemically treat all required areas.

K. Refurnish the blade immediately, using the following procedures and materials.

(1) Paint mixtures by volume:

(a) No. 1 mix – four parts Wash primer MIL-P-8514 plus four parts T54 Reducer plus one part acid diluent.

(b) No. 2 Mix – one part V6627 Sherwin-Williams Polane Catalyst plus seven parts Z99BB510 Polane Satin Finish Back plus two and two-thirds parts R7K69 or R7K84 Polane Reducer.

(c) No. 3 Mix – one part V66V29 Sherwin-williams Polane Catalyst plus seven parts Z99YB572 Polane Yellow plus two and two-thirds parts R7K69 or R7K84 Polane Reducer.

(d) No. 4 Mix – one part V66V27 Sherwin-Wiliams Polane Catalyst plus seven parts Z99AB503 Grey plus two and two-thirds parts R7K69 or R7K84 polane Reducer.

(e) No. 5 Mix – (Optional primer to No. 1 Mix) one part No. 32966 zinc chromate primer plus three parts No. E333 thinner, Spec. TT-T266A (such as one gallon No. 32966 plus three gallons No. E333).

(f) No. 6 Mix – one part V66V27 Sherwin-Williams Polane Catalyst plus seven parts Z99WB612 Polane White plus two and two-thirds parts R7K69 or R7K84 Polane Reducer.

NOTE: Mix thoroughly, then allow at least one hour set time. Mix in reducer before use. Pot life of catalyst mix is approximately 8 hours.

L. Procedure for Refinishing Nicks

(1) Clean surface thoroughly with lacquer thinner at room temperature. Allow to dry.

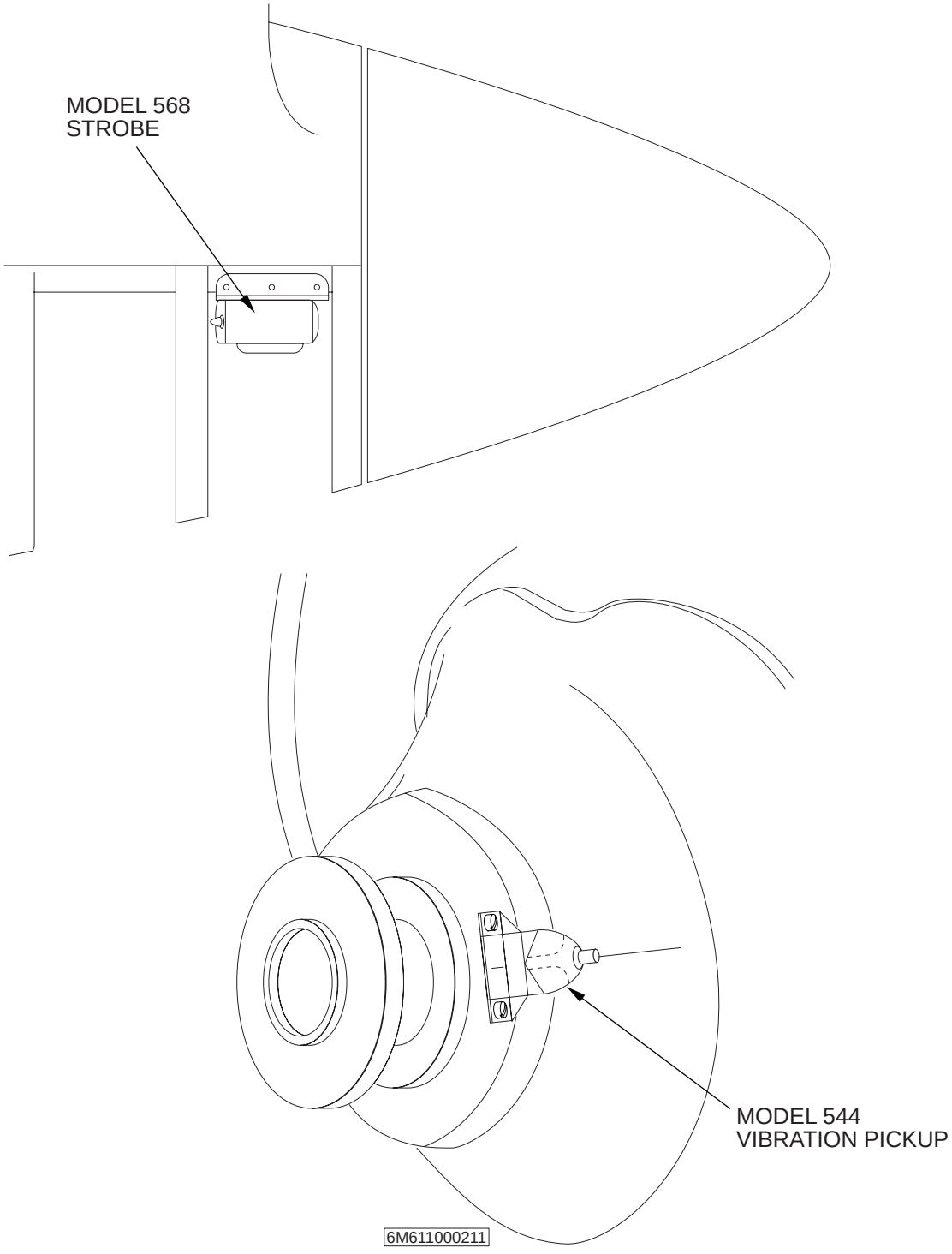
(2) Tape shank and deice boot.

(3) Clean surface with emery cloth. Apply zinc chromate (Mix No. 1) or wash primer (Mix No. 5) to entire surface of blade. Allow to dry.

(4) Apply grey (Mix No. 4) to camber side of blade. Allow finish to set 15 to 20 minutes in 180° heat and for one half hour at room temperature.

(5) Stamp blade design and serial number information on camber side eight inches from end of shank. Install decal when required 16 1/2 inches at center. Locate top edge of decal two and three-fourths inches from leading edge. Spray decal and stamping using pattern with a clear acrylic coating.

(6) Mask off and apply yellow (Mix No. 3) or white (Mix No. 6), whichever required, to tip section of camber. Apply black (Mix No. 2) to face side of blade. Remove tape and allow to dry.



Propeller Balancing
Installation
Figure 211

M. Propeller Balancing

(1) Procedures

(a) The following procedures dynamically balance the propeller using one specific model of equipment and one specific procedure. Many other procedures and types of equipment are available and acceptable as long as the final result is the propeller balanced to 1 mil or less. Additional information related to propeller balancing is contained in section 61-SERVC of this chapter.

(2) Equipment

NOTE: Equipment required to accomplish dynamic propeller balance is a Vibration Analyzer/Dynamic Balancer, International Research and Development Corporation (IRD), Model 330 or Model 350 and following accessories:

(a) Vibration Pickup Unit, Model 544, (P/N 4526),

(b) Pickup Cable, 12 foot, (P/N 4521),

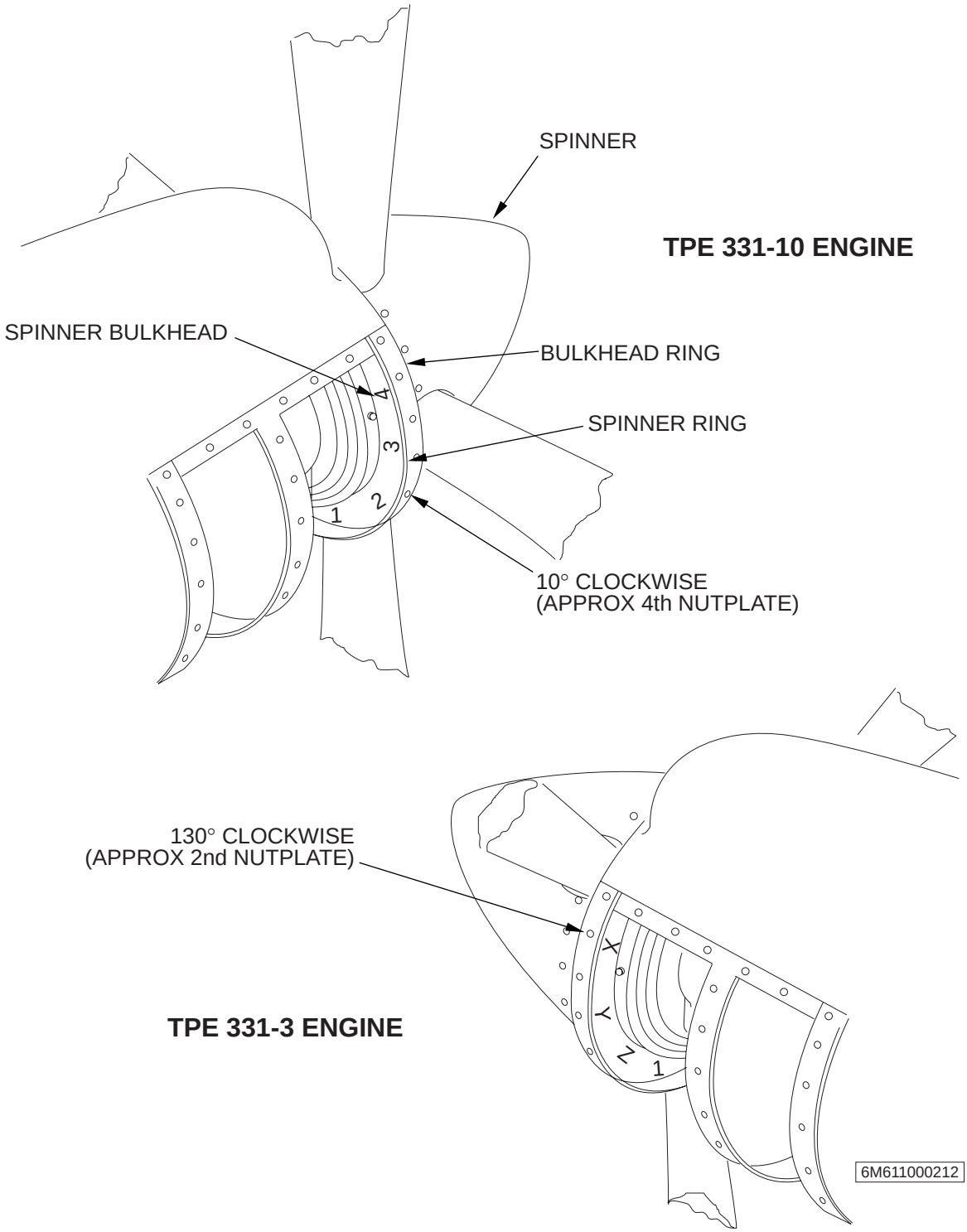
(c) Stroboscopic Light, Model 568, (P/N 4524),

(d) Stroboscopic Light cable, 12 foot, (P/N 4522).

NOTE: The IRD propeller balancing system can be obtained from IRD Mechanalysis Inc., 6150 Huntly Rd., Columus, Ohio 43229, phone 614/885-5376.

N. Power Requirements

(1) Power supply for Vibration Analyzer/Dynamic Balancer (105/129 volt, 50/60 cycle, 25 watts) or internal Battery should be charged.



Propeller Balancing
Figure 212

O. Setup Procedure

(1) At the rear of the spinner bulkhead, mark off twelve equal increments and assign each increment a numerical/letter designator. For example, paint the numerical designators one through nine, and the remaining three increments with the letters X, Y, and Z (see Figure 212).

(2) Attach Model 568 strobe light to the right hand side of the nacelle (see Figure 211).

CAUTION: SECURE THE BULKHEAD RING TO PREVENT IT FROM COMING INTO CONTACT WITH THE SPINNER RING DURING ENGINE RUN.

(3) Attach Model 544 Vibration Pickup Unit at the left side of the engine case (see Figure 211).

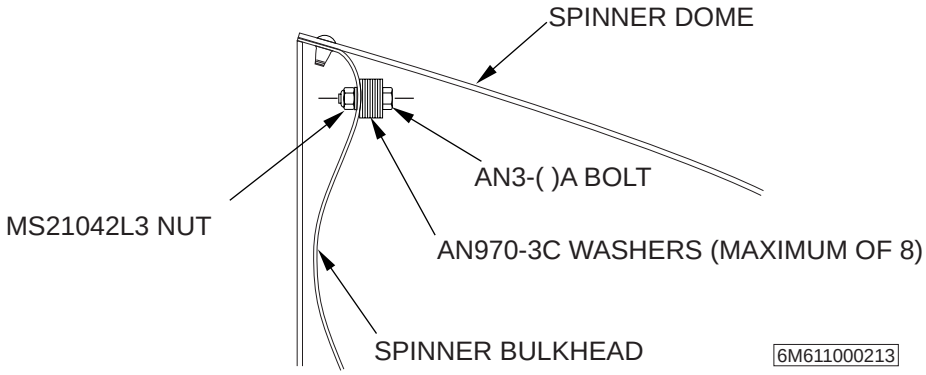
CAUTION: ENSURE BULKHEAD RING DOES NOT COME INTO CONTACT WITH VIBRATION PICKUP UNIT.

(4) Connect strobe light power lead and Vibration Pickup Unit cables to their respective receptacles on the Vibration Analyzer/Dynamic Balancer.

CAUTION: TIE DOWN ALL CABLES TO PREVENT IT FROM COMING INTO CONTACT WITH THE PROPELLER DURING ENGINE RUN.

P. Calibration Procedure

NOTE: Before balancing the propellers, remove the spinner and lubricate each propeller blade. Make sure to note the position of the spinner prior to its removal so that it may be reinstalled in the same position upon completion of lubrication. Recommended lubricants are: MIL-G-81322A or B by Mobil. Mobil Grease #28, Aeroshell Grease #22, or marfak Grease #3 (Texaco). Recommended greasing procedure is to remove one zerk fitting from the blade clamp and then service through the opposite zerk fitting until the clamp is full. The zerk fitting is then reinstalled and one additional shot of grease is added. Reinstall propeller spinner before propeller balancing.



Propeller Balancing
Figure 213

(1) Set the POWER switch on the Vibration Analyzer/Dynamic Balancer to AC. Allow one minute for warmup.

(2) Set the DISP-VEL (Displacement-Velocity) selector switch to DISP. Set the AMPLITUDE RANGE selector to TEST.

(3) Set the FILTER-OSC selector to OUT. A meter reading within the green test area on the amplitude meter indicate normal circuit operation.

(4) Set the FILTER-OSC selector to SHARP. Set the FREQ RANGE CPM selector to 500-5k. Tune the FREQUENCY TUNING DIAL through a reading of 360 on the FREQUENCY TUNING METER and note that the meter reading peaks on the AMPLITUDE METER when the FREQUENCY TUNING METER reads 360. The peak should read within the green test area of the AMPLITUDE METER. Repeat the test with the FILTER-OSC selector set to BROAD.

(5) Set the FILTER-OSC selector to OSC. Switch the strobe light to ON. The strobe light should flash at a rate indicated by the FREQUENCY TUNING DIAL and the FREQ RANGE-CPM selector.

Q. Balancing Procedure

(1) Set the FREQUENCY TUNING DIAL to read 200 CPM on the FREQUENCY TUNING METER. Set the INPUT SELECTOR to L (on the Model 330 unit) or place the PICKUP switch (on the Model 350 unit) to the applicable connected receptacle, either number 1 or number 2. Set the POWER switch to AC. Set the FREQ RANGE-CPM selector to 500-5K. Set the FILTER-OSC selector to OSC. Set the AMPLITUDE RANGE selector to 30 (mils-in/sec). Set the DISP-VEL switch to DISP.

(2) Start the aircraft engine and idle until warm. For the TPE-331-3U-303G engine, place the speed lever into high position and increase power to 1,200 foot-pounds torque. For the TPE-331-10U-501G engine, advance the speed lever and increase power until 50% torque is attained.

(3) Adjust the FREQUENCY TUNING DIAL until a number/letter designation on the spinner bulkhead appears stationary within the flashing strobe light. Record the number/letter nearest in line with the strobe light.

(4) Switch the FILTER-OSC selector to the sharp position.

(5) Read the amplitude meter and change the range as necessary in order to achieve the most accurate MIL reading. Record the amplitude MIL reading.

(6) Shut down the engine. Attach two washers (P/N AN970-3C) per mil (reading from step 5) to the spinner attach screw with a MS35207 (10-32) screw of appropriate length. On the TPE-331-3U-303G engine, attach the washers in line with the screw hole nearest to 130° clockwise from the recorded number/letter designation on the spinner bulkhead (approximately the second nutplate of the bulkhead ring, refer to Figure 212). On the TPE-331-10U-501G engine, attach the washers in line with the screw hole nearest to ten degrees clockwise from the recorded number/letter designation on the spinner bulkhead (approximately the forth nutplate on the bulkhead ring (Figure 212). Use no more than eight washers per screw.

(7) Restart the engine and repeat steps 2 through 5. Adjust position of washers as necessary until amplitude reading in step 5 reads less than 1.0 mils.

NOTE: Static weights may have been previously placed on one or more propellers at the respective propeller clamp by the manufacturer. To reduce the amount of washers required to dynamically balance the propellers, the manufacturer's static weights may be removed.

(8) Feather the propeller. Remove the balancing washers from spinner, noting their previously mounted position and number of washers required for permanent mounting to the spinner bulkhead. Remove the spinner, noting the position of the spinner prior to its removal so that it may be reinstalled in the same position later. Attach the required balancing washers permanently, along the same radial where preciously installed on the spinner, to the spinner bulkhead. Use appropriate length AN-3 ()A bolt securing with MS21042L3 nut (Figure 213).

(9) Reinstall the spinner and remove strobe light and vibration pickup unit from aircraft.

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DEICE BOOTS – DESCRIPTION AND OPERATION

1. General

A. To conserve electrical power, current is cycled to the deice heaters at timed intervals rather than continuously. Each deice has two separate heaters, one for the outer half and one for the inner half. By heating all outer or inner heaters simultaneously, rotational balance is easily maintained.

B. The use of heat for deicing reduces the grip of the ice resulting in removal by centrifugal force and the blast of the airstream. The system may be used continuously in flight if needed.

NOTE: Heating may begin at any phase in the cycle, depending on the position of timer when the switch was previously turned off.

C. Each deice boot consists of an outer and an inner wire mesh protected by fabric plies and encased inside an oil abrasion resistant rubber mat. The side of the deice boot that is to be cemented to the propeller blade has a dull finish, while the outer or air side has a glossy finish.

D. Each boot has three separate electrical leads; one for the outer portion, one for the inner portion, and one which is a common ground. Each of these leads is marked accordingly. An unmarked ground can be identified by using an ohmmeter across the three leads. Two will show identical resistance and the third will show double this resistance. The high resistance lead is the ground.

DEICE BOOTS – MAINTENANCE PRACTICES

1. Removal/Installation – Deicer Boots

A. Remove – Deicer Boots

NOTE: Cushion the jaws of the pulling tool to prevent damage to the deicer, unless it is to be scrapped. If the deicer has failed under warranty and is to be returned, use extreme care to prevent additional damage to the deicer during and after removal.

(1) Remove the propeller spinner.

(2) Disconnect the propeller deicer terminals from the studs on the engine spinner bulkhead.

(3) Remove the clamp which holds the wires to the spinner bulkhead.

(4) Remove the clamp which holds the wires to the propeller hub.

(5) Remove the teflon terminal mount and disconnect the wires.

(6) Use methyl propyl ketone (MPK) or toluol to soften the adhesion line between the deicer and the propeller blade. Starting at one corner, loosen enough of the deicer to grasp in the jaws of a vise grip plier or similar tool. Apply a steady pull to remove the deicer from the propeller surface slowly and carefully while continuing to use MPK or Toluol to soften the adhesion line.

(7) If the condition of deicer after removal is not important, strip it from the blade without solvent. Deicer may tear or come off in pieces.

(8) Remove the residual cement from the blade. Use solvent or Turco No. 3 (Turco Products Co., one pint per six deicers) as necessary.

B. Deicer Boots Material Kit List

(1) Material Required:

(a) One deicer boot for each propeller blade is supplied in each B.F. Goodrich propeller deicing system kit. Replacement deicer boots may be ordered from the nearest service center by specifying the airplane make and model number or deicer part number. The following materials and tools are not supplied in the kit. They should be purchased from local suppliers.

1 Filler EC801, EC1031 (3M Company); three and one-half ounce kit for six blades; or

2 Filler 750 (3M Company); five ounces is required for six blades.

3 Cement 1300L or EC1403 (Minnesota Mining and Manufacturing Company); one pint is required for each six blades. Sealer A56B (the B.F. Goodrich Company); one-half pint is required for each six blades.

4 Tackifying solvent – Toluol or MPK (see note below).

5 Sealer 82076-1 and -2 (B.F. Goodrich Aerospace and Defense Products, Akron, Ohio); one-half pint is required for six blades; or

6 Sealer V66V27, F63B8, and R7K69 (Lowe Bros., Dayton, Ohio); one-half pint is required for six blades; or

7 Sealer kit A-2743 (Hartzell Propeller Co., Piqua, Ohio); one quart kit is provided.

8 Cleaning cloth – any clean, lint-free cloth.

9 One inch paint brush.

10 Two inch rubber hand roller.

11 1/4" metal hand stitcher.

12 Scissors

13 Masking tape

NOTE: MPK may be used instead of toluol to tackify cement, but it provides approximately ten seconds working time for deicer boot applications, whereas toluol provides approximately 40 seconds working time.

C. Blade Preparation

NOTE: The inboard end of the deicer should be approximately one-half inch inside the point where the blade emerges from the spinner. All deicer boots on a single propeller must be located the same distance from the hub for rotational balance.

(1) Place a center mark at the hub end of the blade in line with the blade's leading edge. Determine the location by sighting along the leading edge. Starting at the hub, center the deicer boot on this center mark and fit the strap clamp on the hub. Compare its position with the position of the strap. Adjust the position of the boot as necessary to make the lead strap agree with strap clamp. Hand fit the edges and hold the boot in this position while masking off an area approximately one-half inch from each side of the boot and one-half inch from the outer end.

(2) Remove the boot and any paint within the masked off area down to bare metal. Clean the area thoroughly with MPK or Acetone. For final cleaning, quickly wipe off the solvent with a clean, dry, lint-free cloth to avoid leaving a film.

NOTE: Hartzell propellers have a special protective coating not easily removed by solvents alone. Use 320 grit (or finer) emery paper and MPK or toluol to remove coating down to bare metal. Remove in spanwise direction only. Be careful not to score blade.

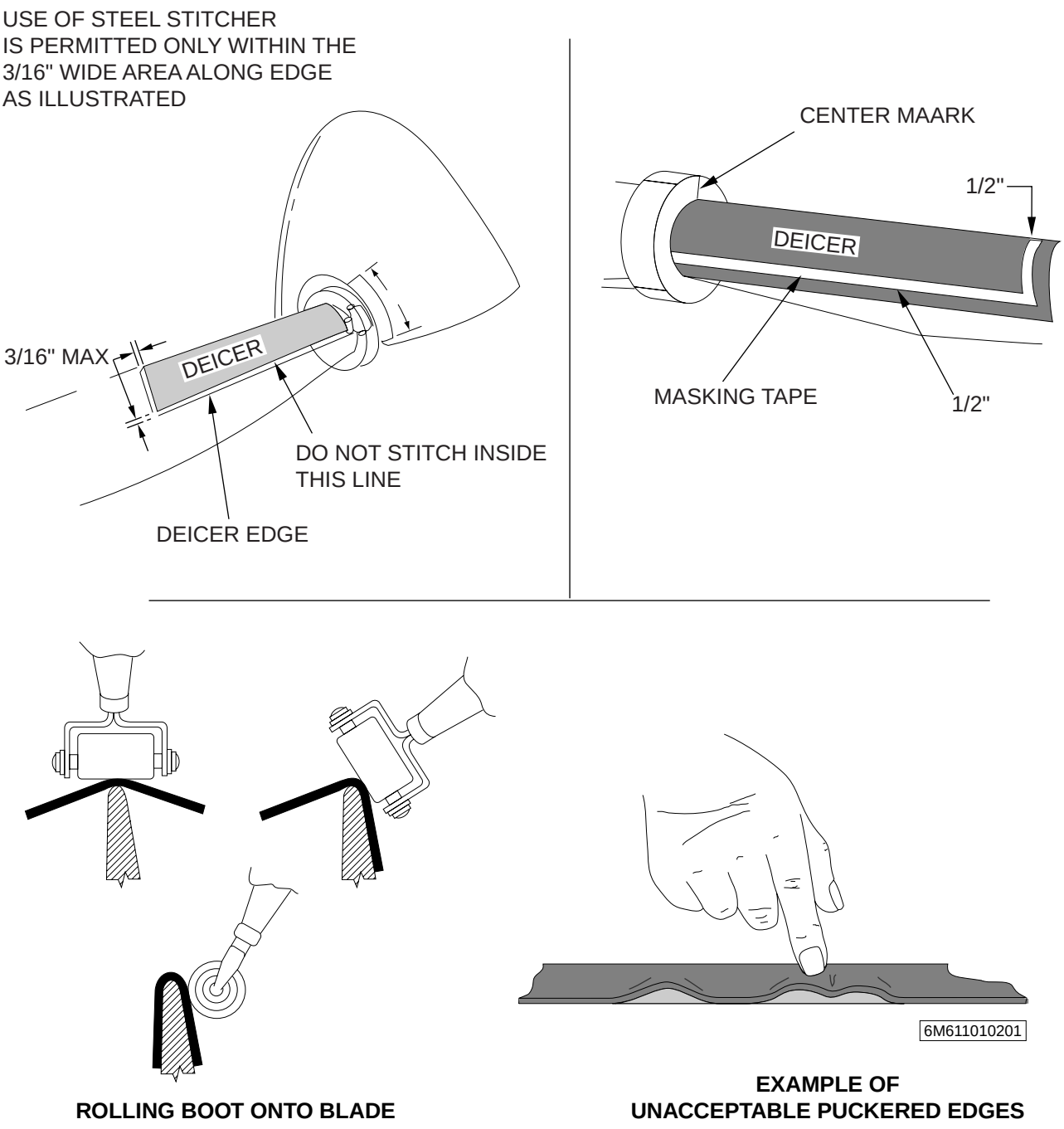
D. Cement Application

(1) With a pencil (or equivalent), very lightly mark a centerline at the hub of the propeller blade and on the tape at the outboard edge of the marked area. Using a silver pencil, also mark a centerline on the glossy (outer) side of boot.

(2) Moisten a clean cloth with MPK or acetone. Clean the dull (back) surface of the boot. Change the cloth frequently to avoid contamination of the clean area.

(3) Thoroughly mix the cement.

NOTE: To prevent curling of the boot edges, apply masking tape to the edges on the glossy side before applying cement to the dull side. Remove the tape before installing the boot.



ROLLING BOOT ONTO BLADE

**EXAMPLE OF
UNACCEPTABLE PUCKERED EDGES**

**Deice Installation
Figure 201**

(4) Apply one even brush coat of cement to dull back surface of boot. Apply cement to the proper length of the lead strap. Allow to air dry for a minimum of one hour at 40°F (4.44°C)or above, when the relative humidity is less than 75%. If the humidity is 75% to 90%, allow two hours drying time. Do not apply cement if the relative humidity is higher than 90%. After the cement is dry (not tacky), evenly apply a second brush coat of cement.

(5) Apply an even brush coat of cement on cleansed surface of the propeller blade immediately after the second coat of cement has been applied to boot. Timing is very important because the cement on both surfaces must reach the tacky stage at the same time.

E. Installation – Deice Boot (Figure 201)

NOTE: For best result, make installation at normal room temperature.

(1) When cement coats are tacky dry on both blade and boot surfaces, position the boot on the blade leading edge. Start at the hub end, using the centerlines as a guide. Make sure that the strap will fall in previously marked position. Working outward toward the tip, tack boot centerline to leading edge of the propeller blade. Use tackifying solvent as necessary to reinstall the boot. When it is positioned correctly, roll firmly along the centerline with a rubber roller.

(2) Gradually tilt the rubber roller by carefully bonding the boot over each side of the blade contour to avoid trapping air between the blade and boot.

(3) Roll from the centerline outward to the boot edges.

CAUTION: BE ESPECIALLY CAREFUL TO WORK OUT EXCESS MATERIAL AT THE OUTBOARD EDGE OF THE BOOT BEFORE THE OTHER EDGES ARE COMPLETELY ROLLED DOWN.

NOTE: If excess material at edges tends to pucker, work the puckers out smoothly and carefully with your fingers.

(4) Roll the tapered edges (especially the inboard edges) of the boot with the metal stitcher.

CAUTION: TO AVOID DAMAGE TO THE BOOT RESISTANCE WIRES, DO NOT USE THE METAL STITCHER ON THE BODY OF THE BOOT.

5) To add filler, locate masking tape approximately one-eighth inch beyond cemented area around deicer to permit filler to contact bare metal. Apply one even coat of filler to area around inboard end, strap, and sides of deicer Filler must cover one-eighth inch of bare metal, cemented area, and about one-eighth inch of deicer. Make sure a filet of filler covers the area between the lead strap and the blade as shown. Immediately remove masking tape and allow filler to dry for six hours (24 hours for 3M 750 filler).

(6) Relocate the masking tape approximately one-eighth inch beyond the cemented deicer boot. This will permit sealer to be applied directly to the metal. Apply one even brush coat of 82D76-1 or -2 sealer around the edges of the installed deice boot to seal the edges and dress up the appearance. Completely cover the cemented area on the blade and approximately one-eighth inch of the boot. Immediately remove the masking tape after the sealer has been applied.

F. Cement Curing Time

(1) Cement will set in one-half hour; however, allow 12 hours for the cement to cure before starting the engine. Allow 24 hours for the cement to cure before operating the deicer boot.

G. Installing Wire Clamps

(1) The deicer wires are secured to the propeller hub by a clamp. When the deicer boots are replaced, the old clamps or retainers may be reinstalled.

(2) Install the clamp with the correct screws through the clamp and into the tapped holes in the propeller hub. Use only the existing holes. Do not install safety wire at this time.

H. Installation of Clamps and Terminals

CAUTION: NEVER USE TYPE B STAR WASHER (TEETH ON OUTER DIAMETER) ADJACENT TO TONGUE OF ELECTRICAL TERMINAL.

(1) The deicer wires are secured to the spinner bulkhead with a metal clip through which the deicer lead strap is inserted. Relocate the parts in the same positions from which they were removed.

(2) The deicer wires terminate in three narrow leads to which electrical terminals are attached. Attach these terminals to studs on the spinner bulkhead.

CAUTION: THE DEICER LEAD STRAP MUST NOT BE TWISTED MORE THAN ONE TURN (180 DEGREES) IN LINING UP THE RESPECTIVE LEADS.

(3) Make sure that there is no slack in the leads between the terminals and the clip.

(4) Check each deicer boot installation to be certain that the propeller blades can be moved throughout the entire pitch range without applying tension to the deicer leads.

I. Balancing Weights

(1) Reinstall the balance weights if they were removed during deicer boot maintenance. The original weights (or equivalent) may be used. The weights must be installed in their original position on the propeller hub, but may be installed above the clamp. The clamp and weights must not interfere with any part of the propeller assembly.

(2) If no balancing weights are placed over the clamp, use a washer under the screw head.

(3) Install safety wire on the screws.

CAUTION: OBSERVE THE 12 HOUR CEMENT CURING TIME BEFORE STARTING THE ENGINE. ALSO OBSERVE THE 24 HOUR CEMENT CURING TIME BEFORE OPERATING THE DEICER BOOTS, HANDLE THE PROPELLER BLADES CAREFULLY TO PREVENT DAMAGE TO THE BOOTS. CHECK THE POSITION OF THE DEICER STRAPS TO BE CERTAIN THAT THEY ARE NOT PINCHED OR SNAGGED BY NEARBY COMPONENTS.

J. Blade Deicer, Lead Strap Repair (Fig 202)

(1) Cut the patch (0.020 inch rubber) to be about one-forth inch larger on all sides that the damaged area.

(2) Use the rubber roller to press the patch into place; a metal roller will damage the wires in the strap.

K. Wrinkled Deicers

(1) If the edge of the deicer is wrinkled or loose, try recementing.

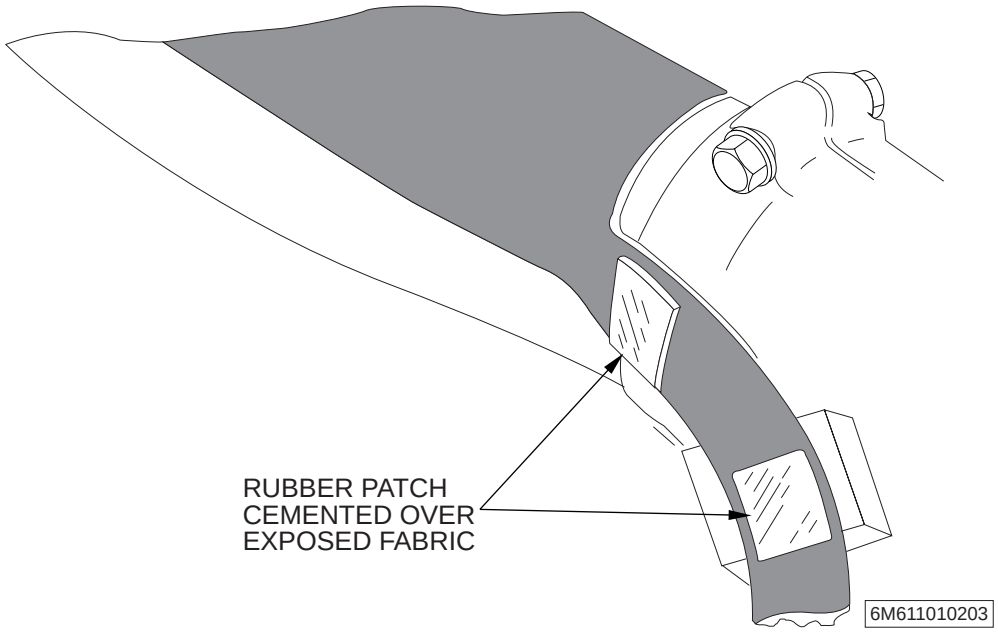
(2) Use MPK or toluol to loosen the bond for an additional one-forth inch beyond the loose or wrinkled area.

(3) Apply one coat of EC-1403 cement to the deicer and propeller bonding surfaces and allow to air dry for one hour.

(4) Then apply a second coat of EC-1403 cement to both the deicer and the propeller bonding surfaces. Retackify with MPK or acetone and press with fingers to work out wrinkles or to secure any loose edges.

(5) If the material has stretched and will not cement flat, replace the deicer element.

NOTE: Use polyethylene or Teflon film between the deicer element and the propeller while cement coats are drying so that they do not stick together.



Deicer Boot Repair Figure 202

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DEICE PLATTER – DESCRIPTION AND OPERATION

1. General

A. The deice slip ring assembly is constructed of machined aluminum with three copper commutator rings bonded to the aft side.

(1) Three deice leads for each propeller blade are located on the forward side of the platter and are attached to the commutator rings on the aft side through holes.

(2) The assembly is secured with bolts through the spinner bulkhead to the propeller hub.

B. The system is electrically operated and powered by the aircraft electrical system.

(1) Electricity is applied to the brush block on the engine gear box housing.

(2) Six brushes, under spring tension, ride on the commutator rings of the deice platter and transfer this energy to the deice leads on the forward side of the assembly.

DEICE PLATTER – MAINTENANCE PRACTICES

1. Servicing – Deice Platter

A. Commutator Repair

(1) Replace the commutator rings with a new assembly if found cracked, damaged structurally, have irreparable damage, or give evidence of electrical shorts. When replacing, align the commutator rings on the brushes (refer to 61-10-30, Figure 201).

(2) Ensure the commutator rings run perpendicular to the brushes. Any misalignment may result in wobble. Wobble in excess of 0.005 (five thousandth) inch will cause arcing and rapid failure of the brushes and commutators. Ensure the brushes and rings are clean and free of grease. Use CRC2-26 solvent (from Corrosion Reaction Consultants, Inc. of Philadelphia, Pennsylvania) to remove any grease or other contaminants.

B. Deice Platter Runout

(1) Attach a dial indicator gauge or equivalent to the engine case with the stylus riding on the commutator ring.

(2) Zero (0) the dial indicator.

(3) Grasp the propeller blade and rotate the deice platter slowly. Note the deviation indicated on the gauge.

NOTE: Due to the loose fit of some propeller thrust bearings, a considerable error may be induced into the readings by pushing the propeller in or out during rotation. Care must be taken to exert a uniform pull on the blade to hold any error to a minimum.

(4) If the total indicator reading is within 0.005 (five thousandth) inch, the deice platter is acceptable.

(5) If platter is not within tolerance, place AN960 washers on the attachment bolts between the deice platter and spinner bulkhead.

(6) Repeat steps 1.B.(1) through (3).

(7) If deice platter still has a deviation greater than 0.005 inch total indicator reading, make small washer shaped shims and place over the washers previously installed.

(8) Check and adjust as necessary.

2. Removal/Installation – Deice Platter

A. Removal – Deice Platter

(1) Remove the propeller assembly from the engine.

(2) Remove the start locks, two bolts each lock (not applicable on HC-B3TN-6M propellers).

(3) Remove the screws from the deice wires.

(4) Remove the remaining attaching hardware.

NOTE: Support the spinner bulkhead when removing the bolts from the deice platter. This will free the platter, spinner, mounting plate, spacer, and spinner bulkhead.

B. Installation – Deice Platter

(1) Assemble the deice platter, mounting plate, spacer, and spinner bulkhead.

(2) Insert the deice wires through the spinner bulkhead.

(3) Install the deice platter with the commutator rings on the aft side and with the spinner bulkhead engine side facing aft. Line up the arrows on the mounting plate and bulkhead. Install the attaching hardware from front to rear. Torque in accordance with instructions contained in this section (propeller station).

(4) Install the start locks with hardware from rear to front. Torque 48 inch-pounds (5.42 nm) (not applicable on HC-B3TN-5M propellers).

(5) Secure the deicer wires to the terminal strip on the spinner bulkhead.

(6) Secure the spinner bulkhead assembly to the propeller assembly. Torque to 48 inch-pounds (5.42 nm).

(7) Install the propeller assembly on the engine.

C. Deice Platter Rework (Figure 201)

(1) Thoroughly clean slip ring or starter ring gear assembly with MPK before beginning resurfacing.

(2) Check flatness of slip ring or starter ring gear assembly mounting surface. It must be flat within 0.005 inch overall.

(3) Locate assembly concentrically in lathe so that there is no more than 0.002 inch wobble or run-out over 360° rotation.

(4) Take a light cut for a smooth finish (29-35 micro-inches). Cut no deeper than required to remove surface damage. Do not machine past 0.475. Measure dimensions from mounting surface.

CAUTION: IN MACHINING, DO NOT ALLOW TEMPERATURE OF SLIP RINGS TO EXCEED 200°F (93.33°C).

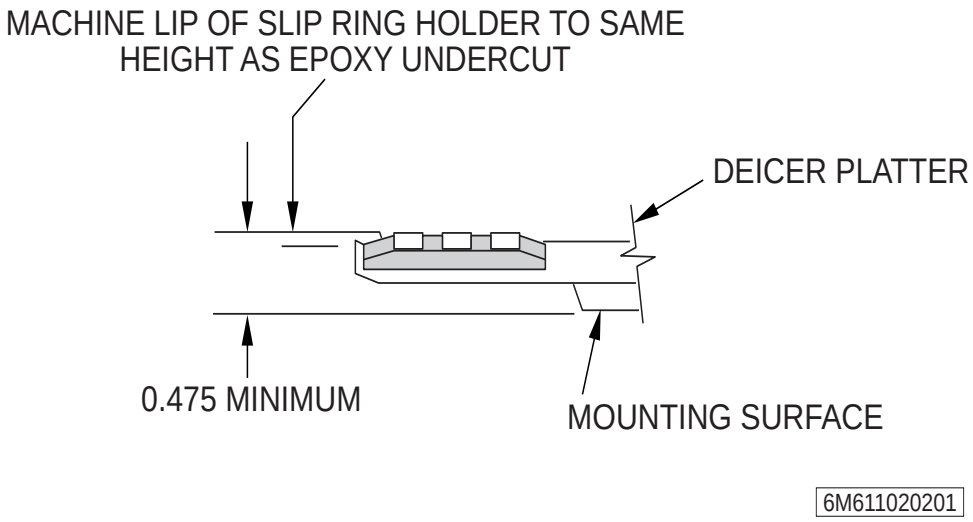
(5) Ensure that contact surfaces of slip rings are parallel within 0.005 inch and flat within 0.005 inch overall. Deviation not to exceed 0.002 inch over a four inch arc.

(6) Undercut insulation between and around slip rings 0.020 to 0.030 inch, as necessary. Slip ring holder outer edge must be undercut to same dimension as insulation. Do not machine outer edge of slip ring holders containing synchrophasing targets.

CAUTION: WHEN UNDERCUTTING INSULATION, WIDTH OF SLIP RINGS MUST NOT BE REDUCED MORE THAN 0.005 INCH.

(7) Deburr slip ring edges.

(8) Check insulation resistance between slip rings and to metal holder with a megger. Ensure that resistance is at least five megohms after one minute.



Deice Platter
Rework
Figure 201

BRUSH BLOCKS – DESCRIPTION AND OPERATION

1. General

A. The brush block assembly retains the brushes that ride on the commutator rings of the deice platter. It transfers electrical power from the aircraft source to the propeller blade deice boots.

2. Modular Brush Assemblies

A. A modular brush assembly may be used instead of a brush block assembly. Each modular brush assembly is comprised of three modules. Each module consists of a plastic housing with an integral brush and spring. Screws hold the modules and spacers together.

BRUSH BLOCKS – MAINTENANCE PRACTICES

1. Adjustment/Check – Brush Blocks

A. Inspection/Adjustment – Brush

(1) Examine the mounting bracket and housing for cracks, deformation, or other physical damage. Assure that the leads are not cracked, chafed, or binding. Check for tight connection.

(2) Check to assure that each brush rides fully on the commutator rings over 360° of rotation. If the alignment is not satisfactory, place shims under the brush block or elongate the mounting bracket holes as necessary to place the brush block in the correct position. If the brushes ride both high and low in relation to the commutator rings throughout the 360° rotation, the commutator assembly is eccentrically mounted and the spinner bulkhead must be replaced.

(3) Check for the correct clearance of the brush block to the slip rings. If it is not correct, loosen the mounting screws and reposition them in the elongated screw holes to correct the brush block position and then tighten the screws. If necessary, place shims between the thrust bearing plate and the mounting bracket.

B. Brush (Module) Replacement

(1) Brush modules should be replaced when three-eights inch of brush material remains; brush modules must be replaced when one-forth inch remains.

(2) When a brush wears out, the module containing it should be replaced as follows:

(a) Remove the modular brush assembly by removing the attaching hardware and disconnecting the engine wire harness.

(b) Remove the assembly screws and separate the modules and spacers.

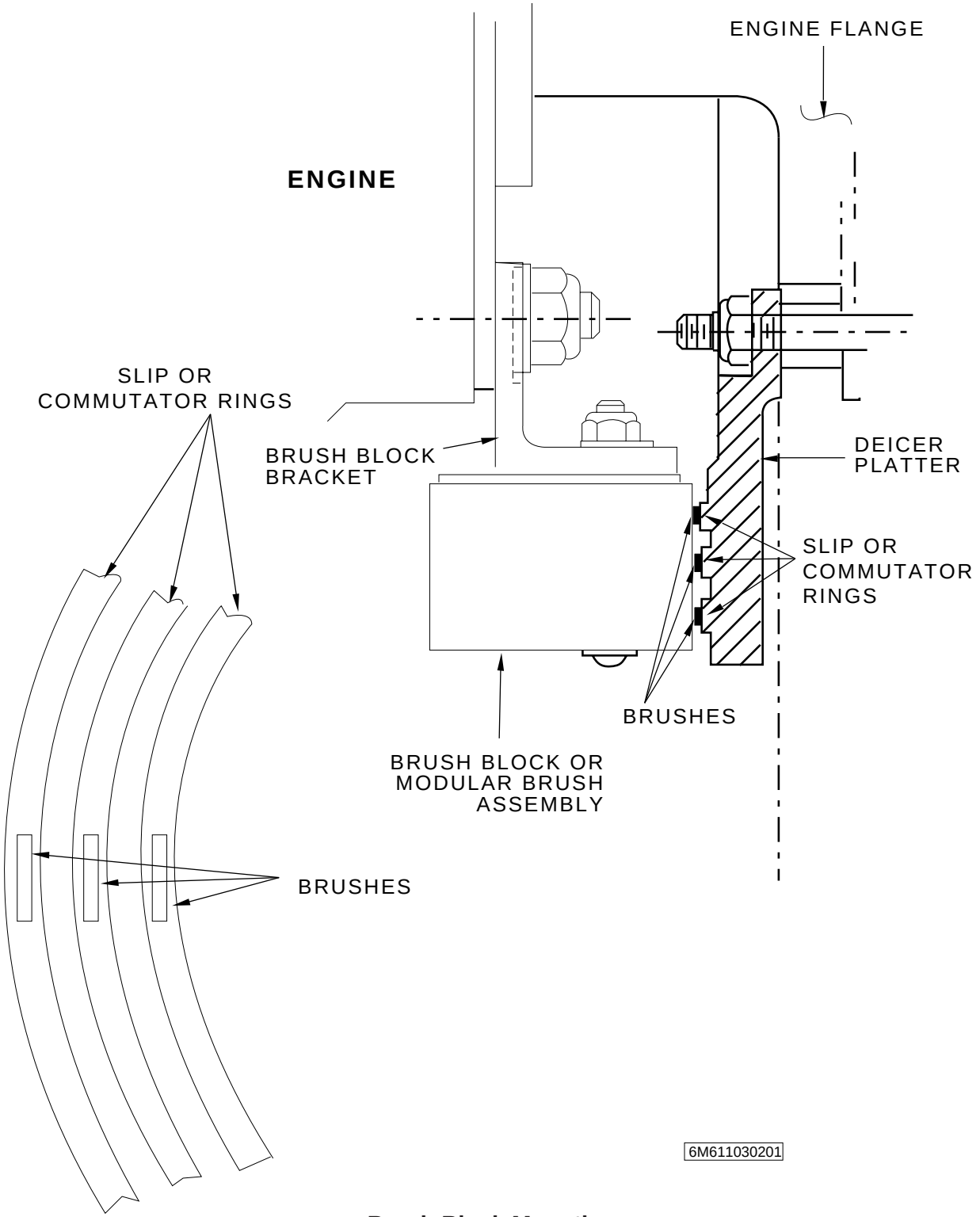
(c) Replace the module with one having the same part number. The part number is etched into the surface of the module plastic housing.

(d) Restack the modules and spacers. Check A/B/C or C/A/B stacking arrangement and location of spacers. Assure that the assembly is square, then tighten screws.

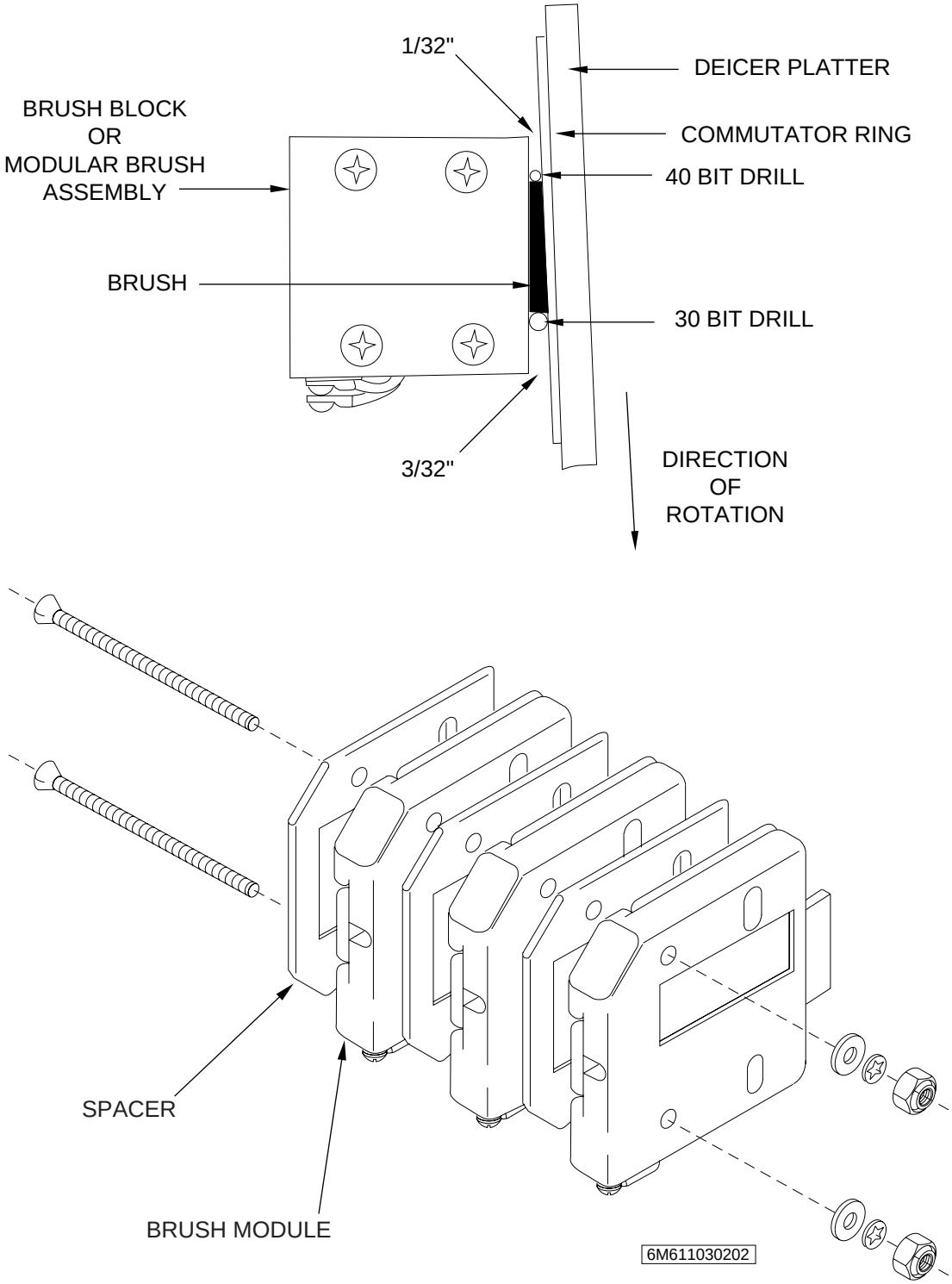
(e) Reconnect the aircraft wire harness to the A, B, and C terminals of the modules. Ensure that the adjacent ring terminals are not touching.

(f) Install the assembly on the aircraft with the attaching hardware from step 1.

(g) Check the assembly for correct alignment and adjust as necessary.



Brush Block Mounting and Alignment
Figure 201



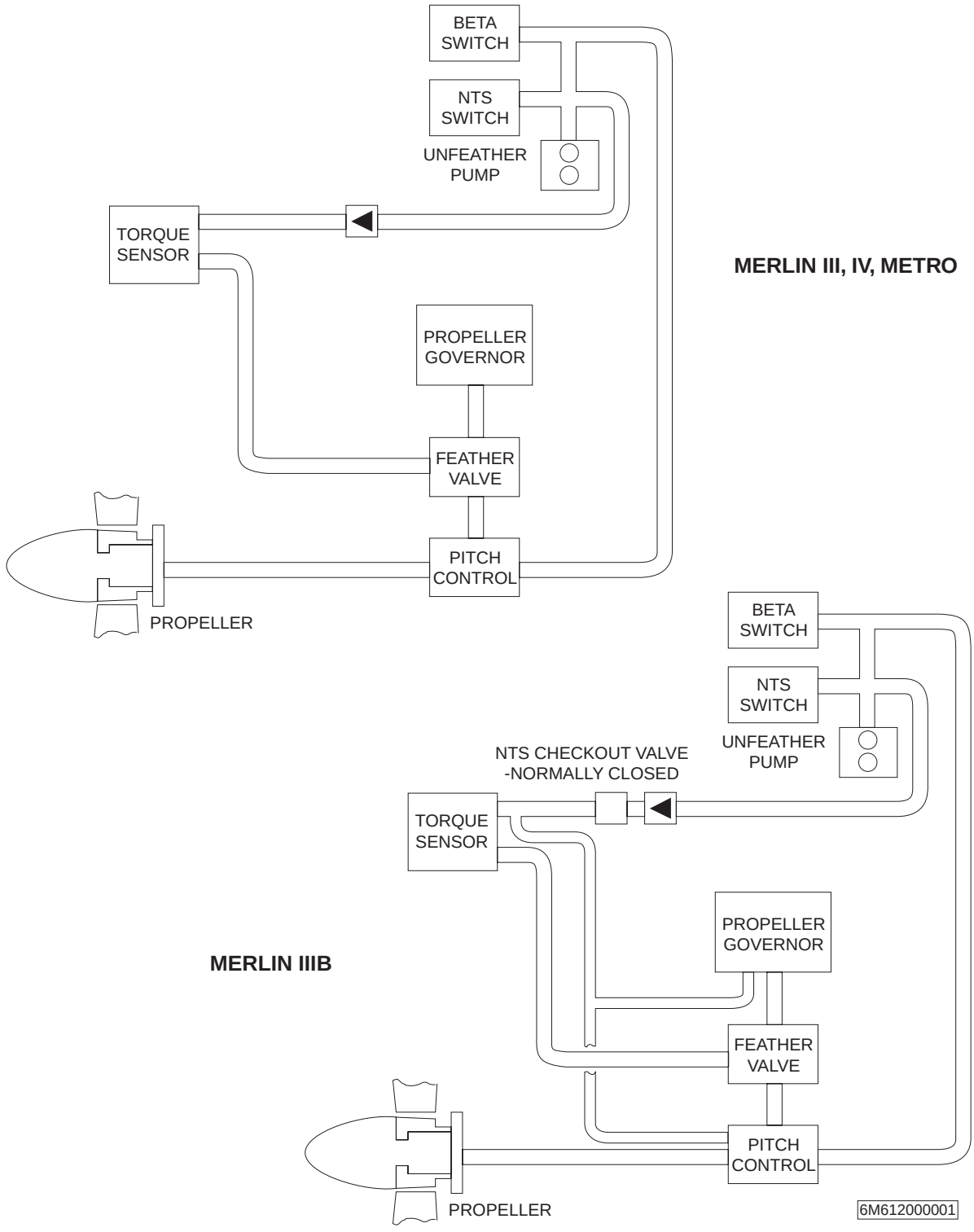
Brush Block installation
Figure 202

PROPELLER CONTROL SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The propeller control system involves an interaction of several engine installed components in the basic engine oil system. As shown in Figure 1, the basic components of the propeller control system are the propeller governor, the pitch control, the feather valve, engine torque sensor and the unfeather pump. The control system operates in two basic modes; propeller governing mode and the beta mode. In both cases the source of high pressure oil for controlling the propeller is the propeller governor. In the propeller governing mode, the governor itself varies the oil pressure to in turn vary the pitch of the blades to maintain the pilot selected RPM. In the beta mode the propeller governor provides the oil pressure and the pilot manually varies the pitch control to change the blade angle. The unfeather pump provides oil pressure for moving the blades out of the feather position on the ground and for air starting the engine. Because these components are part of the basic engine control system, they will not be discussed further in this chapter.

PROPELLER CONTROL SYSTEM – DESCRIPTION AND OPERATION



Propeller Control System
Figure 1

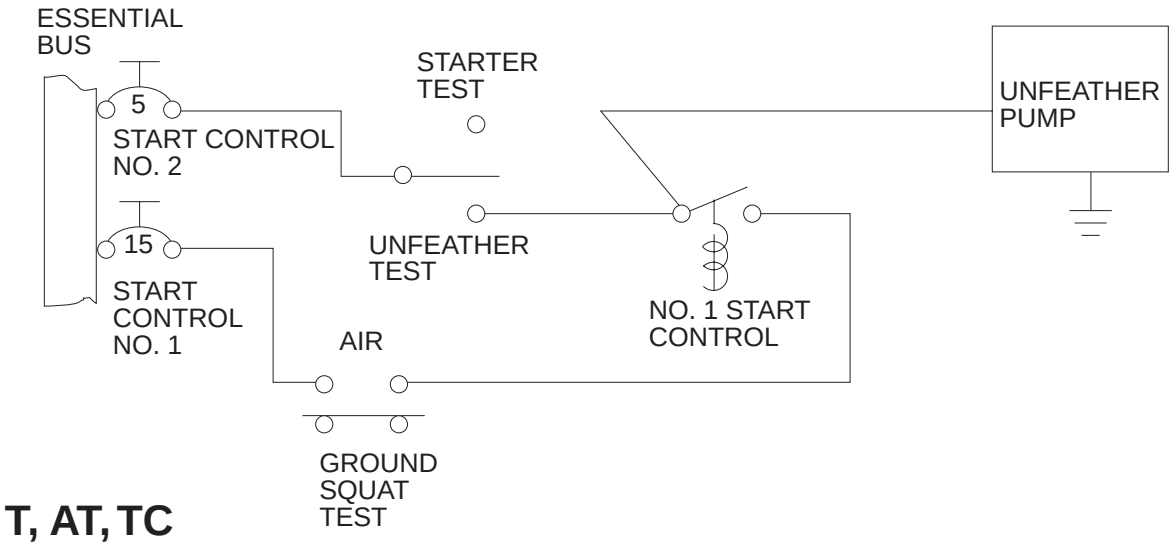
PROPELLER UNFEATHERING PUMP – DESCRIPTION AND OPERATION

1. General

A. The unfeathering pumps are electrically driven, positive displacement, rotor type pumps. They are located in each engine nacelle on the left side of the engine on the engine mount. They supply oil from the engine oil tank, pressurized to 160 psi, to the propeller pitch control unit.

B. The pump is activated on the ground with the unfeather switch located on the left console, the pump is used for propeller unfeathering, if the propeller blades had not been positioned on the start locks during shutdown. In flight, the pump is activated when the start cycle is initiated (as in the case of an airstart after engine shutdown). With the landing gear squat switch closed (strut extended), the number one start control relay delivers power to the unfeathering pump. The pump terminates operation at 50 percent RPM when the start relay opens.

C. Operation of the unfeathering pump is limited to 30 seconds with hot oil and one minute with cold oil to prevent oil buildup in the accessory case. Oil can be scavenged from the accessory case to the oil tank by rotating the propeller by hand approximately 100 revolutions or by motoring the engine for three to five seconds with the starter test switch.



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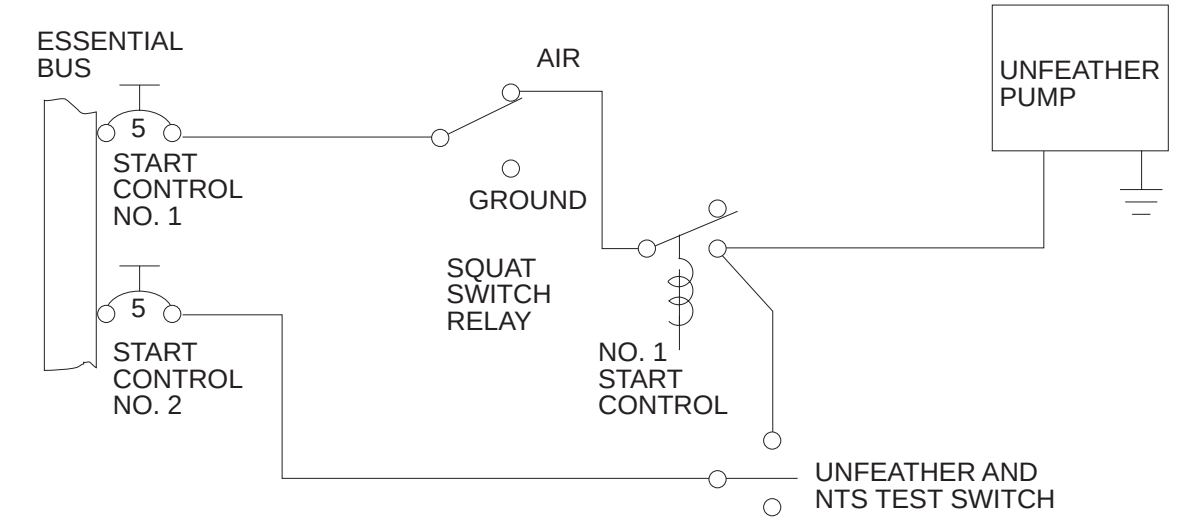

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Propeller Unfeathering Pump Electrical Figure 1

PROPELLER UNFEATHERING PUMP – MAINTENANCE PRACTICES

1. Removal/Installation – Propeller Unfeathering Pump

A. Removal – Propeller Unfeathering Pump

(1) Remove the engine left hand side cowling.

(2) Disconnect the pump motor electrical connection.

(3) Disconnect pump inlet, outlet, and vent lines. Install protective plugs in the lines and ports of the pump to prevent foreign object contamination.

(4) Remove the safety wire from the four mounting bolts, remove the bolts, and remove the pump and motor assembly.

B. Installation – Propeller Unfeathering Pump

(1) Place the pump and motor assembly into position and install the four mounting bolts.

(2) Secure the bolts with safety wire.

(3) Remove the protective plugs (if installed) from the pump inlet, outlet, and vent ports and install the inlet, outlet and vent lines.

(4) Connect the pump motor electrical connector plug, and safety.

(5) Install the engine side cowling.

2. Inspection/Check – Propeller Unfeathering Pump

A. Inspection/Check – Propeller Unfeathering Pump

(1) Remove the left side cowling.

(2) Remove the cap screw from the pump motor.

(3) Install a 0 to 200 psi pressure gauge.

(4) Turn the battery switch ON.

(5) Turn the unfeather pump switch ON.

(6) The correct gauge reading is 150 psi.

(7) If the gauge reading is incorrect, repair as necessary. If or when reading is correct, continue below.

(8) Turn the unfeathering pump switch off.

(9) Turn the battery switch off.

(10) Remove the pressure gauge.

(11) Install the cap screw into pump motor.

(12) Install the left side cowling.

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PROPELLER SYNCHRONIZER – DESCRIPTION AND OPERATION

1. General

A. The synchronizer automatically matches the speed of one engine with that of another. The speed of the slave engine will follow changes in the speed of the master engine over a predetermined limited range. The limited range feature prevents the slave engine from losing more than a fixed amount of RPM in the event the master engine were to be feathered with the synchronizer on. Normal governor speed setting controls and procedures are unchanged. The system is controlled by an ON-Off switch on the cockpit pedestal. The switch is labeled CLIMB-CRUISE for ON and TAKEOFF-LANDING for OFF. The synchronizing system is comprised of the following components: a synchronizer control box, a speed setting actuator, and adjustable rod end assembly, a flexible rotary shaft, two synchronizer governors, electrical connectors, and cables.

2. Synchronizer Control Box

A. The synchronizer control box is mounted on the aft side of the forward pressure bulkhead and contains all the transistorized circuits. The system operates on 28 volts DC.

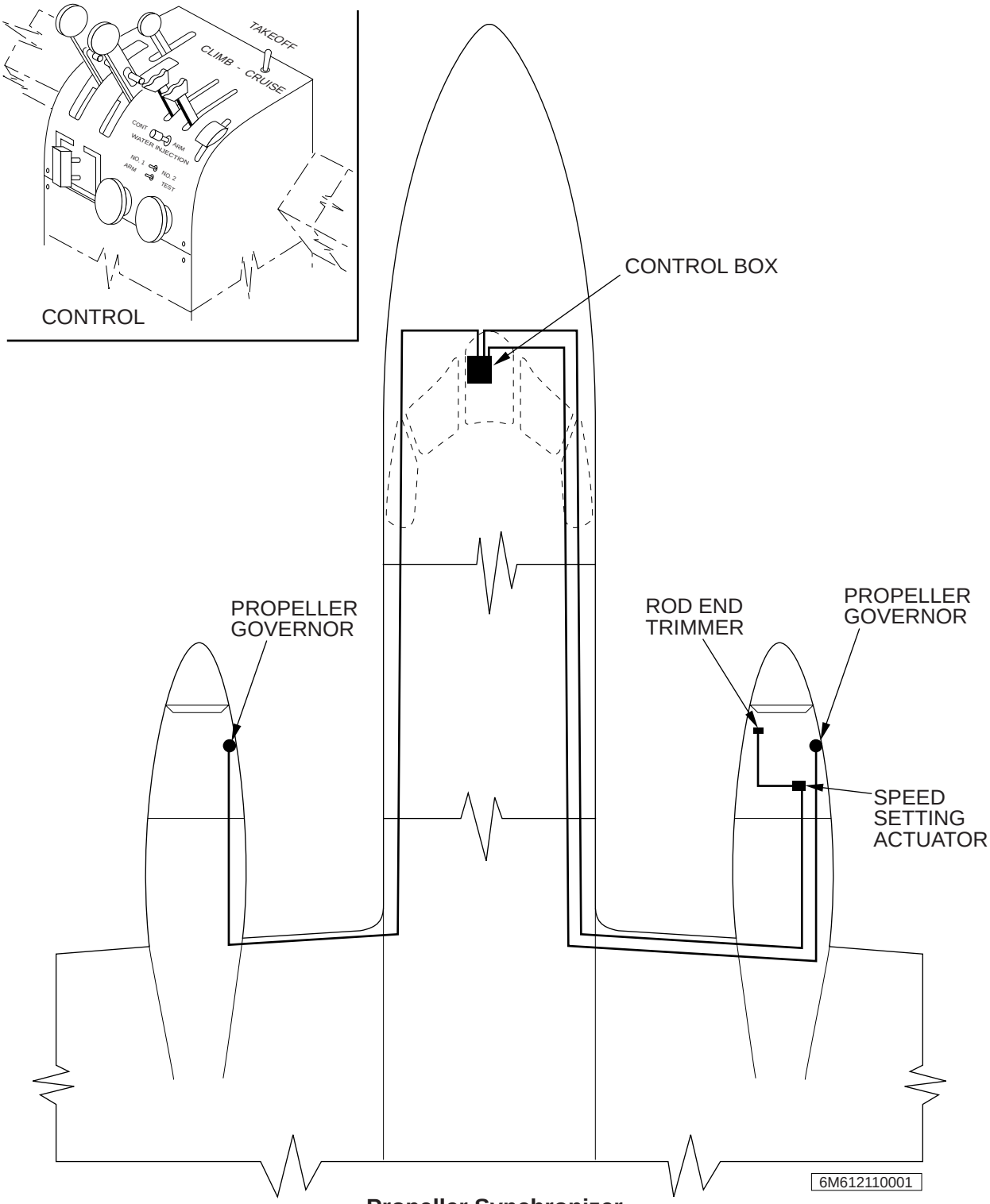
3. Speed Setting Actuator

A. The speed setting actuator is a step type motor that operates on command from the synchronizer control box. It is in the slave (right) engine nacelle.

4. Rod End Trimmer

A. The rod end trimmer mounts on the slave engine speed control and is linked to the actuator by a flexible shaft. The actuator trims the slave engine governor setting through the flexible shaft. A magnetic speed pickup is incorporated in each governor.

B. Alternating current generated by the magnetic pickup in each governor is fed into the control box. If any difference in frequency is detected between the two governors, the control box sends a signal to the actuator which trims the slave governor speed to match that of the master engine. The synchronizer continuously monitors engine speeds and adjusts the slave engine speed accordingly. Operating range of the actuator is approximately 3% RPM.



Propeller Synchronizer System
Figure 1

PROPELLER SYNCHRONIZER – MAINTENANCE PRACTICES

1. Removal/Installation – Propeller Synchronizer

A. Removal – Synchronizer Control Box

(1) Remove the pilot's and copilot's seats if necessary.

(2) Deflect the pilot's right rudder pedal aft and remove two of the four control box mounting screws.

(3) Deflect the copilot's left rudder pedal aft and remove the other two control box mounting screws.

(4) Disconnect both connector (Jones) plugs.

NOTE: On the earlier aircraft, the main wire bundle is located on the left side of the pedestal with no plug to the switch. On later aircraft, do not replace the switch unless it is necessary.

(5) Remove the control box.

B. Installation – Synchronizer Control Box

(1) Remove the pilot's and copilot's seat, if necessary.

(2) Deflect the copilot's left rudder pedal aft and place the control box in position.

(3) Install two of the four control box mounting screws.

(4) Connect both connector (Jones) plugs and secure with plastic tie straps.

(5) Deflect the pilot's right rudder pedal aft and install the other two control box mounting screws.

(6) Install the pilot's and copilot's seats, if they were removed.

C. Removal – Synchronizer Actuator

(1) Remove the right engine top cowling.

(2) Remove the safety wire on the flexible rotary shaft and remove the shaft.

(3) Disconnect the electrical connection.

(4) Remove the four mounting screws.

(5) Remove the actuator.

D. Installation – Synchronizer Actuator

(1) Place the actuator on the support bracket.

(2) Install the four mounting screws.

(3) Connect the electrical connection and safety.

(4) Install the flexible rotary shaft.

(5) Secure the shaft with safety wire.

(6) Install the top cowling.

E. Removal – Synchronizer Adjustable Rod End

(1) Remove the right engine inboard cowling.

(2) Remove the safety wire from the rotary cable and remove the cable.

(3) Remove the safety wire from the rod end and remove the rod end

(4) Back off the jam nut.

(5) Remove the bolt and nut holding the rod end to the fuel control serrated arm.

(6) Unscrew the rod end off of the speed lever cable.

NOTE: To ease engine trimming (after reinstalling the rod end) mark the threaded end of the speed lever and the position of the rod end on the serrated arm.

F. Installation – Synchronizer Adjustable Rod End

(1) Screw the rod end on the speed lever cable, taking note of the mark made before its removal and tighten the jam nut.

(2) Attach the rod end to the fuel control serrated arm and position it as it was before its removal. Secure it with bolt, washer, and nut.

(3) Attach the rotary cable and secure it with safety wire.

(4) Install the inboard cowling.

2. Adjustment/Test – Propeller Synchronizer

A. Adjustment – Trimmer Assembly

(1) The rod end trimming assembly is designed to have more rotary travel that the actuator that it is used with. If both the rod end assembly and the actuator are centered before attaching the flexible shaft, the actuator will provide the stops.

(2) Rotate the shaft in the rod end trimming assembly by hand and count the total number of turns available, Return the shaft to the center of its range.

(3) With the rod end trimming assembly centered, rig the RPM control just as with standard rigging.

(4) Count the total turns available in the speed setting actuator motor and turn it to the center of its range. (Turn it clockwise or counterclockwise by inserting a screwdriver in the actuator drive and turning it by hand). Recenter the trimmer an connect the flexible shaft to the actuator and trimmer.

B. Functional Test

(1) To test the operation of the synchronizer in flight, first synchronize the propellers manually and turn the synchronizer on. Slowly adjust the master engine propeller governor control to a point which is close to the end of this limited travel. Turn the synchronizer off. An unsynchronized condition will develop as the actuator moves to its mid-position. When the synchronizer is turned on again, synchronization will result. If the units do not become synchronized, the actuator has reached the end of its travel and must be recentered as follows:

(a) Turn the switch to TAKEOFF-LANDING (OFF).

(b) Synchronize the engines manually.

(c) Turn the switch to CLIMB-CRUISE (ON).

C. Actuator Travel Check

(1) Remove the flexible rotary shaft at the actuator. Insert a screwdriver in the actuator and rotate it through its range. It should rotate freely except for the ratcheting effect of the detent wheel. Normal output torque is 1.5 inch-pounds. The actuator should have a range of 52 - 54 pulses (3.0 turns), and stop positively at each end. The 5-1/4 turn actuator should have a range of 93.94 pulses. Leave the actuator in the center of its range.

(2) Adjust the governor trimmer by rotating the flexible shaft to check the amount of torque required. An excellent torque level is one that allows the trimmer to be adjusted by turning the squared end of the flexible cable with the fingers. It is more difficult to turn the shaft in the decrease RPM direction. But in no case should a turning fixture of over one-fourth inch diameter be used on the end of the cable to rotate the trimmer freely throughout its full range. Recenter the trimmer and attach it to the actuator. This action verifies an acceptable friction level of the rotating parts.

D. Governor Pickup Checks

(1) Resistance Check

(a) Check resistance of each pickup at governor.

90-110-Old style pickup with two separate wires exiting pickup.

90-110 ohms- Old style pickup with two separate wires exiting pickup.

52-68 ohms-New style pickup with white sheathing exiting pickup.

(2) Output check - check magnetic pickup output with the engines running at any cruise RPM. Check potential between pin receptacles 6 and 8. Also check potential between pin 7 and 8. Voltage must be a minimum of one-half volt at minimum cruise RPM, and a maximum of three volts at maximum cruise RPM. Voltages are read on a 5,000/volt AC Voltmeter. If you do not obtain this voltage, replace governor or have an approved governor repair shop adjust or replace the pickup.

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PROPELLER SYNCHROPHASER SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The synchrophaser system automatically matches the RPM of the two engines and allows the pilot to select any desired phase relationship between the propellers. The RPM of one engine will follow changes in the RPM of the other engine over a predetermined limited range. This limited range feature prevents either engine from losing more than a fixed amount of RPM should the other engine be feathered with the synchrophaser on. Normal governor speed setting controls and procedures are unchanged. Components of the synchrophaser system include the synchrophaser control box, propeller governors, and propeller synchrophaser target.

2. Synchrophaser Control Box

(1) The synchrophaser control box is installed in the cockpit on the left side of the pedestal. It contains an ON-OFF and phase select position and is used to control synchrophaser operation.

3. Propeller Governors

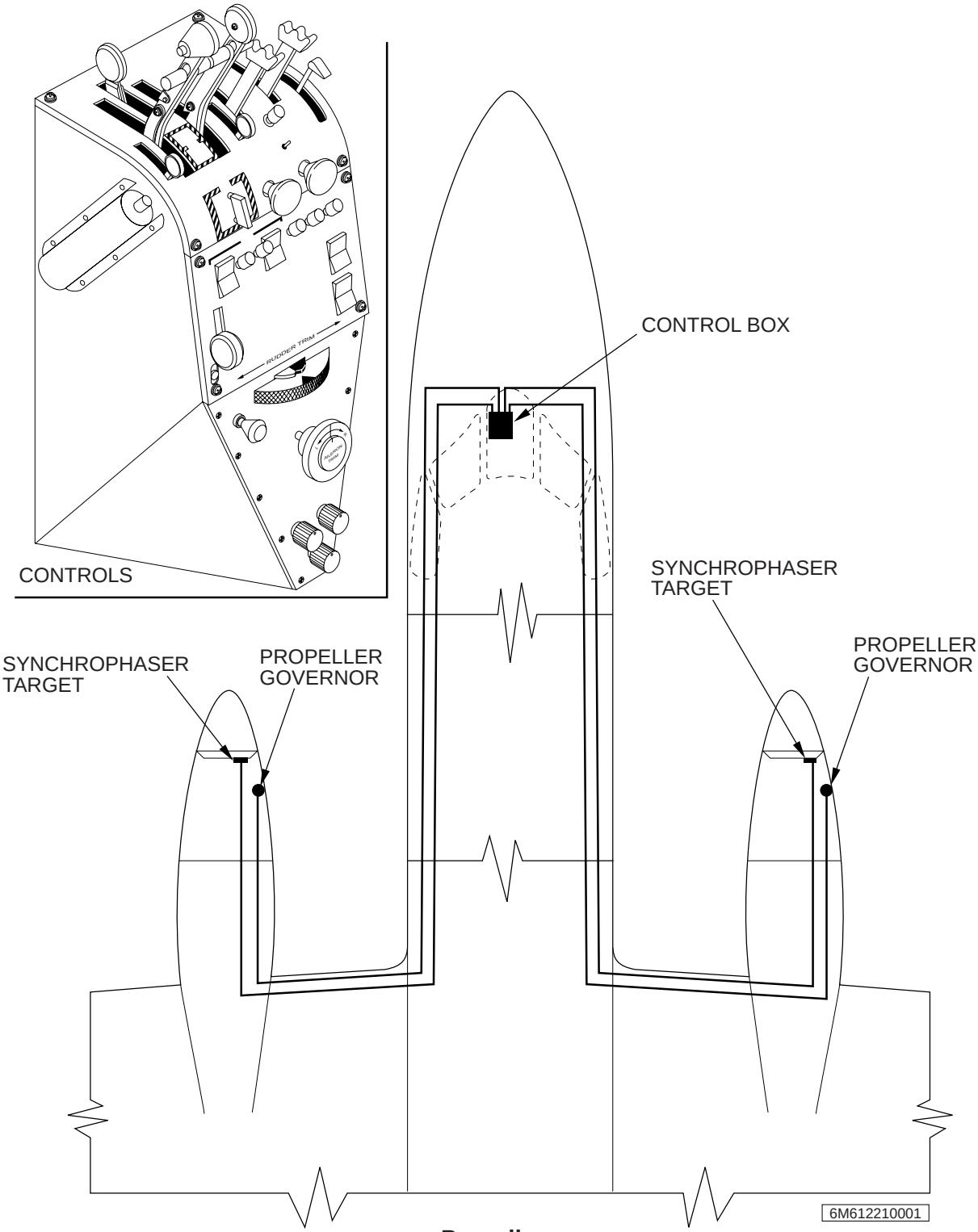
A. Each propeller governor contains an electromagnetic biasing coil installed within the governor. The coil affects the governor fly weights. If current through the coil is increased or decreased, the electromagnetic field created by the coil will affect the governor fly weights. Thus, by increasing the current through the coil, the engine RPM will increase. Decreasing current through the coil will cause the engine RPM to decrease. Each governor also incorporates a magnetic pickup, which consists of a coil of wire and a target disc mounted on the fly weight head which rotates with the governor. The output of this magnetic pickup is used in the single red line system and is not part of the propeller synchrophaser system.

4. Propeller Synchrophaser Target

A. A propeller synchrophaser target is installed on the deicing platter located immediately adjacent to the back of the propeller. A magnetic pickup is installed on the propeller brush block. Each time the target passes the magnetic pickup, a signal is generated and supplied to the propeller synchrophaser control box.

5. Operation

A. The initial rotation of the synchrophaser control knob turns the system on. Electric impulses generated by the target disc passing each magnetic pickup are fed into the control box. Any difference in the pulse rates will cause the control box to change the governor speed setting by varying the voltage supplied to the biasing coil until the engine RPMs exactly match. Normal governor operation is unchanged but the synchrophaser will continuously monitor and bias the governor speed setting to match engine RPMs. Further rotation of the knob allows the pilot to select any desired phase relationship between the propellers. The synchrophaser system will maintain this phase relationship until the speed setting of the engines is again moved by the pilot.



Propeller Synchrophaser System
Figure 1

PROPELLER SYNCHROPHASER SYSTEM – MAINTENANCE PRACTICES

1. Removal/Installation – Propeller Synchrophaser System

A. Removal – Synchrophaser Control Box

(1) Loosen the allen set screw in the OFF-ON knob and remove the knob.

(2) Remove the screws that hold the control box housing onto the pedestal and remove the housing.

(3) Remove the retaining nut that secures the control box into the housing and remove the control box.

(4) Disconnect the control box from its wire bundle.

B. Installation – Synchrophaser Control Box

(1) Connect the control box to its wire bundle.

(2) Insert the control box into the control box housing and secure it with the retaining nut.

(3) Position the control box housing on the pedestal and install and tighten the screws.

(4) Place the ON-OFF knob onto the control box and tighten the allen set screw.

C. Removal – Synchrophaser Magnetic Pickup

(1) Remove engine chin cowl.

(2) Disconnect electrical connections to pickup.

(3) Cut safety wire and loosen jam nut on pickup; unscrew pickup and remove.

D. Installation – Synchrophaser Magnetic Pickup

(1) Screw new pickup into jam nut and mounting and brush block adapter plate.

(2) Adjust pickup until the gap between pickup and target is .025 (twenty five thousandths). Tighten jam nut and torque to 25 inch-pounds.

(3) Connect electrical connections to main wire bundle.

(4) Perform operation check on synchrophaser system.

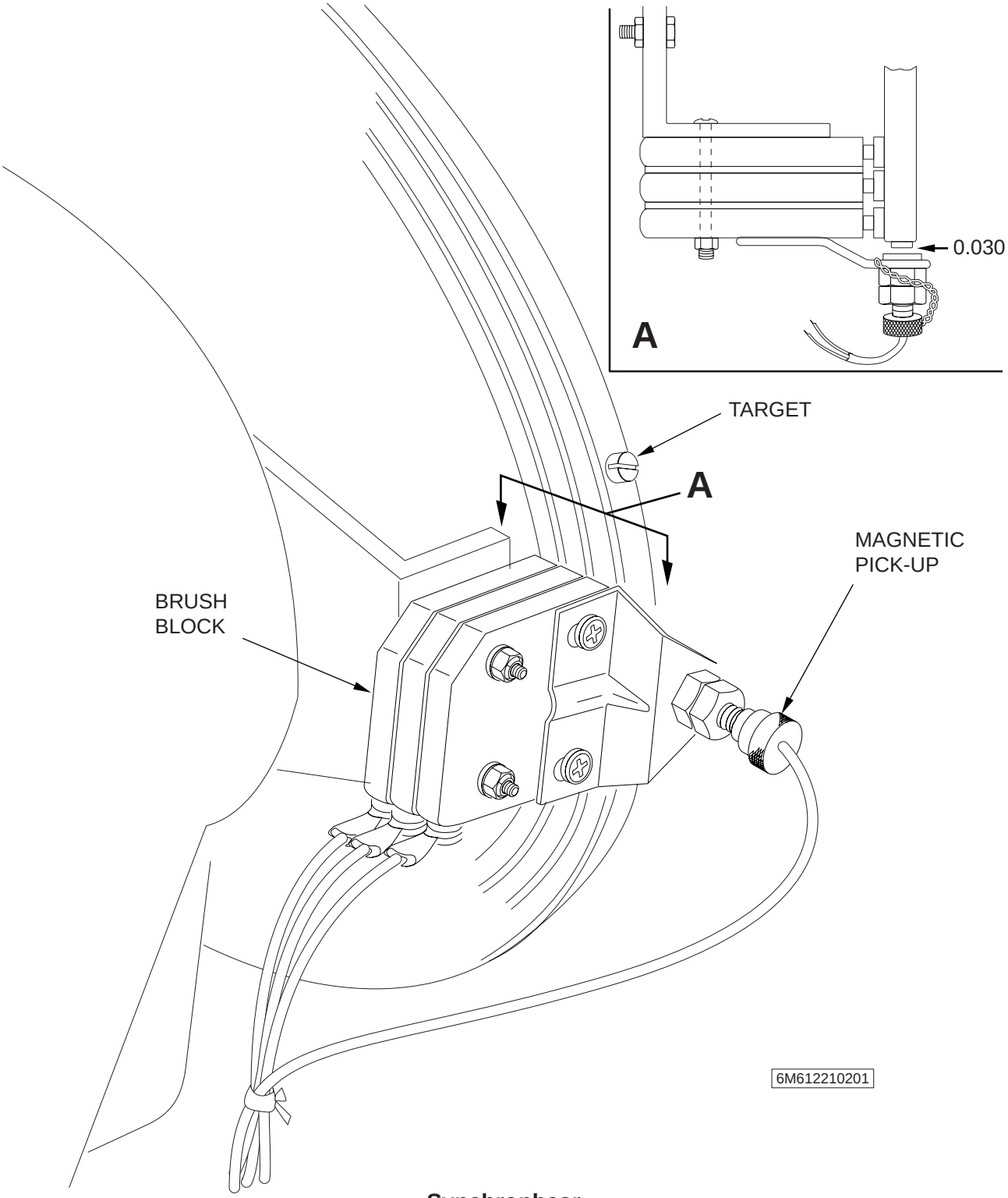
(5) Install engine chin cowl.

2. Inspection/Check – Propeller Synchrophaser System

A. Synchrophaser Check-Out Procedure

(1) Before starting this test, be sure the control box is unplugged, the battery switches are OFF, and the synchrophaser circuit breaker is pulled.

(2) Make the following tests using an ohmmeter on the X1 or X10 scale to the pin receptacles in the Jones plug. Resistance readings may be 20 percent higher during heat soak following shutdown.



Synchrophasr Pickup
Figure 201

TEST BETWEEN RECEPTACLE NUMBERS	OHMMETER READING
1 and aircraft ground	0 - ohms
2 and aircraft ground	open circuit
3 and aircraft ground	52 - 68 ohms
4 and aircraft ground	open circuit
5 and aircraft ground	52 - 68 ohms
6 and aircraft ground	open circuit
7 and aircraft ground	open circuit
8 and aircraft ground	open circuit
4 and 8	112 - 138 ohms
6 and 8	112 - 138 ohms

3. Adjustment/Test – Propeller Synchrophaser System

A. Synchrophaser Setup Procedure

(1) To check voltage output of governor speed pickup, you will need an oscilloscope or Woodward Test Instrument 213599. The voltage between pins 5 and 1, and between pins 3 and 1 should be three volts, peak to peak, at cruise RPM. The voltage should not be less than two, peak to peak at minimum cruise RPM, nor more than ten peak to peak at maximum cruise RPM.

NOTE: You will not be able interpret the peak to peak voltage with a typical R.M.S. reading voltmeter, such as a Simpson or Triplett.

(2) Locate propeller synchrophaser control assembly and disconnect system wire harness connector. Connect 32-82020-1 test box assembly to aircraft wire harness connector and propeller synchrophaser control assembly.

(3) Adjust left and right speed pickup targets as follows:

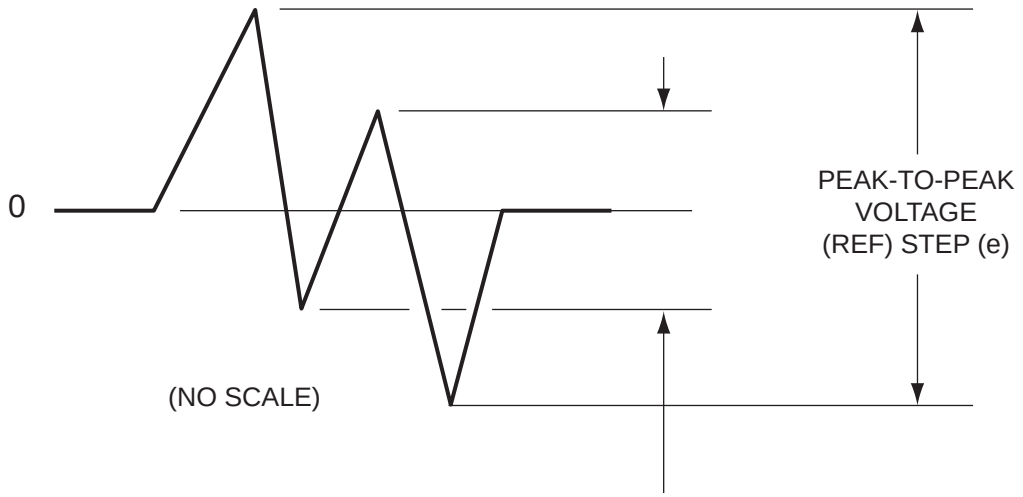
(a) Set magnetic pickup target gap to .025 + or - .002.

(4) Start aircraft engines and turn synchrophaser on (climb and cruise).

(5) Connect oscilloscope to left speed pickup jacks on the 32-82020-1 test assembly to measure speed pickup output voltage.

(6) Advance left engine to indicate 97% RPM on prop speed indicator. Measure left speed pickup output peak-to-peak voltage. See Figure 202 for proper oscilloscope waveform and record.

(7) Repeat steps (5) and (6) above for right engine and record.



NOTE: THIS VALUE MUST NOT BE MORE THAN 1.00 VOLT PEAK-TO-PEAK. BEST RESULTS ARE OBTAINED IF THIS VALUE IS 0.50 OR LESS.

NOTE: LOOK FOR A WAVEFORM APPROXIMATELY AS SHOWN.

6M612210202

Synchrophaser
Oscilloscope Waveform
Figure 202

(8) Magnetic speed pickup output voltages must be 2.5 to 2.8 volts peak-to-peak. If magnetic pickup voltages are not within specs, readjust magnetic speed pickup gap noted in (2) or (3) to obtain correct output.

NOTE: A target gap of .002 change will result in approximately 0.5 volts peak-to-peak change out of speed pickup. An increase in gap setting will decrease peak-to-peak voltage output.

(9) Repeat steps (5) through (8) until correct voltage output noted in step (8) from magnetic speed pickup has been obtained.

(10) Setup procedure acceptance, sign-off required.

(a) Technician name _____ date _____

(b) Q. A. Name _____ Date _____

4. Functional Operation – Propeller Synchrophaser System

A. Synchrophaser Functional Operation Check

(1) Start engines and run engines in propeller governing range at approximately 97% RPM.

(2) Turn synchrophaser on (climb and cruise).

(3) Adjust speed levers until condition indicator light on 32-82020-1 test box assembly extinguishes (indicates props are in sync).

(4) Adjust left speed lever until condition indicator light on 32-82020-1 test box assembly starts to flicker on and off. Leave speed lever at this setting and observe change in RPM as synchrophaser control brings props back into sync. Observe the condition indicator light extinguished again.

(5) Repeat step (4) for right speed lever.

(6) Adjust left or right speed levers out of synch capture range (props out of sync). Observe condition indicator light illuminated. (Note – Capture range for prop sync should be approximately ± 1%, or 10 RPM.

(7) Shut-down engines.

(8) Functional operation check, acceptance sign-off required.

(a) Technician name _____ date _____

(b) Q. A. Name _____ Date _____

5. Setup and Checkout Completion Synchrophaser System

A. Remove test equipment and return aircraft to original configuration.

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CHAPTER 70 – STANDARD PRACTICE – ENGINE

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STANDARD PRACTICE – ENGINE – DESCRIPTION AND OPERATION

1. General

A. Standard practices for the TPE331 turboprop engine includes disassembly and reassembly instructions, replacable component information, cleaning, painting, and special tools, and material. For information pertaining to these standard practices, refer to the following listed Garrett-AiResearch Maintenance Manuals.

B. For information pertaining to engine systems not included in the referenced manuals listed below, refer to applicable chapters within this Maintenance Manual.

C. For information pertaining to Fairchild components installed on the TPE331 engine, refer to Fairchild Powerplant Build-Up Manuals (dash 3 and dash 10).

Maintenance Manual: TPE331-3U-303G
Engine Part Number: 894040-1/2
Applicable Aircraft: Merlin III model SA226-T
Merlin IV model SA226-AT
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valves.

Maintenance Manual: TPE331-3UW-303G
Engine Part Number: 895880-1/-2
Applicable Aircraft: Metro model SA226-TC
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valves.
2. Alcohol/Water Injection System.

Maintenance Manual: TPE331-3U-304G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III model SA226-T
Merlin IV model SA226-AT
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valve.

*See Fairchild Aircraft Service Bulletin 30-011 for conversion procedures between TPE331-3U-303G and TPE331-3U-304G engines.

Maintenance Manual: TPE331-3UW-304G
Engine Part Number: 3102550-1
Applicable Aircraft: Metro model SA226-TC
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valve.
2. Alcohol/Water Injection System.

Maintenance Manual: TPE331-10U-501G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III (B) model SA226-T(B)

STANDARD PRACTICE – ENGINE – DESCRIPTION AND OPERATION

Maintenance Manual: TPE331-10U-501G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III (B) model SA226-T(B)
Identifying Characteristic:
*1. Alcohol/Water Injection System.

*See Fairchild Aircraft Service Bulletin 30-011 for conversion procedures between TPE331-3U-303G and TPE331-3U-304G engines.

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71-30-00	MP	201	AUG 15/10
71-30-00	MP	202	AUG 15/10
71-40-00	DO	1	AUG 15/10
71-40-00	DO	2	AUG 15/10
71-40-00	MP	201	AUG 15/10
71-40-00	MP	202	AUG 15/10
71-40-00	MP	203	AUG 15/10
71-40-00	MP	204	AUG 15/10
71-40-00	MP	205	AUG 15/10
71-40-00	MP	206	AUG 15/10
71-40-00	MP	207	AUG 15/10
71-40-00	MP	208	AUG 15/10
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71-40-00	MP	210	AUG 15/10
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71-40-00	MP	216	AUG 15/10
71-40-00	MP	217	AUG 15/10
71-40-00	MP	218	AUG 15/10
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71-50-00	MP	2	AUG 15/10
71-50-00	MP	201	AUG 15/10
71-50-00	MP	202	AUG 15/10
71-50-00	MP	203	AUG 15/10
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71-50-10	MP	209	AUG 15/10
71-50-10	MP	210	AUG 15/10
71-50-10	MP	211	AUG 15/10
71-50-10	MP	212	AUG 15/10
71-60-00	DO	1	NOV 01/85
71-60-00	DO	2	NOV 01/85
71-70-00	DO	1	NOV 01/85
71-70-00	DO	2	NOV 01/85
71-70-00	MP	201	NOV 01/85
71-70-00	MP	202	NOV 01/85
71-SERVC	SI	1	AUG 15/89
71-SERVC	SI	2	AUG 15/89
71-SERVC	SI	3	AUG 15/89
71-SERVC	SI	4	AUG 15/89

EFFECTIVITY:
T 201-999, AT 001-999
TC 201-999

AN ELBIT SYSTEMS OF AMERICA, LLC, COMPANY

MAINTENANCE MANUAL

CHAPTER 71 – POWERPLANT

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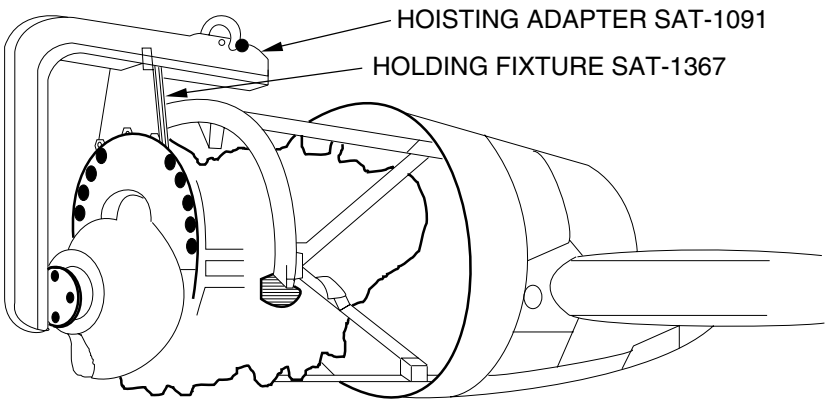
POWERPLANT – SERVICE INFORMATION

1. Introduction

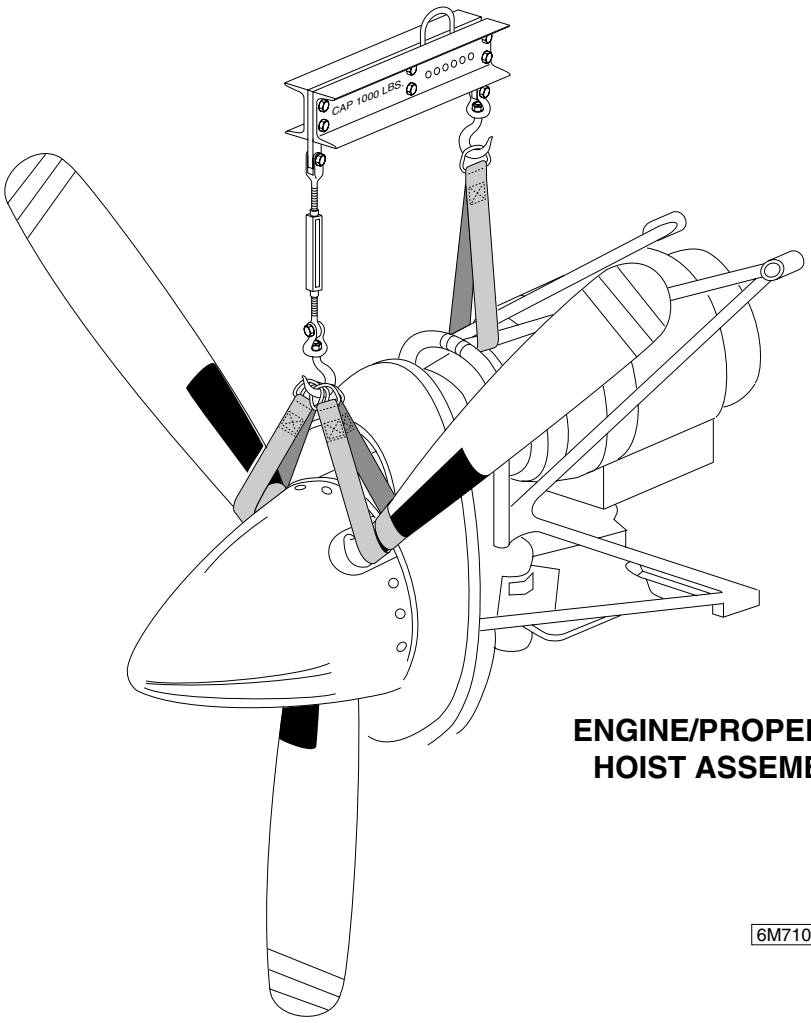
A. The following is supplied as a supplement to this chapter. Subject matter is included which is not applicable to an ATA 100 format, but is still of interest to Merlin and Metro owner/operators. The information in this section is not part of the basic manual, and may be removed or retained at the discretion of the owner/operator.

B. Revisions and additions to this section will be made as additional information is available. Input from individual owner/operators is encouraged, and may be submitted to:

Fairchild Aircraft Corporation
Post Office Box 790490
San Antonio, Texas, 78279-0490
ATTN: Technical Services Department



ENGINE HOIST ASSEMBLY



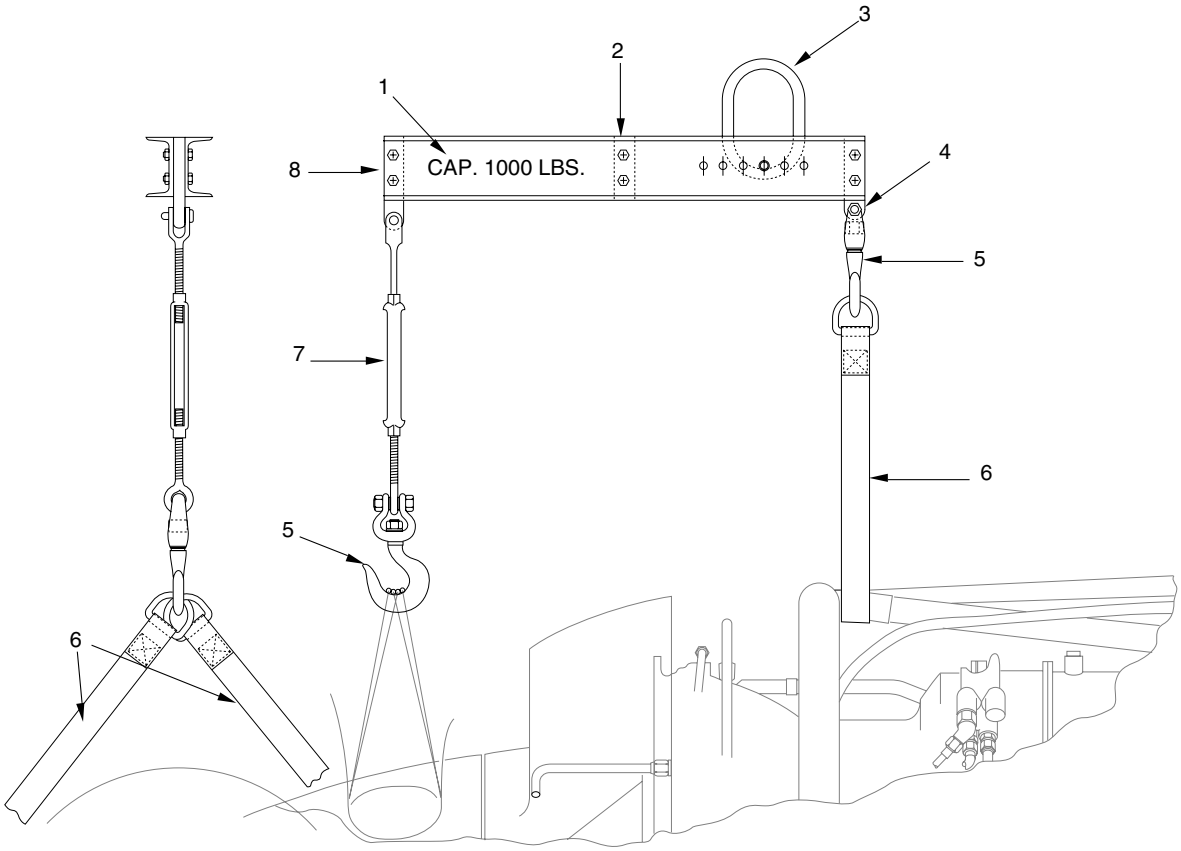
ENGINE/PROPELLER HOIST ASSEMBLY

6M710030201

Engine/Propeller Hoist Assembly Figure 1

2. Engine-Propeller Hoist combo

(1) Figures 1 and 2 provide information for the construction of a hoist for removing propeller and engine from the airframe.



1.	27-95023-011	channel
	–Attaching Parts–	
	AN4-13A	bolt
	AN960-416L	washer
	MS21042L4	nut
	* * *	
2.	27-95023-017	fitting
3.	3566T32	link
	–Attaching Parts–	
	MS24665-319	pin
	AN960C716L	washer
	MS24665-319	pin, cotter
	* * *	
4.	27-95023-013	fitting
5.	3527T1	hook
6.	NAS1212R15AA36	strap
7.	3000T53	turnbuckle
8.	27-95023-015	fitting

7M71 SERVC2

Engine Propeller Hoist
Channel Assembly
Figure 2

POWERPLANT – DESCRIPTION AND OPERATION

1. General

A. This chapter provides maintenance practices for the overall powerplant on SA226 series aircraft using Garrett AiResearch TPE331-3U-303G, TPE331-3U-304G, or TPE331-10U-501 engines.

B. Section one (71-00-00) provides general ground safety practices, separate basic engine ground operation procedures for either the -3 or -10 engine, removal and installation procedures, and standby rocket (JATO) maintenance practices.

C. Section two (71-10-00) covers the engine cowlings to include a separate description and operation of each cowling, their removal and installation procedures, inspection/check procedures, and detailed instructions for the drilling, trimming, and buildup procedures for custom fit of new cowling to replace old cowling, which may be beyond repair.

D. Section three (71-20-00) provides a general description of the different engine mount truss assemblies and engine (Lord) mounts used on SA226 series aircraft. Also provided are their removal, installation, and inspection/check procedures.

E. Section four (71-30-00) provides a general description of the engine fireseal components used on various SA226 series aircraft to include engine heat shrouds (Tamale Cans), heat shroud retainers, ignitor plug heat shields, the aspirator fireseal, and their removal and installation procedures.

F. Section five (71-40-00) provides a general description and removal installation procedure for engine component attach fittings to include engine mount fittings and the engine control cable attach fittings.

G. Section six (71-50-00) provides detailed maintenance procedures for the removal, buildup, and installation of the engine electrical harness for either the -3 or -10 engine. Within the electrical harness buildup procedures for each engine are provided charts which detail the engine component wire lead and terminal/connector part numbers.

H. Section seven (71-60-00) provides general data and reference concerning maintenance practices for the SA226 series aircraft air intake.

I. Section eight (71-70-00) provides maintenance practices concerning the engine drains to include the drain block removal and installation, and mounting location of engine component drain hoses and tubing.

J. For complete engine maintenance information, consult Garrett/ AiResearch Maintenance Manuals. For detailed information concerning the engine operation, consult the respective Aircraft Flight Manual.

K. Ground Safety

(1) Observe the following safety precautions when performing maintenance or operating the engines:

CAUTION: MANY FUELS, LUBRICANTS, AND SOLVENTS IRRITATE THE SKIN AND HAVE TOXIC FUMES. AVOID BREATHING THE FUMES IN CONFINED AREAS. IF SPILLS ON THE SKIN OCCUR, WASH AFFECTED AREA IMMEDIATELY. CHANGE SATURATED CLOTHING IMMEDIATELY.

(a) Clean up all spills immediately and deposit cleaning towels in a specially prepared metal container.

(b) Use suitable drip pans under engines and any other area where drips are likely when aircraft is in hangar.

WARNING: OPERATING ELECTRICAL CIRCUITS CAN INFLICT SERIOUS SHOCK OR BURNS AS WELL AS DAMAGING ARCING WHEN SHORTING OR OTHER MALFUNCTIONS OCCUR. TURN OFF ALL ELECTRICAL POWER WHEN PERFORMING MAINTENANCE UNLESS POWER IS NECESSARY FOR ISOLATED INSTANCES OF CIRCUIT ANAYLSIS.

(c) Before performing maintenance on engines after operation, allow sufficient time for engine cool-down.

(d) Allow only qualified personnel to operate aircraft engines.

(e) Ensure that propellers and tail pipe areas are clear of personnel and material objects before starting engines.

L. Engine Ground Operation

NOTE: See Engine Ground Operation Procedures within this section for the -3 or -10 engine.

When maintenance practices require the operation of one or both engines, perform external and internal prestart check.

M. External Prestart Check:

(1) Nose wheel.....CENTERED

(2) Chocks.....IN PLACE

(3) Moorings and covers.....REMOVE

(4) Access panels, plates, and filler caps.....SECURE

(5) Engine inlet and exhaust.....CLEAR

(6) Propeller condition and rotation.....CHECK

(7) Oil level.....CHECK

(8) Fuel sumps.....DRAIN

(9) Area around aircraft.....CLEAR OF ALL OBJECTS NEAR/PROPELLER SLIP STREAM

(10) Water injection.....FULL - IF TO BE USED

N. Internal Prestart Check:

(1) Cabin, cargo, and baggage doors, and emergency hatches.....AS REQUIRED
(2) Seat and rudder pedals. .....ADJUST
(3) Parking brake .....SET
(4) Controls. .....FREE
(5) Left console bus ties and circuit breakers. .....CHECK IN
(6) Cabin pressure selector valve .....AUTO
(7) Cabin pressure manual control. .....CLOSED
(8) Pilot's light controls. .....OFF
(9) Airspeed static selector. .....NORMAL
(10) Pilot's switch panel - all switches. .....OFF
(11) Elevator, rudder, and aileron trim.. .....CHECK
(12) Flap control.. .....NEUTRAL
(13) Landing gear handle .....DOWN
(14) Oxygen control .....CLOSED
(15) Copilot's switch panel - light switches. .....OFF
(16) Cabin dump .....DUMP
(17) Max flow air controls .....OFF
(18) Fresh air fan .....AS REQUIRED
(19) Air conditioning switches .....AS REQUIRED
(20) Copilot's light controls .....OFF
(21) Right console bus tie and circuit breakers .....CHECK
(22) Radio and electronic switches. .....OFF

ENGINE GROUND OPERATION – TPE331-3U-303G/304G – DESCRIPTION AND OPERATION

1. General

A. The following practices detail the procedures and checks to be performed during normal engine ground run. These practices do not teach elementary engine ground run principles, but are inserted to provide manual users with a general knowledge of ground run procedures under normal engine operation. Ensure that only pilots or qualified personnel perform ground run procedures, accomplished in accordance with the AFM/Pilot's Operating Handbook. Perform normal engine start procedures, operation, and shutdown in accordance with Section II of the AFM. Reference Section III, AFM for Emergency Operating Procedures.

B. Conduct these procedures and satisfactorily perform prior to initial flight. Check procedures need not be performed on both engines simultaneoulsy except where matching of engine is so indicated within the procedures.

C. Initial rigging of the powerplant/controls is described in the AiResearch Maintenance Manual, and M7 Aerospace Maintenance Manual.

D. Each check/functional test procedure and its applicable corrective action is identified by its title and number. A ground run check sheet is also titled and itemized to correspond with each applicable procedure.

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ENGINE GROUND OPERATION – TPE311-3U-303G/304G – MAINTENANCE PRACTICES

1. Adjustment/Test – Engine Ground Operation – TPE331-3U-303G/304G Engine

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP. REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. System Test Procedures and Corrective Action

(1) The corrective actions provided for each procedure are those considered most likely to produce desired results to problems, if any, encountered during the check procedure. An attempt has not been made to cover all situations or provide information for troubleshooting problems that cannot be corrected by minor adjustments due to component malfunction. If during a ground check procedure the acceptance criteria of the check is not met, all pertinent information should be recorded; i.e., record all engine instrument readings at the point of the check and any other data that may aid in troubleshooting or adjustment.

(2) Corrective steps should be performed in the indicated sequence. If after performing the indicated steps, the problem remains unresolved, contact M7 Aerospace Engineering Propulsion Group and/or AiResearch Field Service Representative.

2. Inspection/Check – Engine Ground Operation – TPE331-3U-303G/304G Engine

NOTE: The following table is provided to aid in performing adjustments:

FUNCTION	ADJUSTMENTS	EFFECT
Overspeed Governor	1 turn clockwise	=4.5% Increase (RPM)
Underspeed Governor		
Min. Speed Stop	1 turn clockwise	=5.0% Increase (RPM)
Max. Speed Stop	1 turn clockwise	=3.6% Decrease (RPM)
Propeller Governor		
Low Speed Stop	1 turn counter-clockwise	=1.0% Decrease (RPM)
High Speed Stop		
Screw X	1 turn clockwise	=1.0% Increase (RPM)
Flight Idle Adjustment	1 full turn clockwise or 1 click clockwise	=76 PPH Increase in fuel flow =2 PPH Increase in fuel flow
Max Power Adjustment	1 full turn clockwise	=12 PPH Increase in fuel flow
Beta Tube Assembly	1 turn counter-clockwise	=2.0 degrees increase (blade angle)

Engine Control Adjustment
Table 201

A. Engine Start Time – Check Procedure

(1) Conduct normal start in accordance with Aircraft Flight Manual (AFM) procedures.

(2) Record light-off time (from 10% to light-off).

(a) Ground start – 10 seconds (when not using start fuel pressure regulator).

(b) Ground start – two seconds (when using start fuel pressure regulator).

(c) Air start – 15 seconds.

NOTE: After light-off, observe a smooth continuous acceleration to ground idle (65% RPM).

B. Corrective Action (Engine Start Time – Check Procedure):

(1) Check battery and electrical system.

(2) Reference AiResearch Maintenance Manual, Engine Troubleshooting section.

C. Overspeed Governor – Check Procedure (103.0% to 105.0% RPM):

(1) After normal engine start, operate engine until oil temperature reaches normal range (70 to 80°C).

(2) Place speed lever at LOW.

(3) Advance power lever toward maximum position, observe RPM. When governor setting is reached, observe that the RPM should no longer increases with further movement of the power lever.

CAUTION: DO NOT EXCEED 106% RPM.

(4) Observe RPM stabilizes between 103% to 105% RPM.

(5) Repeat procedure for the other engine.

D. Corrective Action (Overspeed Governor – Check Procedure 103.0% to 105.0% RPM):

NOTE: Do not remove the power lever arm assembly from the power lever shaft.

(1) Disconnect rod-end from power lever arm on fuel control by removing cotter pin, nut, and washer.

(2) Remove intermediate (flight idle) fuel flow adjustment plug.

(3) Position power lever shaft in 100° position to aid in removal of cover from fuel control unit.

(4) Remove the six screws and two 12 point hex head bolts securing cover to the fuel control unit.

(5) Rotate power lever shaft to minimum position, then tap cover loose using soft faced mallet and remove cover.

(6) After clearing alignment pins, rotate cover toward vertical centerline of engine to clear second cam from internal linkage.

(7) Use 5/64 inch swivel tip Allen wrench or equivalent. Adjust overspeed governor as required (see Table 201). One turn clockwise increases RPM 4.5%.

CAUTION: ENSURE MANUAL FUEL CAM FOLLOWER LEVER IS IN SLOT PRIOR TO POSITIONING COVER ONTO FUEL CONTROL. OTHERWISE AN OVER-TEMP CONDITION WILL RESULT ON ENGINE START.

(8) Place manual fuel valve shaft to minimum position, install cover with new gasket (P/N 3054-726).

(9) Install screws, torque 12 to 14 inch-pounds (1.36 to 1.58 nm). Install bolt; torque 60 to 75 inch-pounds(6.78 to 8.47 nm)

(10) Install flight idle access plug and spring, safety plug.

(11) Connect control rod to power lever arm with washer and nut. Torque nut to 12 inch-pounds (1.36 nm), install cotter pin.

E. Underspeed Governors, Low RPM - Check Procedures

(1) With the speed lever low, move power lever slowly toward reverse to attain temperature increase of 10°C.

(2) Observe RPM reads 64.0 to 66.0%.

F. Corrective Action (Underspeed Governors, Low RPM):

(1) Adjust minimum speed stop (LOW RPM) on the underspeed governor. Turning adjustment screw one turn clockwise increases engine RPM 5.0%.

G. Underspeed Governor, High RPM - Check Procedures:

(1) Place speed lever in HIGH position.

(2) Move power lever slowly toward reverse. When minimum temperature is indicated continue movement until temperature increases 10°C. Observe RPM stabilizes between 95% and 97%.

H. Corrective Action (Underspeed Governors, High RPM):

(1) Adjust maximum stop on the underspeed governor.

NOTE: One turn clockwise decreases engine RPM 3.6%.

I. Underspeed Governor Reset - Function Check - Check Procedure

(1) Take propeller off start locks.

(2) Place speed lever to LOW.

(3) Move power lever to full reverse and observe RPM.

(4) Observe RPM minimum is 75%.

J. Corrective Action (Underspeed Governor Reset - Functional Check):

(1) Set power lever shaft at 18° to 20° and adjust screw V to contact the stop with speed lever at minimum.

(2) Adjust speed lever and speed lever rod until underspeed governor resets to protractor position obtained in step 2., J., (1), (18° to 20°).

(3) Start engine and determine underspeed governor resets to 75% with speed lever at minimum and power lever at full reverse.

(4) Recheck step 2., J., (1) and readjust as required.

ENGINE GROUND OPERATION – TPE311-3U-303G/304G – MAINTENANCE PRACTICES

K Flight Idle Flat – Check Procedure (Low Engine Speed Flat Check):

NOTE: Propeller must be on the start locks.

(1) Move speed lever to LOW position.

(2) Advance power lever slowly from minimum fuel flow toward takeoff position in the area of flight idle hard stop. Observe a constant fuel flow. Observe RPM nominally stabilizes between 96 to 98%, with fuel flow approximately 200 pounds per hour.

(3) Record fuel flow at flat.

CAUTION: DO NOT EXCEED TORQUE OR TEMPERATURE LIMITS.

(4) Advance and retard power lever rapidly to flight idle hard stop and record fuel flow. Observe that fuel flow is the same as noted in step 2., K, (3).

NOTE: If engine is rigged correctly, misalignment of flight idle flat can usually be corrected in the aircraft power lever linkage.

L. Corrective Action (Flight Idle Flat):

(1) Check rigging of flight idle in accordance with engine (AiResearch) and Aircraft Maintenance Manual.

(2) For misalignment marks where fuel flow begins increasing, adjust the micro vernier on the manual fuel valve on one or both engines.

NOTE: This adjustment changes the position of the manual fuel valve at flight idle, thus changing the relationship between flight idle flat and power lever position.

(3) For improper RPM with power lever in the flight idle position, adjust the flight idle fuel flow. One click produces a minimal fuel flow change of two pounds per hour. If fuel flow must be adjusted excessively (beyond the range 185-200 pounds per hour), check the flight idle propeller blade angle in accordance with the Aircraft Maintenance Manual, Chapter 61. Observe fuel flow is within tolerance when flight idle RPM is obtained.

M. Propeller Governor, High RPM - Check Procedure and Adjustment

CAUTION: DO NOT EXCEED 102.5% RPM, TEMPERATURE, OR TORQUE LIMITS DURING PROPELLER GOVERNOR CHECK.

(a) Propeller Governor High RPM Check

(1) Place speed lever at HIGH.

(2) Move power lever slowly to 800~C ITT; observe RPM between 99.5% and 100.5% (right engine RPM 0.2% to 0.5% higher than left engine to permit synchronization).

(3) Return power lever to slightly aft of start position. Operate three minutes for cooling and shutdown.

N. Corrective Action (Propeller Governor, High RPM - Check Procedures and Adjustment):

NOTE: Propeller Governor High RPM is set to produce 100% to 101% RPM for takeoff. Both FAC and GTEC Maintenance Manuals permit a PGH RPM range of 99% to 101% for SA227 AT and TC airplanes. In cases where engine oil temperatures are very high and a prevailing tail wind is present, it may be necessary to set PGH RPM as low as 99% to produce 100% takeoff RPM. Every effort should be made to check PGH RPM at normal engine oil temperature and with the nose of the aircraft into the wind.

(1) For each percent of speed more than 100.5%, turn high speed screw one turn clockwise.

(2) Restart engine and repeat steps 2., M., (1) through (3). Continue making adjustment until propeller governor maintains RPM at 99.5% to 100.5% regardless of the amount of power lever advancement.

O. Propeller Governor, Low RPM - Check Procedures:

(1) Speed lever to LOW.

(2) Move power lever to 800°C ITT (propeller governor mode).

CAUTION: DO NOT EXCEED 885°C ITT.

(1) Retard speed lever slowly until propeller governor speed does not drop further with reduction in speed lever setting.

(2) Observe propeller governor maintains RPM at minimum setting of 93.5% to 96% (right engine RPM 95.8% maximum to permit synchronization at 96% in flight).

(3) Shut down engine in accordance with AFM.

P. Corrective Action (Propeller Governor, Low RPM):

(1) Adjust propeller governor low speed stop. One turn clockwise to increase RPM approximately 1.0%.

Q. Reverse Power - High RPM - Check Procedures

(1) Bleed Air On

(a) Place speed lever to HIGH.

(b) Move power lever toward full reverse, while monitoring RPM. In full reverse, observe RPM is 95.5% minimum.

(c) Check that torque split between engines is no more than 100 foot-pounds.

R. Corrective Action (Reverse Power – High RPM):

(1) If RPM drops below 95.5%, recheck flight idle propeller blade angle under hot oil conditions and repeat underspeed governor high RPM check.

S. Reverse Power Symmetry Check – Check Procedure

(1) Place both speed levers to HIGH.

(2) Advance both power levers to 100% RPM.

(3) Pull both power levers smoothly into full reverse. Observe that both engines move into reverse simultaneously. Observe both torque and fuel flow are symmetrical. If one engine is slow to reverse, recheck separation between underspeed governor and propeller governor at high RPM. Speed separation of governors must be 2.5% minimum. If torque and fuel flow are not symmetrical, check propeller blade angle at flight idle and reverse.

T. Reverse Power – Low RPM – Check Procedure

(1) Bleed Air On

(a) Place speed levers in LOW position.

(b) Place power levers in ground idle position.

(c) Slowly move power lever of one engine toward reverse; Observe engine does not bog down or show evidence of compressor stall (beta light on).

(d) Mark on tape, point where RPM begins to increase.

(e) Repeat with other power lever.

(f) Retard both power levers to full reverse. Observe RPM is 75.0% minimum at full reverse. Separation shall not be more than 5% between engines.

(g) Marks on tape where RPM begins to increase may not be separated more than 0.50 inch.

Power Setting	Time Limit (minutes)	Maximum Torque (ft.-lb.)	Maximum ITT (°C) (1)	RPM(%) 100% = 2000 (2)	Oil Pressure (psi)	Fuel Pressure (psi)	Oil Temperature (°C)
Takeoff	–	2206	923	100(3)	70 to 120	20 to 80	55(5) to 110
Maximum Continuous	–	2206	923	100	70(4) to 120	20 to 80	55(5) to 110
Flight Idle	–	–	–	96 to 100	70(4) to 120	20 to 80	55(5) to 110
Ground Idle	–	–	–	65 to 97(6)	40 to 120	15 to 80	-40 to 127
Starting	–	–	1149	–	–	–	-40 minimum
Reverse High	–	–	–	95.5 to 97	70 to 120	20 to 80	55(5) to 110
Reverse Low	–	–	–	75 minimum	70 to 120	15 to 80	-40 to 127

(1) 885°C Maximum ITT at 96% RPM.

(2) Avoid operation between 18 and 28% RPM except for transients occurring during engine start and shutdown. 96% is the minimum RPM during flight.

(3) 101% for 5 minutes and 105% for 5 seconds is allowed.

(4) Above 23,000 feet, minimum oil pressure is 50 psi.

(5) Operation from 22°C to 55°C allowed as a function of ambient temperature.

(6) Typical engine speeds for low and high speed lever positions.

Engine Limits For SA226 – T(A) / AT / TC
TPE 331-3U-303G / 304G Without AWI
Table 202

U. Corrective Action (Power Lever Split at Point of Increasing RPM Greater Than 0.50 inch)

(1) Check propeller pitch control position on rigging plate (approximately 14°) when adjusting screw V (see Table 202), contacts tab with RPM lever in low. Refer to rigging procedures in M7 Aerospace Aircraft Maintenance Manual.

(2) Adjust screw V on one or both engines as necessary to remove power lever split, keeping pitch control position at contact of screw V as near 14° as possible.

(3) Recheck RPM obtained at full reverse, as adjustment of screw V will affect the RPM obtained.

(4) Recheck propeller governor high RPM in accordance with Propeller Governor, High RPM.

V. Corrective Action (When RPM Obtained At Full Reverse Is Not Within Tolerance)

(1) Check propeller pitch control position in accordance with Corrective Action (Power Lever Split At Point Of Increasing RPM Greater Than 0.50 Inch).

(2) Adjust point of attachment of link I to arm F (see Table 202), as necessary to increase engine RPM with power lever in full reverse position.

NOTE: Moving arm F toward maximum RPM stop increases RPM.

(3) Recheck power lever split at point of increasing RPM and RPM obtained at full reverse in accordance with Corrective Action (Power Lever Split At Point Of Increasing RPM Greater Than 0.50 Inch).

(4) Recheck propeller governor high RPM in accordance with Propeller Governor, High RPM. Repeat paragraph Corrective Action (Power Lever Split At Point Of Increasing RPM Greater Than 0.50 Inch) and Corrective Action (When RPM Obtained At Full Reverse Is Not Within Tolerance) until both power levers split and RPM obtained are within the specified tolerances.

ENGINE GROUND OPERATION – TPE311-3U-303G/304G – MAINTENANCE PRACTICES

W. Power Lever Split At Takeoff Power – Check Procedure

(1) Bleed air OFF.

(2) Place speed levers in HIGH position.

(3) Advance one power lever to stabilized takeoff power. (Takeoff power is defined as 923°C ITT or 2,206 foot-pound torque, whichever occurs first.) If ITT is limited, check minimum acceptable torque for SA226 – AC/TC per Table 202.

(4) Mark reference index on power lever and on tape at takeoff power position.

(5) Retard power lever to ground idle position and repeat check with other power lever.

(6) Ensure that index marks on tape do not exceed 0.05 inch between power positions for left and right power levers.

(7) Repeat 2., W., (2) through (6) with bleed air on. Calculate and record loss. Observe that when speed lever at 96%, ITT is 885°C.

X. Corrective Action (Power Lever Split At Takeoff Power, Bleed Air Off)

(1) Check for free play in control cables.

(2) Recheck flight idle flat and correct as necessary in accordance with Flight Idle Flat check procedure.

(3) Adjust maximum fuel flow adjustment on fuel control on one or both engines to remove excessive power lever split.

NOTE: One-fourth (one click) turn clockwise will increase fuel flow four pounds per hour. Do not increase more than one turn.

(4) Note amount of adjustment in the Engine Log Book.

Y. Propeller Governor – Underspeed Governor Separation – Check Procedures

(1) Start engine, take propeller off start locks. Allow oil temperature to reach normal.

(2) Place speed lever to maximum and power lever to mid-range.

(3) Slowly retard speed lever and stabilize RPM at 96%.

NOTE: Do not exceed 885~C ITT.

(4) Retard power lever and obtain minimum fuel flow.

(5) Observe stabilized RPM does not exceed 93.5%.

Z. Corrective Action (Propeller Governor – Underspeed Governor Separation)

(1) Adjust length of propeller governor arm as necessary to obtain 2.5% separation.

NOTE: Turning propeller governor arm out increases RPM separation.

(2) Recheck propeller governor high RPM in accordance with System Test Procedures And Corrective Action of check procedure and repeat Propeller Governor Underspeed Governor Separation - Check Procedure and Propeller Governor, High RPM - Check Procedure And Adjustment until both are in tolerance.

AA. Beta Light Operation – Check Procedure

(1) Place power lever in ground idle position.

(2) Advance speed lever to 85% RPM.

(3) Observe beta light is ON, or comes on before engine speed reaches 85%.

ENGINE GROUND OPERATION – TPE311-3U-303G/304G – MAINTENANCE PRACTICES

BB. Corrective Action (Beta Light Operation):

(1) Conduct reverse power high RPM check per Reverse Power – High RPM – Check Procedures.

(2) If Reverse Power – High RPM – Check Procedures is unacceptable, conduct corrective action in accordance with Corrective Action.

(3) If Reverse Power – High RPM – Check Procedures is acceptable, malfunction of the BETA switch is indicated and the switch, light, and related circuitry should be checked for cause.

CC. Flame Out Check – Check Procedure

(1) Place power lever at ground idle (minimum fuel flow).

(2) Place speed lever to HIGH.

(3) Move speed lever from HIGH to LOW speed stop in one second or less. Repeat procedure twice more.

(4) Observe that engine does not flame out.

(5) Observe deceleration time does not exceed 10 seconds.

DD. Corrective Action (Flame Out Check)

(1) If limits are not met, recheck underspeed governor setting in HIGH and LOW. Reset speeds within limits and repeat Check Procedures, steps 2., CC. (1) through (3).

ENGINE GROUND OPERATION – TPE311-3U-303G/304G – MAINTENANCE PRACTICES

EE. Emergency Shutdown – Check Procedure

(1) Place speed lever in LOW position.

(2) Place power lever at approximately 70% RPM.

CAUTION: IF ANY RISE IN TORQUE OR ITT IS NOTED, PUSH ENGINE STOP BUTTON IMMEDIATELY.

(3) Pull STOP and FEATHER knob until red groove shutoff control cable rod is visible at pedestal. Observe that engine flames out.

NOTE: When RPM decreases to approximately 40%, pull knob all the way aft. (4) Propeller must feather.

(4) Observe that propeller feathers.

FF. Corrective Action (Emergency Shutdown):

(1) Recheck rigging of manual fuel valve shutoff and feathering valve linkage in accordance with M7 Aerospace Maintenance Manual.

GG. Negative Torque Sensing – Check Procedure

(1) Set up for normal engine start in accordance with AFM procedure.

(2) Check NTS annunciator panel lights with PRESS-TO-TEST switch.

(3) Place NTS test switch on selected engine (left or right as appropriate).

(4) Turn on unfeather pump (hold ON).

(5) Check NTS light illuminated.

(6) Depress start button for normal AFM start (release at ignition).

(7) Observe NTS light extinguishes at starter actuation.

(8) Observe NTS light illuminates at 15% to 30% RPM.

(9) ITT monitor (1.149°C maximum).

NOTE: Start button prime as required (after reaching 25% and below 40% RPM).

(10) Turn off unfeather pump (test switch to OFF) at start completion.

(11) Observe NTS light extinguishes at start completion.

(12) Observe that RPM, ITT, fuel pressure, and oil pressure stabilizes.

(13) Repeat steps 2., GG., (1) through (10) for other engine.

HH. Corrective Action (Negative Torque Sensing):

(1) Check installation of system, see Aircraft Wiring Manual and Maintenance Manual, Chapter 77.

II. Alcohol Water Injection (AWI) Systems – Check Procedures

(1) Power lever's above 600 foot-pound torque.

(2) AWI switch to ARMED.

(3) Observe AWI armed light ON, pump light ON, bleed air OFF.

(4) Retard right power lever momentarily below approximately 600 foot-pounds note torque increase on left engine.

(5) Left bleed air ON. AWI switch off, verify AWI armed light OFF, pump light OFF and bleed air ON.

(6) Observe that low water light illuminates when approximately two gallons remain in AWI tank.

ENGINE GROUND OPERATION – TPE311-3U-303G/304G – MAINTENANCE PRACTICES

JJ. Corrective Action (Alcohol Water Injection (AWI) Systems):

(1) Check installation of system in accordance with Wiring Manual and Aircraft Maintenance Manual, Chapter 82.

KK. Dual (Continuous) AWI System Functional Test

(1) Place the PUMP TEST switch to the No. 1 PUMP position.

(2) Observe that the annunciator light, No. 1 PUMP ON, illuminates. Also observe that the No. 2 PUMP ON light does not illuminate.

(3) Place the PUMP TEST switch to the No. 2 PUMP position.

(4) Observe that the annunciator light, No. 2 PUMP ON illuminates. Also observe that the No. 1 PUMP ON light does not illuminate.

(5) Place the AWI switch to the CONT position.

(6) Observe that both No. 1 and No. 2 PUMP ON lights illuminate.

(7) Place the AWI switch to the ARM position.

(8) Observe the annunciator AWI ARMED light illuminates and that both No. 1 and No. 2 PUMP ON lights illuminate.

(9) Return the AWI switch to the OFF position.

(10) Observe that all AWI annunciator lights extinguish.

(11) Read the AWI fluid level on both the AWI tank sight glass and the AWI quantity meter. Verify that these readings agree within plus 3% full scale.

(12) Start both engines in accordance with Aircraft Flight Manual.

(13) Recheck pump operation as described in Dual (Continuous) AWI System Functional Test, steps 2., KK., (1) through (10).

Power Setting	Time Limit (minutes)	Maximum Torque (ft.-lb.)	Maximum ITT(°C) (1)	RPM(%) 100% = 2000 (2)	Oil Pressure (psi)	Fuel Pressure (psi)	Oil Temperature (°C)
Normal Takeoff (Dry power)	5	2363(4)	923	100(6)	70(8) to 120	20 to 80	55 to 110(9)
Emergency Takeoff (Wet power)	5(3)	2468(4)	944(5)	100(6)	70(8) to 120	20 to 80	55 to 110(9)
Maximum Continuous	–	2206	923	100	70(8) to 120	20 to 80	55 to 110(9)
Flight Idle	–	–	–	96 to 100	70(8) to 120	20 to 80	55 to 110(9)
Ground Idle	–	–	–	65 to 97(7)	40 to 120	15 to 80	-40 to 127
Starting	–	–	1149	–	–	–	-40 minimum
Reverse High	–	–	–	95.5 to 97	70 to 120	20 to 80	55 to 110(9)
Reverse Low	–	–	–	75 minimum	70 to 120	15 to 80	-40 to 127

(1) 885°C Maximum ITT at 96% RPM.

(2) Avoid operation between 18 and 28% RPM except for transients occurring during engine start and shutdown. 96% is minimum RPM during flight.

(3) 5 minutes of wet power is allowed the engine. However, a full AWI tank will provide only approximately 1 minute of wet power.

(4) Static takeoff power should be limited to 2206 ft.-lb. to preclude over torque condition due to ram effects on normal takeoff or water injection during a single engine emergency.

(5) Maximum ITT permissible during duration of AWI firing.

(6) 101% for 5 minutes and 105% for 5 seconds is allowed.

(7) Typical engine speeds for low and high speed lever positions.

(8) Above 23,000 feet, minimum oil pressure is 50 psi.

Engine Limits For SA226 – T(A) / AT / TC
TPE-331-3U-303G / 304G with AWI
Table 203

(14) Bleed air switch OFF.

(15) Speed lever high.

(16) Water injection switch check both pump lights ON.

CAUTION: DO NOT EXCEED 923~C ITT OR 2,363 FOOT-POUNDS TORQUE.

(17) Advance power lever to ensure that torque meets the requirement for continuous wet takeoff, see Table 203.

(18) AWI switch OFF.

(19) Observe reduced torque on both engines and that both No. 1 and No. 2 PUMP ON lights extinguish.

(20) Power lever's ground idle.

(21) Speed lever low, idle engine for five minutes prior to shutdown.

(22) Stop engine in accordance with AFM.

LL. Corrective Action (Dual (Continuous) AWI System):

(1) Check installation of AWI Wiring Manual and Aircraft Maintenance Manual, Chapter 82.

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ENGINE GROUND OPERATION (WORKSHEET) – TPE331-3U-303G/304G – MAINTENANCE PRACTICES

1. Preparation – Engine Ground Run (Worksheet)

A. Engine Ground Run Worksheet

NOTE: The engine ground run worksheet is used to record accumulated engine ground run data. For engine ground run adjustments refer to Engine Ground Run (Adjustments), this chapter.

Aircraft S/N_____ **TTA**_____ **DATE**_____

LH Engine #_____ **TTE**_____ **RH Engine #**_____ **TTE**_____

CAUTION: ENGINE STARTING, OPERATION, AND SHUTDOWN PROCEDURES WILL BE PERFORMED BY QUALIFIED PILOTS OR MAINTENANCE PERSONNEL IN ACCORDANCE WITH THE AIRCRAFT FLIGHT MANUAL.

NOTE: The following procedures are intended only as a guide. See FAIRCHILD AIRCRAFT MAINTENANCE MANUAL, ADJUSTMENT/TEST (Engine Ground Run Adjustments) for complete check procedures, arranged under identically numbered steps, with corrective action reference.

STEP	LH ENGINE	RH ENGINE
(1) PRELIMINARY CHECK (Completed).....	YES NO	YES NO
(2) N.T.S. GROUND CHECK (O).....	YES NO	YES NO

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-3U-303G/304G – MAINTENANCE PRACTICES

STEP	LH ENGINE	RH ENGINE

(3) ENGINE STARTING CHECK

(a) Light Off Time (Maximum 10 Sec)

1 Time from 10% to light off..... _____ SEC _____ SEC

(b) Start Time (Maximum 60 Sec)

1 Time from 0% to stable RPM..... _____ SEC _____ SEC

(4) OVERSPEED GOVERNOR CHECK CAUTION: DO NOT EXCEED 106% RPM. DO NOT STAY ON O.S.G. MORE THAN 30 SEC.

(a) Start Locks Engaged.

(b) Oil Temp Normal (55-110°C).

(c) S/L in LOW RPM.

(d) Advance P/L to Max Stable RPM.

(e) Record O.S.G. RPM Settings..... _____% RPM _____% RPM

NOTE: RPM must be 103-105%.

STEP	LH ENGINE	RH ENGINE

(5) FLIGHT IDLE CHECK

(a) Flight Idle Flat Position Check

1 Start locks engaged, S/L LOW RPM, P/L ground idle (MIN fuel flow).

2 Place tape on pedestal between P/L.

3 Advance P/L slowly, observe FF.

4 Mark tape where FF increase stops.

5 Continue to advance P/L, mark tape where FF begins to increase again.

6 Record if flight idle flat begins before and ends after P/L flight idle stop gate.....YES NO YES NO

(b) Power Lever Split Check (Flight Idle)

1 S/L in LOW RPM, advance P/L to 100% RPM, retard abruptly to FI.

2 Hold both P/L even, advance slowly until fuel flow and RPM begin to rise.

3 Mark tape at point where F.F. and RPM begin to increase for each P/L.

NOTE: MAX distance between marks 0.05 inch.

4 Record if distance between marks is within limits.....YES NO YES NO

STEP	LH ENGINE	RH ENGINE

(c) Flight Idle Fuel Flow Check

1 With propeller start locks engaged, S/L LOW RPM. Advance P/L to 100% RPM, then retard abruptly to flight idle.

2 Record RPM (approx. 96-98%)..... _____%RPM ____%RPM
Record fuel flow.
(approx. 185-200 PPH)..... _____ PPH ____ PPH

NOTE: Final adjustment may require flight test of yaw and descent rate.

(6) UNDERSPEED GOVERNOR CHECK

NOTE: Start locks disengaged, P/L in G.I. (MIN fuel flow)

(a) U.S.G. Low And High RPM Check

1 U.S.G. low RPM check

a S/L in LOW RPM.

b Record RPM (64-66%)..... _____%RPM _____%RPM

2 U.S.G. high RPM check

a S/L in HIGH RPM.

b Record RPM (96.0-97.0%)..... _____%RPM _____%RPM

(b) Flame Out Check

1 S/L in HIGH RPM.

2 S/L to LOW RPM abruptly (three times).

3 Record if engine flames out or deceleration time exceeds ten seconds..... .....YES NO YES NO

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-3U-303G/304G – MAINTENANCE PRACTICES

STEP	LH ENGINE	RH ENGINE

(7) PROPELLER GOVERNOR CHECK

CAUTION: DO NOT EXCEED 923°C ITT, 2206 FT.LB. TORQUE, OR 102.5% RPM.

NOTE: P.G. RPM will vary with oil temp. For each three deg above 75°C oil temp., RPM will decrease 0.1%. Start locks must be disengaged.

(a) P.G. High RPM Check

1 S/L in HIGH RPM.

2 Advance P/L until RPM no longer increases with further advancement.

3 Record engine RPM (99.5-100.5%).....—%RPM ———%RPM

NOTE: BETA light extinguished.

(b) P.G. Low RPM Check

1 Retard P/L to 700°C ITT, then place S/L in LOW RPM.

CAUTION: Do not exceed 885°C ITT in step below.

2 Advance P/L until RPM no longer increases with further advancement.

3 Record engine RPM (93.5-94.5%)..... _____%RPM _____%RPM

NOTE: BETA light extinguished.

(8) U.S.G. – P.G. SEPARATION CHECK

NOTE: Start locks disengaged.

(a) S/L in HIGH RPM.

(b) Advance P/L to 700°C ITT.

CAUTION: DO NOT EXCEED 885°C ITT. Adjust P/L as needed to stay in limits.

(c) Retard S/L to 96% RPM.

(d) Retard P/L to Ground Idle (MIN FF).

STEP		LH ENGINE	RH ENGINE
(e)	Record Engine RPM.....	_____%RPM	_____%RPM

NOTE: 93.5% RPM MAX (2.5% RPM separation).

(9) BETA LIGHT OPERATION CHECK

(a) S/L in LOW RPM and P/L in Ground Idle.

(b) Advance P/L to approx. 85% RPM.

(c) Record RPM BETA Light illuminates..... _____%RPM _____%RPM

NOTE: Must illuminate at or before 85% RPM.

(10) REVERSE POWER – LOW RPM CHECK

(a) Reverse Power – LOW RPM P/L Split Check

1 Both S/L in LOW RPM and both P/L in ground idle (MIN. FF)

.

2 Place tape on pedestal between P/L.

3 Hold both P/L even, retard slowly until FF and RPM begin to increase.

4 Mark tape at point where FF and RPM begin to increase for each P/L.

NOTE: MAX distance allowed is 0.25 inch.

5 Record if distance between marks is within limits.....YES NO YES NO

STEP LH ENGINE RH ENGINE

(b) Reverse Power – LOW RPM – Full Reverse

CAUTION: DO NOT ALLOW RPM TO DECREASE (BOG DOWN) WHEN IN REVERSE. DO NOT EXCEED 923°C ITT.

1 Both S/L in LOW RPM and both P/L in ground idle.

2 Retard both P/L to full reverse.

3 Record engine RPM..... _____%RPM _____%RPM

NOTE: RPM 75% MIN, LH/RH split 5% Max.

(11) REVERSE POWER – HIGH RPM CHECK

CAUTION: DO NOT OPERATE ENGINES AT FULL REVERSE OVER 30 SECONDS.

(a) Both S/L in HIGH RPM.

(b) Advance both P/L to stable 100% RPM.

(c) Retard both P/L smoothly into reverse.

(d) Record time to reach full reverse....._____SEC _____SEC
(Five SEC MAX, LH/RH split three SEC MAX)

(e) Record full reverse RPM..... _____%RPM _____%RPM
(RPM 95.5% MIN)
Record full reverse torque..... _____%FT.LB._____%FT.LB.
(LH/RH torque split 100 FT.LB. MAX)

STEP	LH ENGINE	RH ENGINE

(12) POWER LEVER SPLIT CHECK (MAX & CRUISE)

(a) Place tape on pedestal between P/L.

(b) Turn bleed air OFF.

(c) S/L in HIGH RPM.

CAUTION: DO NOT EXCEED 923°C ITT OR 2206 FT.LB. TORQUE.

(d) Advance P/L to takeoff power as determined by AFM/Performance Section, Takeoff Power Check Chart, 0-50 KTAS (DRY- BLEED AIR OFF).

STEP		LH ENGINE	RH ENGINE
(e)	Mark tape at P/L position (MAX power).		
(f)	Retard P/L to approx. 700°C ITT.		
(g)	Retard S/L to cruise RPM (96% RPM).		
(h)	Advance P/L to 885°C ITT.		
(i)	Mark tape at P/L position (CRUISE PWR).		
(j)	Retard P/L to G.I. and S/L to LOW RPM.		
(k)	Repeat (c) through (j) on opposite engine.		
	NOTE: MAX distance allowed .05 inch.		
(l)	Record if P/L split is within limits.....YES NO		YES NO
(m)	Repeat (c) through (l) with bleed air on. Record if P/L split is within limits.....YES NO		YES NO

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-3U-303G/304G – MAINTENANCE PRACTICES

STEP	LH ENGINE	RH ENGINE
(13) MAXIMUM POWER CHECK (DRY)		
CAUTION: DO NOT EXCEED 923~C ITT OR 2206 FT.LB. torque.		
(a) Record outside air temp (OAT).....	_____°C	_____°C
NOTE: Use calibrated thermometer outside aircraft, not aircraft indicator.		
(b) Record pressure altitude (set aircraft altimeter to 29.92 in. hg.).....	_____ FT	_____ FT
(c) Using AFM, Performance Section, Takeoff Power Check Chart 0-50 KTAS (BLEED AIR OFF); determine expected torque.....	_____FT.LB.	_____FT.LB.
(d) Bleed air OFF. S/L HIGH RPM, advance P/L to 923°C ITT or expected torque (DRY-BLEED AIR OFF) is obtained. Record if expected torque is reached.....	.....YES NO	YES NO

STEP		LH ENGINE	RH ENGINE
(e)	Record engine instrument readings at maximum power.		
	ITT (approx.923°C).....	_____°C	_____°C
	RPM (99.5-100.5).....	_____%RPM	_____%RPM
	Fuel Flow.....	_____PPH	_____PPH
	Oil Temperature (70-90~C).....	_____°C	_____°C
	Oil Pressure (70-120 psi).....	_____PSI	_____PSI
	Fuel Pressure (20-80 psi).....	_____PSI	_____PSI
	Deice Pressure (12-19 psi).....	_____PSI	_____PSI
	Suction (4.4-4.8 in.hg.).....	___in.hg.	___in.hg.
(f)	Retard P/L to G.I. and place S/L in low RPM.		

(14) ALCOHOL-WATER INJECTION (AWI) CHECK

NOTE: Several different AWI systems were installed on SA226 series aircraft as standard equipment or customer option. If the aircraft is equipped with an AWI system, see the ENGINE GROUND OPERATION and perform check with correct effectivity.

Record if AWI system checks OK.....YES NO YES NO

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-3U-303G/304G – MAINTENANCE PRACTICES

STEP	LH ENGINE	RH ENGINE
(15) IGNITION MODE CHECK		
(a) Momentarily select continuous ignition. Record if ignition light illuminates.....	YES NO	YES NO
(b) Momentarily select override ignition. Record if ignition light illuminates.....	YES NO	YES NO
(c) Return ignition switch to normal.		
(16) ENGINE INTAKE ANTI-ICE SYSTEM CHECK		
CAUTION: DO NOT OPERATE ENGINE INTAKE ANTI-ICE SYSTEM MORE THAN TEN SECONDS IF OAT IS ABOVE FIVE DEGREES C.		
(a) S/L in HIGH RPM.		
(b) P/L in ground idle (MIN fuel flow).		
(c) ENGINE & PROP HEAT switch to upper ENG & PROP position.		
(d) Record if ITT increases.....	YES NO	YES NO
Record if INTAKE HEAT ON light illuminates for appropriate engine.....	YES NO	YES NO
(e) ENGINE & PROP HEAT switch to OFF.		
(f) Record if ITT decreases.....	YES NO	YES NO
Record if INTAKE HEAT ON light extinguishes for appropriate engine.....	YES NO	YES NO

STEP	LH ENGINE	RH ENGINE

(17) ENGINE SHUTDOWN-PROP UNFEATHER CHECK

NOTE: If power runs have been made, allow three minutes to cool down. Charge EPA kit by maintaining 96% RPM momentarily.

(a) Emergency Shutdown (Stop and Feather)

1 S/L in LOW RPM.

2 Advance P/L to 75% RPM.

CAUTION: IF ANY RISE IN TORQUE OR ITT IS NOTED DURING NEXT STEP, PUSH STOP BUTTON IMMEDIATELY.

3 Pull STOP AND FEATHER knob until red groove in cable shaft is visible at pedestal cover. Record engine flames out, but that propeller does not feather..... YES NO YES NO

CAUTION: DISCHARGE EPA KIT BEFORE ENGINE RPM IS BELOW 40% OR FUEL ACCUMULATION IN PLENUM MAY OCCUR. IF NOT DISCHARGED BEFORE ENGINE STOPS, ALLOW ENGINE TO COOL BEFORE DISCHARGING EPA KIT.

4 Depress STOP button at least five SEC.

5 At 40% RPM pull STOP AND FEATHER full aft, record if propeller feathers..... YES NO YES NO

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-3U-303G/304G – MAINTENANCE PRACTICES

STEP	LH ENGINE	RH ENGINE
(b) Propeller Unfeathering Check		
1 Hold P/L in full reverse.		
2 UNFEATHER TEST switch to L or R. Record if propeller blades drive toward reverse position.....	YES NO	YES NO
3 When propeller blades reach full reverse, release UNFEATHER TEST switch and move P/L to 1/4 inch forward of flight idle.		
4 Record if propeller returns to start locks.....	YES NO	YES NO
(c) Normal Engine Shutdown		
NOTE: Restart engine in accordance with AIRCRAFT FLIGHT MANUAL procedures. Charge EPA kit by maintaining 96% RPM momentarily.		
1 S/L in LOW RPM and P/L in ground idle (MIN fuel flow).		
2 Depress STOP for at least five SEC.		
3 Record if RPM increases approx. five percent as EPA kit discharges.....	YES NO	YES NO
4 Allow RPM to decrease to 50% then retard P/L to full reverse, hold until RPM decreases to 10%. Record if propeller is on start locks.....	YES NO	YES NO

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ENGINE GROUND OPERATION – TPE311-1OU-501 ENGINE – DESCRIPTION AND OPERATION

1. General

A. The following practices detail the procedures and checks to be performed during normal engine ground run. These practices do not teach elementary engine ground run principles, but are inserted to provide manual users with a general knowledge of ground run procedures under normal engine operation. Only pilots or qualified personnel shall perform ground run procedures which shall be accomplished in accordance with the Aircraft Flight Manual/Pilot's Operating Handbook. Normal engine start procedures, operation, and shutdown will be performed in accordance with Section II of the Aircraft Flight Manual. Reference Section III, Aircraft Flight Manual for Emergency Operating Procedures.

B. These procedures shall be conducted and satisfactorily performed prior to initial flight. Check procedures need not be performed on both engines simultaneoulsy except where matching of engine is so indicated within the procedures.

C. Initial rigging of the powerplant/controls is described in the AiResearch Maintenance Manual, and Fairchild Maintenance Manual.

D. Each check/functional test procedure and its applicable corrective action is identified by its title and number. A ground run check sheet is also titled and itemized to correspond with each applicable procedure.

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ENGINE GROUND OPERATION – TPE311-1OU-501G ENGINE – MAINTENANCE PRACTICES

1. Adjustment/Test – Engine Ground Operation – TPE331-10U-501 Engine

WARNING: BLEED OFF ALL RESIDUAL HYDRAULIC PRESSURE BEFORE TURNING ELECTRICAL POWER OFF. THIS CAN BE ACCOMPLISHED BY ACTUATING THE FLAP SELECTOR HANDLE DOWN 1/4 INCH AND THEN BACK UP. REPEAT UNTIL THE HYDRAULIC PRESSURE INDICATOR READS ZERO. ELECTRICAL POWER MAY NOW BE TURNED OFF.

A. System Test Procedures and Corrective Action

(1) The corrective actions provided for each procedure are those considered most likely to produce desired results to problems, if any, encountered during the check procedure. An attempt has not been made to cover all situations or provide information for troubleshooting problems that cannot be corrected by minor adjustments due to component malfunction. If during a ground check procedure the acceptance criteria of the check is not met, all pertinent information should be recorded; i.e., record all engine instrument readings at the point of the check and any other data that may aid in troubleshooting or adjustment.

(2) Perform corrective steps in the indicated sequence. If after performing the indicated steps, the problem remains unresolved, contact M7 Aerospace Engineering Propulsion Group and/or AiResearch Field Service Representative.

2. Inspection/Check – Engine Ground Operation TPE331-10U-501 Engine

NOTE: The following guide may be helpful in making adjustments

FUNCTION	ADJUSTMENTS	EFFECT
Underspeed Governor		
Min. Speed Stop	1 turn counterclockwise	2.4% Reduction (RPM)
Max. Speed Stop	1 turn counterclockwise	3.0% Increase (RPM)
Propeller Governor		
Low Speed Stop	1 turn counterclockwise	1.0% Reduction (RPM)
High Speed Stop	1 turn clockwise	1.0% Increase (RPM)
Screw X	1 turn clockwise	1.0% Increase (RPM)
Flight Idle Adjustment	1 click clockwise	1 PPH Increase (fuel flow)
Beta Tube Assembly	1 turn counterclockwise	2.0 Degrees Increase (blade angle)

Engine Control Adjustment
Table 201

A. Engine Start Time – Check Procedures

(1) Conduct normal start per Aircraft Flight Manual (AFM) procedures.

(2) Record elapsed time between RPM = 10% and light OFF.

(3) Elapsed time may not exceed:

(a) Ground start = 10 seconds.

(b) Air start = 15 seconds.

B. Corrective Action (Engine Start Time)

(1) Check battery and electrical system.

(2) Reference AiResearch Maintenance Manual, Engine Troubleshooting section.

C. Overspeed Governor Check

(1) Start locks must be engaged.

(2) After normal engine start, operate engine until oil temperature reaches normal range (70 to 90°C).

(3) Place speed lever at LOW.

CAUTION: DO NOT EXCEED 106% RPM.

(4) Advance power lever toward maximum position, observe RPM. When governor setting is reached, the RPM should no longer increase with further movement of the power lever.

(5) Observe RPM, it should stabilize between 104% to 105% RPM.

(6) Repeat procedures for other engine.

D. Corrective Action - Overspeed Governor Check

NOTE: Do not remove the power lever arm assembly from the power lever shaft.

(1) Disconnect rod-end from power lever arm on fuel control by removing cotter pin, nut, and washer.

(2) Remove intermediate (flight idle) fuel flow adjustment plug.

NOTE: Position power lever shaft in 100° position to aid in removal of cover from fuel control unit.

(3) Remove the six screws and two 12 point hex head bolts which secure to the fuel control unit.

(4) Rotate power lever shaft to minimum position, then tap cover loose using soft faced mallet and remove cover.

NOTE: During removal (after clearing alignment pins) rotate cover toward vertical centerline of engine to clear second cam from internal linkage.

(5) Use 5/64 inch swivel tip Allen wrench or equivalent. Adjust governor as required, see Table 201. One turn clockwise increases RPM 4.5%.

CAUTION: ENSURE MANUAL FUEL CAM FOLLOWER LEVER IS IN SLOT PRIOR TO POSITIONING COVER ONTO FUEL CONTROL. OTHERWISE AN OVER-TEMP CONDITION WILL RESULT ON ENGINE START.

(6) Place manual fuel valve shaft to minimum position, install cover with new gasket (P/N 3054-726).

(7) Install screws, torque 12 to 14 inch-pounds. Install bolt; torque 60 to 75 inch-pounds.

(8) Install flight idle access plug and spring, safety plug.

(9) Connect control rod to power lever arm with washer and nut. Torque nut to 12 inch-pounds, install cotter pin.

E. Underspeed Governor Low RPM Check

(1) Disengage start locks per Aircraft Flight Manual.

(2) Place speed lever in LOW position.

(3) Place power lever in ground idle (minimum fuel flow) position.

(4) RPM must be 69 plus, minus 1%

F. Corrective Action – Underspeed Governor Low RPM Check

(1) Adjust minimum speed stop (low RPM) on the underspeed governor. Turning adjustment screw one turn clockwise will increase engine RPM 5.0%.

G. Underspeed Governor High RPM Check

(1) Place speed lever in HIGH position.

(2) Place power lever in ground idle position.

(3) RPM must be 96 plus, minus .5%.

H. Corrective Action – Underspeed Governor High RPM Check

(1) Adjust maximum stop on the underspeed governor. Turning the adjustment screw clockwise will reduce engine RPM. Turning the screw one turn will change engine RPM by approximately 3.0%.

(2) Adjustment of the maximum stop on the underspeed governor affects the propeller governor high RPM setting. The governor RPM setting must be checked after adjustment of the underspeed governor.

I. Flight Idle Flat Check

(1) Start locks must be engaged.

(2) Place bleed air mass flow control valve in ON position.

(3) Place speed lever in LOW position.

(4) Place masking tape between power lever on pedestal.

(5) Advance power lever from ground idle position until fuel flow stops increasing. Mark tape with respect to a point on the power lever at this position.

(6) Record fuel flow at the flat.

(7) Continue advancing power lever until fuel flow again begins to increase. Mark tape with respect to same point on power lever at this.

(8) Advance power lever and retard sharply to the flight idle stop. Fuel flow must be the same as previously noted, and reference point on power lever must fall between the two marks on the tape. Marks on the tape where fuel flow begins increasing on each engine must be aligned within .050 inches.

(9) Record fuel flow and RPM. RPM must be 96% to 98%. Fuel flow should be approximately 215 PPH. If fuel flow is not met within tolerances RPM, flight idle propeller blade angle is suspect and should be rechecked per Fairchild Aircraft Maintenance Manual.

(10) Turn bleed air OFF.

NOTE: Adjustment of flight idle fuel flow to 96% to 98% RPM will make the aircraft safe for flight purposes. However, final evaluation of the fuel flow for flight idle shall be determined by the pilot during production flight tests and should be adjusted as requested to obtain proper flight characteristics.

J. Corrective Action – Flight Idle Flat Check

(1) Check rigging of flight idle per Fairchild Aircraft Maintenance Manual.

(2) For misalignment of flight idle flat with power lever in flight idle position or misalignment of marks where fuel flow begins increasing, adjust the micro vernier on the manual fuel valve on one or both engines. This adjustment changes the position of the manual fuel valve at flight idle, thus changing the relationship between flight idle flat and power lever position.

(3) For improper RPM with the power lever in the flight idle position, adjust the flight idle fuel flow. One click produces a nominal fuel flow change of 2 PPH. If fuel flow must be adjusted excessively, check the flight idle propeller blade angle per M7 Aerospace Maintenance Manual. Observe that flight idle RPM is within tolerance when flight idle fuel flow is obtained.

K. Propeller Governors – High RPM Check Procedure

(1) Place speed lever in HIGH position.

(2) Advance power lever past flight idle toward takeoff position until 525°C EGT. Observe that beta lights go out before RPM stabilizes.

(3) Record RPM, Observe that RPM is below 100.0 plus 1.0% minus 0.00% except as noted in number 4 below. Right engine must be higher than left.

(4) RPM value given is for 75°C, decrease value of stabilized RPM 0.1%. For each 3.0~C lower than 75°C, increase value of stabilized RPM 0.1%.

L. Corrective Action Propeller Governor – High RPM Check

(1) Adjust maximum stop to obtain desired RPM. One turn clockwise decreases RPM 1%.

(2) Adjustment of the maximum stop on the propeller governor affects the underspeed governor high RPM setting. Check the underspeed governor high RPM setting after adjustment of the propeller governor.

M. Propeller Governor, Low RPM - Check

(1) Place speed lever in LOW position.

(2) Advance power lever past flight idle toward takeoff position until 525° EGT. Observe beta light goes out before RPM stabilizes.

(3) Observe RPM ise 93.5% to 94.5%.

(4) Correction for oil temperature per Propeller Governor - High RPM Check, No. 4.

N. Corrective Action - Propeller Governor - Low RPM Check

(1) Adjust low speed stop on propeller governor, see Table 201.

NOTE: Turning stop screw clockwise will increase RPM. One turn will change RPM approxi mately 1%.

O. Reverse Power - High RPM - Check - Procedure

(1) Place speed lever in HIGH position and both power levers forward until RPM is stable at 100%.

(2) Retard both power levers smoothly into full reverse.

(3) Observe that the time differential between engines for start of torque rise as the engines are pulled into reverse is less than three seconds.

(4) Observe that no more than five seconds is required for the engine to go from 100% governing to full reverse.

(5) Observe That reverse RPM is 95.5% minimum, and the difference in torque values at full reverse ise less than 4%.

CAUTION: OPERATION AT FULL REVERSE WHILE AT ZERO AIRSPEED SHOULD BE LIMITED TO 30 SECONDS DUE TO POSSIBILITY OF INGESTION OF EX-HAUST GASES.

P. Corrective Action - Reverse Power – High RPM Check

NOTE: Reverse power time differential greater than three seconds or more than five seconds is required to achieve full reverse.

(1) Check proper adjustment of screw W in accordance with M7 Aerospace Maintenance Manual.

(2) Check underspeed governor high RPM value in accordance with step (7) of Test Procedures.

(3) Check general rigging of prop pitch control and manual fuel valve linkage, in accordance with M7 Aerospace Maintenance Manual.

Q. Torque Split at Full Reverse Greater than 4%

(1) Check rigging to determine that prop pitch control linkage travels to full reverse position in accordance with M7 Aerospace Maintenance Manual.

R. Reverse Power – Low RPM - Check Procedure

(1) Place speed lever's in LOW position.

(2) Place power lever's in ground idle position.

(3) Slowly move power lever of one engine toward reverse.

(4) Mark on tape the point where RPM begins to increase.

(5) Repeat with other power lever.

(6) Retard both power lever's to full reverse. Observe RPM is 75.0% minimum at full reverse. Observe separation is not more than 5% between engines.

(7) Ensure marks on tape where RPM begins to increase are not more than 0.50 inch.

S. Corrective Action - Reverse Power – Low RPM Checks

(1) Check propeller pitch control position at approximately 14° when screw V contacts tab of speed lever in LOW position.

(2) Adjust screw V on one or both engines as necessary to remove power lever split, keeping pitch control position at contact of screw V as near 14° as possible.

Power Setting	Time Limit (min)	Maximum Torque 100% = 2972 (ft.-lb.)	Maximum SRL (°C) ET	RPM(%) 100% = 1591 (1)	Oil Pressure (psi)	Fuel Pressure (psi)	Oil Temp. (°C)
Takeoff	–	100(8)	650(6)	100(2)	70 to 120	20 to 80	55(4) to 110
Maximum Continuous	–	100	650(6)	100	70(3) to 120	20 to 80	55(4) to 110
Flight Idle	–	–	–	96 to 100	70(3) to 120	20 to 80	55(4) to 110
Ground Idle	–	–	–	69 to 97(6)	40 to 120	15 to 80	-40 to 127
Starting	–	–	770	–	–	–	-40 minimum
Reverse High	–	–	–	95.5 to 97	70 to 120	20 to 80	55(4) to 110
Reverse Low	–	–	–	75 minimum	70 to 120	15 to 80	-40 to 127
Shutdown	3 min.(7)	–	–	–	–	–	–

(1) Avoid operation between 18% and 28% RPM except for transients occurring during engine start and shutdown. 96% is the minimum RPM during flight.

(2) 101% for 5 minutes and 105% for 5 seconds is allowed.

(3) Above 23,000 feet, minimum oil pressure is 50 psi.

(4) Operation from 22°C to 55°C allowed as a function of ambient temperature.

(5) Typical engine speeds for low and high speed lever positions.

(6) 675°C allowed for 20 seconds. If exceeded, conduct power check. 680°C allowed for up to 5 seconds with power checks, if 5 seconds is exceeded, remove engine for inspection.

(7) 3 minutes minimum at taxi conditions (taxi time included) before shutdown.

(8) Static takeoff power should be limited to 97% to preclude overtorque condition occurring due to ram effects during takeoff.

Engine Limits For SA226 – T(B)
TPE-331-10U-501G / 511G Without AWI
Table 202

(3) Recheck RPM obtained at full reverse, as adjustment of screw V will affect the RPM obtained.

(4) Recheck propeller governor high RPM in accordance with step 8) of Test Procedures.

T. RPM Obtained at Full Reverse Not Within Tolerance

(1) Check propeller pitch control.

(2) Adjust point of attachment of link I to arm F, see Table 202, as necessary to increase engine RPM with power lever in full reverse position. Moving arm F towards maximum RPM stop increases RPM.

(3) Recheck power lever split at point of increasing RPM and RPM obtained at full reverse in accordance with step (11) of Test Procedures.

(4) Repeat step (11)., Corrective Action, until both power lever split and RPM obtained are within tolerance specified in step (11) Test Procedures.

(5) Recheck propeller governor high RPM in accordance with step (8) of Test Procedures.

U. Power Lever Split Check

(1) Ensure that bleed air off.

(2) Place power lever in ground idle position.

(3) Place speed lever's in HIGH position.

(4) Advance one power lever to stabilized takeoff power. (Takeoff power is defined as 650°C EGT or 100% torque, whichever occurs first.) If EGT limited, check minimum acceptable torque in accordance with applicable chart in Table 202.

(5) Mark on tape power lever position at takeoff power.

(6) Retard power lever to ground idle position and repeat check with other power lever.

(7) Ensure marks are not more than 0.05 inch apart.

(8) Repeat steps (3) through (7) with bleed air on. Calculate and record torque loss.

(9) Repeat steps (3) through (7) with bleed air on. Place speed lever at 96%, EGT 650°C.

V. Corrective Action (Power Lever Split at Takeoff Power – Bleed Air Off

(1) Check for free play in control cables.

(2) Recheck flight idle flats and correct as necessary in accordance with paragraph Flight Idle Flat Check.

(3) Adjust maximum fuel flow adjustment on fuel control on one or both engines to remove excessive power lever split.

NOTE: One-fourth (one-click) turn clockwise increases fuel flow four PPH. Do not increasemore than one turn.

(4) Note amount of adjustment in the Engine Log Book.

W. Prop Governor – Underspeed Governor Separation Check

(1) Place speed lever in High position.

(2) Move power lever toward takeoff power until RPM stabilizes at 100.0%.

(3) Slowly retard speed lever until RPM stops decreasing.

(4) Leave speed lever in above position and retard power lever to ground idle position. Observe RPM stabilizes at least 2.5% below steps (3) stabilized RPM.

X. Corrective Action (Prop Governor – Underspeed Governor Separation Check)

(1) Adjust length of propeller governor arm as necessary to obtain 2.5% separation, see Table 201.

NOTE: Turning prop governor arm out increases RPM separation.

(2) Recheck propeller governor high RPM in accordance with step (7) of Test Procedures, and adjust if necessary.

(3) Repeat steps (14) and step (7), Test Corrective Action, until both are within tolerance.

Y. Beta Light Operation Check

(1) Place power lever in ground idle position.

(2) Advance speed lever to 85% RPM.

(3) Beta light must be ON, or come on before engine speed reaches 85%.

Z. Corrective Action (Beta Light Operation Check)

(1) Conduct reverse power high RPM check in accordance with step (12) of Test Procedures.

(2) If step (12) check is unacceptable, conduct Corrective Action per step (11).

(3) If step (11) check is acceptable, malfunction of the beta switch is indicated, and the switch, light, and related circuitry should be checked for cause.

AA. Flame Out Check

(1) Place speed lever in HIGH position.

(2) Place power lever in ground idle position.

(3) Retard speed lever rapidly (less than one second) to LOW position.

(4) Observe that engine does not flame out.

(5) Observe deceleration time does not exceed 10 seconds.

BB. Corrective Action (Flame Out Check)

(1) If limits are not met, recheck underspeed governor setting in HIGH or LOW. Reset speeds within limits and repeat Flame Out Check, steps (1) through (5).

(2) Contact AiResearch Field Service Representative.

CC. Emergency Shut-Down Check

(1) Place speed lever in LOW position.

(2) Set power lever to approximately 75% RPM.

CAUTION: IF ANY RISE IN TORQUE OR EGT IS NOTE, PUSH ENGINE STOP BUTTON IMMEDIATELY.

(3) Pull stop and feather knob until red groove in fuel shutoff cable rod is visible at pedestal. Observe that engine flames out.

(4) When RPM decreases to approximately 40%, pull knob all the way aft.

(5) Observe that propeller feathers.

DD. Corrective Action (Emergency Shut-down Check)

(1) Recheck rigging of manual fuel valve shut off and feathering valve linkage in accordance with M7 Aerospace Maintenance Manual.

EE. Negative Torque Sensing Check

(1) Set up for normal engine start in accordance with AFM Procedures.

(2) Check negative torque sensor annunciator panel lights with Press-To-Test switch.

(3) Place negative torque sensor test switch on selected engine (left or right as appropriate).

(4) Turn on unfeather pump (hold ON).

(5) Observe that negative torque sensor light illuminates.

(6) Depress start button for normal AFM start (release at ignition).

(7) Observe that negative torque sensor light extinguishes at starter actuation.

(8) Observe that negative torque sensor light illuminates at 15% to 30% RPM.

(9) Turn off unfeather pump (test switch to OFF) at start completion.

(10) Observe that negative torque sensor light extinguishes at start completion.

(11) Repeat steps (1) through (10) for other engine.

FF. Corrective Action (Negative Torque Sensing Check)

(1) Check installation of system, see Aircraft Wiring Manual and Maintenance Manual, Chapter 77.

GG. SRL/Temp Limiter Checks

During the SRL test, the EGT difference between SRL ON and OFF for one engine is compared with EGT difference between SRL ON and OFF for the opposite engine.

(1) Left power lever.....Advance to 100 Plus with Propeller on Start Lock

(2) EGT.....Stabilize and Note Value

(3) Left SRL/Delta P/P Test Switch....SRL Off/Note Change in EGT Value/LH SRL Off Light Illuminated

NOTE: Observe that the change in EGT noted between engines with SRL OFF and ON does not exceed 10°C.

EXAMPLE:

Left Engine	Right Engine
SRL ON 640 degrees	635 degrees
SRL OFF 570 degrees	560 degrees
70 degrees	75 degrees

5 degrees difference

NOTE: Observe that the difference between engine does not exceed 10°. If 10° difference is exceeded, check and correct the SRL system.

(4) Left SRL/Delta P/P Test Switch.....Norm (SRL Off Ex-tinguished)

(5) Left SRL/Delta P/P Test Switch.....P/P OFF (Note EGT Increase of Approximately Five Degrees)

NOTE: If no increase is noted, check and correct system.

(6) Left SRL/Delta P/P Test Switch.....Norm

(7) Temp Limiter Test Switch.....L

(Note: Decrease in engine RPM Fuel Flow and Illumination of Fuel Bypass Open Light)

(8) Temp Limiter Test Switch.....Release/Note Increase in Fuel Flow and Fuel Bypass Light Out

(9) Left Power.....Ground Idle

(10) Repeat steps (1) through (9) for right engine.

HH. Continuous Alcohol/Water Injection System Check (CAWI)

(1) Check CAWI quantity indication against quantity shown on sight gauge.

(2) Speed lever LOW.

(3) Power lever ground idle.

(4) CAWI test switch, hold switch in No. 1 position and check that No. 1 pump on light illuminates and No. 2 pump light does not illuminate. Hold switch in No. 2 position and check that No. 2 pump on light illuminates and No. 1 pump on light does not illuminate. Release switch, observe that neither pump on light illuminates.

(5) Position speed lever to high.

(6) Position power lever to 30% torque.

(7) Move CAWI switch to cont. check that both pump on lights illuminate, bleed air cuts off. Note approximately 32°C EGT rise and 25% torque rise on both engines immediately after selecting continuous.

CAUTION: DO NOT EXCEED 650°C OR 100% TORQUE.

(8) Advance power lever's to ensure that torque will meet value required by CAWI takeoff minimum power chart in Aircraft Flight Manual.

(9) Retard power lever's to flight idle; note termination of water firing and return of bleed air.

(10) Turn water injection switch off. Check that pump on lights extinguish.

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-1OU-501 – MAINTENANCE PRACTICES

1. Preparation – Engine Ground Run (Worksheet)

A. Engine Ground Run Worksheet

NOTE: The engine ground run worksheet is used to record accumulated engine ground run data. For engine ground run adjustments refer to Engine Ground Run (Adjustments), this chapter.

Aircraft S/N_____ TTA_____ DATE_____

LH Engine #_____ TTE_____ RH Engine #_____ TTE_____

CAUTION: ENGINE STARTING, OPERATION, AND SHUTDOWN PROCEDURES WILL BE PERFORMED BY QUALIFIED PILOTS OR MAINTENANCE PERSONNEL IN ACCORDANCE WITH THE AIRCRAFT FLIGHT MANUAL.

NOTE: The following procedures are intended only as a guide. See FAIRCHILD AIRCRAFT MAINTENANCE MANUAL, ADJUSTMENT/TEST (Engine Ground Run Adjustments) for complete check procedures, arranged under identically numbered steps, with corrective action reference.

STEP		LH ENGINE	RH ENGINE
(1)	PRELIMINARY CHECK (Completed).....	YES NO	YES NO
(2)	N.T.S. GROUND CHECK (OK).....	YES NO	YES NO

NOTE: See ENGINE GROUNDRUN CHECK step (2) for complete check procedure.

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-10U-501 – MAINTENANCE PRACTICES

STEP			LH ENGINE	RH ENGINE
(3)	ENGINE STARTING CHECK			
	(a)	Light Off Time (Maximum 10 Sec)		
	1	Time from 10% to light off.....SEC		SEC
	(b)	Start Time (Maximum 60 Sec)		
	1	Time from 0% to stable RPM.....SEC		SEC
	(c)	SRL Auto Start Enrichment Check		
	1	Record FF cycle up and down.....YES NO		YES NO
	2	Record EGT range during start		
		(690°C-700°C normal)..... ....._____°C		_____°C
	(d)	Primary Only Solenoid Valve Check		
	1	S/L to HIGH RPM, wait 30-50 sec.		
	2	Record, EGT & FF Fluctuate.....YES NO		YES NO
(4)	OVERSPEED GOVERNOR CHECK			
	CAUTION:	DO NOT EXCEED 106% RPM. DO NOT STAY ON O.S.G. MORE THAN 30 SEC.		
	(a)	Start Locks Engaged.		
	(b)	Oil Temp Normal (55-110~C).		
	(c)	S/L in LOW RPM.		
	(d)	Advance P/L to Max Stable RPM.		
	(e)	Record O.S.G. RPM Settings.....	_____% RPM	_____% RPM
	NOTE:	RPM must be 104-105%.		

STEP	LH ENGINE	RH ENGINE
(5) SRL COMPUTER - DELTA P/P TRANSDUCER CHECK		
(a) Delta P/P Transducer Check		
1 Start locks engaged, S/L LOW RPM, advance P/L to 99.5-100.5% RPM.		
NOTE: Wait 30-50 SEC for primary only solenoid to close.		
2 Record stabilized EGT (DELTA P/P ON).....	_____°C	_____°C
3 SRL – DELTA P/P PWR switch to DELTA P/P OFF.		
4 Record stabilized EGT (DELTA P/P OFF).....	_____°C	_____°C
5 Determine EGT rise (subtract 2 EGT from 4 EGT). (Min. five degrees C).....	_____°C	_____°C
6 SRL – DELTA P/P PWR switch to NORM.		
(b) SRL Computer Check		
1 Record SRL CMPTR OFFannunciator light extinguished above 80% RPM.....	YES NO	YES NO
2 Record stabilized EGT (SRL ON).....	_____°C	_____°C
3 SRL – DELTA P/P PWR switch to SRL OFF.		
4 Record stabilized EGT (SRL OFF).....	_____°C	_____°C
5 Determine EGT change. subtract 4 EGT from 2 EGT.....	_____°C	_____°C
6 SRL – DELTA P/P PWR switch to NORM.		

7 Record LH/RH DIFFERENCE (Ten degrees C maximum difference....._____°C

STEP	LH ENGINE	RH ENGINE
(6) FLIGHT IDLE CHECK		
(a) Flight Idle Flat Position Check		
1 Start locks engaged, S/L LOW RPM, P/L ground idle (MIN fuel flow).		
2 Place tape on pedestal between P/L.		
3 Advance P/L slowly, observe FF.		
4 Mark tape where FF increase stops.		
5 Continue to advance P/L, mark tape where FF begins to increase again.		
6 Record if flight idle flat begins before and ends after P/L flight idle stop gate.....	YES NO	YES NO
(b) Power Lever Split Check (Flight Idle)		
1 S/L in LOW RPM, advance P/L to 100% RPM, retard abruptly to FI.		
2 Hold both P/L even, advance slowly until fuel flow and RPM begin to rise.		
3 Mark tape at point where F.F. and RPM begin to increase for each P/L.		
NOTE: MAX distance between marks		
4 Record if distance between marks is within limits.....	YES NO	YES NO

STEP	LH ENGINE	RH ENGINE

(c) Flight Idle Fuel Flow Check

1 With propeller start locks engaged, S/L LOW RPM. Advance P/L to 100% RPM, then retard P/L to flight idle.

NOTE: Wait 30-50 SEC for primary only solenoid to close.

2 Record RPM (approx. 95-97%)..... _____%RPM ____%RPM
Record fuel flow.
(approx. 215 P.P.H.)..... _____ PPH ____ PPH

NOTE: Final adjustment may require flight test of yaw and descent rate.

(7) UNDERSPEED GOVERNOR CHECK

NOTE: Start locks disengaged, P/L in G.I.

(a) U.S.G. Low And High RPM Check

1 U.S.G. low RPM check

a S/L in LOW RPM.

b Record RPM (70-72%)..... _____%RPM _____%RPM

2 U.S.G. high RPM check

a S/L in HIGH RPM.

b Record RPM (96.5-97.5%)..... _____%RPM _____%RPM

(b) Flame Out Check

1 S/L in HIGH RPM.

2 S/L to LOW RPM abruptly (three times).

3 Record if engine flames out or deceleration time exceeds ten seconds..... YES NO YES NO

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-10U-501 – MAINTENANCE PRACTICES

STEP	LH ENGINE	RH ENGINE

(8) PROPELLER GOVERNOR CHECK

CAUTION: DO NOT EXCEED 650°C EGT, 100% TORQUE, OR 102.5% RPM.

NOTE: P.G. RPM will vary with oil temp. For each three deg above 75°C oil temp., RPM will decrease 0.1%. Start locks must be disengaged.

(a) P.G. High RPM Check

1 S/L in HIGH RPM.

2 Advance P/L until RPM no longer increases with further advancement.

3 Record engine RPM (RPM should be 100-101%)..... _____%RPM _____%RPM

(b) P.G. Low RPM Check

1 Retard P/L to 575°C EGT, then place S/L in LOW RPM.

2 Advance P/L until RPM no longer increases with further advancement.

3 Record engine RPM (93.5-94.5%)..... _____%RPM _____%RPM

(9) U.S.G. – P.G. SEPARATION CHECK

NOTE: Start locks disengaged.

(a) S/L in HIGH RPM.

(b) Advance P/L to 575°C EGT.

CAUTION: DO NOT EXCEED 650~C EGT.

(c) Retard S/L to 96% RPM. Set Friction Lock on S/L.

(d) Retard P/L to Ground Idle (MIN FF).

STEP		LH ENGINE	RH ENGINE
(e)	Record Engine RPM.....	_____%RPM	_____%RPM

NOTE: 93.5% RPM MAX (2.5% RPM separation).

(10) BETA LIGHT OPERATION CHECK

(a) S/L in LOW RPM and P/L in Ground Idle.

(b) Advance P/L to approx. 85% RPM.

(c)	Record RPM BETA Light illuminates.....	_____%RPM	_____%RPM

NOTE: Must illuminate at or before 85% RPM.

(11) REVERSE POWER – LOW RPM CHECK

(a) Reverse Power – LOW RPM P/L Split Check

1	Both S/L in LOW RPM and both P/L in ground idle (MIN. FF).		
2	Place tape on pedestal between P/L.		
3	Hold both P/L even, retard slowly until FF and RPM begin to increase.		
4	Mark tape at point where FF and RPM begin to increase for each P/L.		

NOTE: MAX distance allowed 0.25 inch.

5	Record if distance between marks is within limits.....	YES NO	YES NO

STEP	LH ENGINE	RH ENGINE

(b) Reverse Power – LOW RPM – Full Reverse

CAUTION: DO NOT ALLOW RPM TO DECREASE (BOG DOWN) WHEN IN REVERSE. DO NOT EXCEED 650°C EGT.

1 Both S/L in LOW RPM and both P/L in ground idle.

2 Retard both P/L to full reverse.

3 Record engine RPM..... _____%RPM _____%RPM

NOTE: RPM 75% MIN, LH/RH split 5% max. torque difference.

(12) REVERSE POWER – HIGH RPM CHECK

CAUTION: DO NOT OPERATE ENGINES AT FULL REVERSE OVER 30 SECONDS.

(a) Both S/L in HIGH RPM.

(b) Advance both P/L to stable 100% RPM.

(c) Retard both P/L smoothly into reverse.

(d) Record time to reach full reverse..... _____SEC _____SEC
(Five SEC MAX, LH/RH split three SEC MAX)

(e) Record full reverse RPM..... _____%RPM _____%RPM
(RPM 95.5% MIN)

Record full reverse torque
(4% max. torque difference)..... _____%TOR _____%TOR

STEP	LH ENGINE	RH ENGINE

(13) FUEL BYPASS – TEMPERATURE LIMITER CHECK

CAUTION: DO NOT EXCEED 650 °C EGT OR 100% TORQUE.

(a) Fuel Bypass Check

1 S/L in HIGH RPM.

NOTE: Wait 30-50 SEC for primary only solenoid to close.

2 Advance P/L to 350 PPH fuel flow.

3 TEMP LMTR TEST switch to L or R.

4 Record fuel flow..... _____PPH _____PPH
(FF will decrease approx. 60 PPH)

5 Record if FUEL BYPASS light illuminates.....YES NO YES NO

6 Release TEMP LMTR TEST switch and observe that FF returns to normal.

STEP	LH ENGINE	RH ENGINE

(b) Temperature Limiter Check

1 Temp limiter - Maximum power check

a S/L in HIGH RPM.

b Advance P/L slowly until FUEL BYPASS light illuminates.

c Record EGT..... _____°C _____°C

NOTE: FUEL BYPASS light should illuminate at 650°C EGT.

2 Temp limiter – Cruise power check

a Retard P/L to 575~C EGT then adjust S/L to cruise RPM (98%

b Advance P/L slowly until FUEL BYPASS light illuminates.

c Record EGT..... _____°C _____°C

NOTE: FUEL BYPASS light should illuminate at 650°C EGT.

(14) POWER LEVER SPLIT CHECK (MAX & CRUISE)

(a) Place tape on pedestal between P/L.

(b) Turn bleed air OFF.

(c) S/L in HIGH RPM.

CAUTION: DO NOT EXCEED 650°C EGT OR 100% TORQUE.

(d) Advance P/L to takeoff power as determined by AFM ENGINE POWER SETTING CHART TAKEOFF AND MAXIMUM POWER (DRY-BLEED AIR OFF).

STEP		LH ENGINE	RH ENGINE
(e)	Mark tape at P/L position (MAX power).		
(f)	Retard P/L to approx. 575°C EGT.		
(g)	Retard S/L to cruise RPM (98%).		
(h)	Advance P/L to 650°C EGT.		
(i)	Mark tape at P/L position (CRUISE PWR).		
(j)	Retard P/L to G.I. and S/L to LOW RPM.		
(k)	Repeat (c) through (j) on opposite engine.		
	NOTE: MAX distance allowed .05 inch.		
(l)	Record if P/L split is within limits.....	YES NO	YES NO
(m)	Repeat (c) through (l) with bleed air on.		
	Record if P/L split is within limits.....	YES NO	YES NO

STEP	LH ENGINE	RH ENGINE
(15) MAXIMUM POWER CHECK (DRY)		
CAUTION: DO NOT EXCEED 650°C EGT OR 100% TORQUE.		
(a) Record outside air temp (OAT).....	_____°C	_____°C
NOTE: Use calibrated thermometer outside aircraft, not aircraft indicator.		
(b) Record pressure altitude (set aircraft altimeter to 29.92 in. hg.).....	_____ FT	_____ FT
(c) Using AFM, Performance Section, Takeoff Power Check Chart (DRY-BLEED AIR OFF); determine expected torque.....	_____%TOR	_____%TOR
(d) Using AFM, Performance Section, Takeoff Power Check Chart (DRY-BLEED AIR ON): determine expected torque.....	_____%TOR	_____%TOR
(e) Bleed air OFF. S/L HIGH RPM, advance P/L to 650°C EGT or expected torque (DRY-BLEED AIR OFF). Record if expected torque is reached.....	YES NO	YES NO
(f) Bleed air ON. Readjust P/L until MAX EGT or expected torque (DRY BLEED AIR ON). Record if expected torque is reached.....	YES NO	YES NO

STEP		LH ENGINE	RH ENGINE
(g)	Record engine instrument readings at maximum power. EGT (approx. 650°C).....	_____°C	_____°C
	RPM (100-101%).....	_____%RPM	_____%RPM
	Fuel Flow.....	_____PPH	_____PPH
	Oil Temperature (70-90°C).....	_____°C	_____°C
	Oil Pressure (70-120 psi).....	_____PSI	_____PSI
	Fuel Pressure (20-80 psi).....	_____PSI	_____PSI
	Deice Pressure (12-19 psi).....	_____PSI	_____PSI
	Suction (4.4-4.8 in.hg.).....	___in.hg.	___in.hg.
	(h) Retard P/L to G.I. and place S/L in low RPM.		
(16)	CONT ALCOHOL-WATER INJECTION (CAWI) CHECK		
(a)	CAWI Quantity Indicator Check		
1	Record quantity of CAWI tank indicated on sight gauge.....	_____GAL	_____GAL
2	Record quantity of CAWI tank indicated on cockpit indicator.....	_____GAL	_____GAL

NOTE: Indications must be the same plus or minus 25 gallons.

STEP		LH ENGINE	RH ENGINE
(b)	CAWI Pump Check		
1	AWI PUMP TEST switch to No. 1.		
2	Record if AWI #1 PUMP ON light on, and AWI #2 PUMP ON light off.....YES NO	YES NO	
3	AWI PUMP TEST switch to No. 2.		
4	Record if AWI #2 PUMP ON light on, and AWI #1 PUMP ON light off.....YES NO	YES NO	
5	Release AWI PUMP TEST switch.		
	NOTE: Both AWI #1 PUMP ON light and AWI #2 PUMP ON light extinguish when AWI PUMP TEST switch is released.		
(c)	CAWI Takeoff Power Check		
1	Record outside air temp (OAT)..... _____°C	_____°C	
	NOTE: Use calibrated thermometer outside aircraft, not aircraft indicator.		
2	Record pressure altitude (set aircraft altimeter to 29.92 in.hg.)..... _____FT	_____ FT	
3	using AFM, performance section Takeoff Power Check Chart (WET); determine expected torque..... _____%TOR	_____%TOR	
4	Bleed air ON. S/L HIGH RPM, advance P/L to approximately 30% torque.		
5	WATER INJECTION switch to CONT.		

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-10U-501 – MAINTENANCE PRACTICES

STEP		LH ENGINE	RH ENGINE
6	Record bleed air cut off.....	YES NO	YES NO
	AWI PUMP ON lights illuminate.....	YES NO	YES NO
	EGT rise (approx. 32°C).....	YES NO	YES NO
	Torque rise (approx. 25%).....	YES NO	YES NO
	CAUTION: DO NOT EXCEED 650°C EGT OR TORQUE LIMITS (100%).		
7	Advance both P/L to MAX EGT or expected torque (WET). Record if expected torque is reached.....	YES NO	YES NO
8	Retard P/L to approx. 30% torque.		
9	Pull TEMP LIMITER CB one at a time. Record water stops flowing (drop in EGT and torque).....	YES NO	YES NO
	Record AWI PUMP ON light extinguish on corresponding engine.....	YES NO	YES NO
	Reset CB, record water return (increase in EGT and torque).....	YES NO	YES NO
10	Retard both P/L to ground idle. Record bleed air come on.....	YES NO	YES NO
11	Retard S/L to LOW RPM. Record AWI PUMP ON light extinguish as engine RPM drops below 90%.....	YES NO	YES NO
12	WATER INJECTION switch to OFF.		

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-10U-501 – MAINTENANCE PRACTICE

STEP		LH ENGINE	RH ENGINE
(17) IGNITION MODE CHECK			
(a)	Momentarily select continuous ignition. Record if ignition light illuminates.....	YES NO	YES NO
(b)	Momentarily select override ignition. Record if ignition light illuminates.....	YES NO	YES NO
(c)	Return ignition switch to normal.		
(18) ENGINE INTAKE ANTI-ICE SYSTEM CHECK			
CAUTION:	DO NOT OPERATE ENGINE INTAKE ANTI-ICE SYSTEM MORE THAN TEN SECONDS IF OAT IS ABOVE FIVE DEGREES C.		
(a)	S/L in HIGH RPM.		
(b)	P/L in ground idle (MIN fuel flow).		
(c)	ENGINE & PROP HEAT switch to upper ENG & PROP position.		
(d)	Record if EGT increases.....	YES NO	YES NO
	Record if INTAKE HEAT ON light illuminates for appropriate engine.....	YES NO	YES NO
(e)	ENGINE & PROP HEAT switch to OFF.		
(f)	Record if EGT decreases.....	YES NO	YES NO
	Record if INTAKE HEAT ON light extinguishes for appropriate engine.....	YES NO	YES NO

STEP	LH ENGINE	RH ENGINE

(19) ENGINE SHUTDOWN-PROP UNFEATHER CHECK

NOTE: If power runs have been made, allow three minutes to cool down. Charge EPA kit by maintaining 96% RPM momentarily.

(a) Emergency Shutdown (Stop and Feather)

1 S/L in LOW RPM.

2 Advance P/L to 75% RPM.

CAUTION: IF ANY RISE IN TORQUE OR EGT IS NOTED DURING NEXT STEP, PUSH STOP BUTTON IMMEDIATELY.

3 Pull STOP AND FEATHER knob until red groove in cable shaft is visible atpedestal cover.
Record engine flames out,
but that propeller does not feather..... YES NO YES NO

CAUTION: DISCHARGE EPA KIT BEFORE ENGINE RPM IS BELOW 40% OR FUEL ACCUMULATION IN PLENUM MAY OCCUR. IF NOT DISCHARGED BEFORE ENGINE STOPS, ALLOW ENGINE TO COOL BEFORE DISCHARGING EPA KIT.

4 Depress STOP button at least five SEC.

5 At 40% RPM pull STOP AND FEATHER full aft, record if propeller feathers..... YES NO YES NO

ENGINE GROUND OPERATION (WORKSHEET) – TPE331-10U-501 – MAINTENANCE PRACTICES

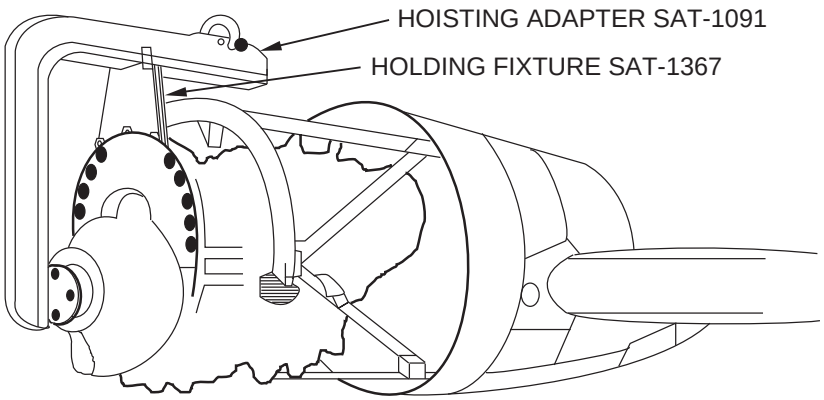
STEP		LH ENGINE	RH ENGINE
(b)	Propeller Unfeathering Check		
1	Hold P/L in full reverse.		
2	UNFEATHER TEST switch to L or R. Record if propeller blades drive toward reverse position.....	YES NO	YES NO
3	When propeller blades reach full reverse, release UNFEATHER TEST switch and move P/L to 1/4 inch forward of flight idle. 4 Record if propeller returns to start locks.....	YES NO	YES NO
(c)	Normal Engine Shutdown		
	NOTE: Restart engine in accordance with AIRCRAFT FLIGHT MANUAL procedures. Charge EPA kit by maintaining 96% RPM momentarily.		
1	S/L in LOW RPM and P/L in ground idle (MIN fuel flow).		
2	Depress STOP for at least five SEC.		
3	Record if RPM increases approx. five percent as EPA kit discharges.....	YES NO	YES NO
4	Allow RPM to decrease to 50% then retard P/L to full reverse, hold until RPM decreases to 10%. Record if propeller is on star locks.....	YES NO	YES NO

MAINTENANCE MANUAL

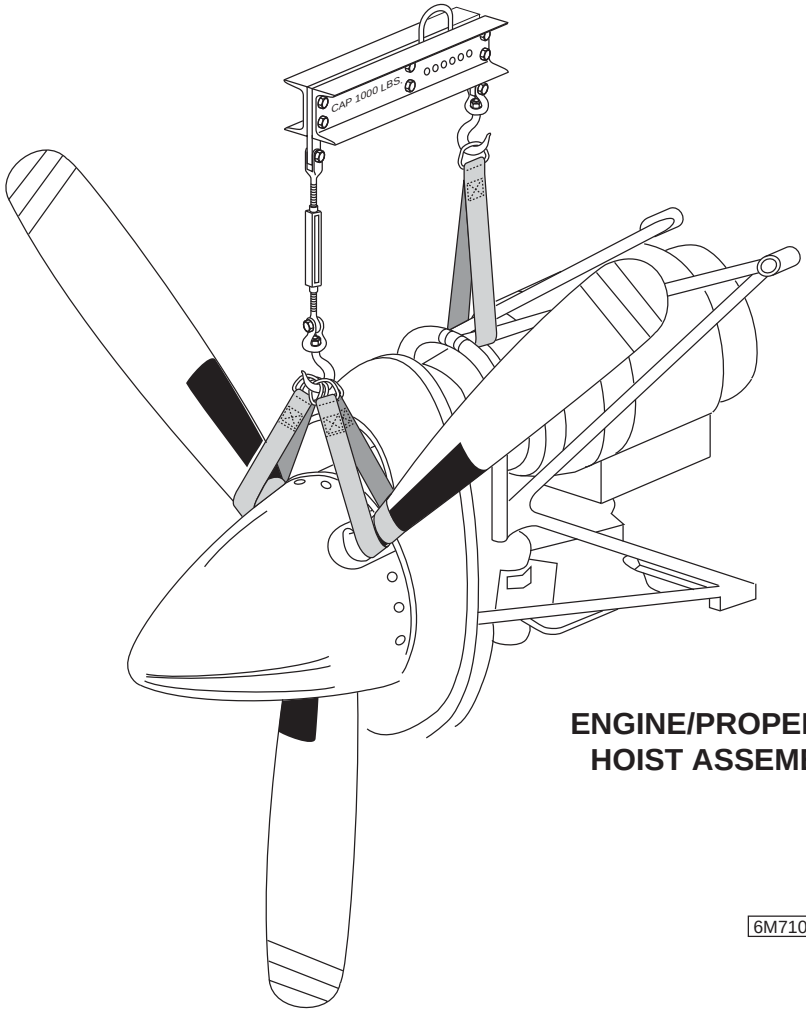
-3 AND -10 ENGINE CHANGE – DESCRIPTION AND OPERATION

1. General

A. The following maintenance practices detail removal and installation procedures for TPE331-3U-303G, TPE331-3U-304G, and TPE331-10U-501 Garrett AiResearch engines on SA226 series aircraft. The practices include the removal and installation procedures for both the quick engine change (Q.E.C.) and regular, or bare, engine change. Quick engine change implies that an engine is previously built-up with M7 Aerospace component parts and is removed and installed with all component parts, including the forward nose cowling section; whereas, regular engine change involves the building up of the engine prior to its installation.



ENGINE HOIST ASSEMBLY



ENGINE/PROPELLER HOIST ASSEMBLY

6M710030201

Engine Hoisting
Figure 201

1. Removal/nstallation -3 and -10 Engine Change

A. It is recommended that engine changes be accomplished by removing the engine and mount from the aircraft firewall attach points. Preliminary steps should be taken to facilitate engine removal and protect the aircraft.

CAUTION: WITH BOTH ENGINES REMOVED FROM THE AIRCRAFT, IT IS NECESSARY THAT 300 POUNDS OF COMPACT BALLAST BE PLACED IN THE FORWARD BAGGAGE COMPARTMENT TO PREVENT THE TAIL FROM SETTLING TO THE RAMP. WITH ONLY ONE ENGINE REMOVED, THE AIRCRAFT MAY BE TOWED WITHOUT BALLAST IF DONE CAUTIOUSLY. IT IS ADVISABLE TO INSTALL A TAIL STAND WHILE THE AIRCRAFT IS PARKED WITH ONE ENGINE REMOVED. AN ALTERNATE METHOD IS TO INSTALL A CRADLE UNDER THE AFT FUSELAGE WHEN THE ENGINES ARE REMOVED.

(1) Install a tail stand or cradle under the aft fuselage; or add 300 pounds of ballast to the forward baggage compartment.

(2) Drain the engine oil.

(3) Ensure all electrical power to the aircraft is disconnected.

B. Removal – Engine

(1) Remove top, side, and lower aft engine cowling, see Section 71-20-00.

(2) Remove propeller, see Maintenance Manual, Chapter 61.

(3) On regular engine change only, remove nose cowling section, air scoop assembly, and cowling support rings, see Section 71-20-00.

(4) Remove aspirator access cover and disconnect engine bleed air tube.

(5) Disconnect all electrical leads at the firewall or the engine and remove any harness clamps necessary to permit engine removal. Cap all connectors to prevent contamination, see Section 71-50-00, Electrical Harness.

(6) Remove starter-generator and ducts, see Maintenance Manual, Chapter 24.

(7) Disconnect engine drain hoses and tubing at drain block, see Section 71-70-00.

(8) On regular engine change only, remove engine torque transmitter/transducer, see Maintenance Manual, Chapter 77.

(9) Disconnect the engine power, RPM, and fuel shutoff (stop and feather) control cables from their respective terminal points.

NOTE: Remove these control cables from their bracket attach points by removing the forward jam nuts and withdrawing the cable rearward from their attach points. Do not disturb setting of rear jam nuts to minimize rigging procedures upon reinstallation of engine.

(10) Disconnect the three hoses (inlet, outlet, and drain) to the hydraulic pump from the firewall attach point.

(11) Disconnect the three fuel hoses at the firewall attach point.

(12) Disconnect the four oil hoses from the oil tank and the oil hose to the oil transmitter at the firewall.

(13) Disconnect AWI (Alcohol/Water Injection) hoses at firewall, applicable aircraft.

(14) Cap all lines and hoses to prevent contamination.

(15) For regular engine change only, remove all M7 Aerospace accessories, hoses, harnesses, brackets, and clamps for installation of these components onto new engine. See Section 71-20-00 (Engine Mounts), 71-30-00 (Fireseals), 71-40-00 (Attach Fittings), 71-50-00 (Electrical Harness), 71-70-00 (Engine Drain).

(16) Attach the hoisting sling to the engine, see Figure 201.

(17) Refer to Figure 201 for hoisting options and 71-SERV Figures 1 and 2 for part numbers and component specifications for Engine/Propeller hoist assembly.

(18) Position A frame or hoist, above the sling. Attach lifting tackle and take up any slack.

(19) Remove lower firewall mounting bolts from engine mount first, then the upper bolts, while an assistant carefully transfers engine weight to hoisting cable. Remove mounting bolts.

(20) Move engine away from the aircraft carefully, avoiding damage to the tail cone or exhaust pipe.

(21) On regular engine change only:

(a) Remove lower engine mount truss assembly.

(b) Lower engine onto shipping container. Remove upper engine mount truss assembly. See Garrett Turbine Engine company Maintenance Manual for shipping container data.

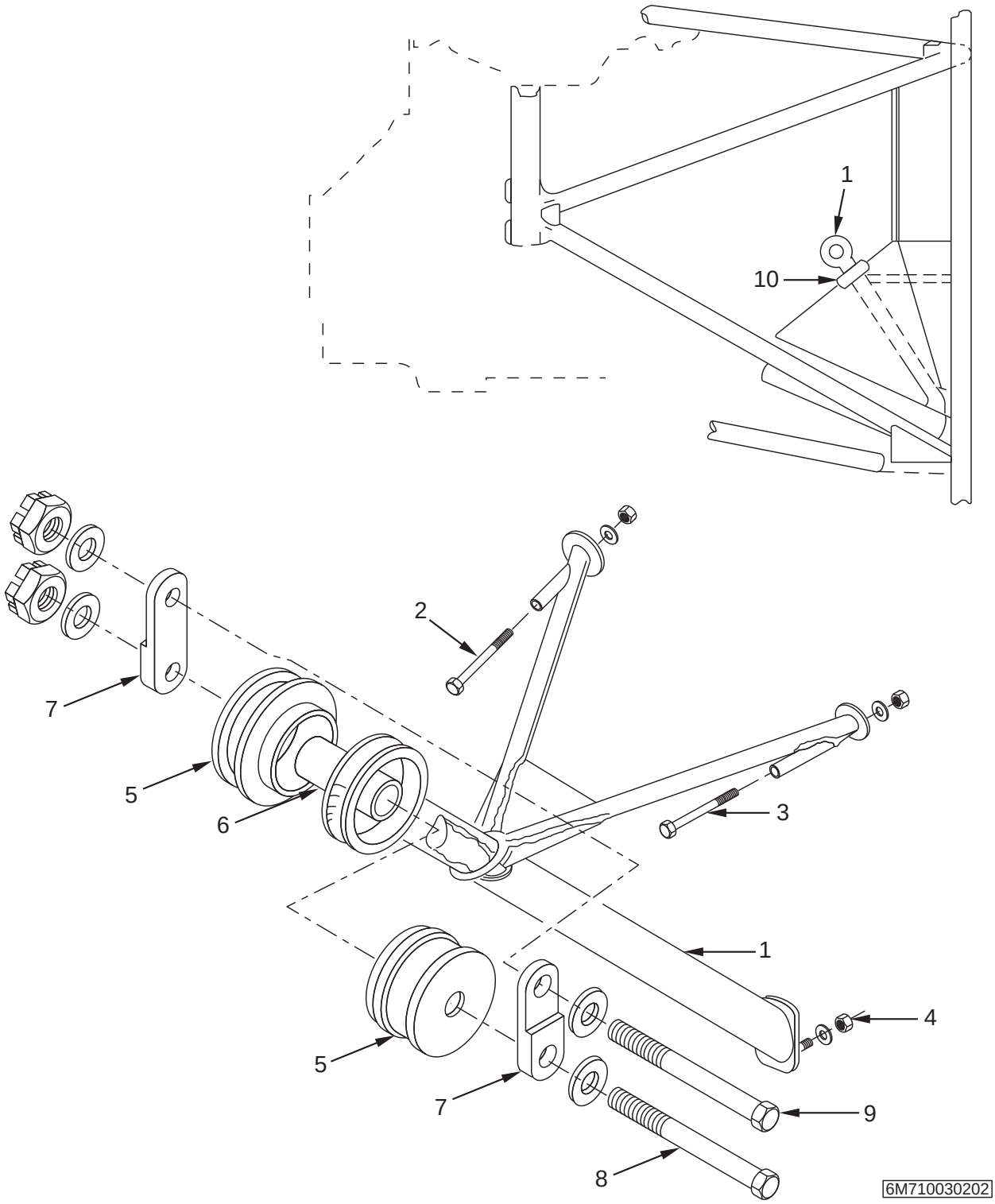
C. Installation – Engine (Reference Figures 201 and 202)

(1) Attach engine sling, and hoist engine.

(2) Swing engine, less the starter-generator, into position while inserting tail cone into the exhaust pipe.

CAUTION: WHILE POSITIONING ENGINE FOR INSTALLATION, GUIDE SPEED LEVER CONTROL CABLE THROUGH THE INSIDE OF ENGINE MOUNT TRUSS MEMBER AND HOSE ASSEMBLIES TO PREVENT KINKING OR DAMAGE TO CONTROL CABLE.

(3) Align upper mounting holes to firewall holes, and insert the mounting bolts (MS20006-12 bolt, NAS143-6C washer, AN960-616 washer, and 12NB-064 nut at four places).



Engine Mounts
Figure 202

NOTE: Anytime mount bolts are removed they must be replaced.

Install washers (with countersunk side of washer facing head of bolt) onto mount bolt prior to installing bolts. When installing engine mount bolts, install all four bolts, washers, and nuts. Torque top bolts Take weight of engine off hoist, then torque bottom bolts.

(4) Install washer and locknut to upper mount bolts and tighten.

NOTE: To facilitate easier installation of mount bolt locknuts, apply gluing substance to washers. Glue washer onto each locknut prior to their installation.

Access for installation of mount bolt locknuts to the backside of the firewall is gained through the wheel well.

(5) Install lower mount bolts, washers, and locknuts.

NOTE: It may be necessary to carefully slacken the hoisting cable to insert the lower mounting bolts.

(6) Torque engine mount bolts from bolt side 350 to 390 inch-pounds.

NOTE: It is recommended that a 5/16 inch hex stock extension be used to torque the mount bolts and a 1/2 inch hex stock extension be used to retain the mount bolt nuts while torquing.

(7) On SA226 series T(B) aircraft only, install rear engine mount as shown in Figure 202. Tighten nuts on mount bolts (items 8 and 9) securely, and install cotter pins into castellated nuts. See Illustrated Parts Catalog, Section 71-22-00 for engine mount and component part numbers.

(8) Check for a minimum clearance of 0.25 inch between the tailcone and the exhaust pipe. Rotate the tailcone or realign the engine to obtain minimum clearance.

(9) Remove hoist and sling from engine.

D. Installation – Engine Oil Hoses (Reference Figure 203)

(1) Install hose assembly (item 1) onto union (item 2) on oil tank. Torque 210 to 240 inch-pounds.

NOTE: Elbow fitting at opposite end of hose assembly (item 1) is installed onto oil cooler manifold.

(2) Install hose assembly (item 3) to oil cooler manifold (90° elbow fitting to manifold). Torque 210 to 240 inch-pounds.

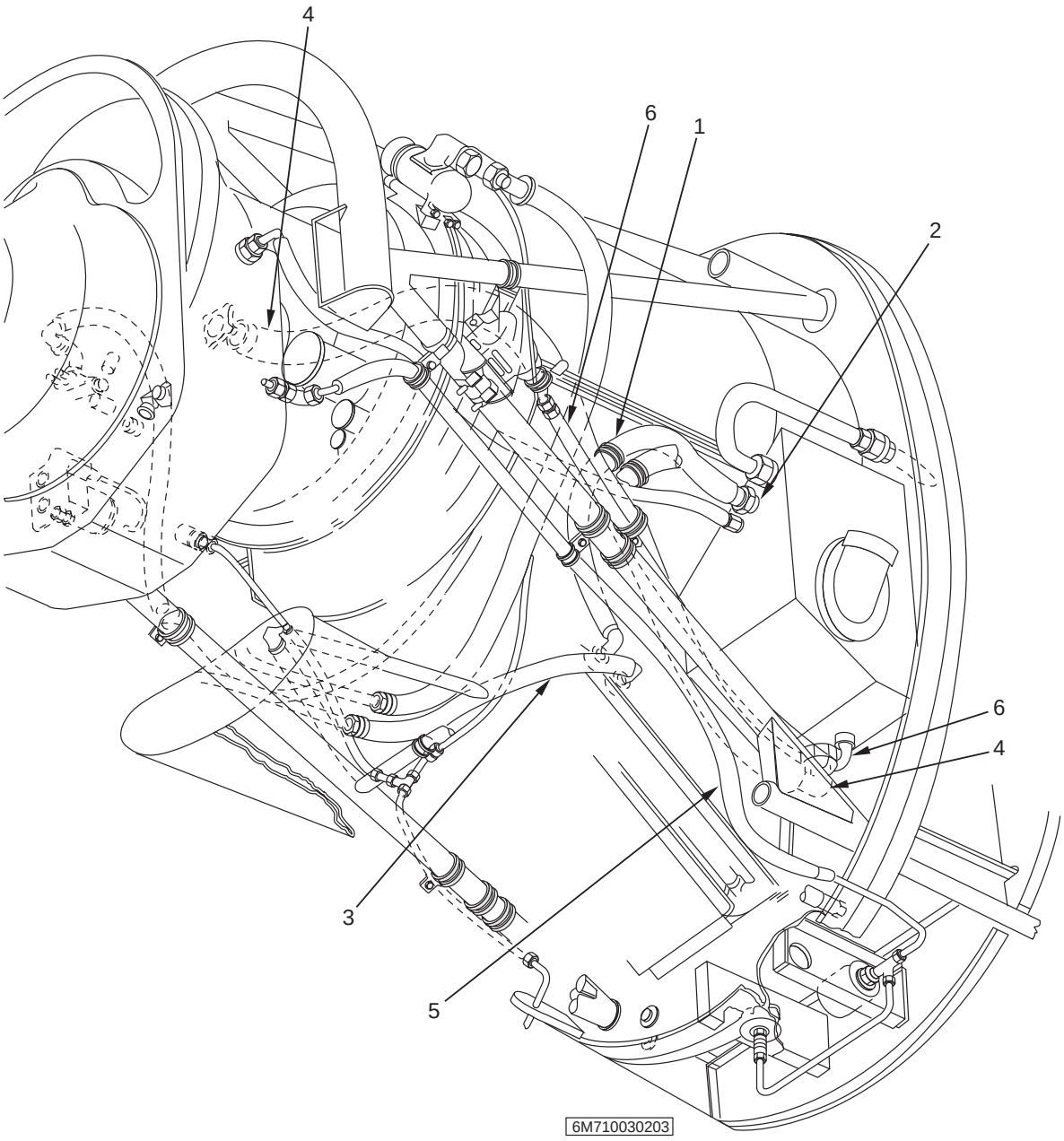
CAUTION: ROUTE HOSE SUCH THAT IT DOES NOT INTERFERE WITH RPM LEVER CONTROL CABLE.

(3) Install main oil supply hose assembly (item 4) to oil tank. Torque 700 to 1,150 inch-pounds.

(4) Install hose assembly (item 5) from gear case oil pressure port to fitting on firewall. Torque 70 to 120 inch-pounds.

(5) Install hose assembly (item 6) from unfeather pump to fitting on bottom of oil tank. Torque to 100 to 250 inch-pounds.

NOTE: On aircraft using Alcohol/Water Injection (AWI), install AWI hoses.



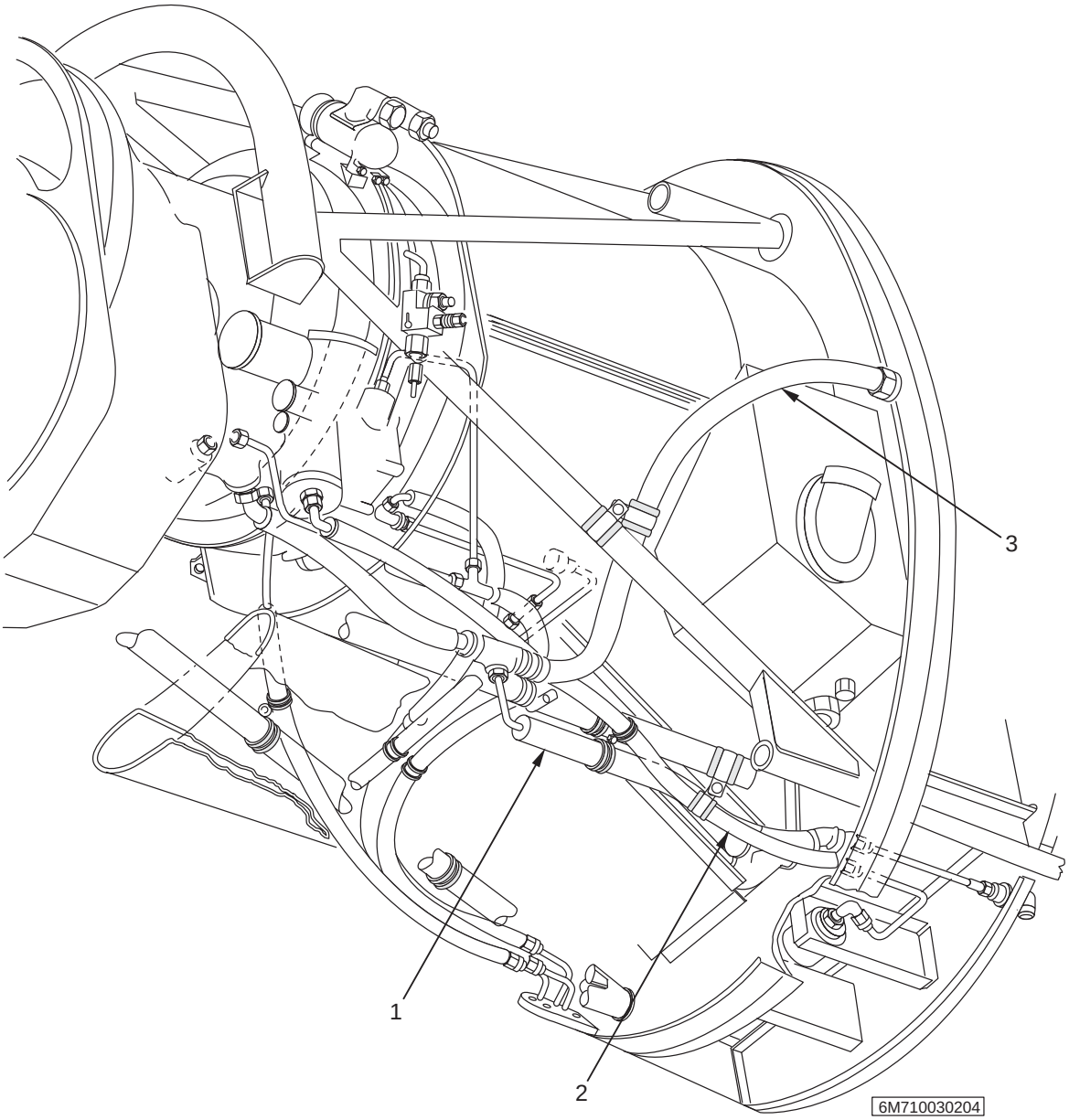
Oil Hoses
Figure 203

E. Installation – Engine Fuel Hoses (Reference Figure 204)

(1) Install fuel drain hose assembly (item 1) from tee fitting to 45° elbow fitting on firewall. Torque 70 to 100 inch-pounds.

(2) Install fuel hose assembly (item 2) from fuel filter bowl assembly to fitting at the firewall. Torque 70 to 120 inch-pounds.

(3) Install fuel hose assembly (item 3) to fitting at firewall. Torque 210 to 240 inch-pounds.

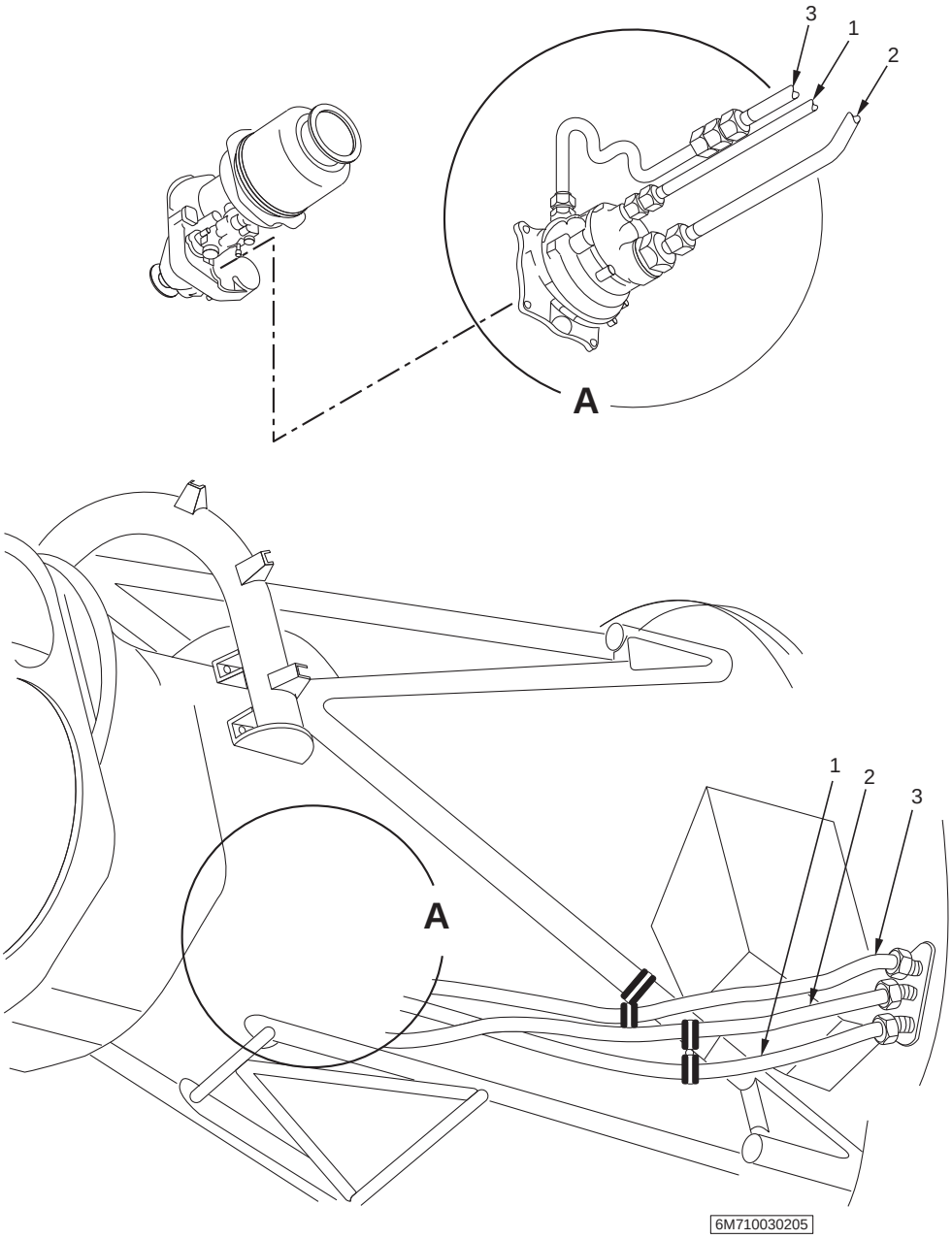


Fuel Hoses
Figure 204

F. Installation – Engine Hydraulic Hoses (Reference Figure 205)

(1) Install hose assembly (item 1) from outlet of hydraulic pump to lower (cluster of three) 45° fitting on firewall. Torque 100 to 250 inch-pounds.

(2) Install hose assembly (item 2) from inlet of hydraulic pump to middle (cluster of three) 45° fitting on firewall. Torque 210 to 240 inch-pounds.

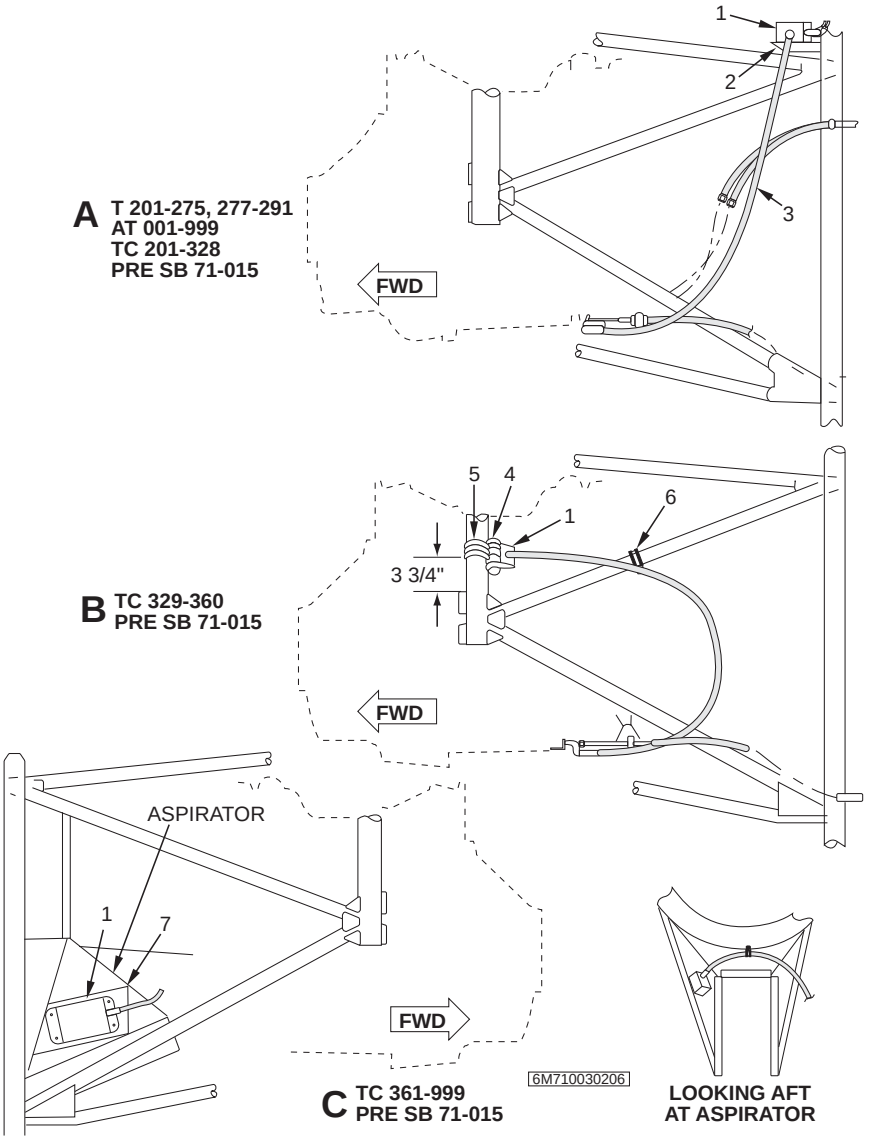


Engine Hydraulic Hoses
Figure 205

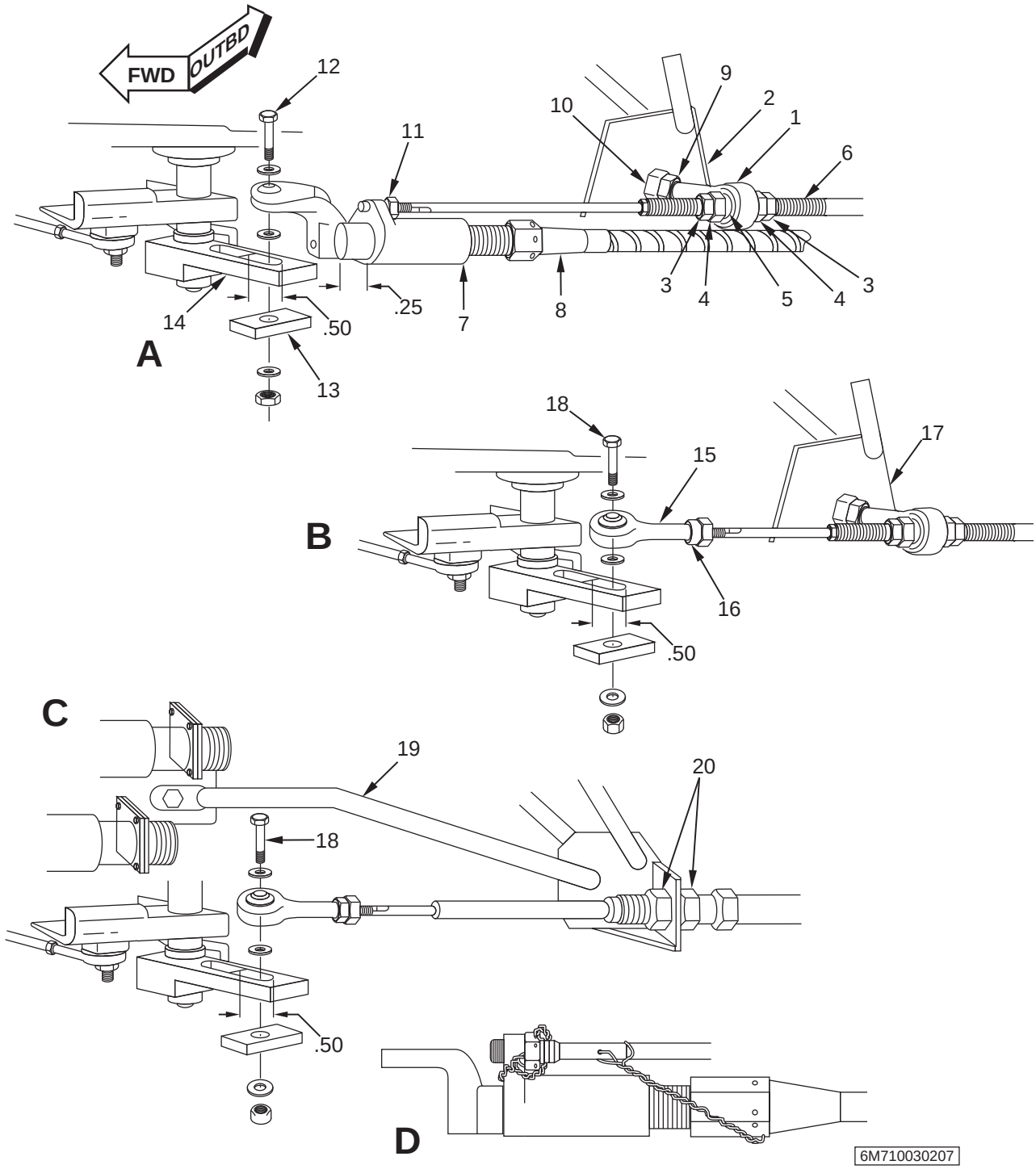
(3) Install hose assembly (item 3) from hydraulic pump bypass to upper (cluster of three) fitting on firewall. Torque 70 to 120 inch-pounds.

NOTE: Certain SA226 series aircraft have incorporated a propeller synchronizer actuator, mounted on the right engine only. See Figure 206 for production installation location and respective applicable effectivities of the synchronizer actuator and mounting bracket. For installation instructions of propeller synchronizer actuator to the right side of the aspirator as shown in Figure 206, inset C, refer to M7 Aerospace Service Bulletin 71-015, Heat Shroud Removal.

See Illustrated Parts Catalog, Section 76-10-03 for RPM control and 76-10-04 for power control installation part numbers.



Engine Propeller Synchronizer and RPM Control, RH Engine
Figure 206



RPM Control
Figure 207

G. Installation of Heim Bearing Rod-End (Left and Right Engines, T201-383, AT001-069, and TC201-384) reference Figure 207.

(1) Remove Heim bearing rod-end (item 1) from bracket assembly in order to install it onto RPM lever control cable (item 6) and o avoid kinking of the control cable during installation.

(2) Install Pal nut (item 3), jam nut (item 4), then spacer (item 5) onto control cable (item 6).

NOTE: Pal nut, jam nut, and spacer is installed on both sides of Heim bearing rod-end typically.

(3) Install Heim bearing rod-end (item 1) onto the RPM lever control cable (item 6) then install spacer (item 5), nut (item 4), and Pal nut (item 3).

NOTE Leave Pal nuts and jam nuts on each side of Heim bearing rod-end loose until rigging is accomplished (see Rigging Procedures, Chapter 76).

(4) Reinstall Heim bearing rod-end and control cable to bracket assembly (item 2).

NOTE: Nuts and Pal nuts (two each) that are identical to those used to secure the Heim bearing rod-end to the control cable are also used to secure the Heim bearing rod-end to the bracket assembly.

(5) Apply Locktite 242 (Locktite Corp. - Newington, Conn. 06111) onto threads of portion of RPM lever control cable (item 6), which slides over bayonet, and then threads into housing of RPM lever cable assembly.

NOTE: Tighten snuggly using 3/8" open-end wrench on flat surface of RPM lever cable portion which slides over bayonet, and use 1/2" open-end wrench on flat surface of cable housing.

H. Installation of Propeller Synchronizer Actuator Rod-End on Right Engine Only (T202-275, T277-291, AT001-999, and TC201-999) Reference Figure 207, Inset A and D

(1) Adjust propeller synchronizer shaft rod-end so that the actuator of the rod end (item 7) extends .25 as shown in illustration.

NOTE: The rod-end can be extended by connecting the synchronizer shaft housing (item 8) to the splines within the rod-end splines, so as not to disturb the rod-end setting until final installation of shaft is accomplished.

(2) With synchronizer shaft housing disconnected, adjust propeller synchronizer actuator (item 1 Figure 206, Insert C) by turning shaft within actuator fully clockwise until shaft stops turning, then turn shaft counterclockwise 27 graduation (audible clicks).

(3) Propeller Synchronizer Actuator Installation (reference Figure 206):

(a) At top of aspirator (see inset A), install propeller synchronizer actuator (item 1) to top of aspirator just forward of firewall and secure it to mounting (item 2) using four screws and washers, see Illustrated Parts Catalog, Section 76-10-03 for item part numbers.

(b) At engine truss member (see inset B) install propeller synchronizer actuator (item 1) onto engine truss member, 3-3/4 inches from top of left engine mount. Install using bracket assembly, clamps, screws, and washers (item 4) see Illustrated Parts Catalog, Section 76-10-03 for item part numbers.

(c) At right side of aspirator (inset C), install propeller synchronizer actuator onto doubler attached to side of aspirator with four screws and washers, see Illustrated Parts Catalog, Section 76-10-03 for item part numbers.

NOTE: See M7 Aerospace Service Bulletin 71-015 for installation procedures of doubler (item 7) to side of aspirator.

(4) Install jam nut (item 11) onto RPM lever cable bayonet; then install propeller synchronizer rod-end (item 7, Figure 207) onto bayonet for approximately three threads of bayonet extending beyond rod-end. Safety jam nut with .025 safety wire as shown in inset D.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

(5) Install rod-end (item 7) onto fuel control RPM lever arm (item 14). Install bolt, three washers (item 12), nut, and plate (item 13).

NOTE: When installing mounting bolt (item 12), center the rod-end (item 7) onto the RPM lever arm (item 14) such that the distance from center of mounting bolt to end of RPM lever arm is .50 inches. Ensure that the serrations of the plate (item 13) align with the serrations on the bottom of the RPM lever arm (item 14).

CAUTION: USE A NEW LOCKNUT FOR BOLT (ITEM 12) EACH TIME THE NUT IS REMOVED.

(6) Propeller Synchronizer Actuator Cable Shaft Installation:

(a) Install propeller synchronizer actuator shaft (item 8) by installing into synchronizer actuator first.

NOTE: Avoid disturbing adjustment of actuator while connecting shaft.

(b) Install opposite end of shaft to synchronizer actuator rod-end.

(c) Safety both fittings at each end of synchronizer actuator shaft housing with .025 safety wire (see Figure 207, inset D).

(7) Install applicable clamps (as shown in Figure 206, items 3 through 6), see Illustrated Parts Catalog, Section 76-10-03 for item part numbers.

(8) Connect connector and adapter (cannon plug) to propeller synchronizer actuator and safety connector with .025 safety wire.

I. Installation of RPM Lever Rod-End on Right Engines (T276, T292-383) and Left Engine (T201-383, AT001-069, AT071-999, and TC201-384) Reference Figure 207

(1) Install jam nut and rod-end locking washer (item 16) onto RPM lever control bayonet. Install rod-end (item 15) onto bayonet.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

(2) Safety rod-end (item 15) to locking washer (item 16) with .025 safety wire.

(3) Install rod-end (item 15) onto fuel control RPM lever (item 14). Install bolt (item 18), three washers, nuts, and plate (item 13).

CAUTION: USE A NEW LOCKNUT FOR BOLT EACH TIME THE LOCKNUT IS REMOVED, SEE ILLUSTRATED PARTS CATALOG, SECTION 76-10-03 FOR ITEM PART NUMBERS.

(4) Operate RPM lever control from the cockpit and accomplish general rigging requirements, see Maintenance Manual, Chapter 76.

J. Installation of RPM Control Cable and Control Cable Rod-End on Left Engine (T384-999, AT070, and TC385-999), and Right Engine (T384-999).

NOTE: The above effectivities reflect latest installation of multiple piece control cables and attaching bracket for installation procedures of the propeller synchronizer actuator (inset D) on right engines for AT070 and TC385-999.

(1) Install the RPM control cable segment connecting it to the wing RPM cable segment at the engine firewall.

NOTE: It will be necessary to advance the RPM cable forward from the cockpit pedestal RPM control. Remove the clevis pin attaching the PRM control cable to the RPM lever. Push the cable forward so that the cable will advance and protrude beyond the engine firewall to facilitate connection.

(2) Connect the RPM control cable segment at the engine firewall. Safety the firewall jam nut and engine RPM control cable nut with MS333540 safety wire.

(3) Install engine RPM control cable through bracket assembly (item 19).

(4) Safety jam nuts (item 20) with MS33540 safety wire.

(5) Install rod-end (item 15) onto control cable.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation. Ensure tab on lockwasher (item 16) mates with grooves in rod-end.

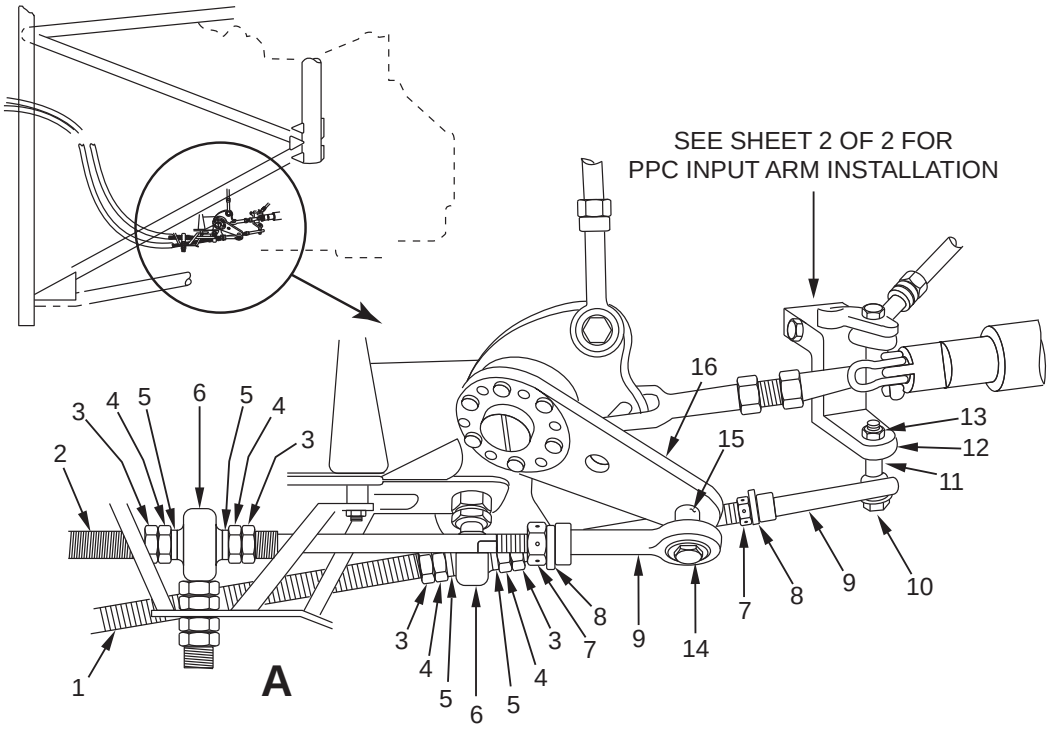
(6) Safety rod-end to lockwasher with MS35540 safety wire.

(7) Install rod-end (item 15) onto fuel control RPM lever arm (item 17). Install bolt (item 18), three washers, nut, and plate (item 13).

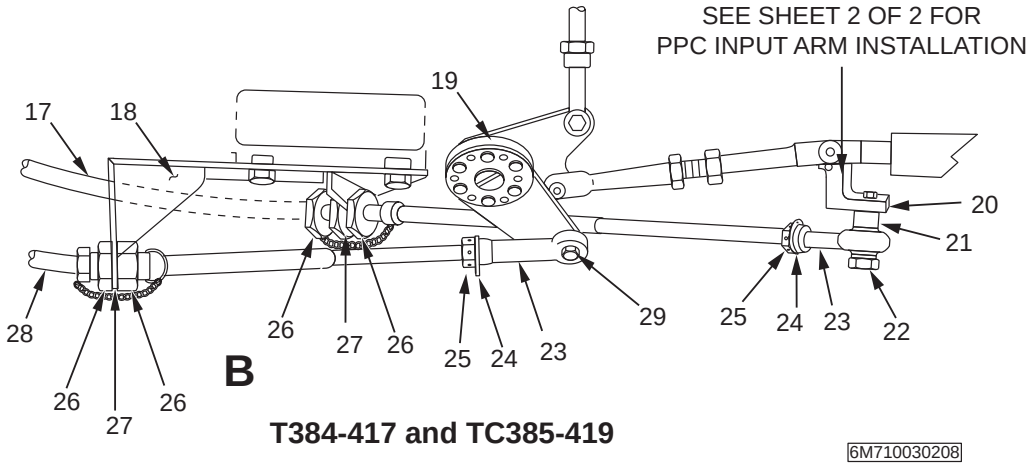
NOTE: When installing mounting bolt, center the rod-end onto the RPM lever arm (item 17) such that the distance from center of mounting bolt (item 18) to end of RPM lever arm (item 14) is .50 inches. Ensure that the serration of the plate (item 13) align with the serrations on the bottom of the RPM lever arm.

CAUTION: USE A NEW LOCKNUT FOR BOLT EACH TIME THE NUT IS REMOVED.

(8) Install RPM cable lever. Operate RPM lever control from the cockpit and accomplish general rigging requirement, see Maintenance Manual, Chapter 76.



T201-383, AT001-074 and TC201-384



T384-417 and TC385-419

6M710030208

Power Control and Fuel Shutoff Control
Figure 208 (Sheet 1 of 2)

K. Installation of Engine Compartment Power Control Cable and Fuel Shutoff Control Cable (T201-383, AT001-074 and TC201-384) (Refer to Figure 208, Sheet 1 of 2, Insert A):

(1) Apply Locktite 242 (Locktite Corp., Newington, Conn. 06111) onto threads of power control cable (item 1) and fuel shutoff control cable (item 2) which slides over bayonet, then threads into housing of each control cable assembly.

(2) Tighten snuggly by using 3/8" open-end wrench on flat surface of each control cable portion which slides over bayonet, and use 1/2" open-end wrench on flat surface of each cable housing.

(3) Install Pal nut (item 3), jam nut (item 4), and spacer (item 5) onto each control cable (items 1 and 2), and install each control cable through their respective Heim bearing rod-ends (item 6). Install spacer, jam nut, and Pal nut onto each control cable on opposite side of each Heim bearing rod-end.

NOTE: Leave Pal nuts and jam nuts on each side of Heim bearing rod-ends on each control cable loose until rigging is accomplished.

NOTE: On power control cable (item 1), install lockwasher (item 8) so that it faces upwards for safetying purposes. On fuel shutoff control cable (item 2), install lockwasher so it faces downward for safetying purposes.

(4) Install jam nut (item 7), then rod-end lockwasher (item 8) onto each control cable bayonet.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation. Ensure tab on lockwashers mate with grooves in rod-ends.

(5) Install rod-end (item 9) onto bayonet of each control cable.

(6) If necessary, install prop pitch control (PPC) input arm onto PPC shaft as follows (Refer to Figure 208, Sheet 2 of 2 for this step (6) only):

(a) For engines with 27-19216-005 tachometer generator installation, refer to Illustrated Parts Catalog 76-10-05 and/or Service Note 226-SN-077 for trimmed PPC rig plate (item 6) to ensure minimum clearance between tachometer generator and rig plate.

(b) With nut (item 5) and cinch bolt (item 4) loose and pointer (item 7) on PPC at 40º on protractor, position the PPC input arm (item 1) onto PPC shaft splines so the index mark on PPC input arm is aligned with center of rig pin hole in PPC pointer (Refer to 76-00-10, page 201 in this manual for detailed power control cable rigging instructions).

(c) For Post SB 226-76-012 PPC with internally threaded shaft, secure PPC input arm (item 1) to PPC shaft with spacer (item 3) and screw (item 2). Torque screw 20 to 25 inch-pounds (2.26 to 2.82 Nm).

(d) Torque nut (item 5) on cinch bolt (item 4) 20 to 25 inch-pounds (2.26 to 2.82 Nm).

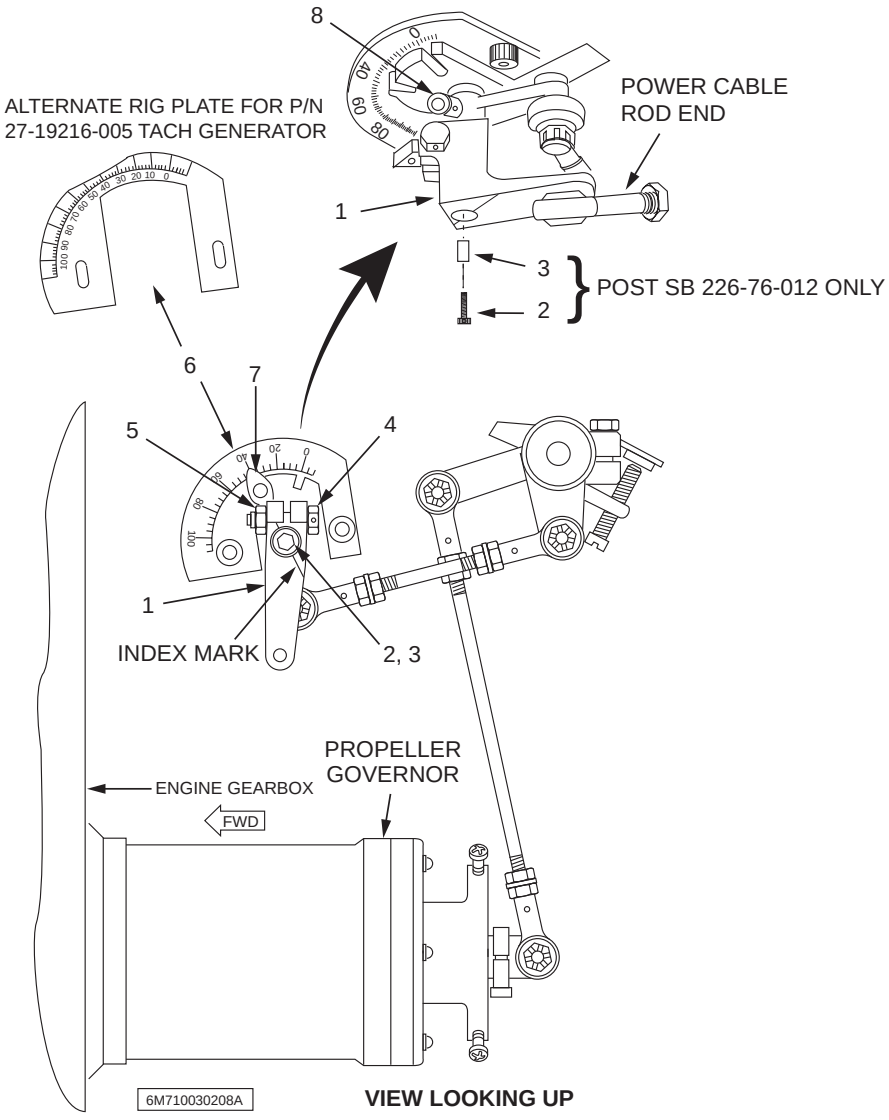
(e) For Pre SB 226-76-012 PPC, safety wire cinch bolt (item 4) to PPC pointer bolt (item 8) IAW MS33540.

(f) For Post SB 226-76-012 PPC, safety wire screw (item 2) to cinch bolt (item 4) IAW MS33540.

(7) Safety wire rod-ends (item 9) to lockwashers (item 8) IAW MS33540.

WARNING: USE A NEW SELF-LOCKING NUT EACH TIME THE POWER CONTROL CABLE IS ATTACHED TO PPC INPUT ARM.

(8) On the power lever control cable (item 1), install bolt (item 10) upwards through rod-end (item 9), install spacer (item 11) onto bolt; then, install to propeller pitch control power lever arm (item 12). Install washer and locknut (item 13).



Power Control and Fuel Shutoff Control
Figure 208 (Sheet 2 of 2)

WARNING: USE A NEW SELF-LOCKING NUT EACH TIME THE FUEL SHUTOFF CONTROL CABLE IS ATTACHED TO BELLCRANK.

(9) On the fuel shutoff control cable (item 2), install bolt (item 14) upwards through rod-end (item 9), install spacer (item 15) onto bolt; then, install bolt (item 14) to bellcrank (item 16). Install washer and locknut.

(10) Operate power lever and fuel shutoff controls from the cockpit and accomplish general rigging adjustments (see Maintenance Manual, Chapter 76-00-00).

L. Installation of Engine Compartment Power Control Cable and Fuel Shutoff Control Cable (T384-417 and TC385-419, Multiple Piece Control Cables) (Refer to Figure 208, Sheet 1 of 2):

(1) Install the engine fuel shutoff (FSO) cable segment (item 28) by connecting it to the wing fuel shutoff cable at the engine firewall.

NOTE: It will be necessary to advance the cable forward from the cockpit pedestal fuel shutoff control. Remove the knob from the pedestal fuel shutoff control, and with a rod, push the fuel shutoff control cable forward so that the cable will advance and protrude beyond the engine firewall to facilitate connection.

(2) Connect the FSO control cable segment at the engine firewall. Safety wire the firewall jam nut and engine fuel shutoff control cable nut IAW MS35540.

NOTE: Flanged bushing is used on Controlex cables only.

(3) Install engine FSO control cable through bracket assembly (item 18) up to aft jam nut (item 26). Install flanged bushing (item 27) and forward jam nut (item 26).

(4) Tighten and safety wire jam nuts (item 26) IAW MS35540.

(5) Install jam nut (item 25) onto FSO control cable.

(6) Ensure tab on lockwasher (item 24) mates with grooves in FSO cable rod-end (item 23).

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

(7) Install rod-end (item 23) onto FSO control cable.

(8) Safety wire FSO cable rod-end to lockwasher (item 24) IAW MS33540.

WARNING: USE A NEW SELF-LOCKING NUT EACH TIME THE FUEL SHUTOFF CONTROL CABLE IS ATTACHED TO BELLCRANK.

(9) Install bolt (item 29) upwards through FSO control cable rod-end; install spacer onto bolt; then, install bolt into bellcrank (item 19). Secure bolt with washer and locknut.

(10) Install the engine power cable segment (item 17) by connecting it to the wing power cable segment at the engine firewall.

NOTE: It will be necessary to advance the cable forward from the cockpit pedestal power control. Pull power lever to full reverse so that the power cable will advance and protrude beyond the engine firewall to facilitate connection.

(11) Connect the power control cable segment at the engine firewall. Safety wire the firewall jam nut and engine power control cable nut IAW MS35540.

(12) Install power control cable through bracket assembly (item 18) up to aft jam nut (item 26).

(13) Install flanged bushing (item 27) and forward jam nut (item 26).

NOTE: Flanged bushing is used on Controlex cables only.

(14) Tighten and safety wire jam nuts (item 26) IAW MS33540.

(15) Ensure tab on lockwasher (item 24) mates with grooves in power cable rod-end (item 23).

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

(16) Install rod-end (item 23) onto power control cable.

(17) Safety wire power cable rod-end to lockwasher (item 24) IAW MS33540.

(18) If necessary, install prop pitch control (PPC) input arm onto PPC shaft IAW step 1.K.(6) in this section.

WARNING: USE A NEW SELF-LOCKING NUT EACH TIME THE POWER CONTROL CABLE IS ATTACHED TO PPC INPUT ARM.

(19) Install bolt (item 22) upwards through power control rod-end; install spacer (item 21) onto bolt (item 22) then install to propeller pitch control power lever arm (item 20). Install washer and locknut.

(20) Operate power lever and fuel shutoff controls from the cockpit and accomplish general rigging adjustments (see Maintenance Manual, Chapter 76-00-00).

M. Installation – Engine Torque Transmitter Assembly (Reference Figure 209, T201-237, AT001-011, and TC201-207)

(1) Position transmitter (item 5) in nacelle cutout, aft of firewall, with threaded boss of transmitter through hole in frame.

(2) Secure transmitter with washer and nut. Torque nut 500 to 850 inch-pounds.

(3) Install two elbow fittings into transmitter, one in each end.

(4) Install tube to inlet and outlet ports of transmitter. Torque end fittings 40-65 inch-pounds.

(5) Connect inlet and outlet tubes to firewall unions and torque 40-65 inch-pounds.

(6) Install hoses (item 4) at firewall unions and torque hose end fittings 70-120 inch-pounds.

(7) Connect hoses (item 4) to engine adapter fitting (item 1) and torque 70-120 inch-pounds.

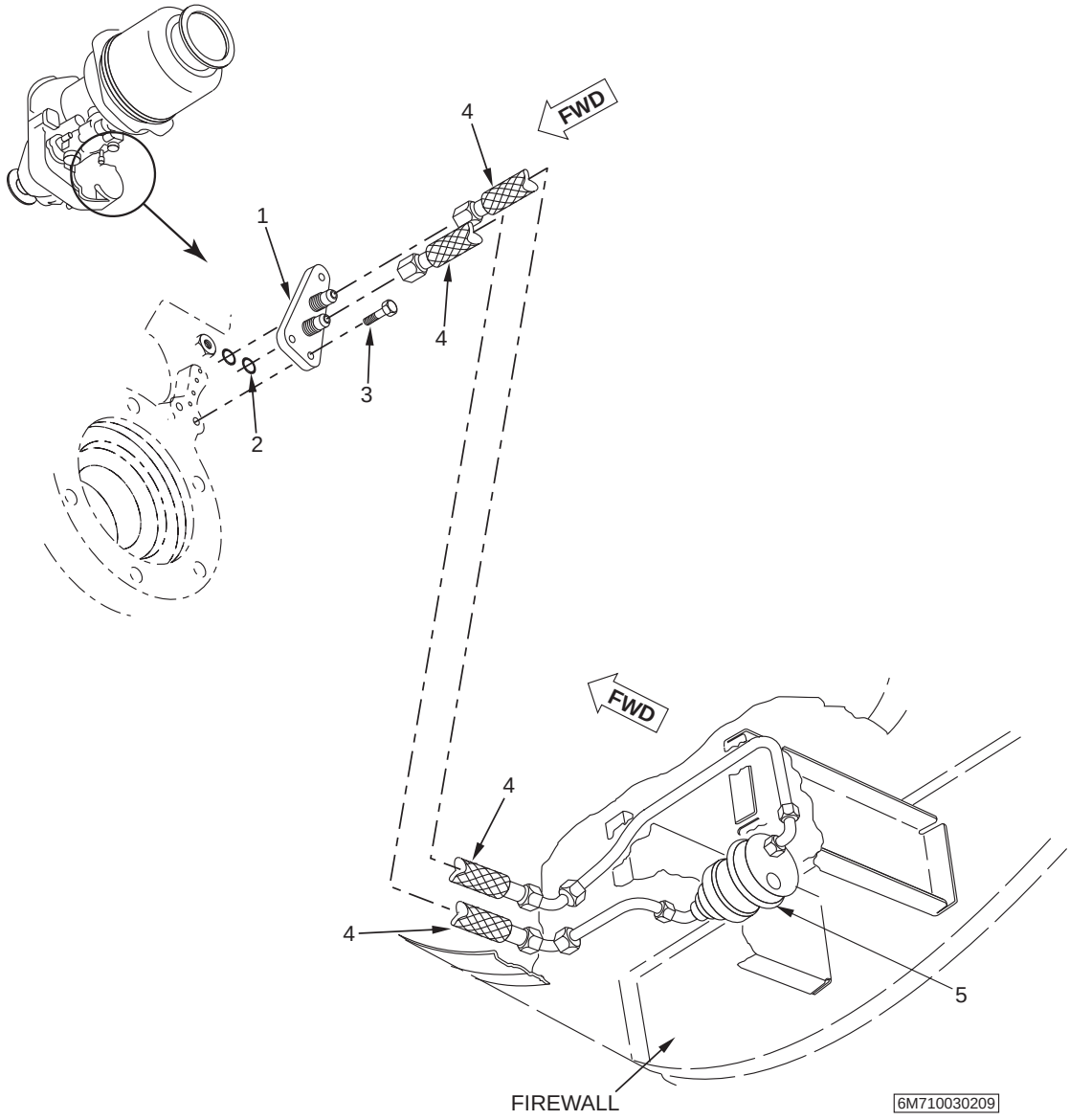
(8) Clamp hoses using two small clamps (MS21919DF12) and one large clamp (MS21919DF20) at both ends of engine truss member.

N. Installation – Engine Torque Transmitter Assembly (Reference Figure 210, T238-276, 278-291, AT012-999, and TC208-999)

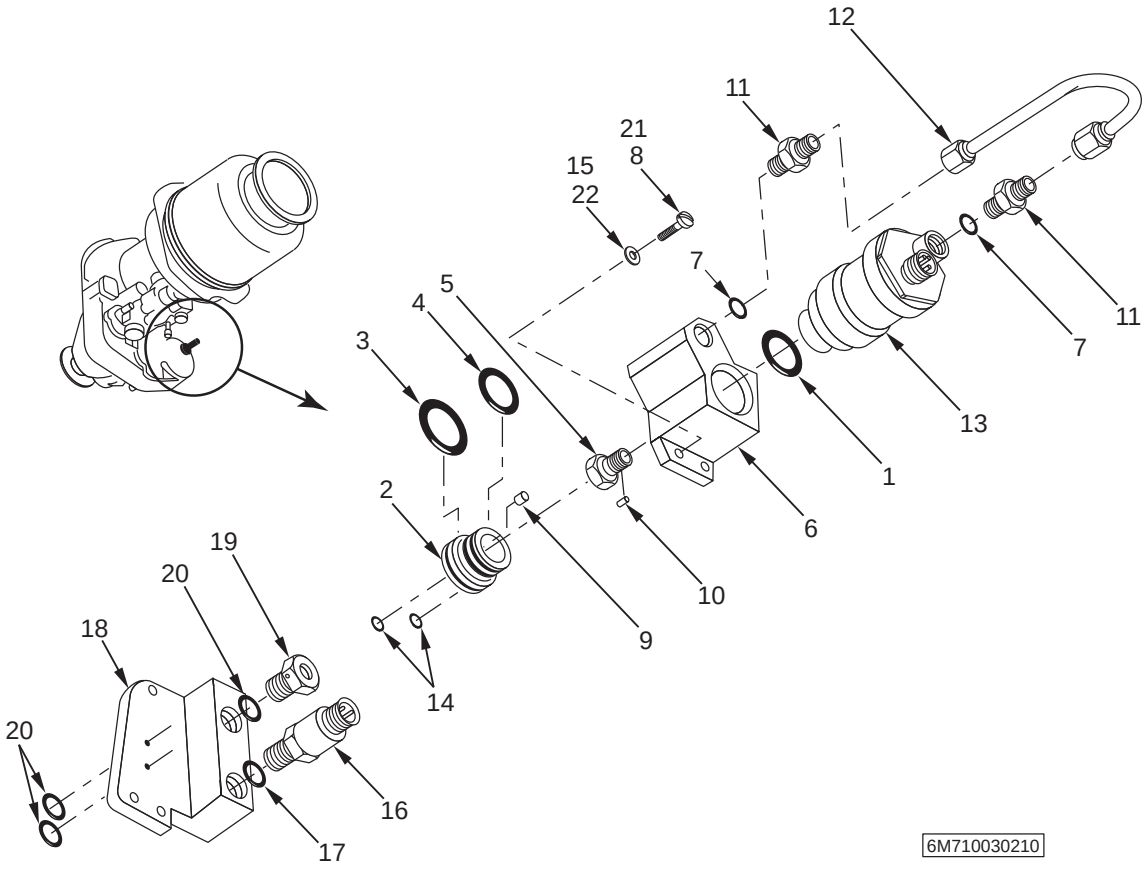
NOTE: Lightly lubricate O-rings with engine oil prior to their installation.

(1) Install O-ring (item 1) onto transmitter (item 13).

(2) Install large O-ring (item 3) and small O-ring (item 4) onto plug assembly (item 2).



Torque Transmitter
Figure 209



Torque Transmitter
Figure 210

(3) Install bolt assembly (item 5) into adapter (item 6).

(4) Install transmitter (item 13) into adapter (item 6); tighten bolt (item 5), and torque 30 to 40 inch-pounds.

(5) Install O-ring (item 7) onto each union (item 11), and install unions into adapter (item 6) and transmitter (item 13). Torque unions 40 to 50 inch-pounds.

(6) Install tube assembly (item 12) onto unions installed during step (5). (7) Install plug assembly (item 2) into adapter (item 6) aligning pins on plug assembly to mate with recessed holes in adapter while installing.

(7) Install O-ring (item 17) onto pressure switch (item 16). Install pressure switch into space block (item 18).

NOTE: Above step is only applicable to aircraft using Alcohol/Water Injection.

(8) Install one O-ring (item 20) onto plug (item 19). Install plug into space block (item 18).

NOTE: Above step is only applicable to aircraft using Alcohol/Water Injection.

(9) Install two O-rings (item 14) into recesses of plug assembly (item 2).

NOTE: It is recommended that a slight amount of lubricant, such as Crisco, be applied to the O-rings to prevent their becoming dislodged during installation.

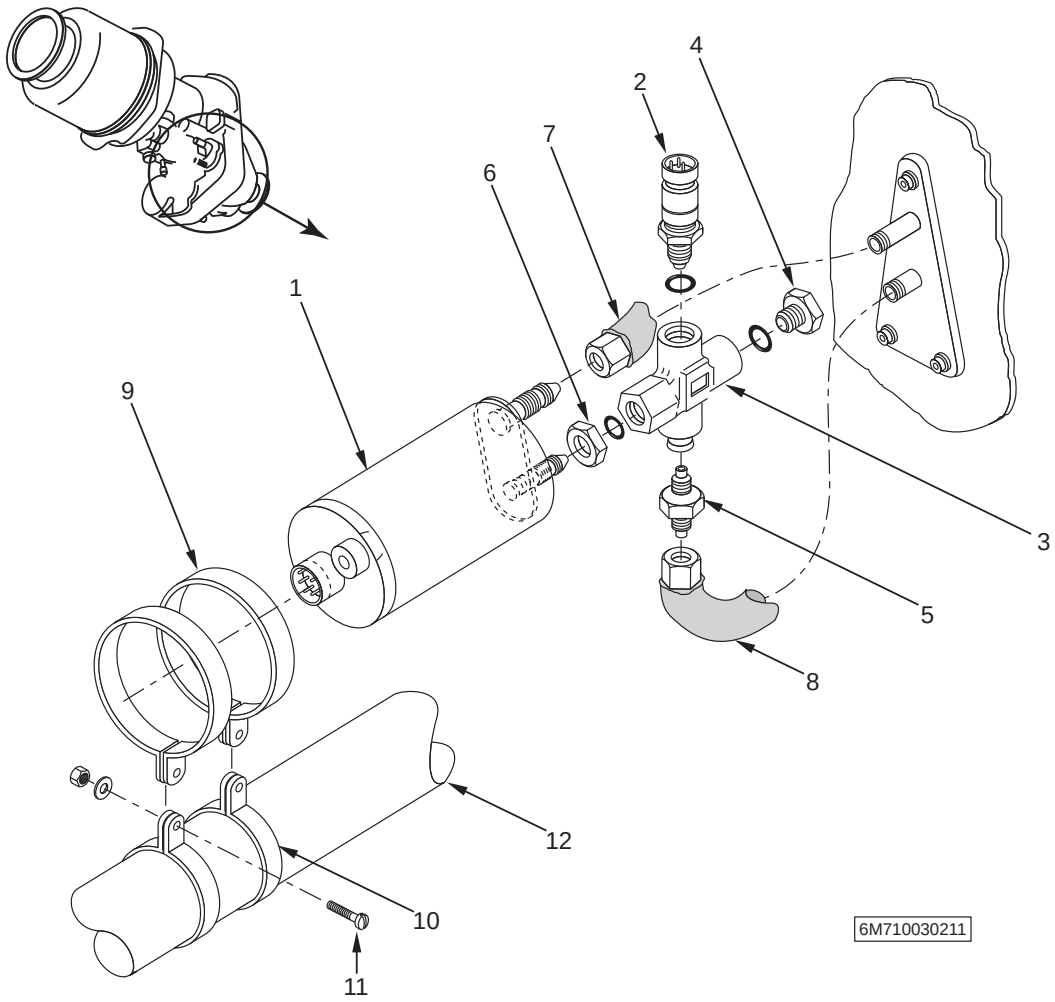
(10) Install two O-rings (item 20) into the recesses of the rear side of the spacer block (item 18).

NOTE: Above step is only applicable to aircraft using Alcohol/Water Injection.

(11) Install entire transmitter assembly to engine using three washers (items 15 and 22) and three screws (item 8 or 21), and torque screws 25 to 30 inch-pounds.

(12) Safety pressure switch (item 16) to plug (item 19) with .032 safety wire.

NOTE: Above step is only applicable to aircraft using Alcohol/Water Injection.



Torque Transducer
Figure 211

O. Installation - Engine Torque Transducer Assembly (Figure 211, T276, 292-999)

(1) Install plug and O-ring (item 4) into cross fitting (item 3).

(2) Install reducer and O-ring (item 5) into cross fitting (item 3).

(3) Install cross fitting (item 3) onto torque transducer (item 1) with nut and O-ring (item 6).

(4) Clamp torque transducer (item 1) to engine mount (item 12) with clamps (item 9 and 10) and secure with bolt, washer, and nut (item 11).

(5) Install hose assemblies (items 7 and 8).

NOTE: Hose (item 8) to cross fittings is routed on inside of engine mount truss member. Hose (item 7) to transducer is routed on outside of engine mount truss member.

(6) Install switch and O-ring (item 2) into cross fitting (item 3). Safety plug (item 4) to jam nut on switch (item 2) with .025 safety wire.

(7) Install connector to torque transducer.

(8) Install cannon plug to switch (item 2), and safety to cross fitting (item 3) with .025 safety wire.

P. Installation – Engine Drain (Reference Figure 212)

(1) Ensure drain block support assembly (item 1) is secure at the cowling flange at the lower right of the firewall.

(2) Install hose (item 2) connecting cowling drain tube assemblies (item 3 and 4) together, and secure with clamps (item 5). Install tube assembly (item 3) into drain block assembly (item 1).

(3) Install union (item 6) onto unfeather pump drain hose (item 7). Install tube assembly (item 8) to union; then, install tube assembly into drain block assembly (item 1).

(4) Install union (item 9) onto engine case water drain hose (item 10). Install tube assembly (item 11) onto union; then, install tube assembly into drain block assembly (item 1).

(5) Install union (item 12) onto fuel control and pump case drain hose (item 13). Install tube assembly (item 14) onto union; then, install tube assembly into drain block assembly (item 1).

(6) Install union (item 15) onto combustion case overboard drain hose (item 16). Install tube assembly (item 17) onto union; then, install tube assembly into drain block assembly (item 1).

Q. Installation – Engine Starter-Generator Assembly (Reference Figure 213) Buildup of Starter-Generator/Alternator (Item 4, insert A) T201-248

NOTE: The following procedures apply to SA226 serial numbers T201 through 248.

(1) Lubricate the interconnecting spring and O-ring between the starter-generator/alternator

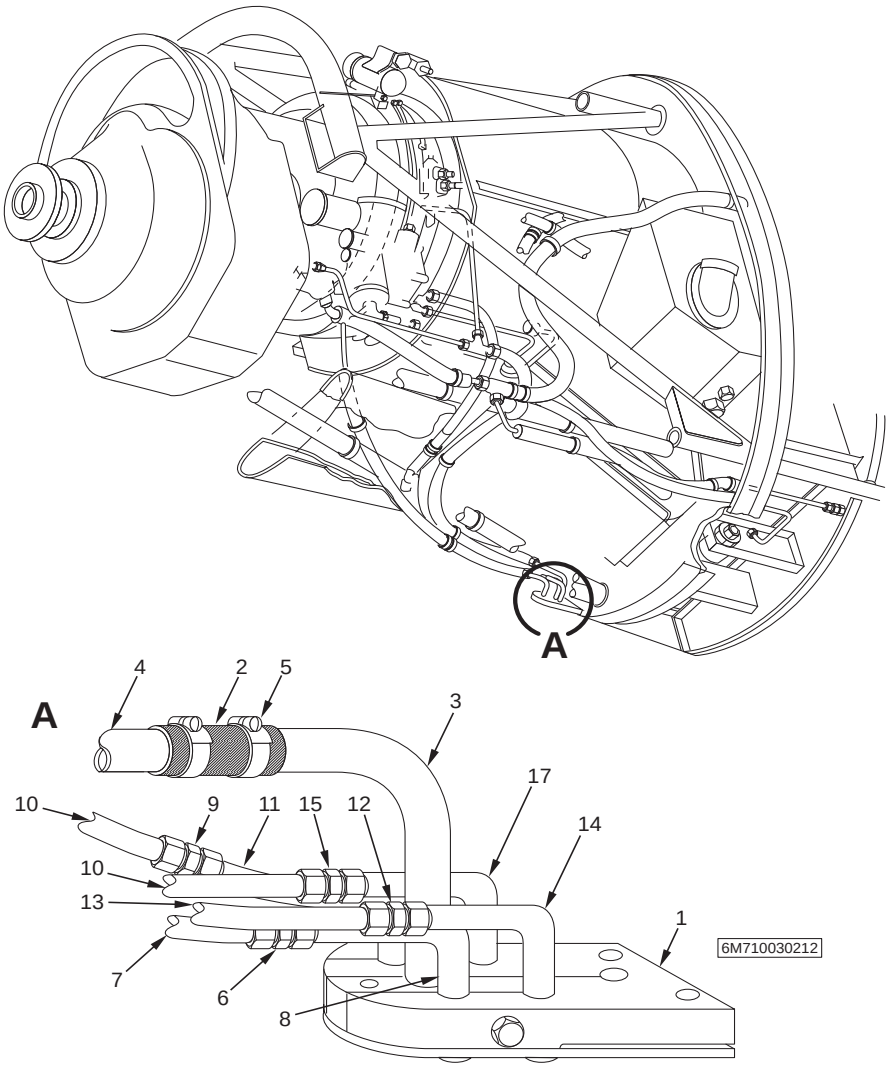
(item 4), and assemble the starter-generator/alternator together.

NOTE: Lubricate with Royco No. 27A, Aero Shell No. 7, Mobil No. 27, or Texaco No. TG6982.

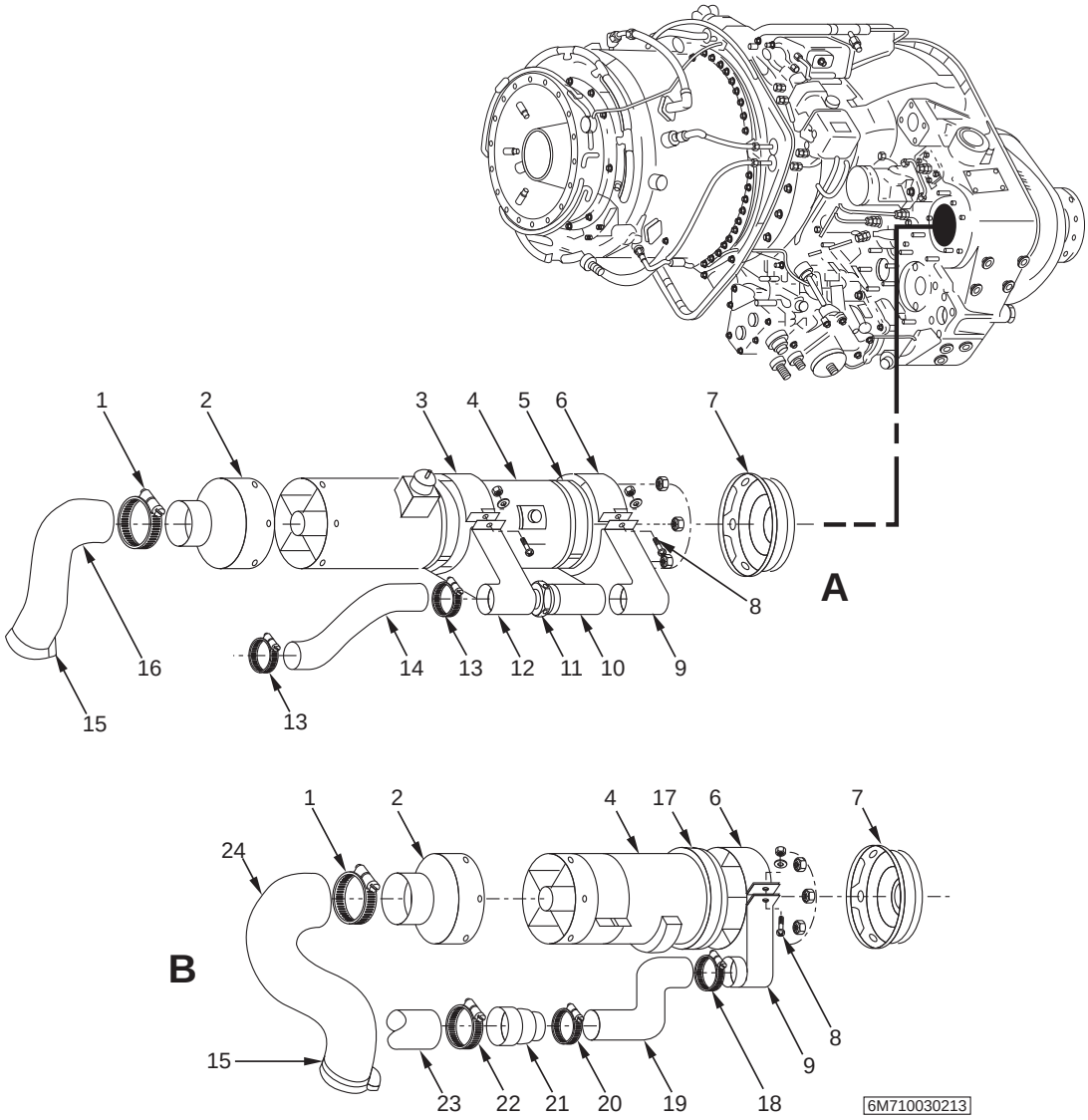
(2) Install duct assembly (item 2) and collar (item 3) onto starter-generator/alternator securing duct and collar with two screws, washers, and nuts.

(3) Install intake duct assembly (item 5) onto starter-generator/alternator.

NOTE: Remove cover screen from starter-generator before installing duct, and discard screen.



Engine Drain Block
Figure 212



Starter-Generator Assembly
Figure 213

(4) Install both collars (items 6 and 9) to end-bell mount (item 7) on engine. Secure both collars with two screws, washers, and nuts (item 8).

(5) Lubricate shaft of starter-generator/alternator with Molybdenum Disulfide (MIL-G-21164), and install starter-generator with clamp (item 7). Torque 70 to 90 inch-pounds.

(6) Install hose (item 10) between duct assembly (item 6) and duct assembly (item 3) securing hose with clamp (item 11) at each end of hose.

(7) Install hose (item 14) to duct assembly (item 3) with clamp (item 13). Install opposite end of hose to aspirator with clamp (item 13).

(8) Install hose (item 16) to intake duct assembly (item 2) with clamp (item 16). Install opposite end of hose (item 16) to scoop assembly with clamp (item 15).

(9) Secure end bell (item 7) to engine accessory drive gearbox with six (6) nuts and washers. Torque nuts 95 to 110 inch-pounds.

(10) Connect electrical terminals, refer to Chapter 24, Wiring Manual for lead and terminal connections.

R. Installation – Starter-Generator (Reference Figure 213, Inset B)

NOTE: The following procedures apply to SA226 series aircraft serial numbers; T249-275 and T277-291, AT001-069, and TC201-264.

(1) Install intake duct assembly (item 2) onto starter-generator (item 4).

NOTE: Remove cover screen from starter-generator before installing duct, and discard screen.

(2) Install both collars (item 6 and 9) to end-bell mount (item 7) on engine. Secure both collars with two screws, washers, and nuts (item 8).

(3) Secure end bell (item 7) to engine accessory drive gearbox with six nuts and washers. Torque nuts 95 to 110 inch-pounds.

(4) Lubricate shaft of starter-generator with Molybdenum Disulfide (MIL-G-21164), and install starter-generator with clamp (item 17). Torque clamp 70 to 90 inch-pounds.

(5) Install hose (item 19) to duct assembly (item 9) with clamp (item 18).

(6) Install reducer assembly (item 21) to opposite end of hose (item 19), and secure with clamp (item 20).

(7) Install hose (item 23) to reducer assembly (item 21) with clamp (item 22). Install opposite end of hose (item 23) to aspirator with clamp.

(8) Install hose (item 24) to intake duct assembly (item 2) with clamp (item 1). Install opposite end of hose to scoop assembly with clamp (item 15).

(9) Connect all engine electrical terminals and connectors. Reference Chapter 24, Wiring Manual, for lead and terminal connections.

* Not required for those aircraft that have complied with M7 Aerospace Service Bulletin 24-017 (delete Generator Cooling Exhaust Duct).

S. Installation – Engine Starter – Generator Assembly (Reference Figure 214)

NOTE: The following procedures apply to SA226 series aircraft serial numbers; T276, 292-999, AT070-999, and TC265-999.

(1) Install intake duct assembly (item 3) onto starter-generator (item 4).

NOTE: Remove cover screen from starter-generator before installing duct, and discard screen.

(2) Lubricate shaft of starter-generator with Molybdenum Disulfide (MIL-G-21164), and install starter-generator to end-bell mount (item 6) with clamp (item 5). Torque clamp to 70 to 90 inch-pounds.

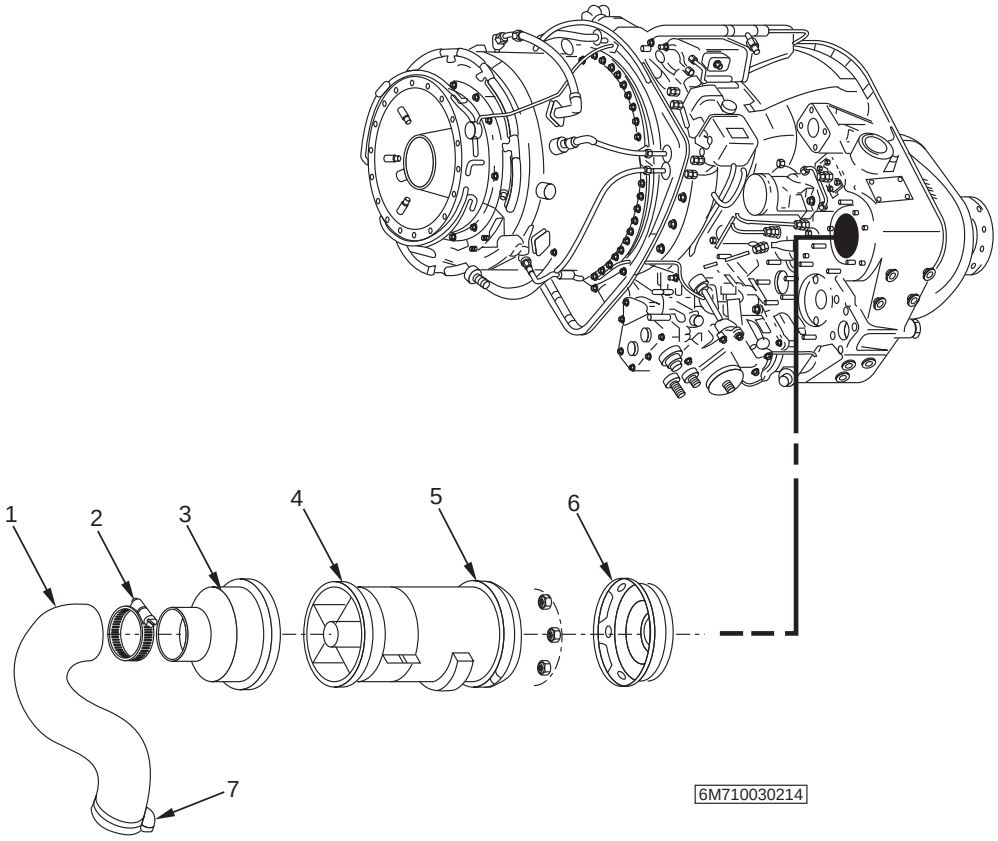
(3) Install hose (item 1) to intake duct assembly (item 3) with clamp (item 2). Install opposite end of hose to scoop assembly with clamp (item 7).

(4) Connect all engine electrical terminals and connections. Reference Chapter 24, Wiring Manual, for lead and terminal connections.

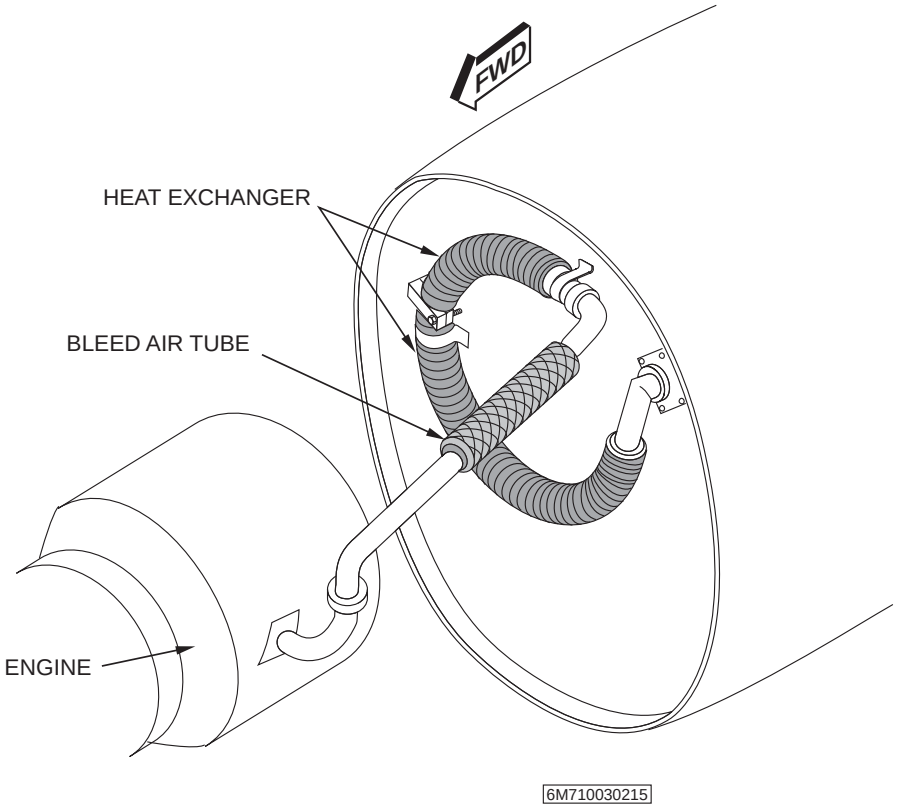
T. Installation - Engine Bleed Air Tube Connection and Aspirator Access Covers (Reference Figure 215)

(1) Apply C5A anti-seize compound (or MIL-A-907 equivalent) and install engine bleed air tube to upper finned tube assembly. Torque clamp to 40 inch-pounds.

(2) Install access cover over pressurization coupling area and fasten.



Starter-Generator Assembly
Figure 214



Engine Bleed Air Tube Connection
Figure 215

U. Engine Installation – Engine Shrouds (Reference Figure 216)

NOTE: Ensure ignitor plug shield fits easily through access hole in lower cover assembly (item 5) before installing shroud.

For easier maintenance, the combustion case overboard drain line should be removed, see Figure 212, Engine Drain Block Installation for drain line location.

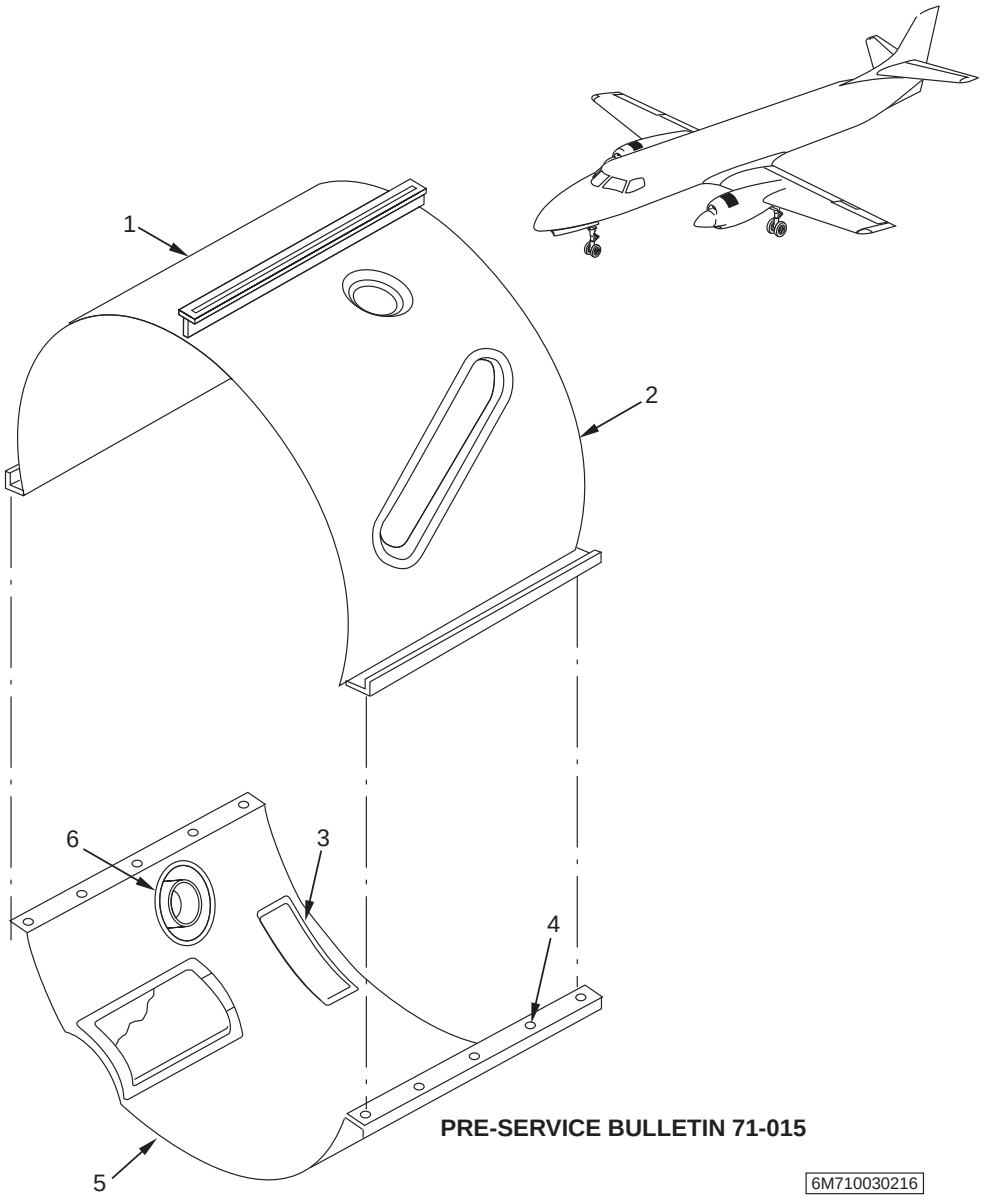
V. Installation – Engine Lower Cover Assembly (item 5)

(1) Install lower cover assembly (item 5) by inserting the aft portion of the assembly between the oil cooler and heat exchanger assembly fireseal. Ensure correct alignment and fit. Make sure that the front of the cover assembly aligns at the front of the heat shroud retainer flange.

NOTE: A 1/4 inch clearance must be maintained between the lower cover assembly and the oil tank. On new installation, remove approximately 10/100 inches, if necessary, from aft portion of lower cover assembly in order to obtain correct alignment and fit.

(2) Install Camloc fastener plate on sides of lower cover assembly, using five screws for each plate.

NOTE: For easier maintenance, install Camloc fasteners by starting in the middle of the lower cover assembly and by working upwards.



Engine Shroud Assembly
Figure 216

W. Installation – Engine Left and Right Cover Assemblies (Items 2 and 1)

(1) Install left and right cover assemblies by installing Camloc fasteners.

NOTE: For easier maintenance, first install fasteners at the cover assembly sides. Ensure ignitor plug shield fits easily through access hole in cover assembly before installing. On new installations, trim sides of cover assembly as required for flush fit with front of engine flange shroud retainers, for contact with fireseal, and for clearance between cover assembly and cover over heat exchanger assembly access area.

(2) Reinstall combustion case overboard drain, if previously removed, to facilitate heat shroud installation.

X. Installation – Engine Nose Cowling Assembly (Reference Figure 217)

(1) Place nose cowling assembly into position, and install new mount bolts (item 5) at the cowling's center frame to brackets on lower front of reduction gear case.

(2) Install bolts (item 2) at cowling's aft frame to brackets at Lord mounts.

(3) Install nuts (item 8) at front of engine air intake.

(4) Install mount bolts (item 10) on rear of engine air intake.

(5) Install four mount bolts (item 9) at air intake. Connect anti-ice lines.

(6) Install propeller (see Chapter 61, Maintenance Manual).

(7) Install lower forward cowl.

Y. Installation – Engine Upper Cowling

(1) Install upper cowling into place, install all screw fasteners.

Z. Installation – Engine Lower Cowling

(1) Disconnect and remove elbow or fitting between oil hoses, which transfers oil from oil cooler to oil tank.

NOTE: If engine was to be operated with lower cowling off, an elbow or fitting would have been installed between the hoses from the oil cooler to lower cowling air scoop lip heating tube in order to transfer oil from oil cooler to oil tank.

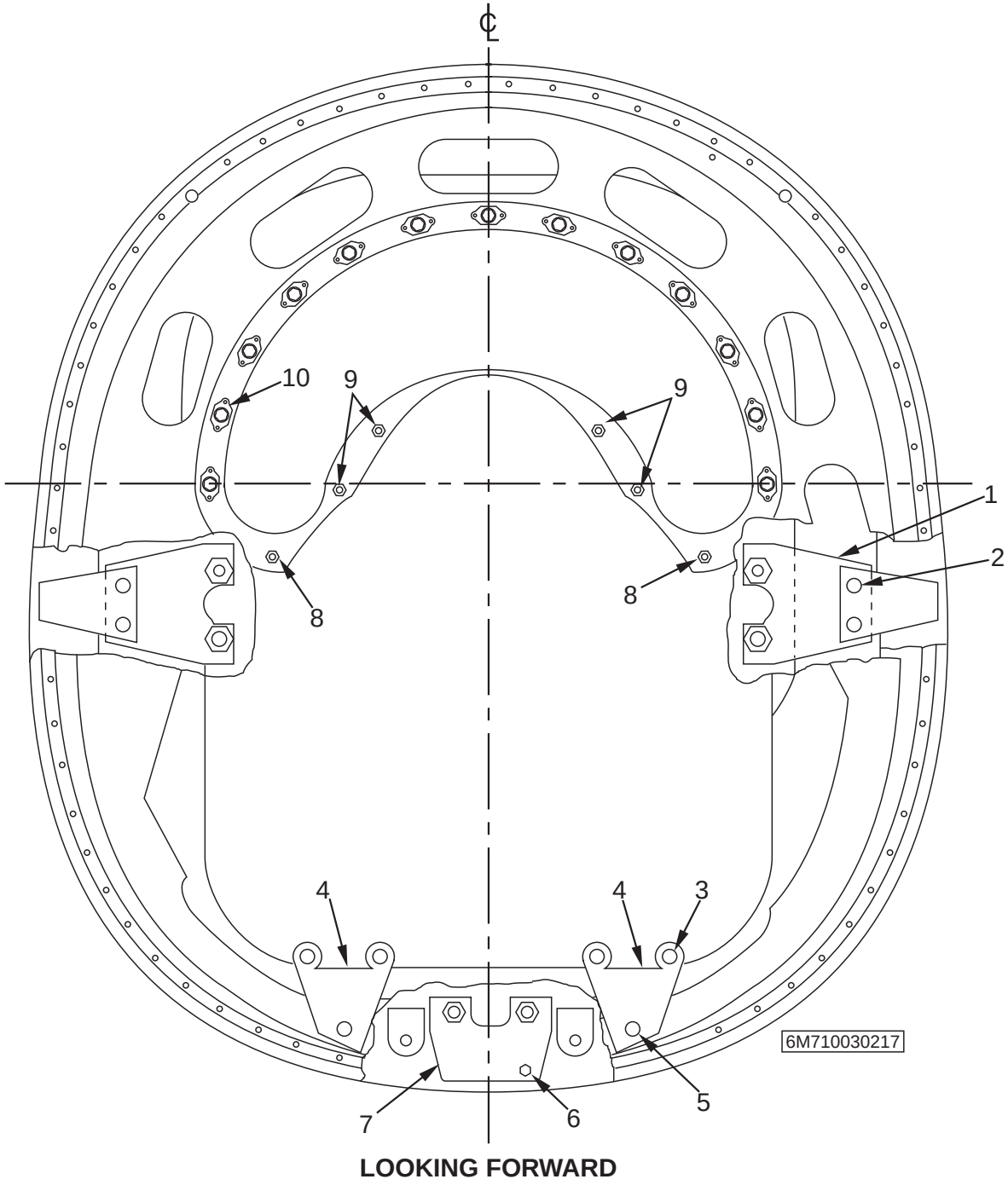
(2) Connect oil hoses to the air scoop lip heating tubes.

(3) Install air scoop assembly to base of oil cooler.

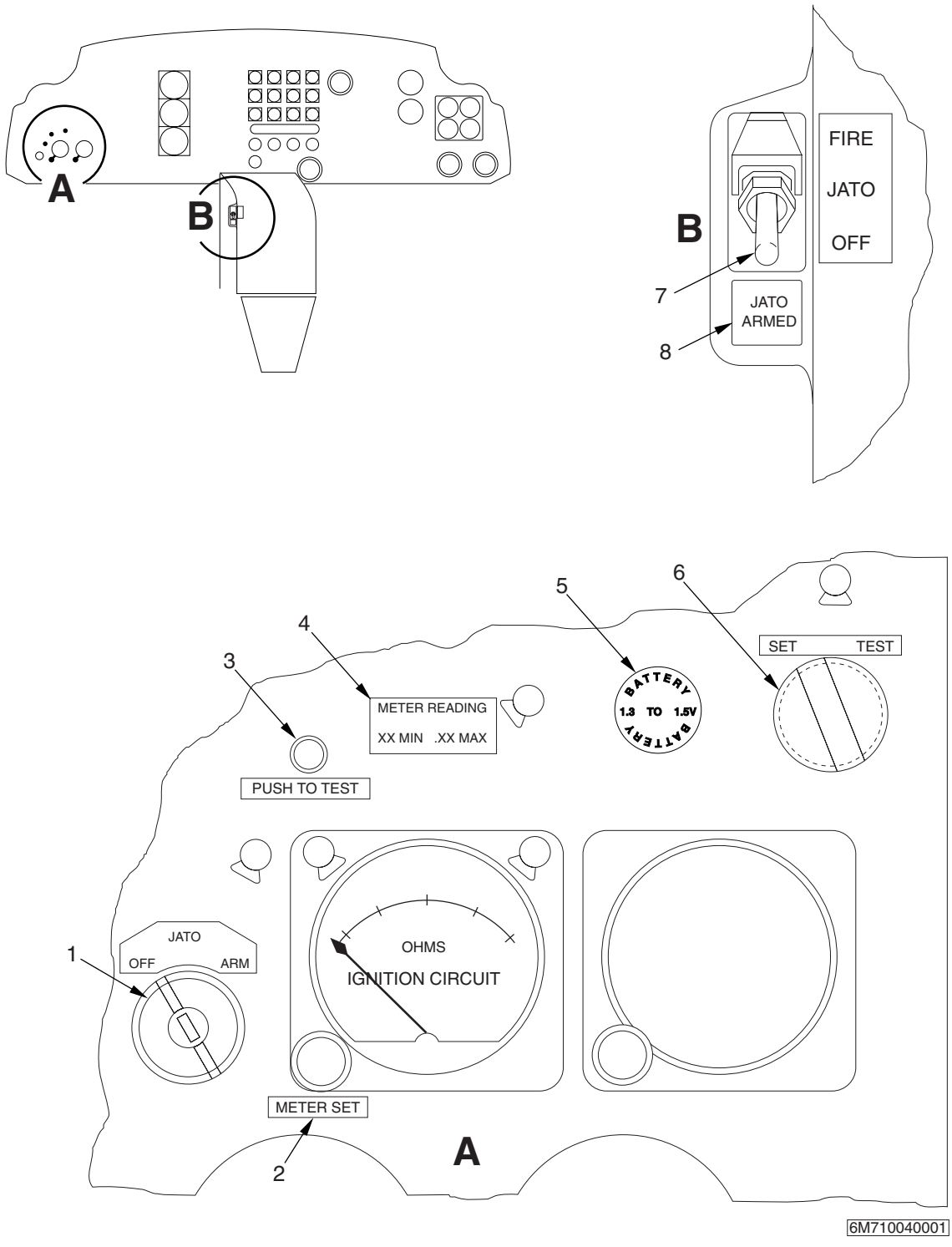
(4) Install lower cowling and Camloc fasteners.

AA. Side Cowling

(1) Install side cowling into place, install all Camloc fasteners.



Forward Cowling Brackets
Figure 217



Jato Controls
Figure 1

STANDBY ROCKET (JATO) – DESCRIPTION AND OPERATION

1. General

A. When selected as a customer option, a standby rocket (commonly referred to as JATO) is installed on the rear pressure bulkhead. The rocket engine (JATO unit) is a model 12NS-350CBA or a model 12NS-350 capable of delivering 350 pounds of thrust for 12 seconds on a standard day at sea level. This power is used to augment existing thrust in the event of an engine failure during takeoff and climb. Cockpit controls allow the pilot to test, arm, and should the need arise, fire the JATO unit. Power to fire the JATO unit is provided by the aircraft bus; an independent battery is provided for the test circuit.

2. Jato System

A. The following controls are available in the cockpit for operation and control of the system (reference Figure 1):

(1) Key-Operated Selector Switch (item 1). The two-position switch applies power to the JATO FIRE switch and the JATO ARMED light when the key is moved to the ARM position.

(2) JATO ARMED Light (item 8). Installed on the left side of the pedestal, immediately below the JATO FIRE switch, the JATO ARMED light serves two functions.

(a) When the light is illuminated, the pilot knows that power is available at the JATO FIRE switch.

(b) When the system is fired, the JATO ARMED light will be extinguished, providing the pilot with a visual indication that the JATO unit has fired.

(3) JATO FIRE Switch (item 7). The guarded, spring loaded, two-position switch applies power to the JATO unit ignitor when the switch is moved to the FIRE position.

WARNING: WHEN THE KEY-OPERATED SELECTOR SWITCH IS IN THE ARM POSITION AND THE JATO FIRE SWITCH IS MOVED TO THE FIRE POSITION, THE JATO UNIT WILL FIRE. WHEN THE AIRCRAFT IS IN THE HANGAR OR HAVING MAINTENANCE PERFORMED THE JATO CIRCUIT BREAKER ON THE LEFT CENTER CONSOLE MUST BE PULLED, THE KEY OPERATED SELECTOR SWITCH MUST BE IN THE OFF POSITION, AND THE KEY MUST BE REMOVED.

B. The operation of the JATO system is as follows:

(1) Closing the JATO circuit breaker provides DC power from the left essential bus to the key-operated JATO selector switch.

(2) Placing the key-operated JATO selector switch in the ARM position arms the system by providing DC power to the JATO FIRE switch. In addition, power is provided to illuminate the JATO ARMED light.

WARNING: WHEN THE KEY-OPERATED SELECTOR SWITCH IS IN THE ARM POSITION AND THE JATO FIRE SWITCH IS MOVED TO THE FIRE POSITION, THE JATO UNIT WILL FIRE. WHEN THE AIRCRAFT IS IN THE HANGAR OR HAVING MAINTENANCE PERFORMED, THE JATO CIRCUIT BREAKER ON THE LEFT CENTER CONSOLE MUST BE PULLED, THE KEY OPERATED SELECTOR SWITCH MUST BE IN THE OFF POSITION, AND THE KEY MUST BE REMOVED.

(3) Placing the JATO FIRE switch in the FIRE position provides DC power to the JATO igniter and fires the JATO unit. When the JATO unit is fired, the exhaust gases will break the wire that completes the ground circuit to the JATO ARMED light, extinguishing the light and providing the pilot with a visual indication that the JATO unit has fired.

C. Test System

NOTE: The following components comprise the self-test system for the JATO ignition circuit:

(1) Ignition Circuit Ohmmeter (item 2). The ohmmeter provides a direct reading of the resistance of the JATO ignition circuit. A METER SET adjustment is provided for initial and recurrent calibration of the ohmmeter. A placard (item 4) immediately above the meter indicates minimum and maximum allowable readings.

(2) PUSH-TO-TEST Switch (item 3). A momentary contact push button is used to connect the battery to the ignition circuit during test.

(3) Battery (item 5). A 1.5 volt battery provides power for the test system. An independent power source is provided to ensure accuracy, as aircraft bus voltage will vary slightly according to electrical load, output voltage of the APU being used, or the condition of the aircraft batteries. Such variance of the bus voltage will not affect the operation of the JATO system, but will compromise the accuracy of the ignition circuit test system.

(4) SET-TEST Switch (item 6). The two-position switch is used to select the mode in which the ohmmeter will operate. With the switch in the SET position, the ohmmeter is disconnected from the JATO ignition circuit and may be calibrated. With the switch in the TEST position, the meter is connected to the JATO ignition circuit for testing.

D. Test System

NOTE: The operation of the test system is as follows (reference Figure 1):

(1) To ensure accurate testing, the first step is to adjust the ohmmeter. When the SET-TEST switch (item 6) is placed in the SET position, the ohmmeter circuit is disconnected from the aircraft system. When the PUSH-TO-TEST switch (item 3) is depressed, the test battery is connected to the ohmmeter circuit. The pointer on the ohmmeter must read exactly on the SET position; the METER SET knob on the ohmmeter is used to adjust the pointer to an exact reading.

NOTE: The ohmmeter must be set as described above prior to every test of the JATO ignition circuit.

(2) When the SET-TEST switch is placed in the TEST position, the ohmmeter is connected to the JATO ignition circuit. When the PUSH-TO-TEST switch is depressed, the ohmmeter measures the resistance of the JATO ignition circuit. This reading must fall between the minimum and maximum figures on the placard immediately above the meter. Failure of the meter to indicate between these limits signifies that system maintenance is required.

E. Emergency Procedures – Fire

(1) JATO units should not be stored with flammable or combustible materials or in areas where floors might collapse in case of fire. The storage area should be well ventilated to allow the release of pressure should a JATO unit ignite. In case of fire in an area where JATO units are stored, the following procedures are recommended:

CLEAR THE AREA OF ALL PERSONNEL NOT INVOLVED IN FIRE FIGHTING.

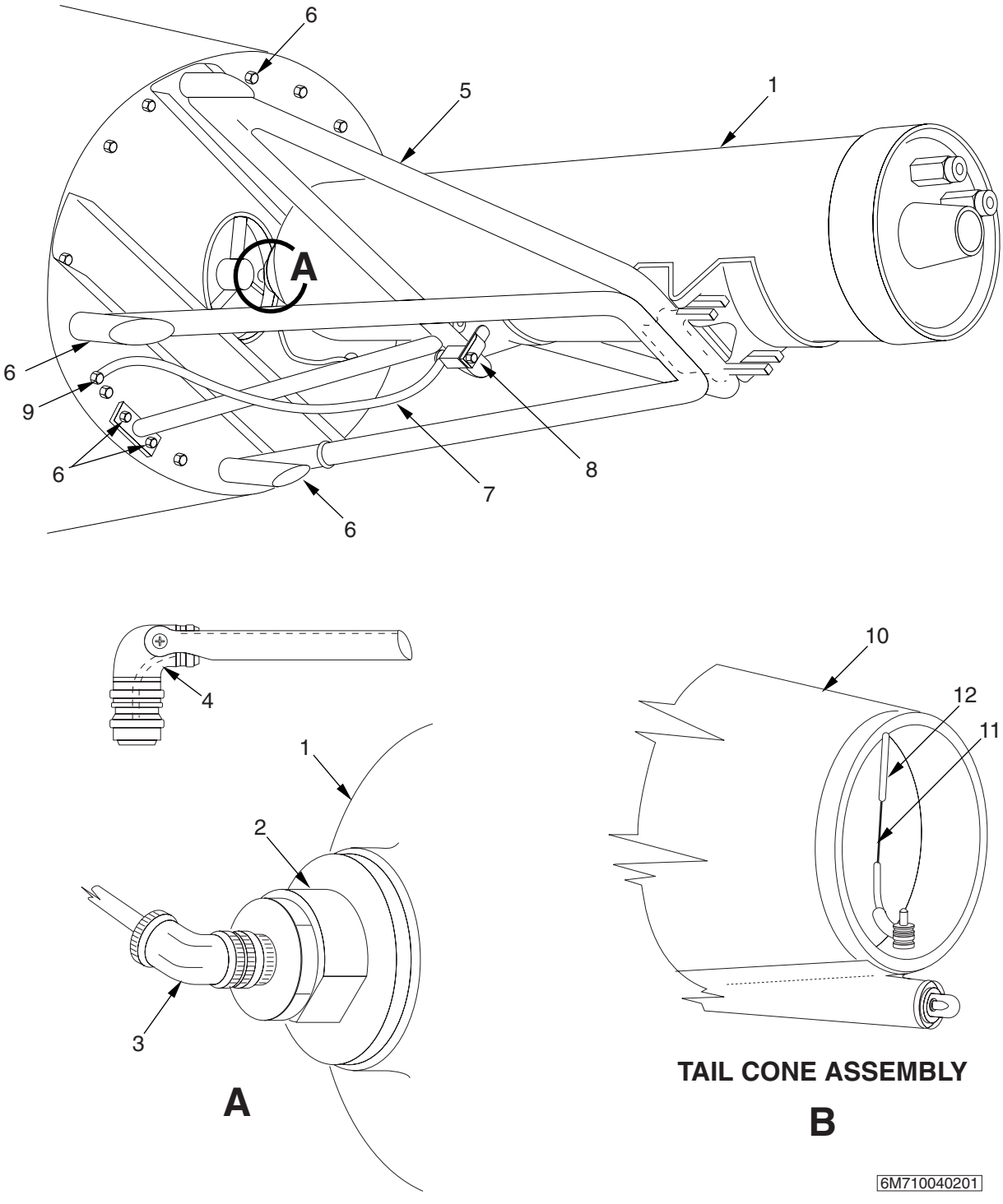
PROVIDE FIRE FIGHTING PERSONNEL WITH SUITABLE RESPIRATOR MASKS IN CASE AREA VENTILATION IS INADEQUATE.

IF POSSIBLE REMOVE ALL JATO UNITS AND AIRCRAFT EQUIPPED WITH JATO UNITS FROM THE FIRE AREA.

IF JATO UNITS OR AIRCRAFT EQUIPPED WITH JATO UNITS CANNOT BE REMOVED FROM THE FIRE AREA, CONCENTRATE FIRE FIGHTING EFFORTS ON COOLING THESE UNITS TO PREVENT IGNITION.

USE WATER OR FOAM TO CONTROL ANY BURNING MATERIAL IN THE VICINITY OF THE JATO UNITS.

DO NOT ATTEMPT TO EXTINGUISH IGNITED JATO UNITS. THE PROPELLANT CONTAINS ITS OWN OXIDIZER AND CANNOT BE EXTINGUISHED BY NORMAL SMOTHERING TECHNIQUES. THE ONLY EFFECTIVE MEANS OF FIGHTING THE FIRE IS BY VENTING THE STORAGE AREA AND CONFINING THE FIRE BY FLOODING THE SURROUNDING AREA WITH WATER OR FOAM.



Jato Installation
Figure 201

STANDBY ROCKET (JATO) – MAINTENANCE PRACTICES

1. Servicing – Standby Rocket (Jato)

A. When fired, the JATO unit produces a high pressure, high temperature exhaust stream. This exhaust can be hazardous to both people and equipment. The following precautions are strongly recommended by the JATO unit manufacturer and by Fairchild Aircraft:

IN THE INTEREST OF SAFETY, ALWAYS ASSUME THAT THE JATO UNIT WILL FIRE AT ANY SECOND.

WHEN POSSIBLE, THE AIRCRAFT SHOULD BE PARKED WITH THE TAIL CLEAR OF ALL OBSTRUCTIONS.

DEVELOP THE HABIT OF AVOIDING THE AFT END OF THE AIRCRAFT. THIS DOES NOT IMPLY DISREGARD FOR MAINTENANCE, AS MOST MAINTENANCE AND SERVICING CAN BE DONE WITHOUT EXPOSING PERSONNEL AND EQUIPMENT DIRECTLY IN LINE WITH THE JATO UNIT.

STANDBY ROCKET (JATO) – MAINTENANCE PRACTICES

IF THE AIRCRAFT IS TO BE IN THE HANGAR FOR SEVERAL DAYS, REMOVE THE TAIL-CONE AND DISCONNECT THE AIRCRAFT ELECTRICAL CONNECTOR FROM THE JATO IGNITER.

WARNING: WHEN THE AIRCRAFT ELECTRICAL CONNECTOR HAS BEEN DISCONNECTED, THE JATO IGNITER MUST BE SHORTED TO PREVENT ACCIDENTAL IGNITION (SEE FIGURE 4 FOR SHORTING PLUG). IF THE IGNITER IS NOT SHORTED, STRAY HIGH TENSION ELECTRICAL SIGNALS, SUCH AS THOSE PRODUCED BY A STROBE LIGHT, MAY IGNITE THE JATO UNIT.

NEVER USE AN ORDINARY VOLT-OHMMETER TO CHECK THE JATO IGNITER. THE ONLY SAFE WAY TO CHECK THE RESISTANCE OF THE IGNITER CIRCUIT IS WITH THE AIRCRAFT INSTALLED OHMMETER SYSTEM. USE OF A STANDARD VOLT-OHMMETER MAY CAUSE IGNITION.

SINCE THE JATO SYSTEM IS AN EMERGENCY SYSTEM, THE GREATEST OF CARE MUST BE EXERCISED IN HANDLING AND MAINTAINING THE SYSTEM AND ITS COMPONENTS. IF A JATO UNIT OR IGNITER IS DROPPED, REPLACE IT.

EVEN DROPPING FROM A SMALL HEIGHT MAY DAMAGE THESE UNITS. IF THERE IS ANY REASON TO SUSPECT IMPROPER OPERATION OF THE JATO SYSTEM, INVESTIGATE AS SOON AS POSSIBLE.

WHEN STORING THE JATO UNIT(S), ALWAYS REMOVE THE IGNITER(S). THE SAFEST PLACE FOR REMOVAL AND INSTALLATION OF THE IGNITER IS WITH THE JATO UNIT SECURED IN ITS MOUNT, AND THE MOUNT INSTALLED ON THE AIRCRAFT.

WHEN THE JATO IGNITER IS REMOVED, ALWAYS RESEAL THE END OF THE JATO UNIT TO PREVENT MOISTURE FROM ENTERING THE PROPELLANT.

JATO UNITS SHOULD ONLY BE TRANSPORTED IN THEIR ORIGINAL PACKING CONTAINERS. THESE CONTAINERS FEATURE A SEPARATE HOLDER FOR THE IGNITER.

2. Removal/Installation –- Standby Rocket (Jato)

A. The following steps must be performed prior to any maintenance:

BATTERY SWITCHES ..... OFF

JATO CIRCUIT BREAKER ..... PULLED

JATO KEY SWITCH ..... OFF, KEY REMOVED

JATO FIRE SWITCH ..... OFF, GUARD DOWN

(1) Remove the tail cone.

(2) Remove the electrical connector from the JATO igniter and install a shorting plug (see Figure 201).

(3) Remove the JATO unit.

(a) IF THE JATO UNIT HAS BEEN FIRED: Remove the bolt securing the JATO unit to the mount and slide the unit out of the mount. Install the new unit and torque the securing bolt to 45 inch-pounds. Install the igniter and torque 225 to 250 inch-pounds.

Install a new ground wire in the tail cone, see inset B, Figure (201). Ensure that the wire is attached in such a way that it is grounded only at the upper terminal. Refer to paragraph RECALIBRATION AFTER REPLACEMENT for JATO system recalibration.

(b) IF THE JATO UNIT IS BEING REMOVED TO FACILITATE REMOVAL OF THE REAR PRESSURE BULKHEAD: Loosen all six JATO mount bolts (item 6) and disconnect the ground strap (item 7) at the pressure bulkhead. Remove the mount and JATO unit as an assembly. Store the mount and unit assembly in place where it will not be disturbed. To reinstall position the mount in place and install the six retaining bolts. Torque the bolts to 30 inch-pounds. Attach the ground strap to the rear pressure bulkhead and torque the retaining nut to 30 inch-pounds.

(c) IF THE JATO UNIT IS BEING REMOVED FOR STORAGE OR SHIPMENT AND HAS NOT BEEN FIRED: Remove the igniter by holding the igniter mounting boss with a 1 1/2 inch open-end wrench and unscrewing the igniter (item 2) using a 1 3/8 inch socket. Plug the end of the JATO unit where the igniter was removed to prevent moisture from entering propellant. Remove the two bolts securing the JATO unit to the mount and slide the unit out of the mount. To reinstall, position the unit within the mount and torque the securing bolt to 45 inch-pounds. reinstall the igniter and torque 225 to 250 inch-pounds.

(4) Reconnect the electrical connector and perform the standard preflight check of the system. If the rocket engine igniter was replaced, the system must be recalibrated in accordance with paragraph B.

(5) Reinstall the tail cone.

B. Recalibration After Replacement

(1) Place the JATO ARM switch in the OFF position, remove key.

(2) Place the JATO FIRE switch to the OFF position, with the guard down.

(3) Install the aircraft electrical connector to the JATO igniter.

(4) Move the aircraft to a suitable area; assure that the area behind the exhaust is free and clear of obstructions.

(5) Operate the aircraft engine, turn generators and avionics ON.

(6) Move the SET-TEST switch to the SET position.

(7) Depress and hold the push-to-test button.

(8) Observe that the ohmmeter reading is SET. If the pointer does not indicate properly, adjust the METER SET knob. If satisfactory reading cannot be obtained, the test system battery is weak and must be replaced.

(9) Move the SET-TEST switch to the TEST position and observe the ohmmeter reading. The reading obtained must be between 0.35 and 0.65 ohms. Record the value obtained during this test. The limits to be marked on the cockpit JATO placard will be computed as follows:

MIN = XXX (value obtained) minus 0.15 ohm MAX = XXX (value obtained) plus 0.15 ohm

NOTE: Mark the two new values on the placard and install the placard on to the instrument panel. (10) If the value obtained does not fall within the 0.35 and 0.65 ohm limits, a defect exists in either the igniter itself or the aircraft wiring system. Replace the ignitor and recheck the system. If the new igniter does not correct the problem, the aircraft wiring must be tested.

C. Aircraft Wiring Test

(1) Remove the electrical connector from the JATO igniter, install shoring plug.

(2) Using a suitable pin and connector, short the igniter connector center pin to ground.

(3) Perform steps (1) through (9) listed in Recalibration After Replacement section. The final meter resistance reading must be between 0.15 and 0.35 ohm. If the reading does not fall within these limits, a problem exists in the aircraft wiring and must be corrected.

D. Emergency Procedures – Jato Does Not Fire

DO NOT APPROACH THE AFT END OF THE AIRCRAFT FOR THIRTY MINUTES AFTER THE FIRING ATTEMPT.

DO NOT ALLOW PASSENGERS OR CARGO BEHIND THE AIRCRAFT.

PARK THE AIRCRAFT AWAY FROM BUILDING, VEHICLES, AND OTHER AIRCRAFT. PARK THE AIRCRAFT SO THAT THE AFT END POINTS TOWARD FIFTY YARDS OF OPEN AREA.

(1) Perform the standard cockpit preflight check; carefully observe and record the reading obtained.

(2) Perform the following steps:

BATTERY SWITCHES. ....OFF

JATO CIRCUIT BREAKER ....PULLED

JATO KEY SWITCH .....OFF, KEY REMOVED

JATO FIRE SWITCH .....OFF, GUARD DOWN

(3) Remove the tail cone and conduct all maintenance operations clear of the exhaust nozzle of the JATO unit.

(4) Remove the electrical connector from the JATO igniter. Return to the cockpit and perform the standard preflight check - the reading should be SET (far right of the scale). If a reading other than this is obtained, a problem exists within the JATO ignition circuitry.

(5) With the connector removed from the igniter, use a voltmeter to measure voltage from the connector pin to ground. With the BATTERY switch ON and the key operated selector switch in ARM, move the JATO FIRE switch to FIRE. The voltmeter should read bus voltage (27.5 to 29 volts). If the reading is not correct the wiring between the cockpit and the tail of the aircraft is at fault and must be corrected.

(6) If the readings taken in steps 4 and 5 are correct, the problem is in the JATO unit or igniter. Remove the igniter from the JATO unit. If the igniter cannot be removed, unscrew the igniter mounting boss from the JATO unit. Examine the igniter, if it has not fired, proceed to step (7), if the igniter has fired, proceed to step (8).

(7) If the igniter has not fired, recheck the ignition circuitry as in steps (4) and (5).

(8) If the igniter has fired, the JATO unit is defective. Break the moisture barrier at the nozzle end and flood the inside of the unit with water. Remove the unit from its mount and place it in a suitable container. Keep the unit completely submerged in water during transportation to a suitable disposal site.

E. Jato Unit Disposal

(1) When the JATO unit has reached its time change interval (1,000 flight hours or 12 months) the simplest means of disposal is to fire the JATO as part of a pilot proficiency flight. Before this flight, the tail cone should be removed and the JATO unit and mount inspected. After the engine has been fired, wait at least thirty minutes before approaching the aft end of the aircraft. After the JATO unit has been removed, flush the inside with water and then dispose of the unit as scrap. If the JATO unit is to be disposed of without being fired, it will have to be through the use of a government approved deep-water disposal area for materials of this classification. Remove the igniter and fill the inside of the unit with water. Reseal the JATO unit and ship the unit and igniter in the original shipping container. The shelf life of the JATO unit is three years.

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ENGINE COWLING – DESCRIPTION AND OPERATION

1. General

A. The engine cowling consists of a nose cowling assembly, an upper cowling panel, a lower cowling panel, and left and right side cowling panels. All cowlings are custom fitted and trimmed to the aircraft during manufacture. See drilling, trimming, and buildup procedures for custom fit of new cowling within this section.

2. Nose Cowling Assembly

A. The engine air intake is an integral function of the nose cowling assembly. The assembly is riveted to three frames and is bolted to the engine and engine mount forward of the engine mount truss assembly. The aft frame is bolted to brackets at each side Lord mount; the center frame is bolted to brackets on the lower front of the reduction gear case and the forward frame is attached to the cowl only. Fittings are provided in the upper area of the cowl for the attachment of bleed air lines for anti-icing of the air intake. On some models, an Alcohol/Water Injection manifold is installed at the center frame area. Later nose cowling assemblies have a removable upper aft access skin held to the cowling by screw fasteners. All nose cowling assemblies have a removable lower forward access skin which is held to the cowling frames with screw fasteners.

3. Upper Cowling Panel

A. The upper cowling panel is attached at its forward and aft end with screws and nutplates. Channels, fitted with Camloc receptacles on each side of the upper cowling panel, provide as attach points for the upper edges of each side panel.

4. Lower Cowling Panel

A. The oil cooler air scoop cowl comprises the air scoop for the oil radiator (cooler) and the tubing to heat the intake of the scoop for anti-icing purposes, all of which forms the integral function of the lower cowling panel. Fittings on the heating tube provide a connection to the oil supply system. The cowl is attached forward and aft with screws and nutplates. Channels, fitted with Camloc receptacles on each upper side, provide attaching points for the lower edge of each side panel.

5. Side Cowling Panel

A. The left and right side panels attach to the upper panel at the firewall flange, at the lower cowling panel, and at the forward frame of the nose cowling assembly with Camloc fasteners. The side panels provide the easiest access to the majority of the engine components most frequently checked or serviced.

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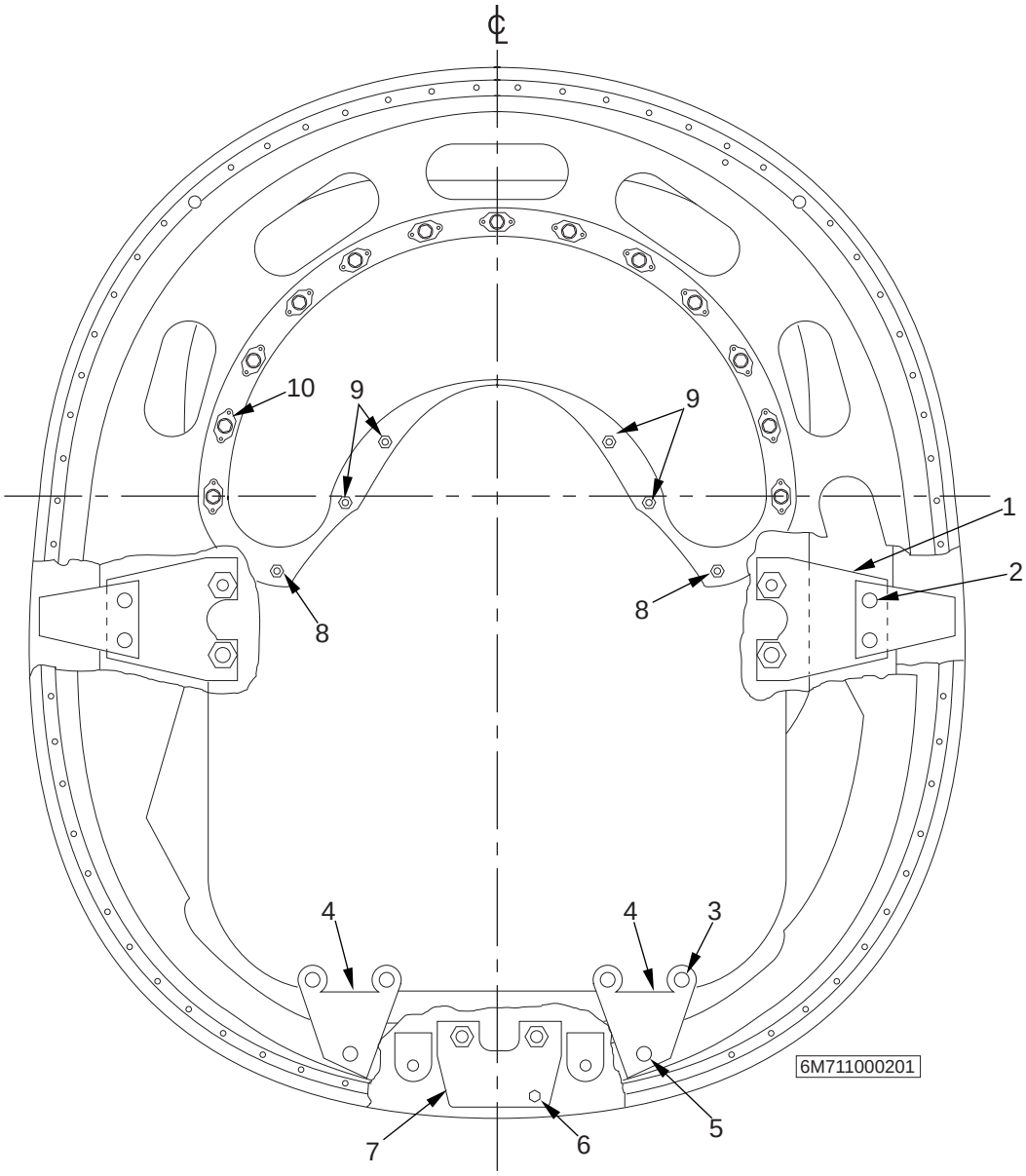

Nose Cowling
Figure 201

1. Removal/Installation – Engine Cowling

A. Removal – Nose Cowling (Figure 201)

(1) Remove propeller, see Chapter 61, Maintenance Manual.

NOTE: The propeller deicing brush blocks remain on the engine.

(2) Remove the lower forward cowling.

(3) Disconnect the anti-icing tube at the upper portion of the nose cowling assembly.

(4) Remove mounting bolts (item 10) from the rear of the engine air intake.

(5) Remove nuts (item 8) from the mounting bolts on the front of the engine air intake.

(6) Remove mount bolts (item 2) at cowling's aft frame to brackets at Lord mounts.

(7) Remove mount bolts (item 5) at cowling's center frame to brackets on lower front of reduction gear case.

(8) Remove mount bolts (item 9) aft front of engine.

(9) Move the cowling forward and remove it from the engine.

B. Installation – Nose Cowling

(1) Place nose cowling assembly into position and install mount bolts at the cowling's center frame to brackets on lower front of reduction gear case.

(2) Install mount bolts (item 9) at front of engine.

(3) Install mount bolts (item 2) at cowling's aft frame to brackets at Lord mounts.

(4) Install nuts (item 8) at front of engine air intake.

(5) Install mount bolts (item 10) on rear of engine air intake.

(6) Connect anti-ice lines.

(7) Install propeller, see Chapter 61, Maintenance Manual.

(8) Install lower forward cowling.

C. Removal - Upper Cowling Panel

(1) Remove all screw fasteners.

(2) Remove upper cowling panel.

D. Installation - Upper Cowling Panel

(1) Install upper cowling into place.

(2) Install all screw fasteners.

E. Removal - Side Cowling Panels

(1) Loosen all Camloc fasteners.

(2) Remove side panel.

F. Installation - Side Cowling Panels

(1) Install side panel into place.

(2) Secure all Camloc fasteners.

G. Removal - Lower Cowling Panel

NOTE: The lower aft cowling panel includes the oil cooler air scoop and the heated air intake duct lip. In normal removal, the air scoop will be removed with the cowling. The heating element is welded to the cowling.

(1) Motor the engine to scavenge as much oil as possible.

(2) Remove the side cowling panels.

(3) Disconnect the scoop lip heating line at the unions.

(4) On older model aircraft, loosen the latches which secure the air scoop to the base of the oil cooler.

NOTE: Reference Fairchild Aircraft Corporation Service Bulletin 71-003 for installation instructions of flexible oil cooler duct for later model aircraft.

(5) Remove the fasteners at the front and rear of the cowling, remove cowling.

NOTE: If engine is to be operated with lower cowling off, an elbow or fitting (AN833-8D) should be installed between the hoses from the oil cooler to lower cowling air scoop lip heating tube in order to transfer oil from oil cooler to oil tank.

H. Installation – Lower Cowling Panel

(1) Disconnect and remove elbow or fitting between oil hoses, which transfers oil from oil cooler to oil tank.

NOTE: If engine was to be operated with lower cowling off, an elbow or fitting would have been installed between the hoses from the oil cooler to the lower cowling air scoop lip heating tube in order to transfer oil form oil cooler to oil tank.

(2) Connect oil hoses to the air scoop lip heating tubes.

(3) Install air scoop assembly to base of oil cooler.

(4) Install lower cowling and fasteners.

CAUTION: WHEN INSTALLING LOWER COWL ENSURE OIL HOSES TO AIR SCOOP LIP HEATING TUBES ARE NOT KINKED.

2. Inspection/Check – Engine Cowling

(1) Inspect for raised rivets, repair as necessary.

(2) Inspect for cracks, repair as necessary.

(3) Inspect all welded, brazed, or soldered assemblies for security of joints, repair damage.

(4) Inspect all tubes for kinks, cracks, excessive wear, signs of corrosion or other damage, replace damaged or worn parts.

(5) Inspect all fittings for broken threads, deterioration, and cleanliness. Replace damaged parts.

(6) Inspect all fasteners for security, repair as necessary.

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ENGINE COWLING BUILDUP – MAINTENANCE PRACTICES

1. Engine Cowling Buildup

NOTE: The following procedures detail the trimming, fitting, and buildup of the engine cowling assemblies.

A. Trimming and Fitting of Nose Cowling (26-62315-059) and Frame (27-62029-021) Reference Figures 201 through 205.

(1) Install rubber seal (Figure 201, item 1) onto bottom of flange below firewall for lower cowling panel. Trim seal at each end to fit length of lower cowling panel, and trim holes into seal to provide for Camloc receptacle holes where required. Ensure flange area is free of lubricant and debris. Glue seal onto flange with 3M 947 glue, or equivalent.

(2) Enlarge cutout (see Figure 201), if required, at bottom of frame so that the frame will clear the bottom Lord mount when frame is installed.

(3) Temporarily install frame onto engine mount truss member to ensure cutout at bottom of frame clears engine Lord mount, trim as required.

(4) Install nose cowling into place. Ensure cowling fits over frame and that frame aligns with all engine mount truss member and frame attach points.

(5) Temporarily bolt nose cowling into place by installing at least four retaining bolts. Ensure that the nose cowling is securely installed in order to determine required trimming.

(6) Drill No. 10 hole through bottom engine mount clips and frame, and secure frame to clips by installing Cleco fasteners.

(7) Starting at top, backdrill through holes (seven places) of engine mount truss member (Figure 202, item 1) and frame (Figure 202, item 2) attach points. Secure frame to truss member by installing Cleco fasteners.

(8) Backdrill No. 40 pilot hole through nutplate hole of bracket and through aft frame of nose cowling assembly. Remove lower forward panel from nose cowling assembly (see Figure 203, inset A), and complete drilling of pilot hole in nose cowling aft frame assembly using No. 10 drill. Install bolt and washer securing aft frame of nose cowling to bracket.

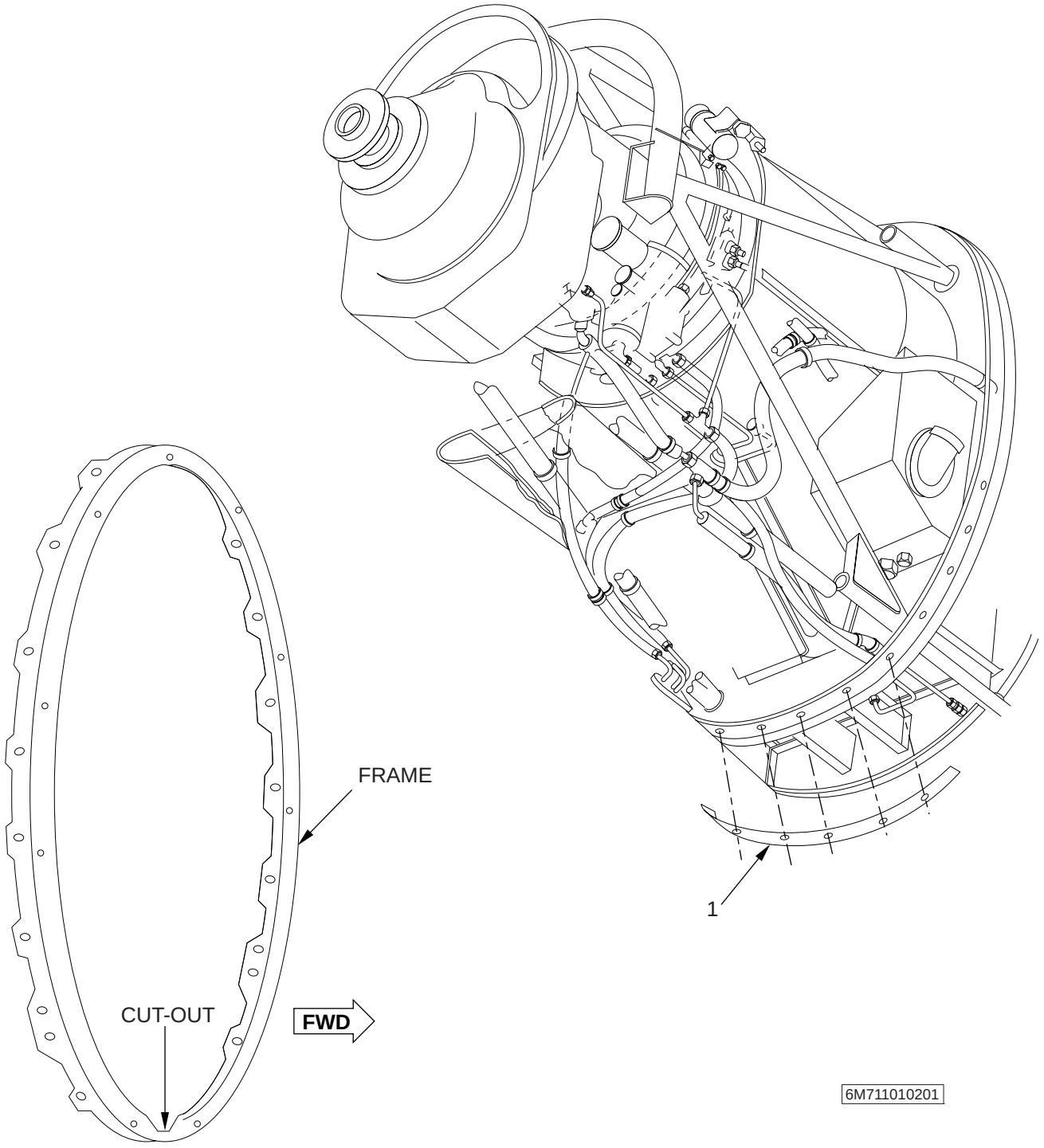
(9) Install bracket, two each, onto 26-62309-009 brackets with two retaining bolts and washers.

NOTE: Prior to installing brackets (26-62328-001), drill three equally spaced No. 40 pilot holes through each bracket so that the brackets may later be permanently riveted to the forward nose cowling assembly.

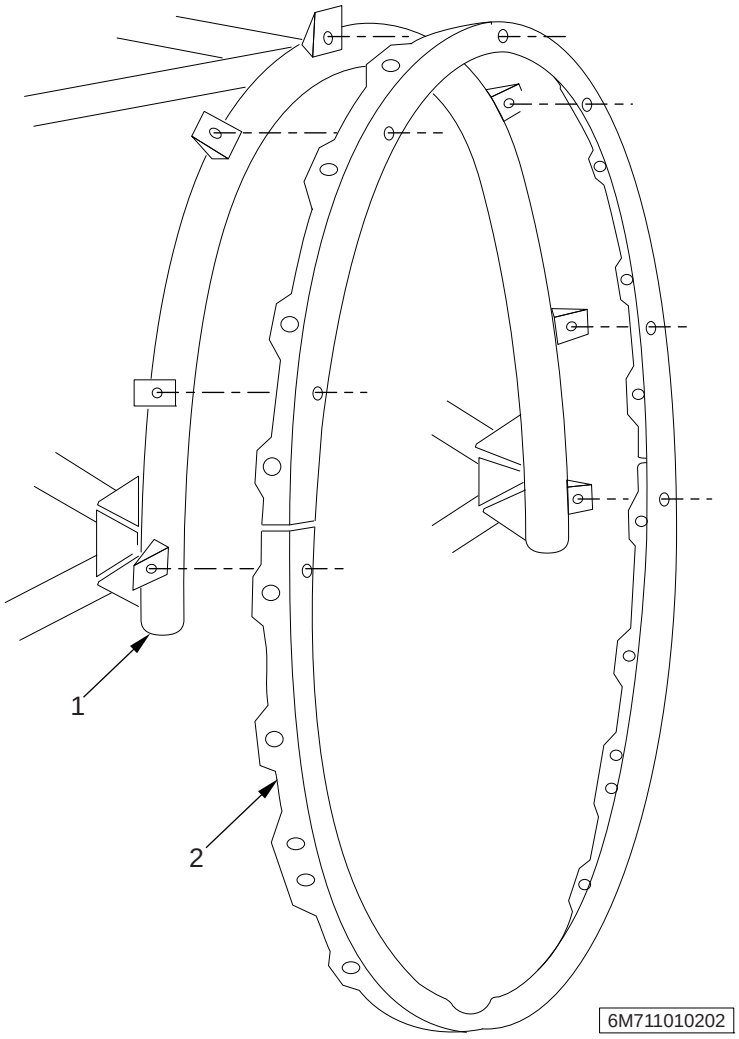
(10) With brackets (26-62328-001) installed onto 26-62309-009 brackets, drill through No. 40 pilot holes and aft frame of nose cowling assembly with No. 30 angle drill.

(11) Install one Cleco fastener onto each bracket to hold it in place.

(12) Remove retaining bolts on brackets (26-62328-001), which was securing the 26-2328-001 brackets to the 26-62309-009 brackets. Remove bolt and washer securing aft frame of nose cowling to 26-62309-013 bracket.



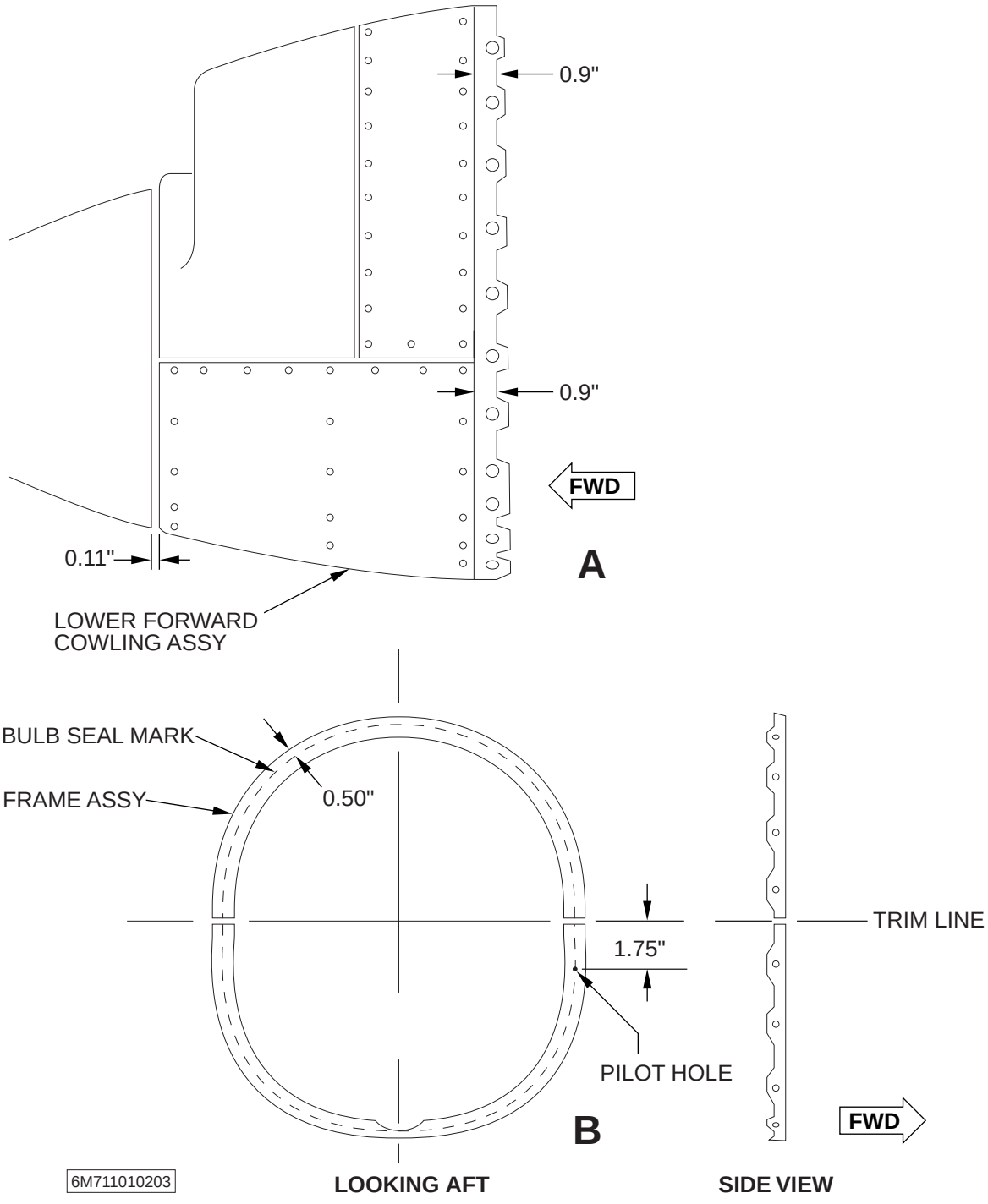
Cowling Frame Installation
Figure 201



Cowling Collar Installation
Figure 202

(13) Reinstall lower forward panel onto nose cowling assembly.

(14) Temporarily install propeller and propeller spinner (reference Chapter 61, Propeller Installation). Rotate propeller by hand and determine amount of trimming required of front of nose cowling in order to obtain 0.11 to 0.16 inches clearance between front of nose cowling and propeller spinner (see Figure 203, inset A). Mark area required for trimming onto front of nose cowling assembly, so that when nose cowling is removed the front area may be trimmed.



Nose Cowling Installation
Figure 203

(15) Determine amount of trimming required of aft portion of nose cowling in order to obtain 0.9 inch clearance from aft end of nose cowling to aft of frame (27-62029-021) below Camloc fasteners. Mark area required for trimming onto aft portion of nose cowling, so that when nose cowling is removed, the aft area may be trimmed.

(16) Remove propeller spinner, propeller, and nose cowling assembly.

(17) With nose cowling assembly removed, install rivets (three each MS20470AD44 per bracket) into brackets (26-62328-001). Rivet the brackets to the aft frame of the nose cowling assembly. Trim aft and forward areas of nose cowling where required.

(18) Complete buildup of nose cowling assembly.

(19) Make china mark around frame assembly for bulb seal and retainer installation. Mark 0.5 inch from outside circumference of frame. Mark all around frame as shown in Figure 203, inset B.

(20) To allow removal of frame (27-62029-021) without removing propeller and forward nose cowling assembly, the frame may be split at a point approximately 1.75 inches from the existing pilot hole (as shown in Figure 203, inset B). Make china split line mark at this point onto frame at both sides. Do not split frame yet during this step. NOTE: Frame assembly should be split halfway between the Camloc fasteners, as shown in side view of Figure 203, inset B.

(21) Position clips (27-62029-011) by placing clip at midpoint at frame over the china split line mark. Drill four equally spaced No. 30 rivets holes through clip and upper frame above split line. Drill two equally spaced No. 10 bolt holes through clip and frame below split line, as shown in Figure 204, inset A.

(22) Drill four No. 40 rivet holes, countersink the holes with a No. 40 countersink on the side of the clip, which is placed against the frame for fastening of two nutplates (MS21048-L3) over the bolt holes of each clip, as shown in Figure 204, inset B.

(23) Install nutplates (MS21048-L3) two each, to each clip with rivets (MS20426-3-3).

(24) Locate and drill rivet hole with No. 40 drill for each single wing nutplate (see Figure 205) to be mounted over each pilot hole (nine places) within frame (27-62029-021). Dimple each No. 40 rivet hole with 3/32 dimpler so flush head of rivet sets on aft side of frame. Install single wing nutplate (MS21080-3K) with MS2042-3-3 rivet.

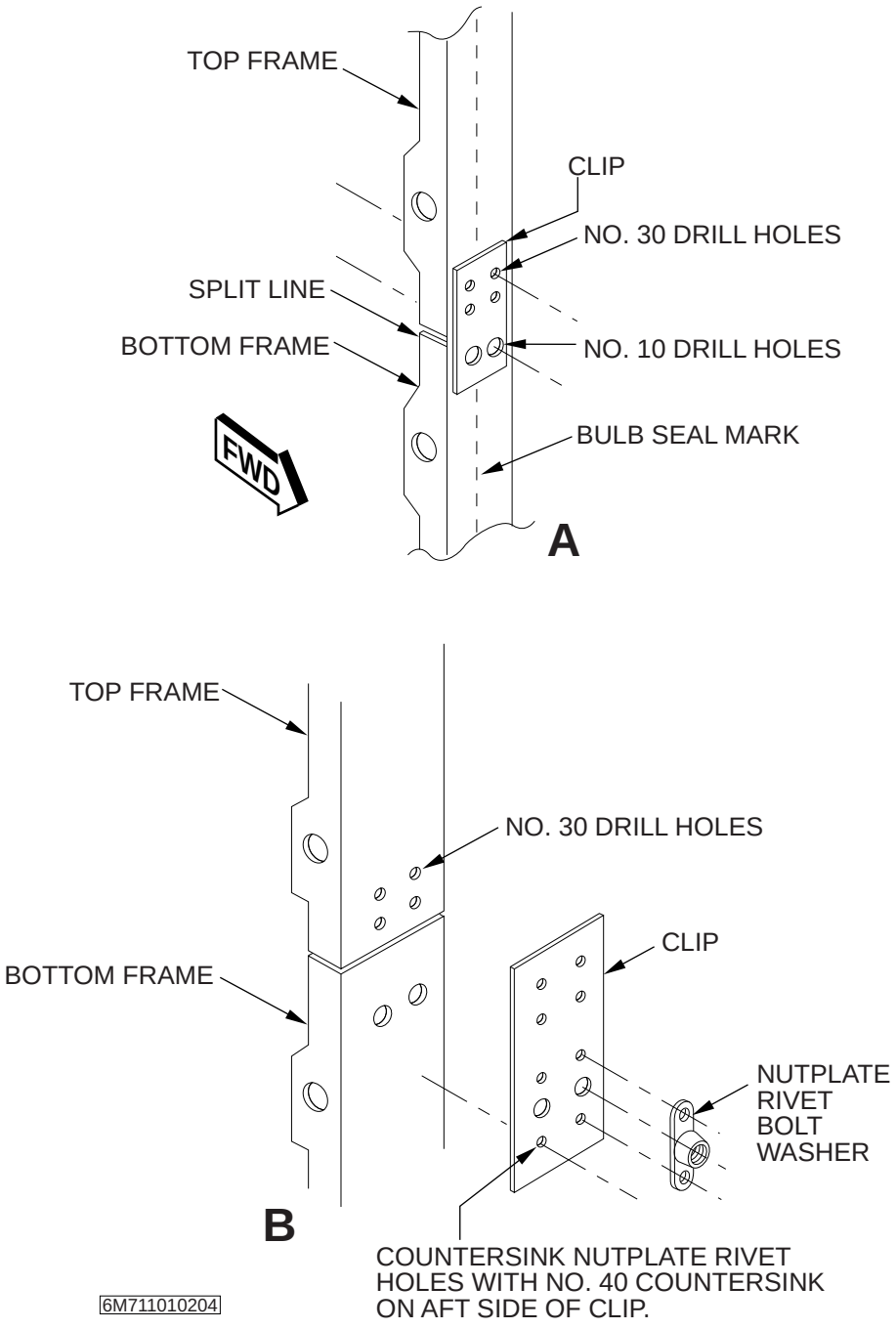
(25) Split the frame at the split line mark made during step (20). File frame ends to obtain clearance between where the frame halves will join again when clip is installed.

(26) Install clips (27-62029-011) to top portion of each side of frame at split line. Fasten each clip with two brazier head rivets (MS20470AD44) each clip at location as shown in Figure 204, inset B.

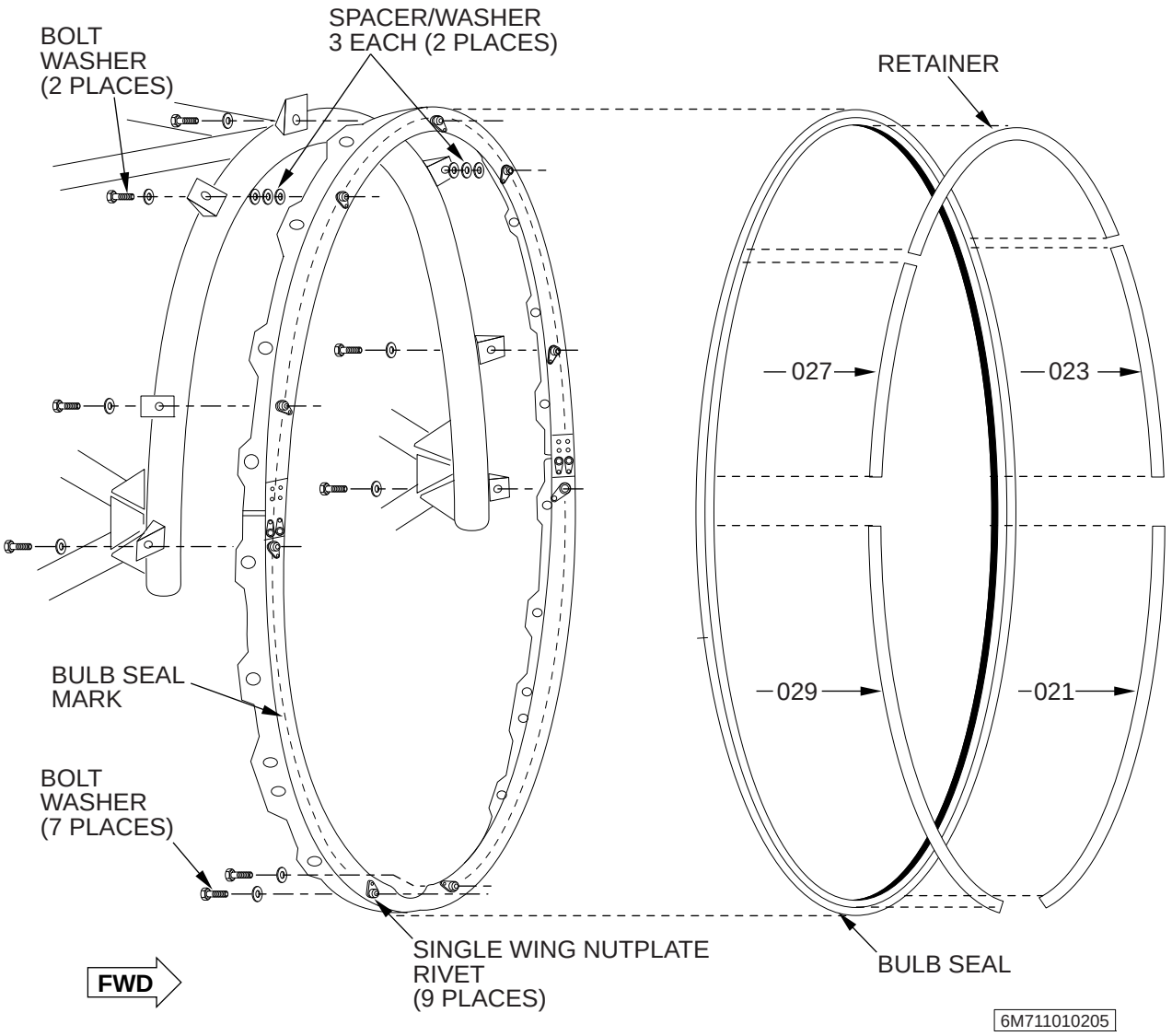
(27) Join the top to the bottom frame. Install bolts (AN3-3A) and washers (AN960-10) through the frame and the clip at bottom half of frame (see Figure 204, inset B).

(28) Install top bulb seal retainer (26-623325-025) into place centering the pilot holes within the retainer along the previously marked china line on the frame. Drill through pilot holes and frame with No. 30 drill (see Figure 205). Install Cleco fasteners to hold retainer into place.

NOTE: Cut retainers as necessary to clear nutplates on frame.



Cowling Clip Installation
Figure 204



Cowling Retainer Installation
Figure 205

(29) Install retainers (26-62332-025 and 26-62332-027) into place. Center pilot holes of retainers along marked china line, and drill through pilot holes and frame with No. 30 drill. Install Cleco fasteners.

NOTE: Allow space between retainers, and trim retainer ends where necessary to clear nutplates and clips at frame split line.

(30) Install retainers (26-62332-021 and 26-62332-029) into place; drill holes and install Cleco fasteners.

(31) Install bulb seal (approximately 8 1/2 feet of P/N 1380 seal) by starting at the clip of the frame split line, removing Cleco fastener, inserting seal between frame and retainers, and reinserting Cleco fastener.

NOTE: Install seal so that the bulb of seal faces outer circumference of frame.

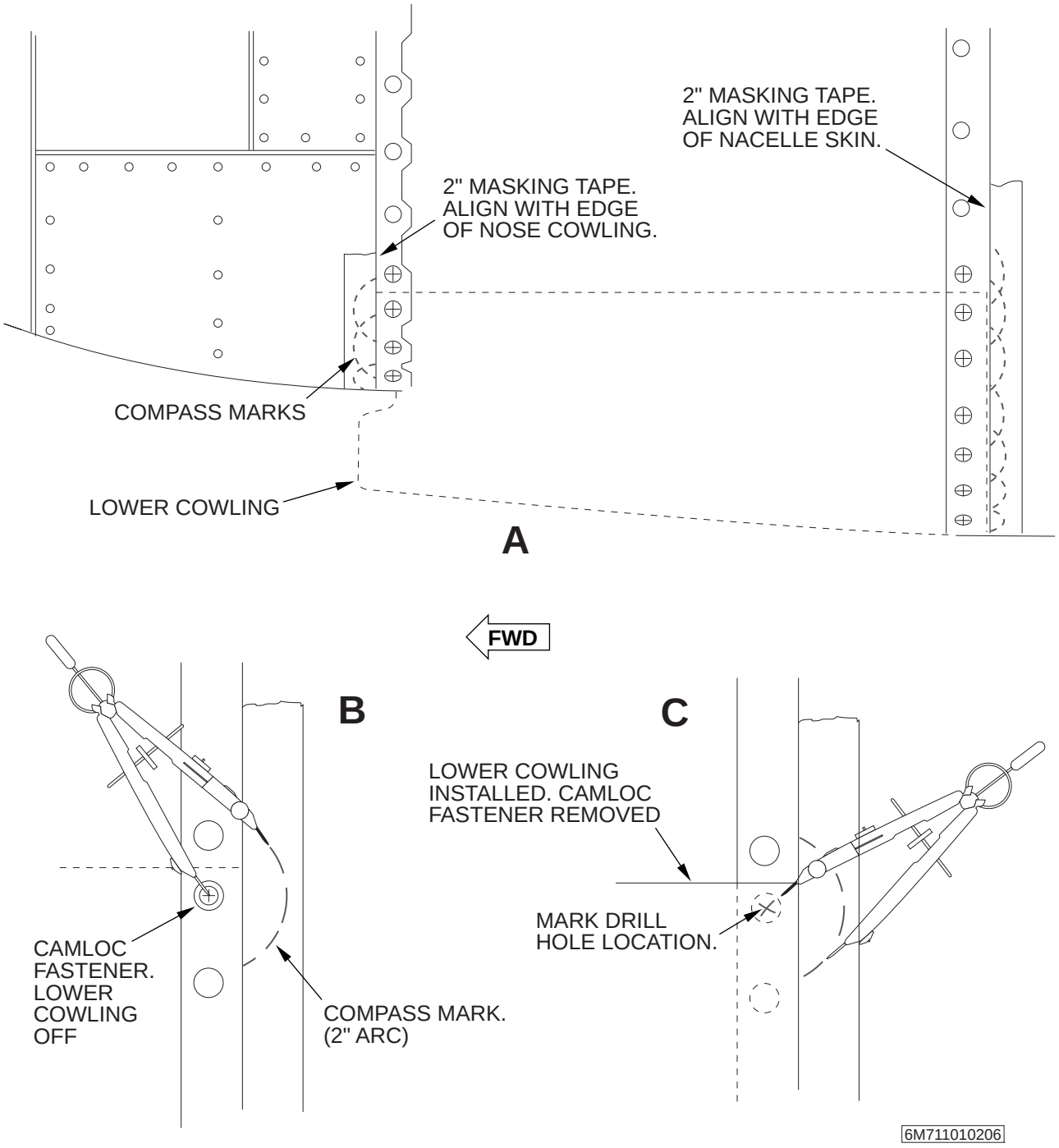
(32) With bulb seal in place, drill out the holes at the retainer pilot holes and through the bulb seal with No. 30 drill.

(33) Install pop rivets (MS20600AD4W2) from the aft side of frame through bulb seal and retainer (approximately 69 places).

NOTE: On the two pop rivets for the bulb seal located over each clip at the frame split line, (reference Figure 204, inset B), install washer (AN960-6L) on the back of each rivet before bucking rivet in order to hold bulb seal in place.

(34) Install frame assembly onto engine truss member using seven bolts (AN3-3A) and washers (AN960-10), two bolts (AN3-5A), washers (AN960-10), and six spacers/washers (AN970-3).

NOTE: Three spacers/washers (AN970-3) are placed between the left and right upper ears of the engine truss member and frame assembly, see Figure 205.



Lower Cowling Installation
Figure 206

B. Drilling and Trimming – Lower Cowling (Reference Figures 206 and 207)

(1) In preparation for locating and drilling of Camloc fastener holes in lower cowling with nose cowling assembly installed, place strip of two inch masking tape around bottom of engine nacelle skin at the firewall area. Carefully align edge of tape with edge of nacelle skin, see Figure 206, inset A. Place strip of two inch masking tape around bottom of nose cowling. Carefully align edge of tape with aft edge of nose cowling, as shown in Figure 206, inset A.

(2) With lower cowling off, install at lower cowling Camloc fasteners into their respective receptacles (fasteners will be used as center points for determining Camloc fastener hole locations).

NOTE: The lower cowling uses 15 Camloc fasteners at its aft end and six Camloc fasteners at its forward end. However, install an additional Camloc fastener above each corner of the lower cowling (four places) so that 17 fasteners will be installed at the lower cowling's aft end but eight fasteners at its forward end. Therefore, 17 holes will be marked, see step (3), but only 15 will be drilled at the aft end of cowling, and eight holes will be marked, but only six drilled at its forward end. The extra Camloc fastener markings will later be used to determine amount of trimming required of the lower cowling's sides.

(3) As shown in Figure 206, inset B, adjust compass for two inches, and using Camloc fastener at center point, make circumference mark onto masking tape. Do for all installed Camloc fasteners.

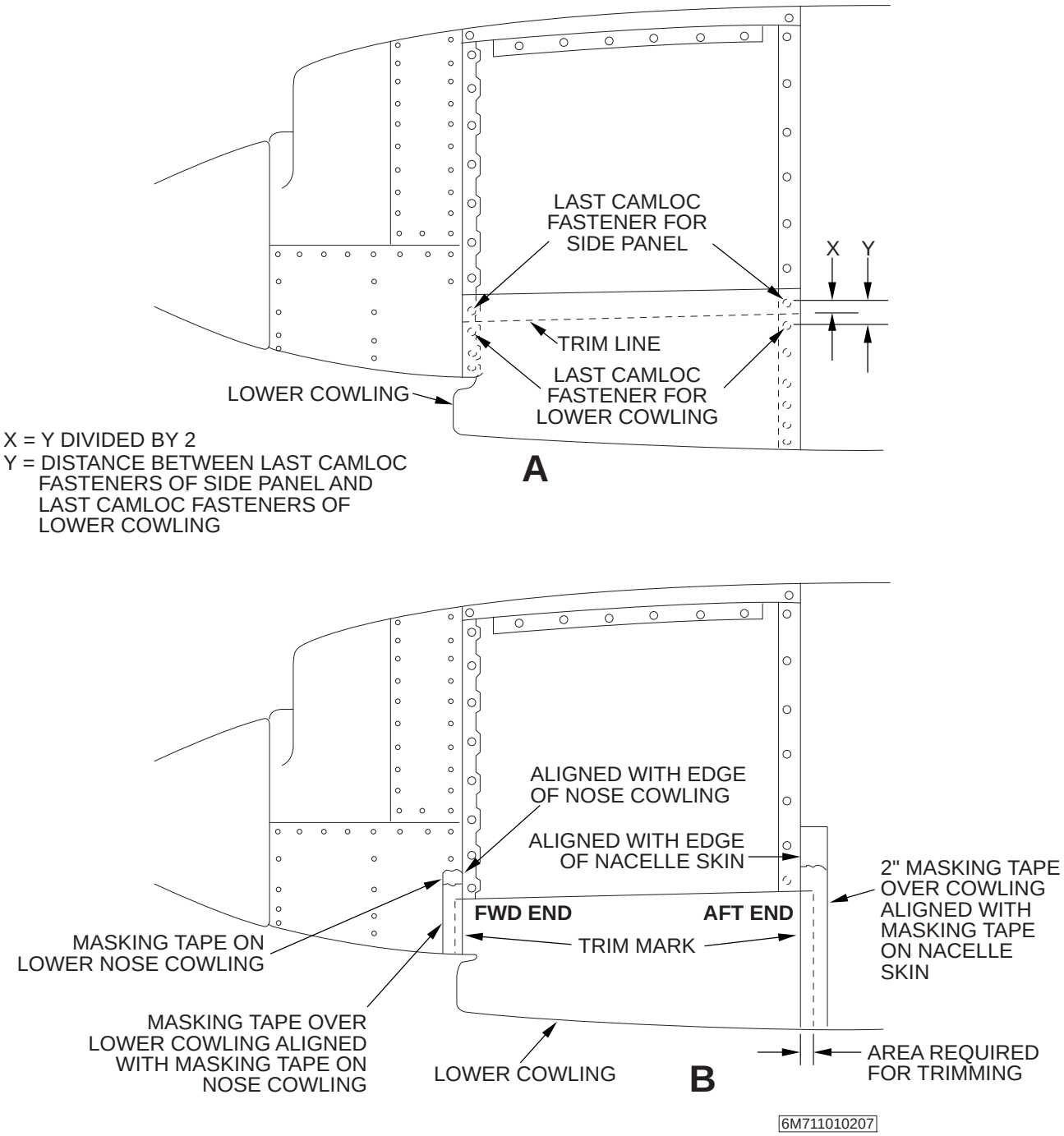
(4) Remove all Camloc fasteners.

(5) Install lower cowling into position, and secure into place by placing a seat belt type strap (or suitable equivalent) around the lower cowling securing the strap to the engine truss member. NOTE: Ensure lower cowling (1) is not coming into contact with fire detector at lower forward right side of engine, (2) has sufficient equal overlap at both the forward and aft ends of panel to cover aft edge of nose cowling assembly, as well as the nacelle fuselage skin at the opposite end, (3) center weld at scoop area on forward end of lower cowling is aligned with center of bottom engine mount.

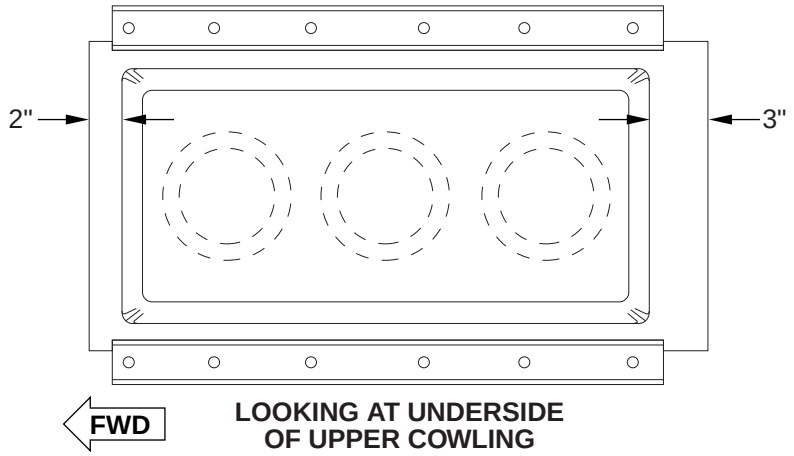
(6) As shown in Figure 206, inset C, adjust compass for two inches. Using circumference mark made on masking tape, make Camloc fastener hole location marks onto the lower cowling panel beginning at the panel's aft edge center Camloc fastener, working outwards, drilling Camloc fastener hole with 15/32 spot face drill, inserting Camloc fastener into each newly drilled hole in order to pull panel down in place before proceeding on to the next Camloc fastener. Complete drilling and installation of Camloc fasteners for remainder of aft edge of panel and, using same procedure as described above, complete Camloc fastener drilling and installation of forward edge of panel.

(7) With lower cowling installed , as shown in Figure 207, inset A, determine the distance between the Camloc fastener for the left and right panels on each side, and the last Camloc fasteners of the lower cowling. Split this distance, and this will be the amount to be trimmed along the lower cowling at its upper sides.

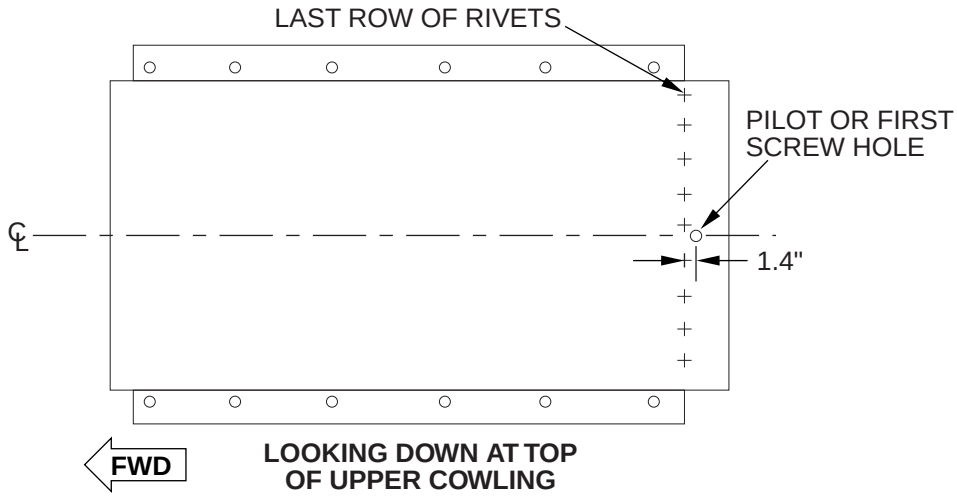
NOTE: Use straight edge to make trim mark on both sides of lower upper sides (reference Figure 207, inset A).



Cowling Installation
Figure 207



A



B

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Cowling Installation
Figure 208

(8) To trim forward and aft ends of lower cowling, place strips of two inch masking tape over edges of forward and aft ends of the lower cowling. Align the edge of the tape with the edge of the tape previously installed on both the nose cowling aft edge and lower nacelle skin edge at firewall area, as shown in Figure 207, inset B. NOTE: Ensure tape being used comes from same roll or that tape used is of exact equal width.

(9) Make required trim mark at the opposite edge of masking tape onto the cowling panel at forward and aft panel ends, as shown in Figure 207, inset B.

(10) Remove lower cowling, and trim excess from both its ends and sides. NOTE: Masking tape may be removed.

(11) Reinstall lower cowling by securing it with all fasteners. Ensure that a gap of between 0.02 and 0.04 inches exists between the aft end of lower cowling and nacelle skin. Ensure that a gap of between 0.18 to 0.25 inches exists between forward end of lower cowling and nose cowling. Remove lower cowling, and trim as required.

(12) Complete buildup of lower cowling, see Lower Cowling Buildup Procedures within this chapter.

C. Drilling and Trimming - Upper Cowling Assembly (Reference Figures 208 through 210)

NOTE: Steps (1) through (11) detail the procedures for trimming and fitting the upper cowling panel with the side cowling panels installed. However, if the side cowling panels also need trimming and fitting and are not available for installation, steps (12) through (20) detail the procedures for upper panel installation without side cowling panels installed.

(1) Accomplish initial trimming of upper cowling assembly by trimming approximately three inches from the aft end of the panel and two inches from the forward end of the panel between the indicated dimensions at its aft and forward ends, as shown in Figure 208, inset A.

NOTE: The upper cowling assembly consists of a panel with a Camloc fastener channel assembly attached to its sides. A slight incline at one end of each channel indicates the forward end of the upper cowling assembly.

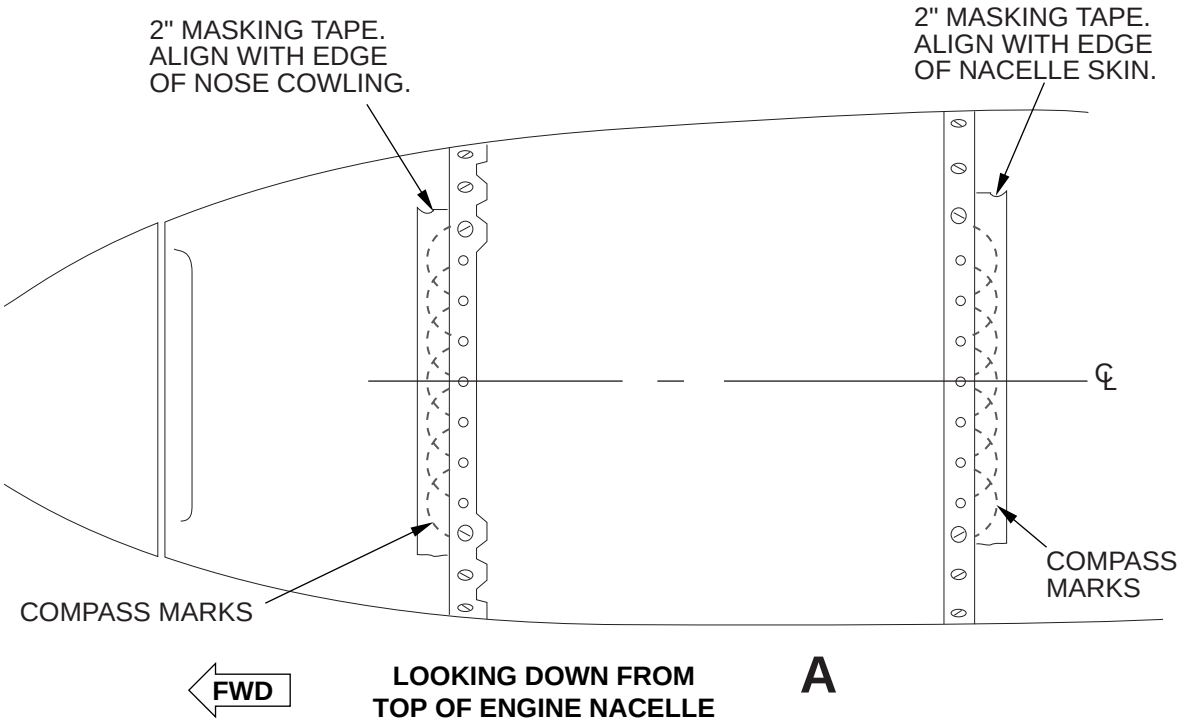
(2) With the upper cowling off and side cowlings installed, install all 14 screws into the screw receptacles for the upper cowling. (Screws will be used as centerpoints for determining screw hole locations for drilling of upper cowling panel.)

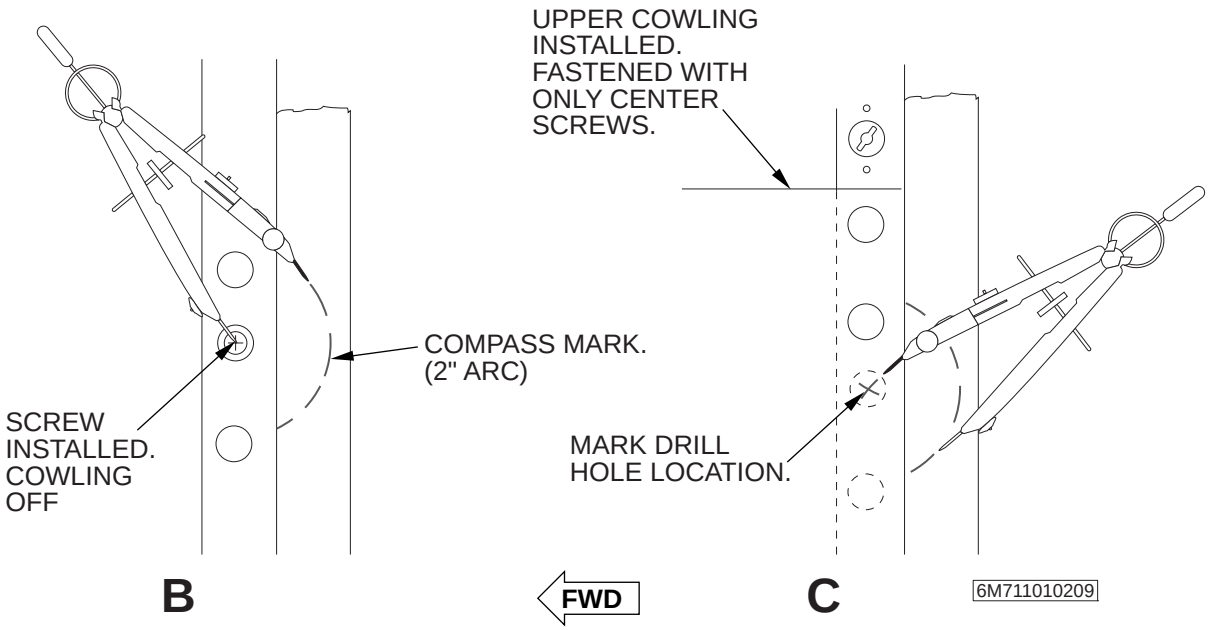
NOTE: Seven screws are required for each end of upper cowling panel.

(3) Place strip of two inch masking tape at top of engine nacelle skin at the firewall area. Carefully align edge of tape with edge of nacelle skin (see Figure 209, inset A). Place strip of two inch masking tape at top of nose cowling. Carefully align edge of tape with edge of nose cowling, as shown in Figure 209, inset A.

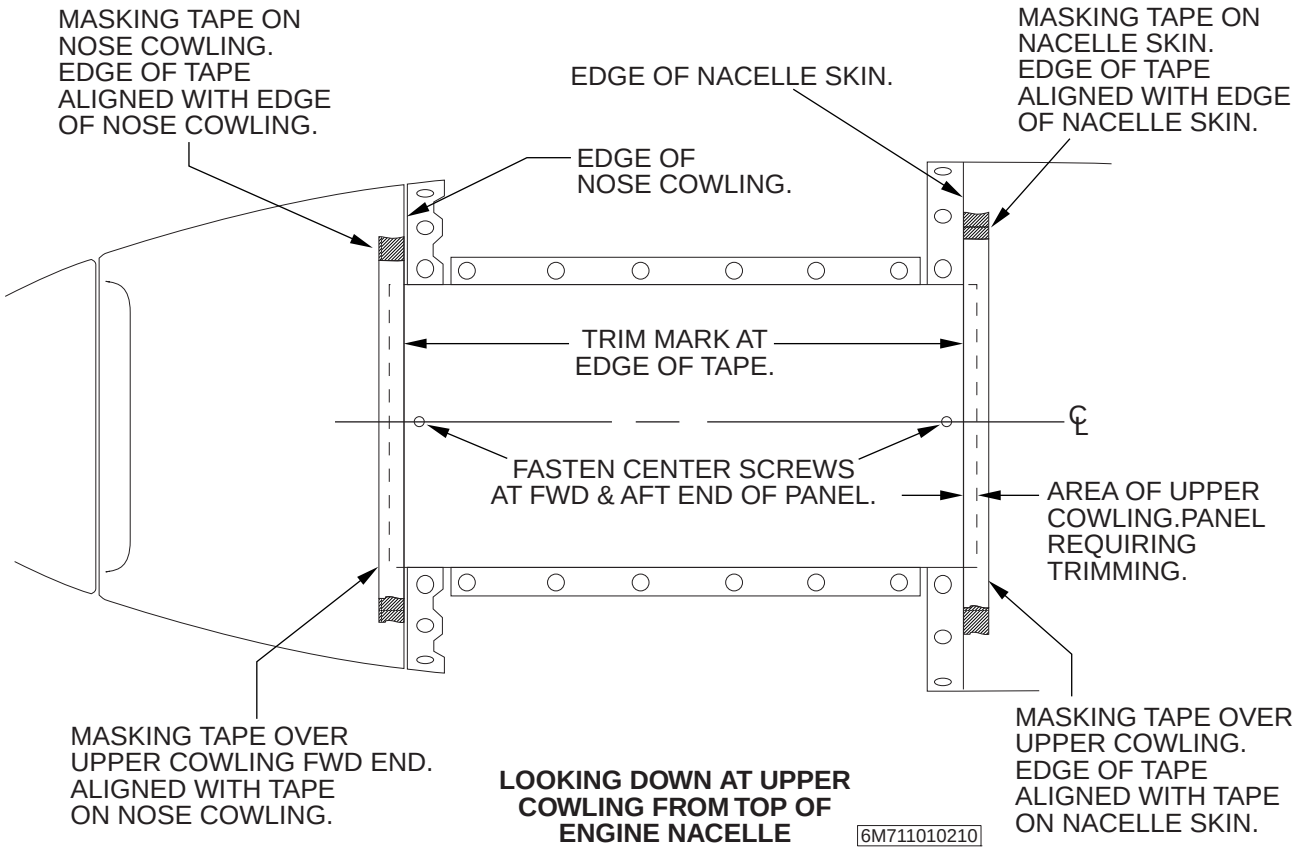
(4) As shown in Figure 209, inset B, adjust compass for two inches, and using installed screw as centerpoint, make circumference mark onto masking tape. Do for all screw fasteners.

(5) Remove screw fasteners. Install upper cowling into position, and fasten panel to side cowlings.





Cowling Installation
Figure 209



Cowling Installation
Figure 210

(6) Place strips of two inch masking tape over edges of forward and aft ends of upper cowling panel aligning the edge of the tape with edge of tape previously installed, see step (3) on both the nose cowling aft edge and the upper nacelle skin edge, as shown in Figure 210.

NOTE: Ensure tape being used comes from same roll or that the tape being used is of exact width.

(7) Make required trim mark at the opposite edge of masking tape onto the cowling panel at its forward and aft ends, as shown in Figure 210.

(8) Remove upper cowling panel, and trim excess from both its ends where required.

NOTE: Masking tape over panel ends may be removed. Leave masking tape on nacelle skin and tape on the nose cowling in place.

(9) Reinstall upper cowling, and fasten it to the side cowling panels. Ensure gap of between 0.02 to 0.04 inches exists between aft end and upper cowling panel and nacelle skin, and ensure 0.18 to 0.25 inches exists between forward end of upper cowling panel and aft edge of nose cowling, remove panel, and trim/file if required.

(10) Reinstall upper cowling panel, and fasten to side cowling panels. Adjust compass for two inches, and using circumference marks made on masking tape, make screw fastener hole location marks onto upper cowling panel; beginning at the panel's center screw hole drilling hole with No. 19 drill, working outward; installing retaining screw into each newly drilled hole in order to pull panel down into place as the work proceeds. Complete drilling of all holes and installation of all fasteners for remainder of panel edge and, using same procedure as described above, complete fastener drilling and installation of opposite edge of panel.

(11) Remove upper cowling panel, and complete its buildup. See Cowling Buildup Procedures within this chapter.

NOTE: Steps (12) through (20) Trimming and Fitting of Upper Cowling Panel Without Side Cowling Panels Installed

(12) Accomplish initial trimming of upper cowling assembly, as described in step (1).

(13) If no pilot hole exists at aft end of new upper cowling panel, determine centerline of panel. Make mark for first screw hole (pilot hole) to be drilled by measuring 1.4 inches aft of panel's last row of rivets at centerline, as shown in Figure 208, inset C. Drill hole with No. 19 drill.

(14) With upper cowling off, install all 14 screws into the screw receptacles for the upper cowling. (Screws will be used as centerpoints for determining screw hole locations when drilling upper cowling panel, see step (19).

NOTE: Seven screws are required for each end of upper cowling panel.

(15) Accomplish steps (3) and (4).

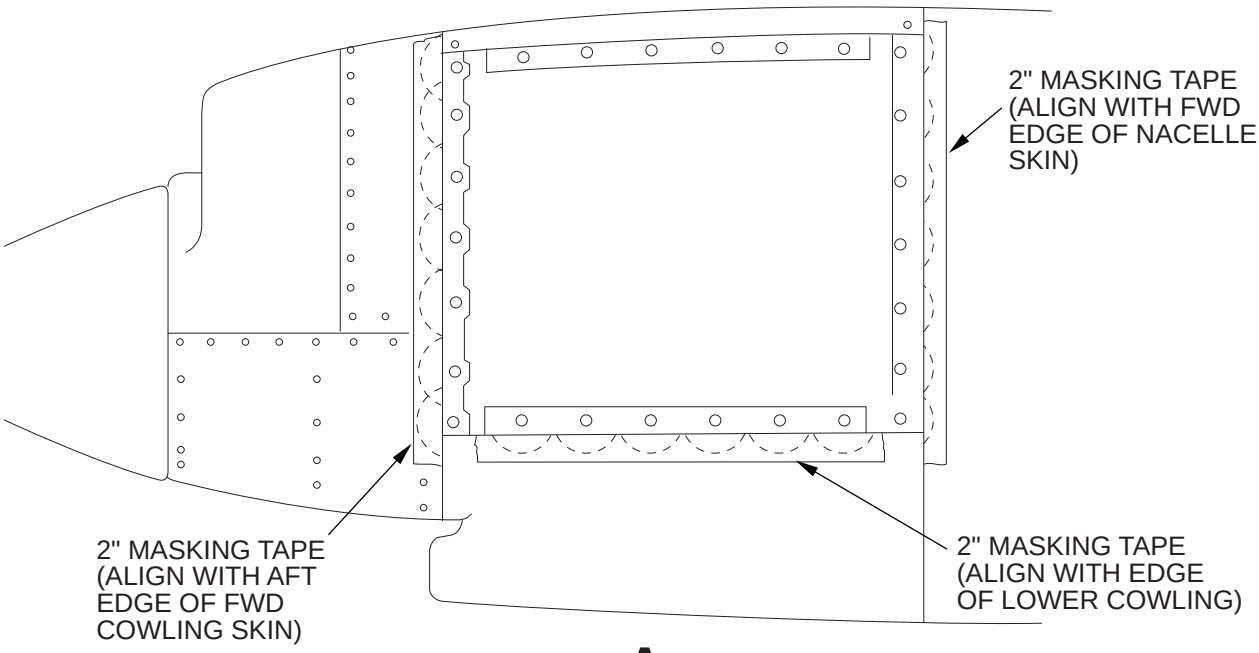
(16) Remove screw fasteners. Install panel into position, and fasten panel by installing the one screw at the center aft end of panel. Locate screw hole location at centerline for the center forward end of the panel; drill No. 19 hole, and install screw fastener securing forward end of panel.

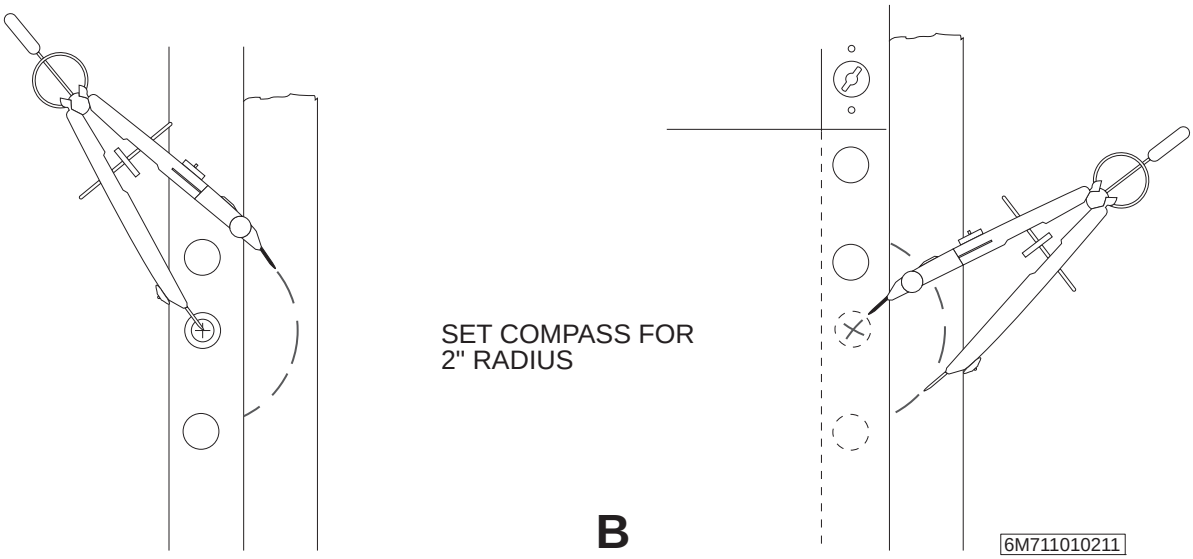
(17) Accomplish steps (6) through (8).

(18) Reinstall cowling panel securing it with screw fasteners at both its ends. Ensure gap of between 0.02 to 0.04 inches between aft end of panel and nacelle skin. Ensure 0.18 to 0.25 inches between forward end of panel and nose cowling. Remove panel and trim/file if required.

(19) Reinstall upper cowling, and install the screw fasteners at each end of panel, as shown in Figure 209, inset C. Adjust compass for two inches, and using circumference marks made on masking tape, make remaining screw fastener hole location marks on to upper panel; beginning from the panel's center screw hole, drilling hole with No. 19 drill, working outwards; installing retaining screw into each newly drilled hole in order to pull panel down into place as the work proceeds. Complete drilling of all holes and installation of all fasteners for remainder of panel edge and, using same procedures as described above, complete fastener drilling and installation of opposite edge of panel.

(20) Accomplish step (11).





Engine Cowling Trim
Figure 211

D. Drilling and Trimming - Left and Right Cowling Panels (Reference Figures 211 and 212)

NOTE: Camloc fasteners consists of the 4002-N collar with the 40S-4 Camloc stud installed in it with an R4G lockring. The R4G lockring is not used until the Camloc's are installed permanently in the finished cowling.

(1) With all cowlings except side cowlings installed, install all side cowling Camloc fasteners into their receptacles. (Camloc fasteners will be used as centerpoints for determining Camloc fastener hole locations when drilling side panels, see step (3).

(2) Place strip of two inch masking tape at side of engine nacelle skin at firewall area. Carefully align edge of tape with edge of nacelle skin, as shown in Figure 211, inset A. Place strip of two inch masking tape at sides of nose cowling, as shown in Figure 211, inset A. Carefully align edge of tape with aft edge of nose cowling. Place strip of two inch masking tape at sides of lower cowling assembly, as shown in Figure 211, inset A. Place two inch strip of masking tape at sides of upper cowling assembly, as shown in Figure 212, inset A.

(3) As shown in Figure 211, inset B, adjust compass for two inches and using installed Camloc fasteners as centerpoint, make circumference mark onto masking tape. Do this for all fasteners; then, remove Camloc fasteners.

(4) Install side panel into place, as shown in Figure 211, inset C. Adjust compass for two inches, and using circumference marks on masking tape, locate and make Camloc fastener hole mark by picking up the center Camloc fastener holes at both the forward and aft sides of the panel.

NOTE: Position panel such that the row of rivets around the inner side of the panel do not come into contact with Camloc receptacle frame at any point on the front, top, or rear.

When fitting left side panel, position panel such that easy accessibility is available to the oil tank filler cap through the side panel's oil filler access door.

Two seat belt type straps, positioned front and back on the panel to hold it in place, will facilitate easier maintenance.

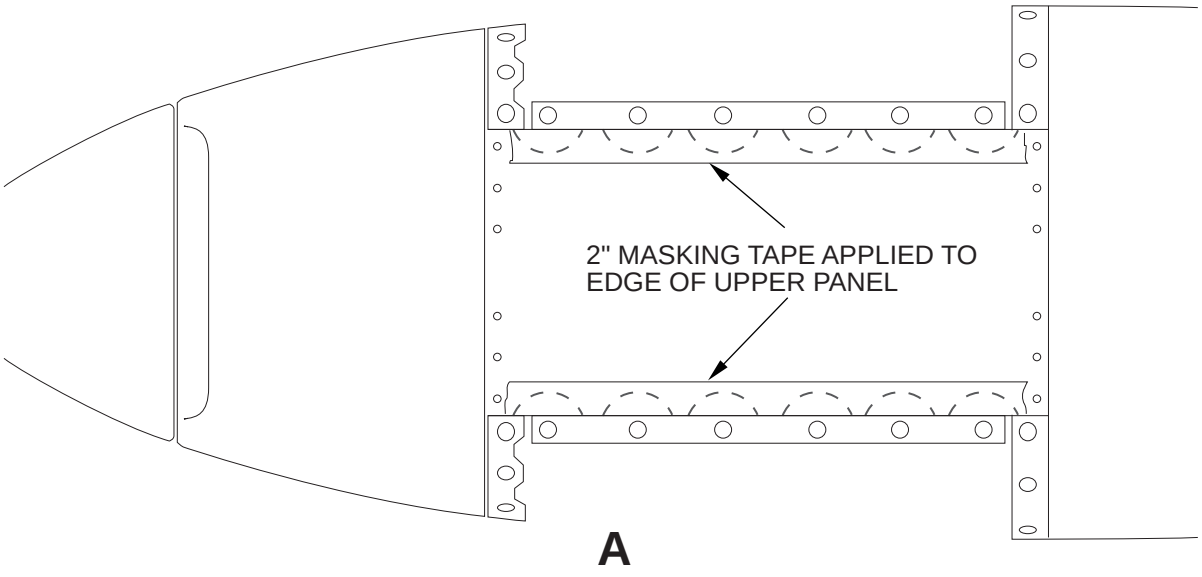
(5) Drill the center Camloc fastener hole at the forward and aft sides of panel using 15/32 spot face drill.

(6) Fasten side panel into place with center aft and center forward Camloc fasteners.

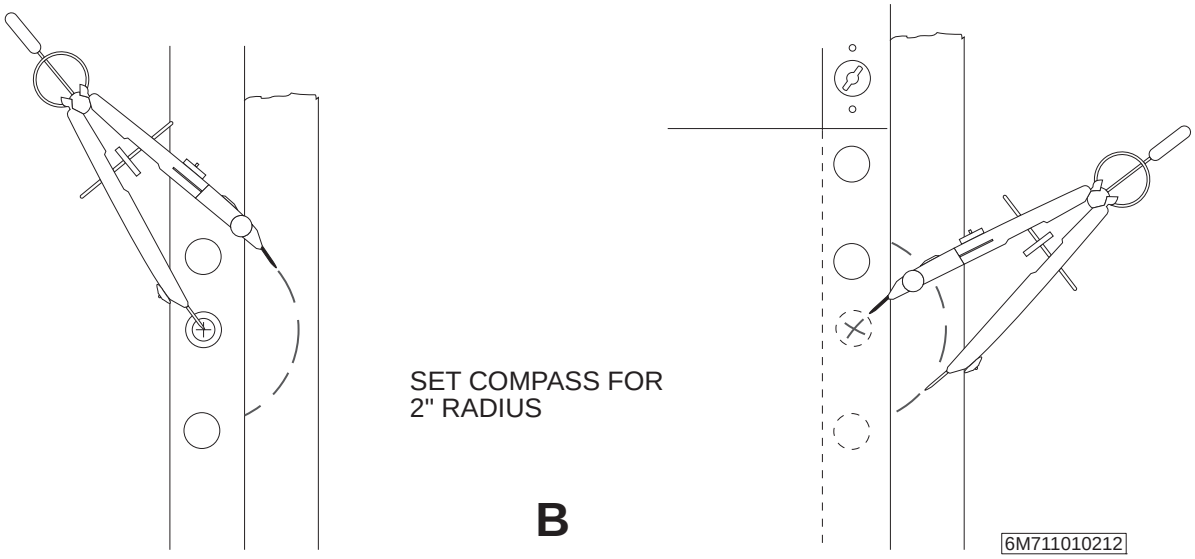
(7) To trim the forward and aft ends of the side panel, place strip of two inch masking tape over edges of forward and aft ends of panel. Align the edge of the tape with the edge of the tape previously installed on both the nose cowling aft edge and the lower nacelle skin edge, as shown in Figure 213.

NOTE: Ensure tape being used comes from same roll, or ensure ensure that the tape used is of exact equal width.

(8) Make required trim mark at the opposite edge of masking tape onto the side panel at its forward and aft edges, as shown in Figure 213.

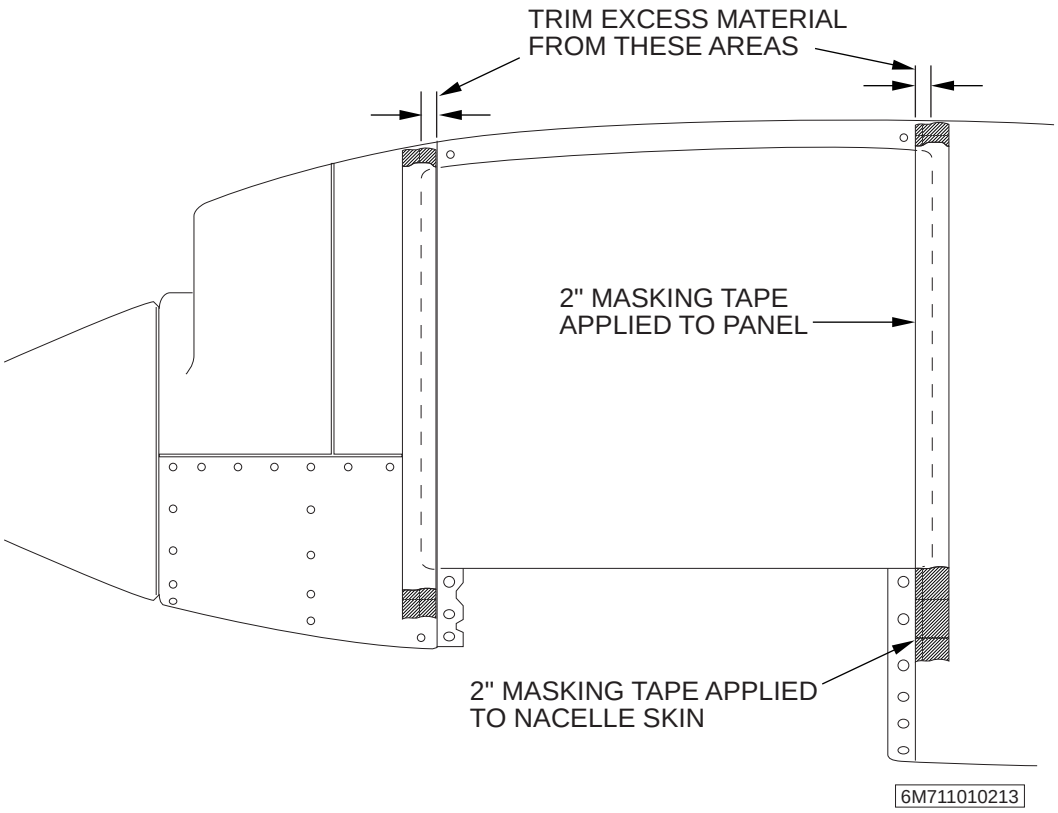


A



B

Engine Cowling Trim
Figure 212



Engine Cowling Trim
Figure 213

(9) Remove side panel, and trim excess from both its aft and forward edges.

NOTE: Do not trim front of panel further back than 0.70 inches from the center of the front Camloc fasteners.

Masking tape over panel edges may be removed, leave masking tape on the sides of nacelle skin and nose cowling in place.

(10) Reinstall side panel fastening in place with center and forward Camloc fastener. Ensure gap of between 0.02 to 0.04 inches exits between aft end of side panel and nacelle skin, and ensure gap of between 0.18 to 0.25 inches exits between forward edge of panel and nose cowling. Remove panel and trim if required.

(11) With side panel installed in place, complete drilling of Camloc fastener holes along its aft side. Adjust compass for two inches, and using circumference marks made on masking tape, make Camloc fastener hole location marks on side panel; beginning from aft center Camloc fasteners, working outwards, drilling Camloc fastener hole with 15/32 spot face drill; inserting Camloc fastener into newly drilled hole in order to pull panel down in place before proceeding on to the next Camloc fastener. Complete drilling and installation of Camloc fastener for remainder of aft edge of panel and, using same procedure as described above, complete Camloc fastener drilling and installation of forward edge of panel.

(12) To trim top of side panel, as shown in Figure 214, inset A, place strip of two inch masking tape over top edge of side panel by aligning the edge of the tape with edge of tape previously installed on the upper panel, as shown in Figure 214, inset B. Make required trim marks at opposite edge of masking tape onto the side panel, as shown in Figure 214, inset B. Remove side panel and trim excess from its top edge.

NOTE: Masking tape over side panel's top edge may be removed, leave masking tape on cowling panel in place.

(13) Reinstall side panel by securing it with its Camloc fasteners at its forward and aft sides. Ensure gap of between 0.02 to 0.04 inches exist between top edge of side panel and edge of upper cowling panel, remove side panel and trim if required.

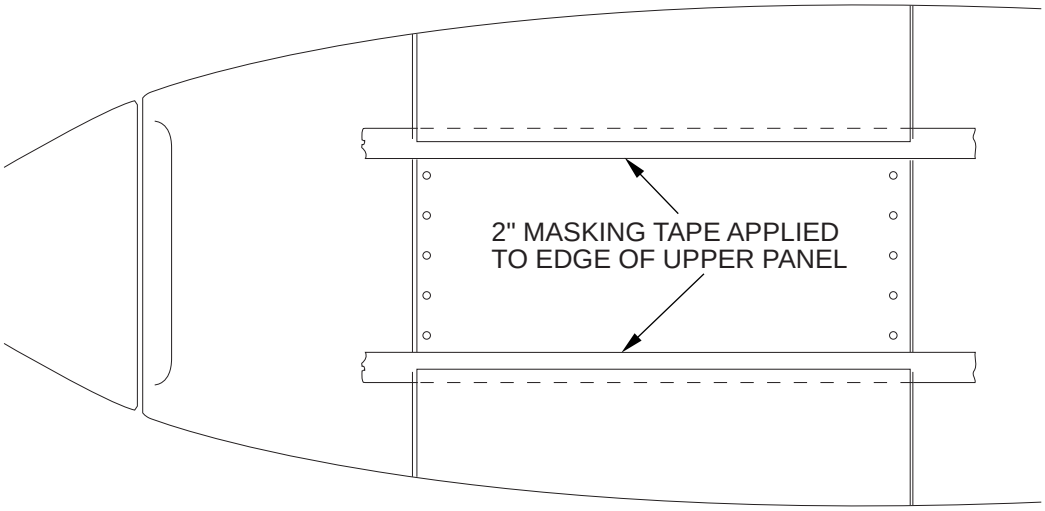
(14) With side panel installed in place, adjust compass for two inches, and using circumference marks made on masking tape on upper cowling, as shown in Figure 213, inset A, make remaining Camloc fastener hole location marks onto top edge of side panel, as shown in Figure 212, inset B. Start with hole nearest center and work outwards. Drill Camloc fastener hole with 15/32 spot face drill; insert Camloc fastener in each newly drilled hole in order to pull panel down in place before proceeding on to the next Camloc fastener. Complete drilling and installing of Camloc fasteners for remainder of top edge of side panel using same procedure as described above.

(15) To trim and drill bottom edge of side cowling with lower cowling installed, place strip of two inch masking tape over bottom edge of side panel, as shown in Figure 215, inset A. Align the edge of the tape with edge of tape previously installed on the lower cowling assembly's top edge, as shown in Figure 215, inset B, see step (2).

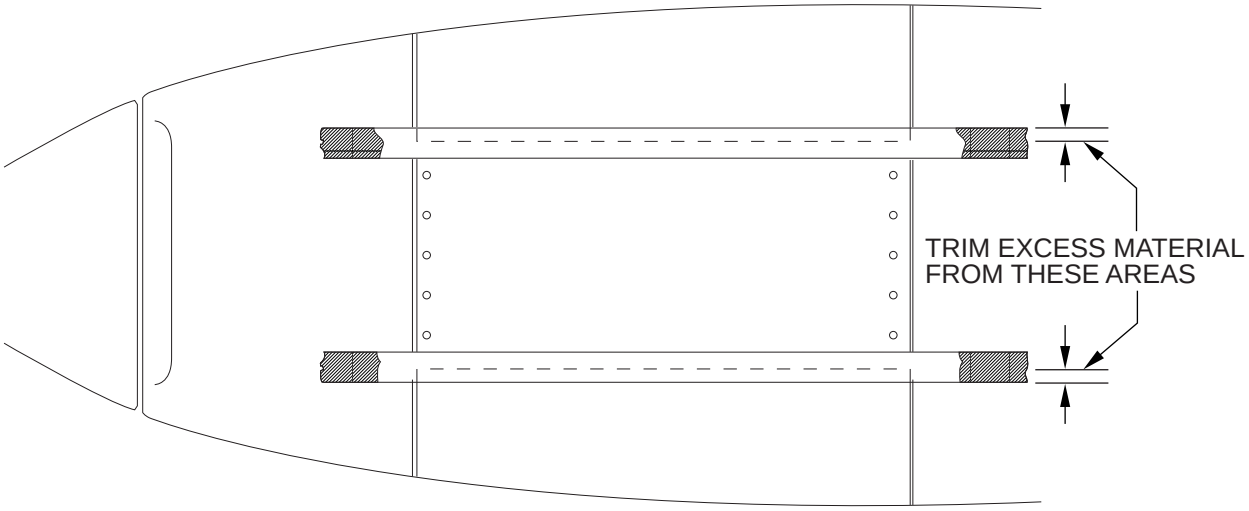
(16) With side panel installed, make required trim mark at the opposite edge of masking tape onto the side panel at its bottom edge, as shown in Figure 215, inset B. Remove side panel, and trim excess from its bottom edge.

NOTE: Masking tape over side panel bottom edge may be removed, leave masking tape on top edge of lower cowling in place.

(17) Reinstall side panel, and secure all fasteners. Ensure gap of between 0.02 to 0.04 inches exists between bottom edge and side panel and top edge of lower cowling assembly. Remove side panel, and trim if required.

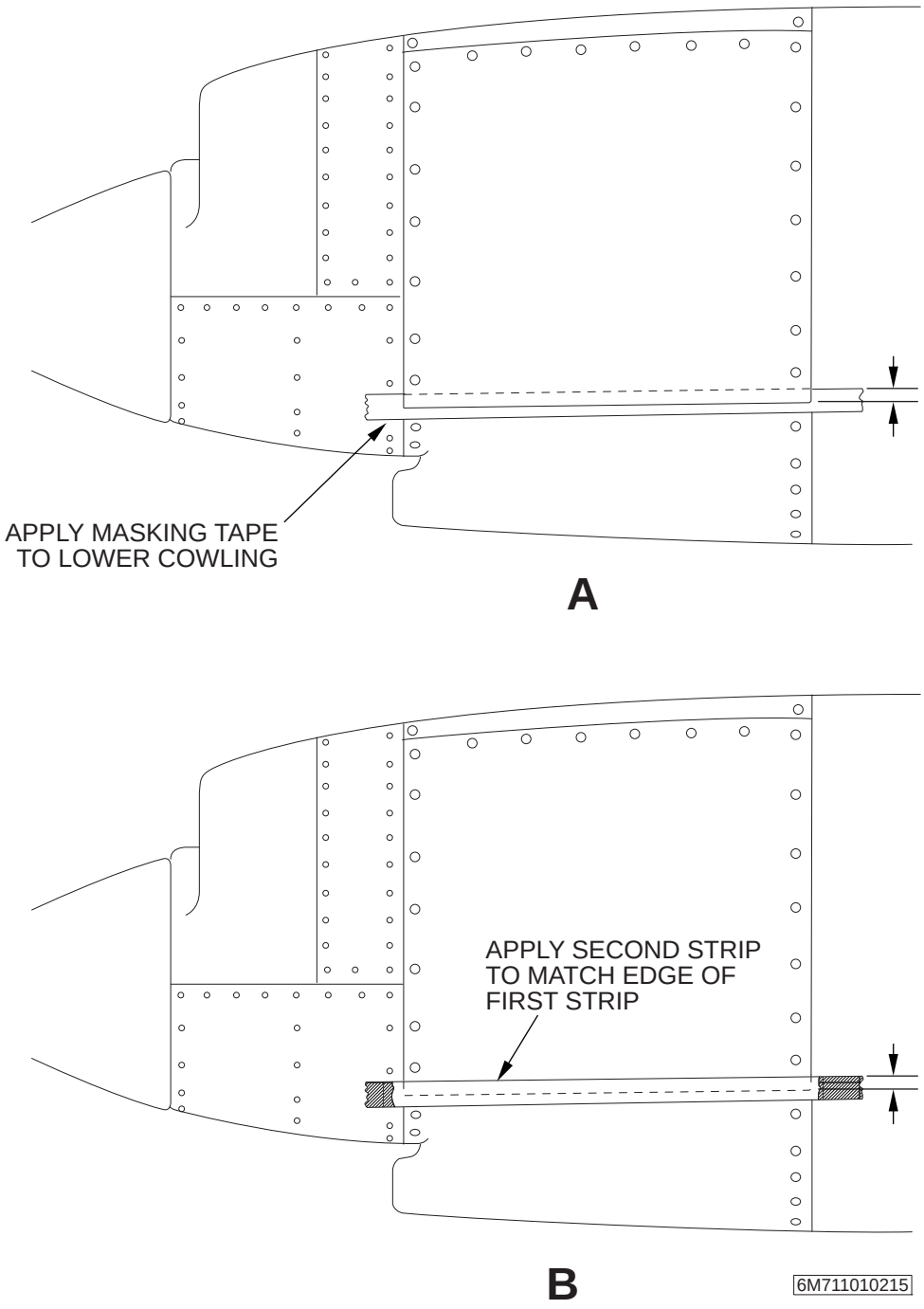


A



B

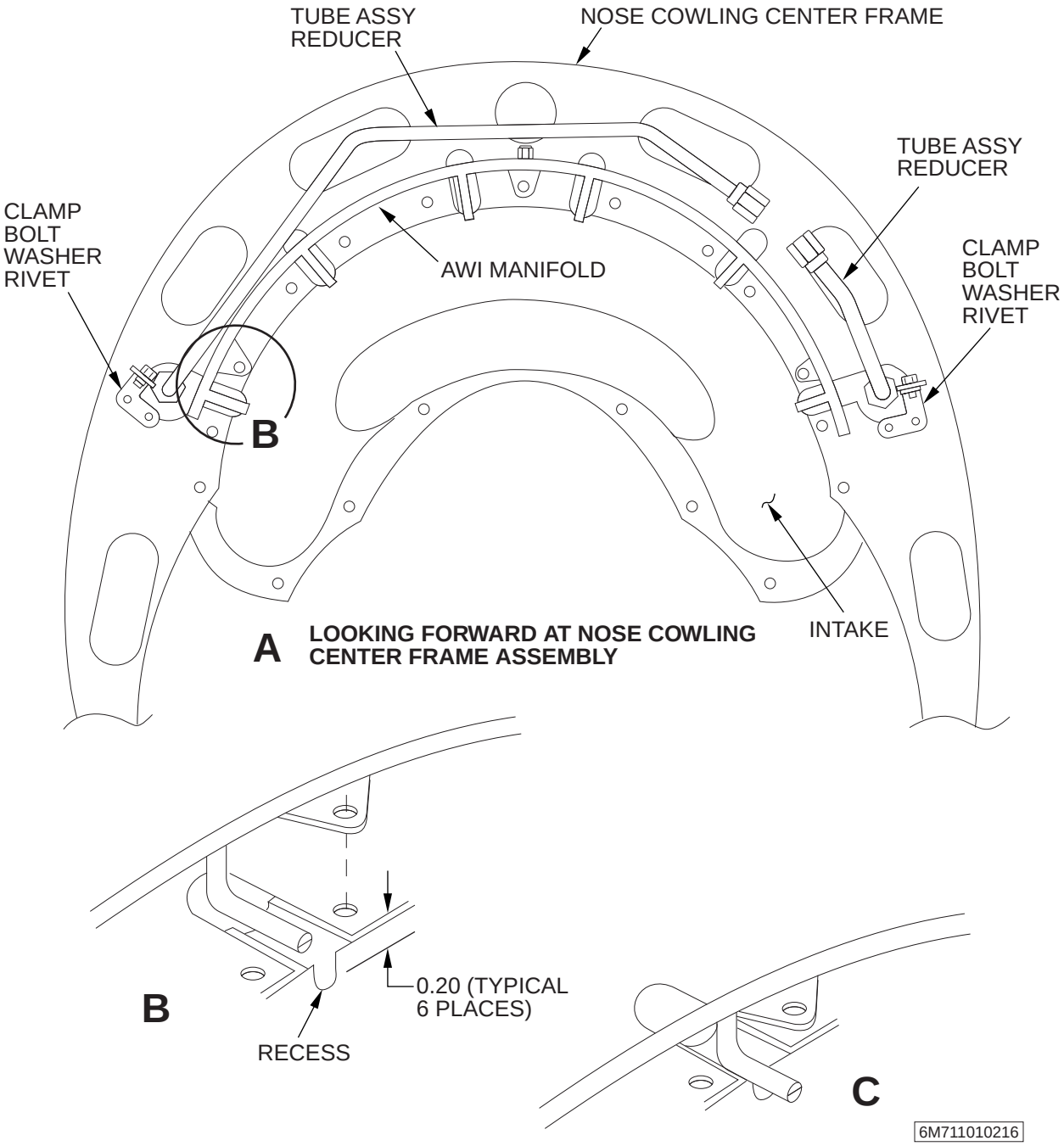
Engine Cowling Trim
Figure 214



Engine Cowling Trim
Figure 215

(18) With side panel installed, adjust compass for two inches, and using circumference marks made on masking tape on top edge of lower cowling assembly, make remaining Camloc fastener hole location marks onto bottom edge of side panel, as shown in Figure 211, inset B, by starting with the fastener nearest center and working outwards. Drill Camloc fastener hole with 15/32 spot face drill; inserting Camloc fastener in each newly drilled hole in order to pull panel down in place before proceeding on to next Camloc fastener. Complete drilling and installation on Camloc fastener for remainder of bottom edge of side panel using same procedure as described above.

(19) Complete buildup of side cowling, see Side Cowling Buildup Procedures within this chapter.



Nose Cowling Buildup
(TPE-331-U-303G / 304G Engine
Figure 216

E. Nose Cowling Buildup Procedures (Reference Figures 216 through 218)

NOTE: The following procedures detail the nose cowling buildup for TPE331-3U-303G/304G and TPE331-10U-501 engines. Figures 216 and 217 illustrate identical procedures for installation of the intake anti-ice tube assemblies and the clamp installation for both type engines. On aircraft using Alcohol/Water Injection (AWI), Figure 216 illustrates the procedures for fitting of the AWI manifold on TPE331-3UW-303G/304G engines, and Figures 217 and 218 illustrate AWI manifold fitting procedures for TPE331-10UW-501 engines.

(1) Installation - Intake Anti-Ice Tube Assembly (Reference Figures 216 and 217).

(a) Install reducer (AN919-12J) onto each intake anti-ice tube assembly attached to the cowling. Install anti-ice valve tube assembly (27-62026-135) onto reducer (AN919-12J) at left of nose cowling assembly. Install tube assembly (27-62026-137) onto reducer at right side, as shown in Figure 216 or 217, inset A.

(b) Install clamp (TAD2H713H08FC) to cowling assembly intake anti-ice tube at left, and install clamp (TAD2LH713H08FG) to intake anti-ice tube at right side, as shown in Figure 216 or 217, inset A. Drill clamp holes (two each) with No. 30 drill, and install clamp with Brazier head rivets (MS20470AD44). Secure tube assemblies to clamps with bolt (AN3-3A) and washer (AN960-10).

(2) Fitting of Alcohol/Water Injection Manifold for TPE331-3UW-303G/304G Engines (See Figure 216).

(a) Install Alcohol/Water Injection manifold into position, as shown in Figure 216, to determine where to cut recess (six places) in cowling for manifold nozzles so that the nozzles will enter the intake area when installed. Place scale (straight edge) along side of nozzle to mark area for recessing to the dimensions, as shown in Figure 216, inset B. Trim excess areas, and reposition nozzles as necessary so that they will center within the recessed area when installed.

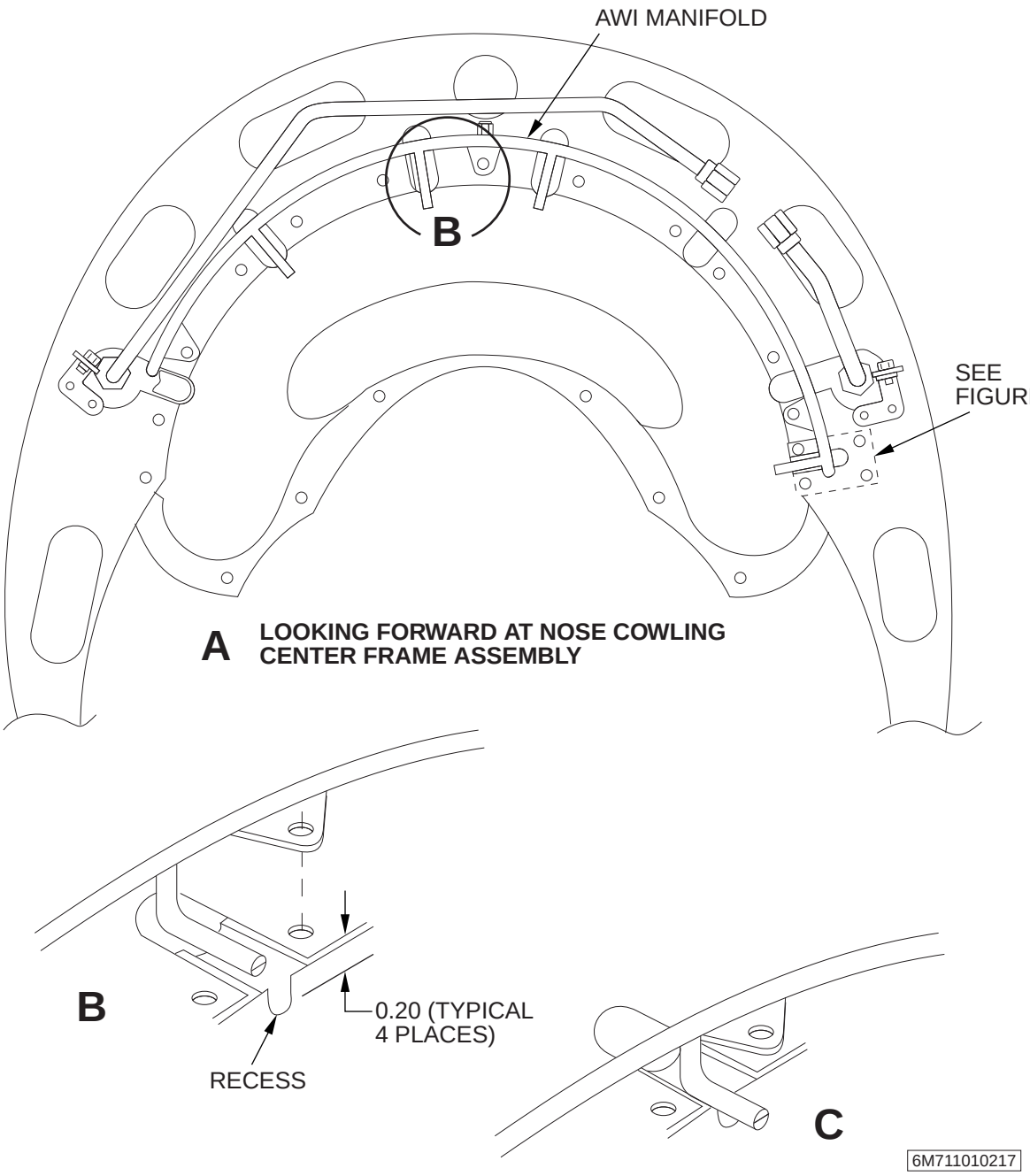
NOTE: Calibrate Alcohol/Water Injection manifold prior to its final installation on aircraft, reference Calibration Procedures in Maintenance Manual, Chapter 82.

(3) Fitting of Alcohol/Water Injection Manifold or TPE331-10UW-501 Engines (Reference Figures 217 and 218)

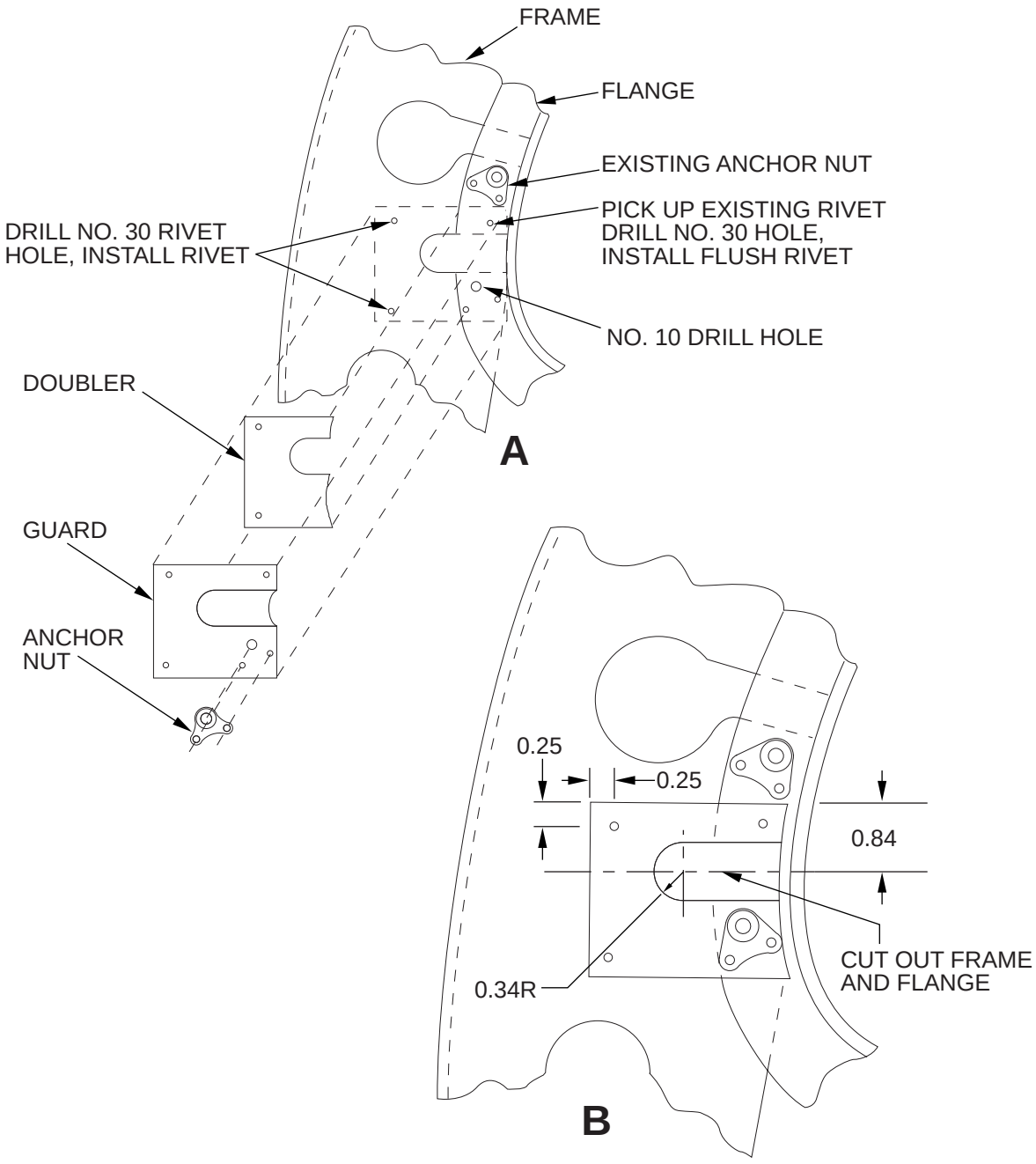
(a) As shown in Figure 218, remove 22NA21-22-02 anchor nut; cut out frame and flange area, and drill No. 30 rivet holes. Install doubler (26-62315-S2) and guard (26-62315-S1) with given rivets, and install corner anchor nuts (52LHTA525-02).

(b) Install Alcohol/Water Injection manifold into position, as shown in Figure 2017, to determine where to cut recesses (four places) in cowling for manifold nozzles so that the nozzles will enter the intake area when installed. Place scale (straight edge) along side of nozzle to mark area for recessing to dimensions, as shown in Figure 217, inset B. Trim recess area and reposition nozzles as necessary so that they will center within the recessed areas when installed, as shown in inset C.

NOTE: Calibrate Alcohol/Water Injection manifold prior to its final installation on aircraft, reference Calibration Procedures in Maintenance Manual, Chapter 82.



Nose Cowling Buildup
(TPE-331-10U-501 Engine
Figure 217



LOOKING AFT AT NOSE COWLING CENTER FRAME ASSEMBLY

Nose Cowling Buildup
(TPE-331-10UW-501 Engine
Figure 218

(4) Upper Cowling Panel (Reference Figure 219)

NOTE: The following procedures detail the buildup of the upper cowling assembly which include riveting of the panel's forward and aft edges, shrinking of the edges to assure of uniform contour when installed, and hot dimpling of the screw fastener holes.

(a) Establish rivet pattern along edges of panel in accordance with dimensions given in Figure 219. Drill No. 40 rivet holes, and install MS20426-3-3 rivets. Install additional rivet halfway between screw fastener hole and channel at each corner of panel, as shown in Figure 219.

(b) Shrink edges of panel to assure of uniform contour with other nacelle cowling when cowling panel is installed.

(c) Hot dimple all screw fastener holes after shrinking edges.

(5) Side Cowling Buildup (Reference Figure 220)

NOTE: The following procedures detail the buildup of the left or right side cowling assembly, which include riveting of the panel's edges and shrinking of the panel's edges to assure uniform contour when installed.

(a) Establish rivet pattern along all four edges of panel, in accordance with dimensions given in Figure 220. Drill No. 40 rivet holes, and install MS20426-3-3 rivets.

(b) Shrink edges of panel, and deburr edges to assure of uniform contour with other nacelle cowlings when side cowling panel is installed.

(6) Installation - Lower Cowling Buildup (Reference Figures 220 through 229 - Doubler 27-63020-009)

(a) Cut 0.50 inch off from aft edge of doubler.

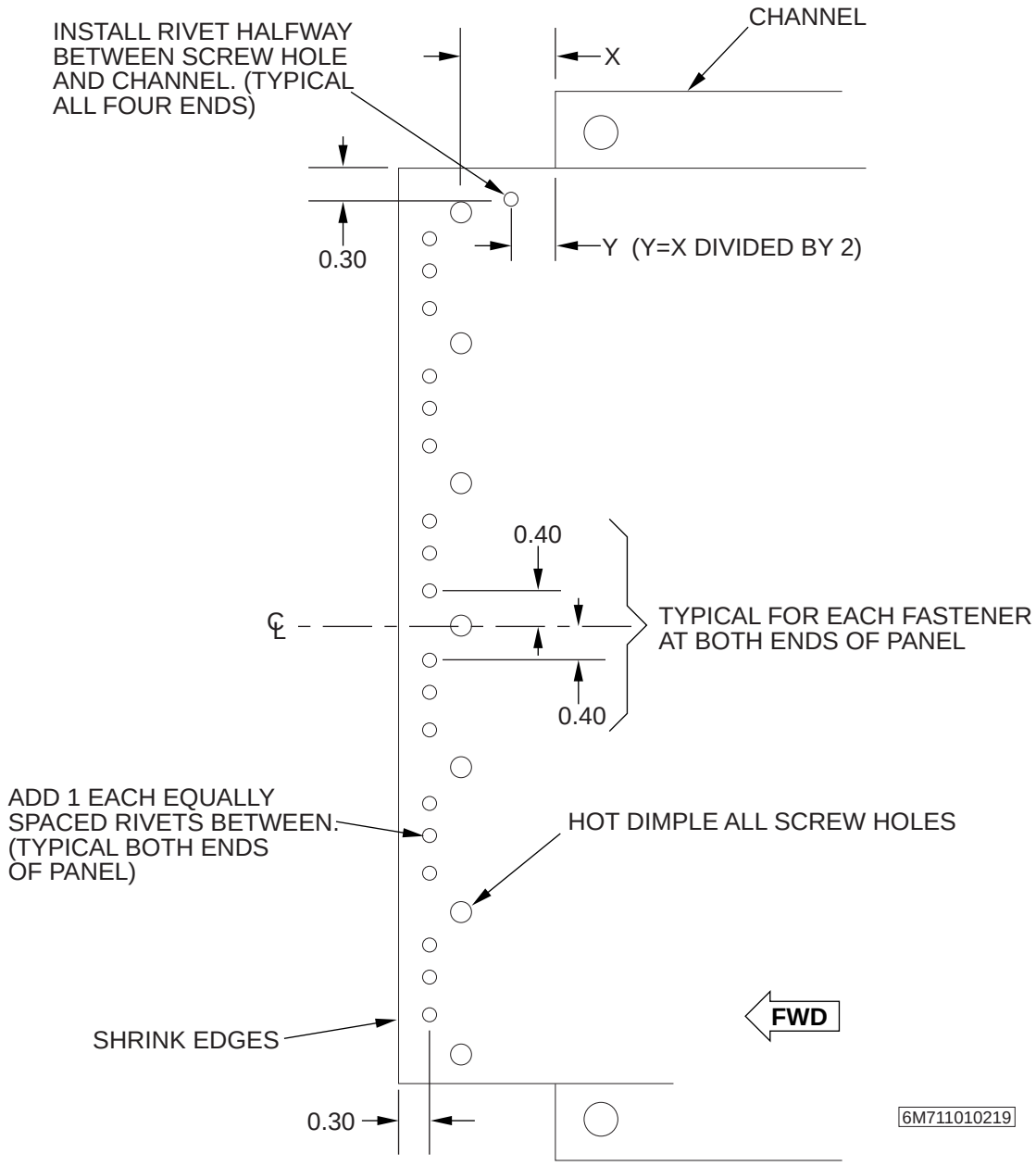
(b) Establish rivet hole location onto inside edge of doubler by drawing china mark onto doubler 0.30 inch from inside edge of doubler, as shown in Figure 220.

(c) Establish centerline of drain block cutout area of doubler by measuring the distance between the center of the Camloc fastener holes at each side of cutout area, and splitting this distance in half, see Figure 221. To ensure that the drain block cut out area of doubler clears drain block when cowling is installed, draw trim mark by marking inside edge of cut out area 0.80 inch from centerline. Using front plate (27-64012-007) of drain block as a template, mark radiused front of drain block cutout area onto doubler.

(d) Install doubler (27-63020-009) into location on cowling ensuring drain block cutout area of doubler aligns with markings made in step (3). Align radiused edge of doubler with marking first, then align the edges of the doubler equally at both ends of the cowling. Trim excess along 0.80 inch marking. Clamp doubler into place onto cowling.

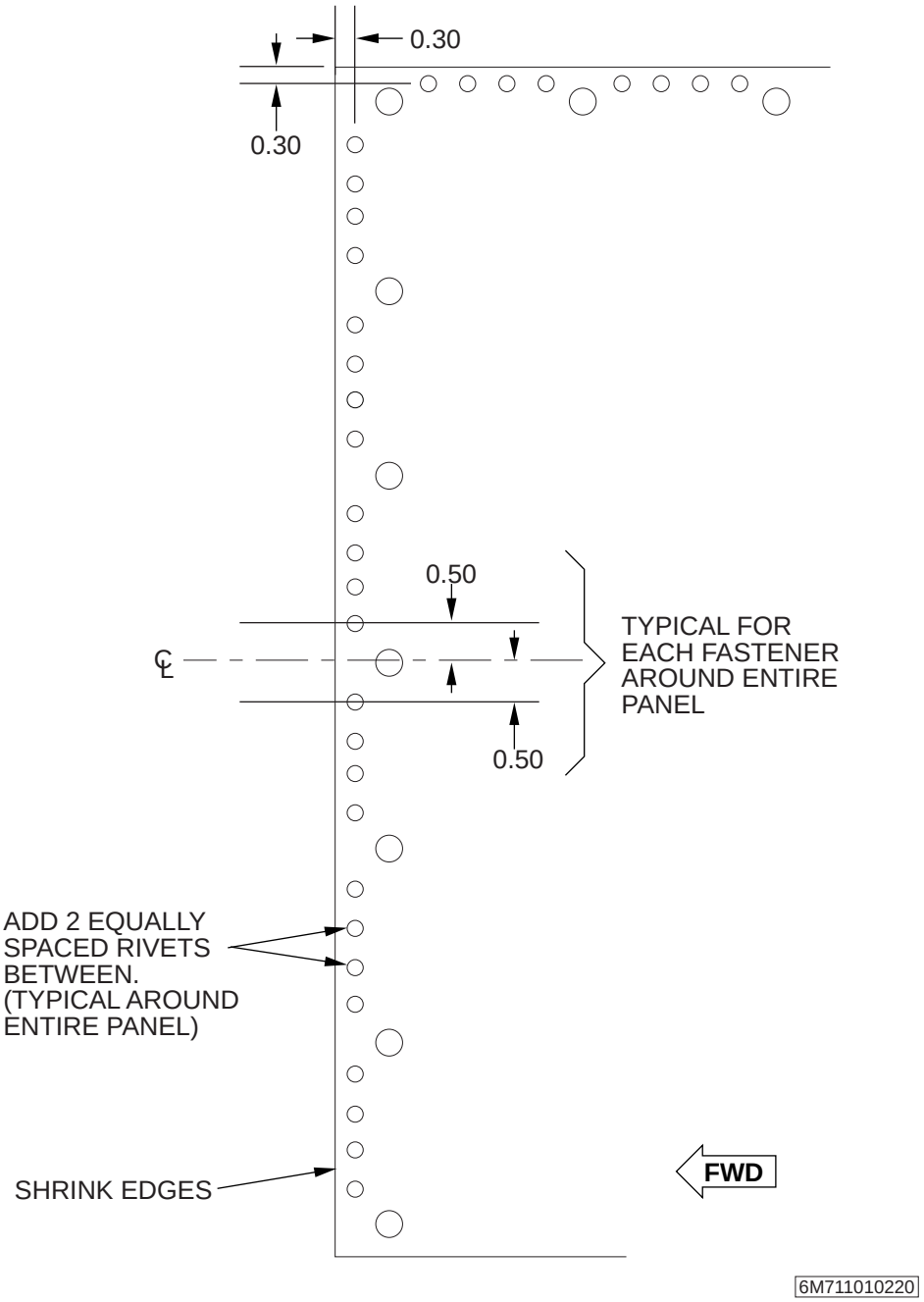
(e) Establish rivet hole location onto doubler at inside edge of drain block cutout area by drawing china mark onto doubler 0.30 inch from inner edge, as shown in Figure 220.

(f) Establish centerline of cowling. Establish and mark initial rivet hole location at inboard edge of doubler at centerline; mark remaining rivet locations one inch apart working outboard from centerline rivet location.



LOOKING DOWN AT UPPER COWLING PANEL

Upper Cowling Panel Buildup
Figure 219



LOOKING AT FORWARD EDGE OF LH SIDE COWLING PANEL

Side Cowling Buildup
Figure 220

(g) Establish and mark initial rivet location at inside of drain block cutout area of doubler at cutout centerline, as shown in Figure 221. Mark remaining rivet locations for inside of cutout area one inch apart from rivet mark established at cutout centerline. Mark locations only as far as the 0.30 inch mark at inner edge of doubler.

(h) Drill thus far marked rivet holes on inboard edge of doubler with No. 40 drill, working outwards from centerline of cowling. Insert Cleco fasteners to fasten doubler to cowling.

(i) Drill rivet holes for rivets marked at drain block cutout area with No. 40 drill and insert Cleco fasteners.

(j) Turn cowling over to its outer side.

(k) To establish rivet hole locations for aft edge of doubler, draw china mark line 0.30 inch from aft edge of cowling across the back of the cowling, as shown in Figure 222.

(l) Rivet hole locations on aft edge of cowling are to be located 1/2 inch from center of Camloc fastener hole at each side of each Camloc fastener hole along the 0.30 inch marking. An additional rivet location lies halfway between the rivets at both sides of Camloc fastener holes. Measure and mark rivet locations, as shown in Figure 224.

(m) Establish rivet locations at corners of cowling at drain block cutout area by aligning with 0.30 inch mark, step (11) at aft edge of cowling and the rivets along inner edge of drain block cutout area, step (5).

(n) Determine last two rivet locations at inboard edge of doubler at left and right sides of cowling. Mark one rivet location in line with edge of channel at 0.30 inch from edge of cowling, and the second rivet hole location halfway between it and the last rivet hole drilled for the inside forward edge of doubler.

(o) Starting from centerline of cowling, drill marked rivet holes with No. 40 drill and insert Cleco fasteners along doubler securing it to the cowling.

(p) Mark all overlapping between doubler and cowling for trimming purposes.

(q) Remove Cleco fasteners and doubler from cowling.

(r) Countersink all drilled rivet holes in doubler with deburring tool (DB8-90 degrees-HS, Weldon Co.) and trim excess from doubler at marked areas, step (16).

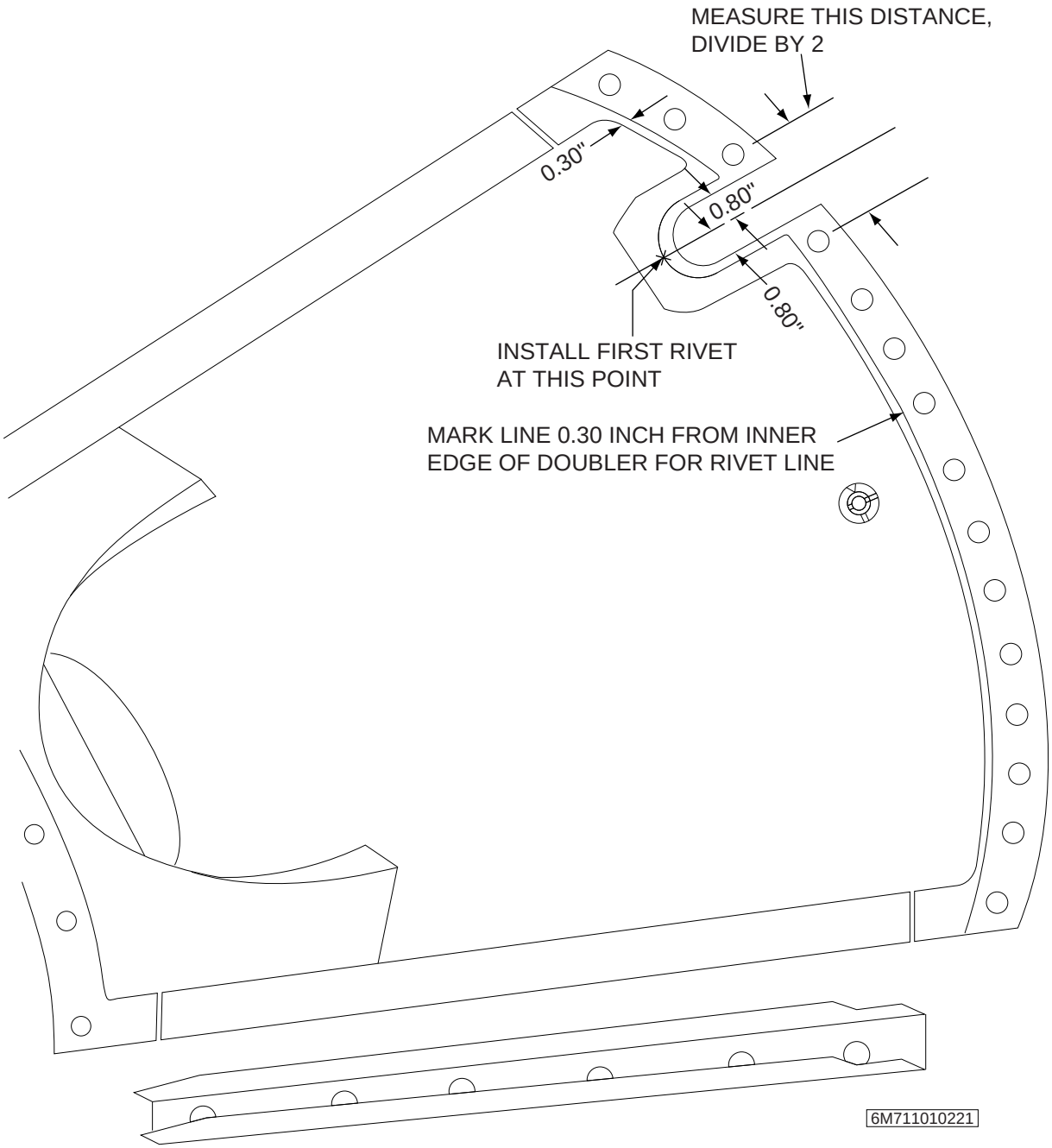
(s) Reinstall doubler onto cowling and fasten with Cleco's.

(t) On cowling outerside, countersink rivet holes with No. 40 countersink.

(u) Install doubler by installing approximately 91 rivets (MS20426-AD4-3) starting at centerline of cowling working outwards.

(v) Shave back row of rivets on inside aft edge of cowling with rivet shaver (micro-miller).

(w) Drill out Camloc fastener holes in doubler using 15/32 spotface drill (counterbore).



Aft Edge Cowling Doubler
Figure 221

(7) Installation - Doubler (27-63020-014)

(a) Establish rivet hole locations onto inner edge of doubler by drawing china mark onto doubler 0.30 inches from inside edge of doubler, as shown in Figure 223.

(b) Place doubler into position on inside front end of cowling and align the doubler centered with each edge of cowling and even with cowling's front edge. Clamp doubler to cowling.

(c) Establish centerline of cowling. Establish and mark initial rivet hole at inside edge of doubler at cowling centerline intersecting at 0.30 inch mark from edge of doubler. Mark remaining rivet locations one inch apart working outboard from centerline rivet location, as shown in Figure 223.

(d) Drill marked rivet holes with No. 40 drill working outwards from centerline of cowling. Insert Cleco's to fasten.

(e) Turn cowling over to its outerside, see Figure 224. To establish rivet hole locations for outer edge of doubler, make china mark line 0.30 inch from outer edge of cowling across the front of the cowling.

(f) Rivet hole locations on forward edge of cowling shall be 1/2 inch from center of Camloc fastener hole at each side of each fastener hole along the 0.30 inch mark at cowling's forward edge. Two additional rivet locations lie equally spaced, halfway between the rivets at both sides of the Camloc fastener holes. Measure and mark rivet locations.

(g) Establish rivet locations at each corner of forward end of cowling by marking locations that are equally spaced between center of last Camloc fastener hole and edge of cowling. Starting at centerline of cowling, drill marked rivet holes with No. 40 drill and insert Cleco Cleco fasteners along doubler securing it to the cowling.

(h) Mark all overlapping between doubler and cowling for trimming purposes. Remove doubler and trim excess from doubler at marked areas.

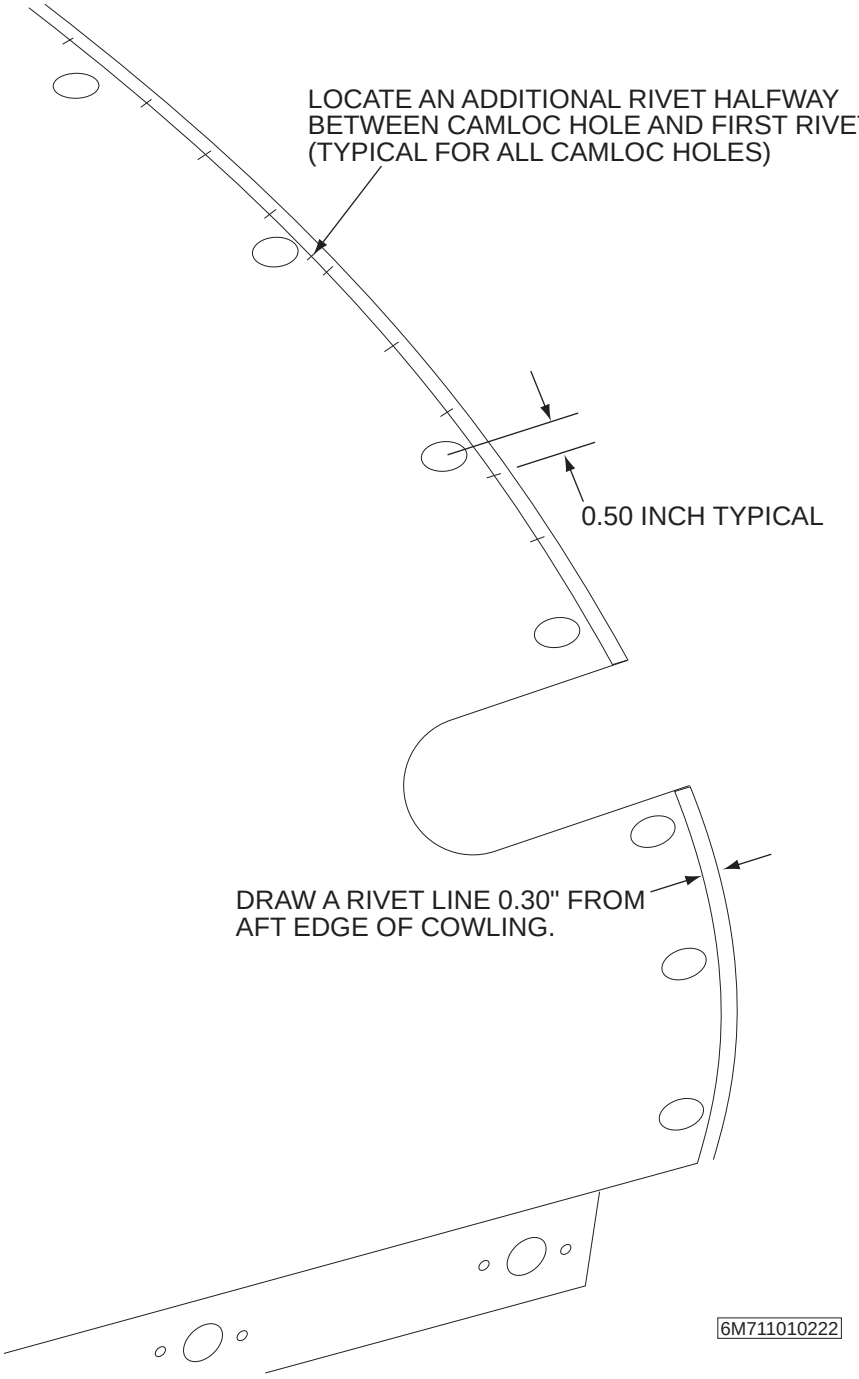
(i) Countersink all drill rivet holes in doubler with deburring tool (DB8-90 degrees-HS, Weldon Co.).

(j) Reinstall doubler onto cowling, fasten with Cleco's and countersink all rivet holes on outside of cowling with No. 40 stop sink.

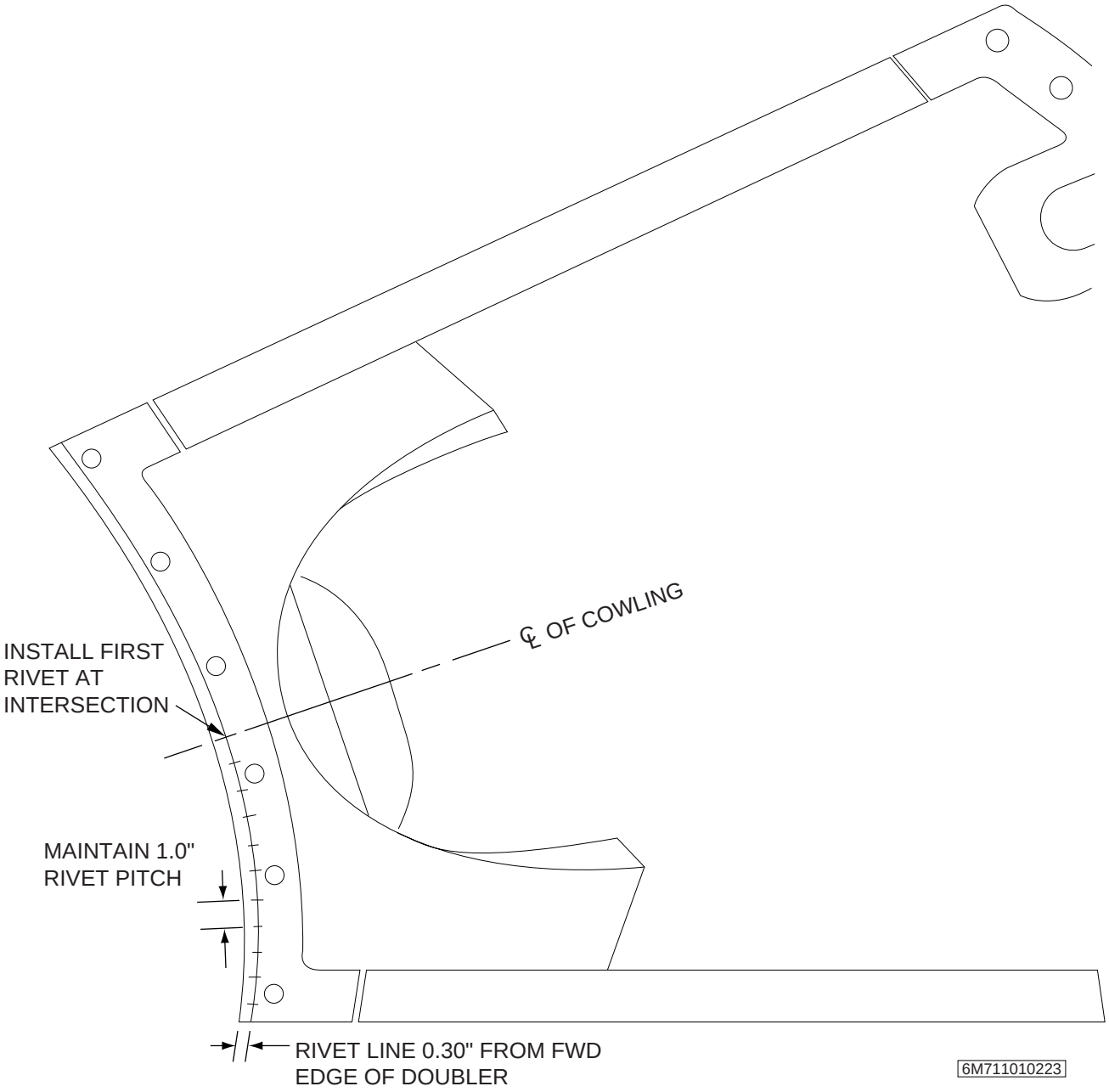
(k) Install doubler by installing rivets (MS20426-AD4-3) starting at centerline of cowling and working outwards.

(l) Shave all rivets of doubler on inside of cowling with rivet shaver (micro-miller).

(m) Drill out the Camloc fastener holes in doubler using 15/32 spot face drill (counterbore).



Camloc Location
Figure 222



Forward Edge Engine Cowling Doubler
Figure 223

(8) Installation - Doubler (27-62020-003), Left Channel (27-63005-005), and Right Channel (27-63005-006)

(a) Position doubler onto cowling and align doubler evenly with edge of cowling, see Figure 225. Draw china mark onto panel along inner edge of doubler. Remove doubler and grind the weld area on cowling flush beneath area where doubler is to be installed.

(b) Draw a straight edge lengthwise china mark that is 0.30 inch above inboard pilot holes of channel assembly across its inner surface, see Figure 225, inset C. Position channel onto cowling and align china mark on channel with edge of cowling. Center the channel evenly between the cowling edges, clamp channel into place.

(c) Drill four of the pilot holes (equally spaced across the channel) with No. 30 drill. Install a Cleco fastener as each hole is drilled to keep channel aligned with edge of cowling.

(d) Remove Cleco's and channel. Install doubler flush with edge of cowling and back drill the four holes through the doubler. Reinstall channel with doubler and insert Cleco's to hold the doubler and channel in place.

(e) Drill all remaining pilot holes through doubler, channel, and cowling with No. 30 drill.

(f) Remove channel, leave doubler installed. Trim doubler so that edges of front doubler (27-63020-014) and side doublers (27-62020-003) end between rivet holes, as shown in Figure 225, inset B. Trim aft end of channel, as shown in Figure 225, inset C. Make cut on each side of channel half the distance of the width of the channel's sides and cut aft along its sides taking dimension from center of Camloc hole.

(g) Remove doubler, trim ends if required.

(h) Dimple all the rivet holes on the outside of the cowling with 1/8 (No. 30) dimpler.

(i) Put doubler and channel together with Cleco's (not installed onto panel) and countersink all rivet holes to accommodate the dimples on the cowling.

(j) Install doubler and channel onto cowling and install all rivets (MS20426AD4-4.5).

NOTE: Preceeding procedures are typical for either left or right channel and doubler.

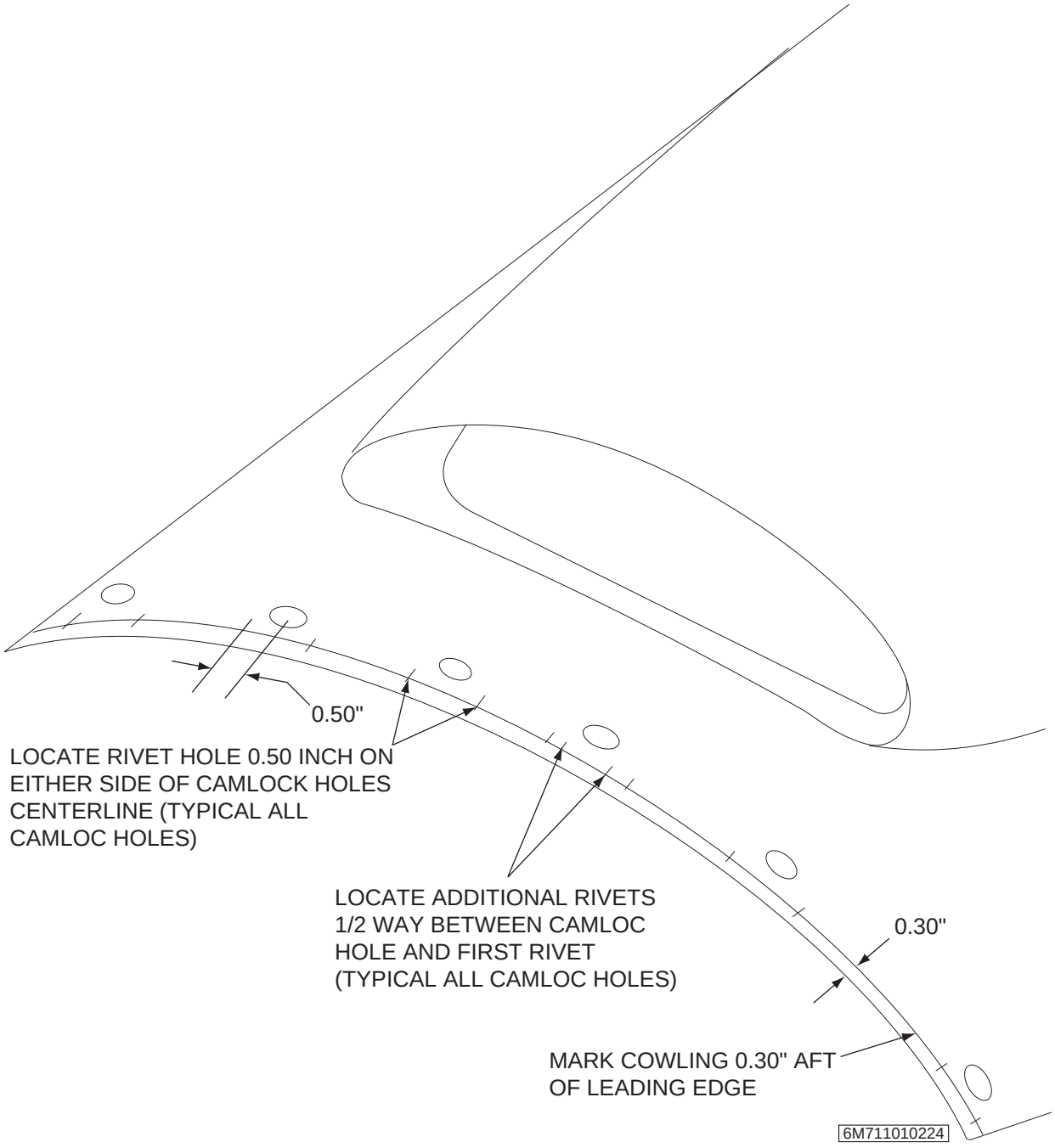
(9) Installation - Oil Ring Assembly (27-63021-005) and Inlet Duct (Boot) (27-19021-001) (Figure 226)

(a) Remove inlet duct boot from oil ring assembly.

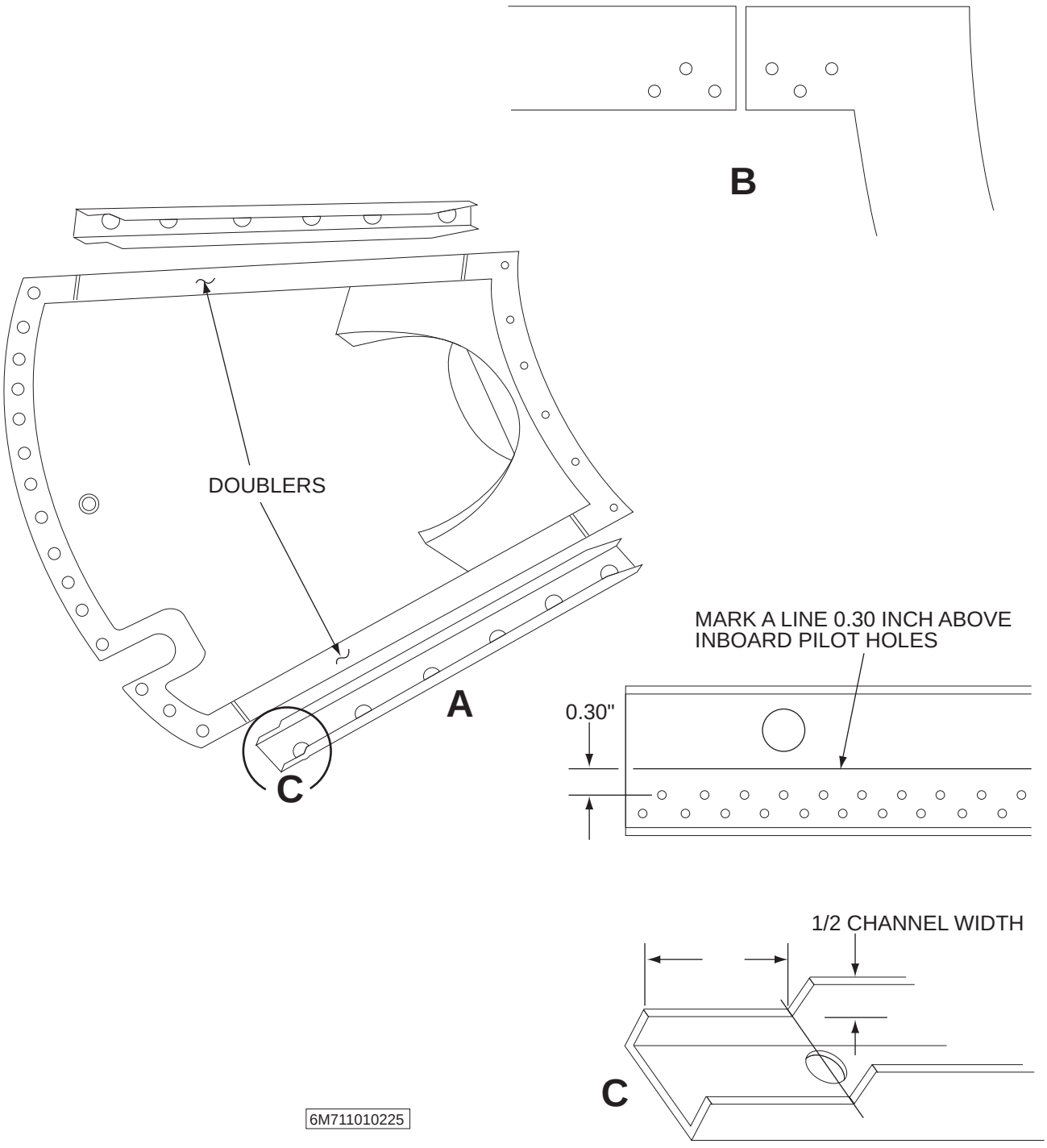
(b) Establish rivet hole location on oil ring by drawing china mark 0.20 inch from aft edge of oil ring assembly around its entire lip circumference (see inset A).

(c) Establish centerline of oil ring assembly and draw centerline china mark intersecting mark made during step (2). Draw centerline mark at both top and bottom of oil ring assembly.

(d) At bottom of oil ring, the initial rivet hole location will be at the centerline intersection. Mark six more rivet hole locations along the 0.20 inch mark made during step (2) on both sides of the centerline intersect, each rivet hole location 1.45 inches apart starting from centerline (13 places total on bottom only, see inset A).



Installation of Forward Doubler
Lower Engine Cowling
Figure 224



Installation of Channels Lower Engine Cowling
Figure 225

(e) At top of oil ring, Inset B, mark initial screw hole location 2.00 inches from centerline; typical for left and right side of centerline. Mark remaining screw hole locations 1.5 inches apart thereafter (six screw holes total, three at left of centerline, and three at right of centerline). Do not drill screw hole at centerline intersect.

(f) Drill bottom 13 rivet holes with No. 30 drill. Drill top six screw holes with No. 19 drill.

(g) Using outer lip of oil ring assembly, fashion a cardboard template that will fit within the inner circumference of the lower cowling assembly scoop area in order to mark trim line for trimming excess from cowling scoop area.

(h) Obtain trim line location by measuring 0.80 inch from weld area beneath lower cowling scoop inlet, taking measurement from either left or right of scoop centerline, and transfer the dimension onto inside of scoop inlet. Place a cardboard template along transferred dimension within inlet area and draw trim mark around entire inner circumference of inlet.

(i) Cut and file excess from lower cowling scoop inlet area.

(j) Install oil ring into place (without inlet duct boot) on lower cowling inlet as far as it will go to assure sufficient edge distance for drilling of holes into cowling inlet.

(k) Drill bottom centerline rivet hole of oil ring through cowling inlet first, with No. 30 drill and install Cleco fastener.

(l) Drill one rivet hole on each end of oil ring assembly with No. 30 drill and install Cleco fasteners.

(m) Drill remaining ten rivet holes for bottom of oil ring with No. 30 drill.

(n) Turn lower cowling upright (as when installed on aircraft) and working between the lower cowling inlet scoop, scribe mark the top six screw holes of the oil ring with a 90° scribe to mark screw hole locations into lower cowling inlet.

(o) Remove oil ring, turn cowling over, and transfer the dimensions of the scribed holes, done in step (14) to the inside lip of the cowling inlet. Drill these six screw holes with No. 19 drill.

(p) Dimple the 13 each No. 30 rivet holes at bottom of cowling inlet with No. 30 dimpler. (17) Dimple all drilled holes of the oil ring.

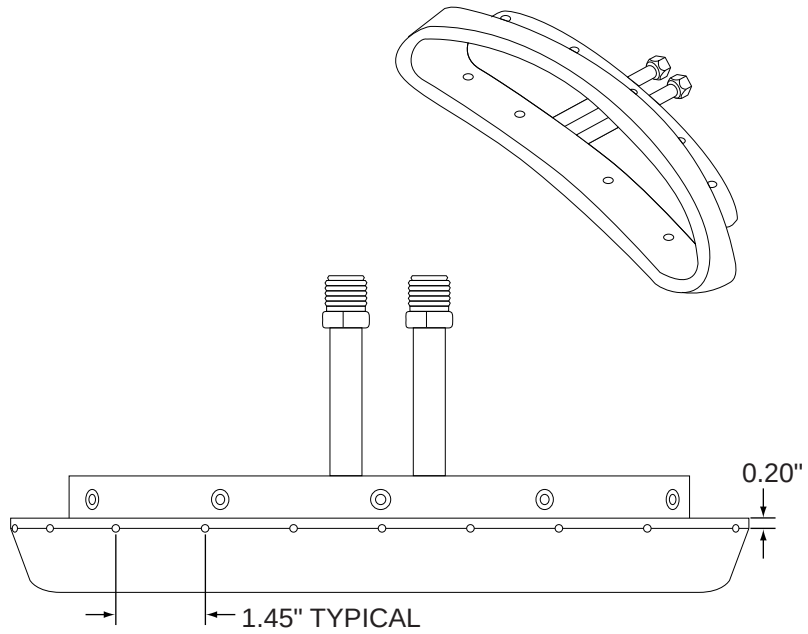
NOTE: It is recommended using a piece of steel (as a bucking bar) with a No. 30 hole drilled into its end, and countersink with No. 30 countersink. Place bucking bar behind hole; and using a No. 30 male-set, hit male-set with rivet gun to dimple holes.

(q) Reinstall oil ring. Install six screws (MS35206-227), washers (AN960-6L), and nuts (AN363-632). Install screws through top of oil ring and install washers and nuts from inside of cowling scoop inlet.

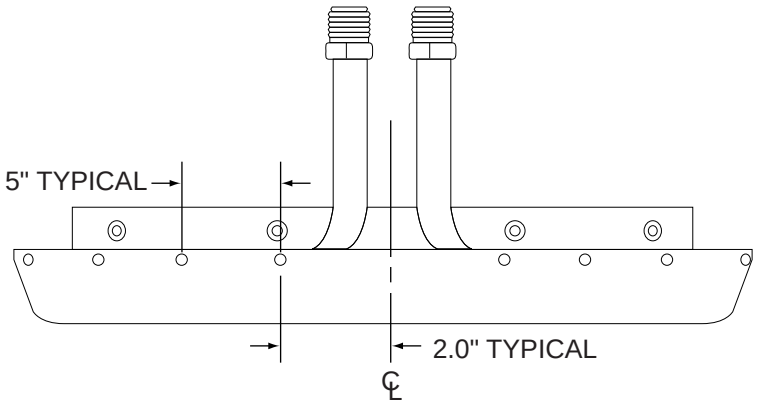
(r) Countersink the 13 dimpled rivet holes on bottom of oil ring with No. 30 countersink, in order for rivets to fit flush when installed.

(s) Install 13 huck rivets (MLS100-B04-02).

(t) Install inlet duct boot onto cowling assembly with screw fasteners.



A



B

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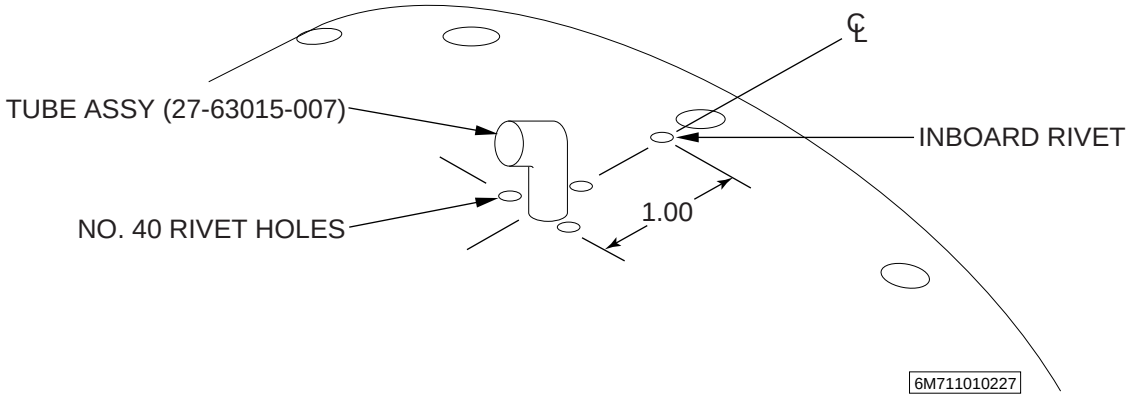
Installation of Oil Ring Assembly
Lower Engine Cowling
Figure 226



Water Drain Tube
Figure 227

(10) Installation – Drain Tube Assembly (Reference Figure 227)

(a) Establish drain tube location on lower cowling, see Figure 227, inset B. Mark drill hole location one inch from inboard rivet, along cowling centerline at aft portion of cowling.

(b) Drill 0.375 diameter hole and install drain tube assembly. Drill three No. 40 rivet holes to match drain tube attaching plate.

(c) Remove drain tube assembly and countersink rivet holes at outer surface of cowling with No. 40 countersink.

(d) Reinstall tube assembly and install rivets (three each, MS20426AD4-3).

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ENGINE MOUNTS – DESCRIPTION AND OPERATION

1. General

A. The engine mounts used on SA226 series aircraft consist of a tubular upper engine mount truss assembly, a lower engine mount truss support assembly, a left and right shock mount assembly, a forward bottom shock mount assembly, and on SA226 series aircraft T276, T292 through T999, a rear bottom shock mount assembly.

B. The tubular engine mount truss assembly and its lower support assembly is reinforced with gussets and welded at each joint. It resembles a U shaped yoke, and fits over the engine in an inverted position immediately aft of the engine side mounts. It is bolt mounted at four places on the nacelle firewall, at the forward and aft ends, and at the left and right bottom shock (Lord) mounts.

C. On SA226 series aircraft T276, and T292 through T999, the engine is also mounted to a rear engine mount. The rear engine mount assembly is bolted to the firewall within the lower section of the aspirator assembly.

ENGINE MOUNTS –- MAINTENANCE PRACTICES

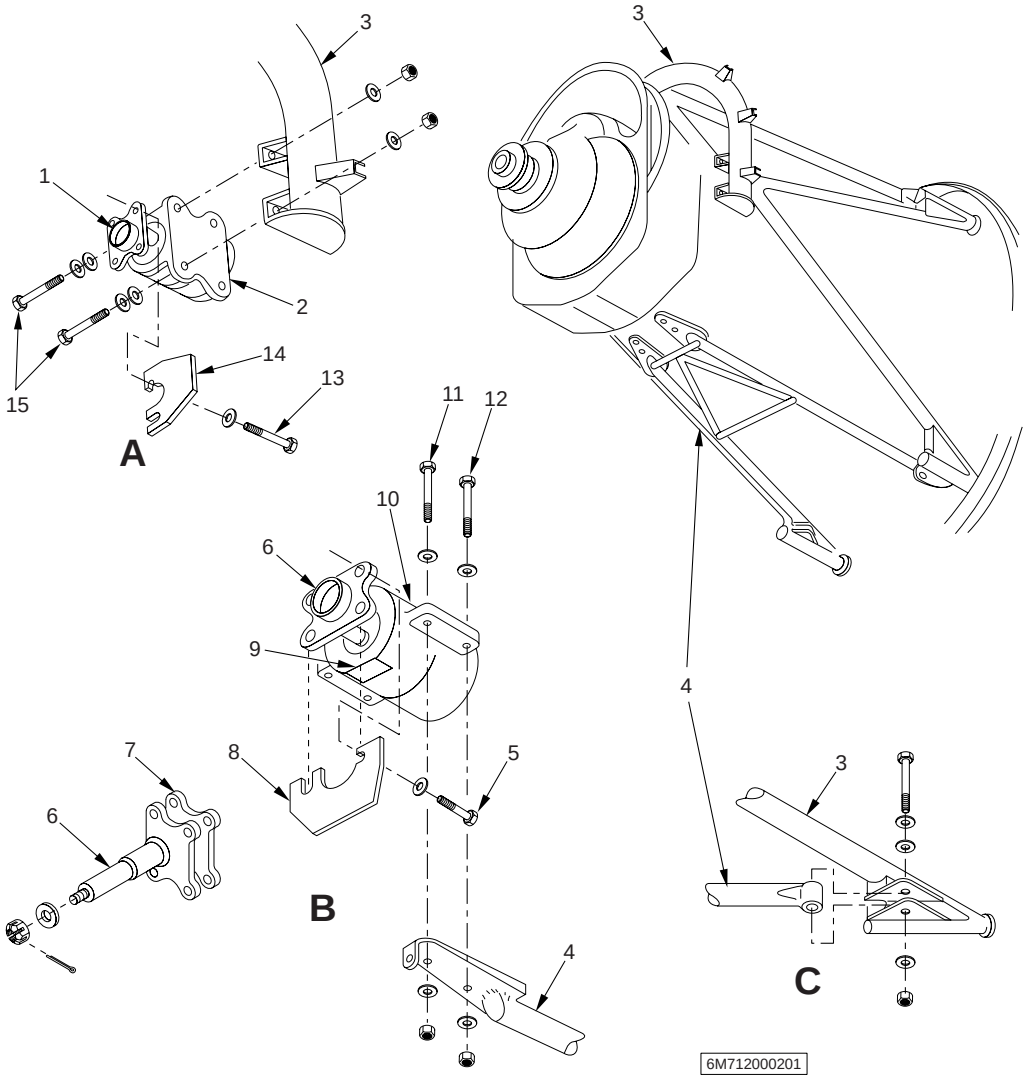
1. Removal/Installation – Engine Mounts

CAUTION: ATTACH HOISTING SLING TO ENGINE AND REMOVE WEIGHT OF ENGINE FROM ENGINE MOUNTS. PLACE JACK UNDER ENGINE GEAR CASE FOR PURPOSES OF ACCOMPLISHING MINUTE ADJUSTMENT AS NECESSARY. REMOVAL OF MORE THAN ONE ENGINE MOUNT WHILE ENGINE IS ON AIRCRAFT IS PROHIBITED.

A. Removal – Lord Mounts (Figure 201)

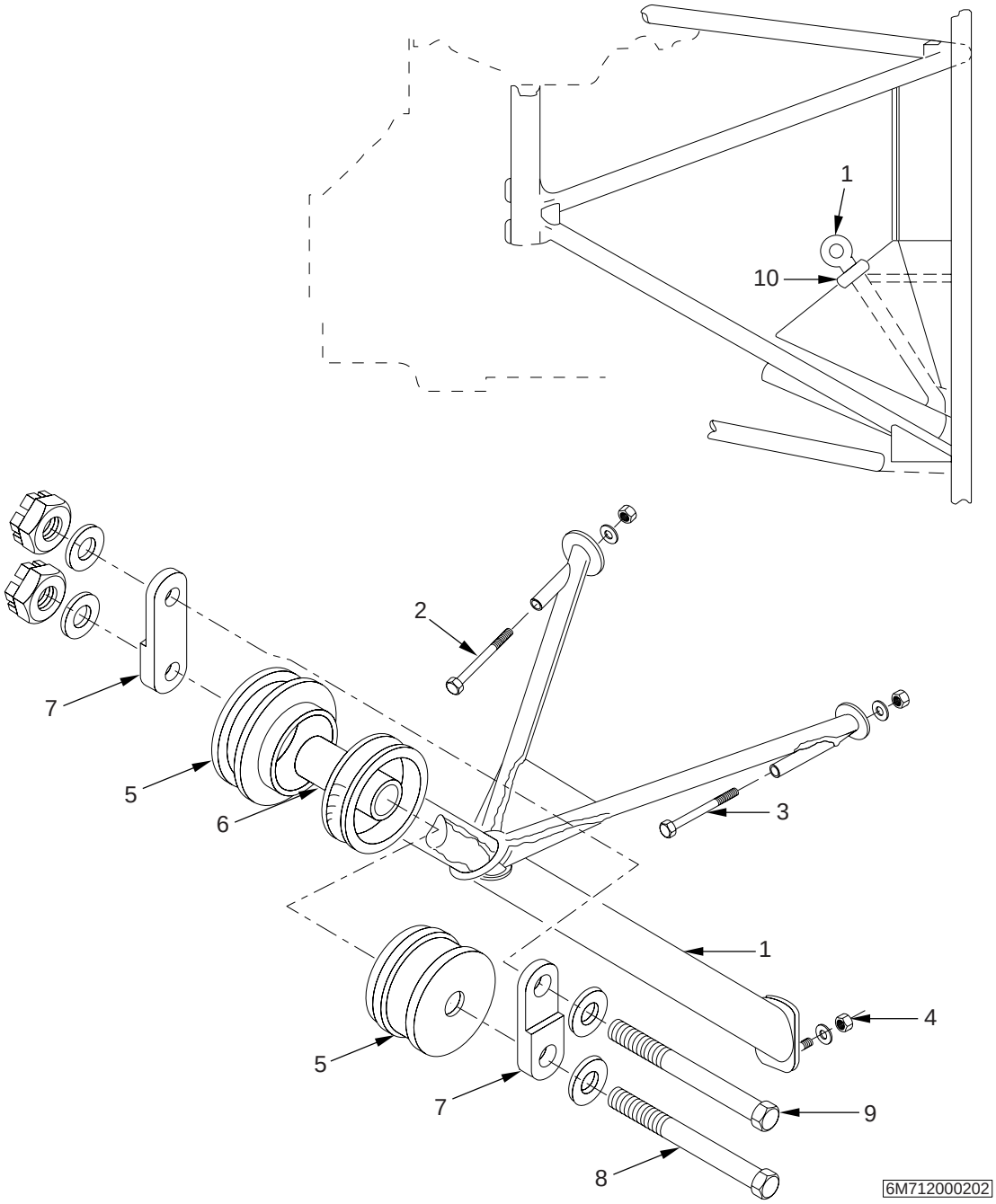
(1) Side Lord Mounts.

(a) Remove retaining bolts (item 15, inset A) securing side mount to engine mount truss assembly (item 3).



Engine Mounts
Figure 201

(b) For ease of removal, remove cotter key and nut from side mount center shaft (item 7, inset B). Remove side mount housing assembly from center shaft.



Rear Engine
Lower Isolation Mount
Figure 202

(c) Remove retaining bolts (item 13, inset A) securing side mount center shaft to engine and remove from engine.

(2) Bottom Lord Mount.

(a) Remove retaining bolts (items 11 and 12, inset B) securing bottom mount to engine mount truss support assembly (item 4).

(b) For ease of removal, remove cotter key and nut from bottom mount center shaft (item 6, inset B). Remove bottom mount housing assembly from center shaft.

(c) Remove retaining bolts (item 5, inset B) securing bottom mount center shaft to engine and remove from engine.

(3) Rear Engine Mount – SA226-T(B) Only (Figure 202).

(a) Remove cotter key, nut, and retaining bolts (items 2 and 3) and remove mounts as shown in Figure 202.

B. Installation – Lord Mounts (Figure 201)

(1) For ease of installation and safetying, remove center shaft (item 1, inset A) from side Lord mount (item 2, inset A); remove center shaft (item 6, inset B) from bottom Lord mount (item 10, inset B), by removing the cotter key and nut of each center shaft.

NOTE: On the right Lord mount, remove the cotter key, but only loosen the nut of the center shaft, because this particular mount and shaft must be installed together due to limited installation space.

(2) Assemble bracket (item 14, inset A) to each of the two disassembled side mount center shafts (item 1, inset A).

(3) Assemble bracket (item 8, inset B) to the disassembled bottom mount center shaft (item 6, inset B).

NOTE: Install brackets onto center shafts such that the nuts on the brackets faces towards aft of the engine.

On SA226 series aircraft AT073-up and TC293-up with TPE331-3U-303G or TPE331-3U-304G engines, use left and right bracket (26-62309-809) for the two side Lord mounts and bracket (26-62309-813) for the bottom Lord mount. See Illustrated Parts Catalog, Section 71-20-00 for further information on item part numbers for all other SA226 series aircraft.

(4) Install the left Lord mount center shaft (item 1, inset A) and bottom Lord mount center shaft (item 6, inset B) and gasket (item 7, inset B) using four bolts (item 5, inset B) and (item 13, Inset A) and washer per each mount. Torque bolts 450 to 500 inch-pounds (50.84 to 56.49 N·m) and safety using 0.041 safety wire.

CAUTION: INSTALL CENTER SHAFTS SO THAT THE OFFSET OF THE SHAFT SHANK IS INSTALLED TOWARDS OUTBOARD OF THE ENGINE.

(5) Assemble left and bottom Lord mounts onto center shafts. Install washer and nut, see Engine Mount Assembly (Truss) Installation procedures, **1.**, D., (6) for torque values.

CAUTION: INSTALL LORD MOUNTS SUCH THAT THE SNAP RING OF EACH MOUNT IS FACING AFT OF ENGINE.

INSTALL BOTTOM LORD MOUNT SUCH THAT THE MILLED FLAT AREA OF THE LORD MOUNT FACES TOWARDS THE ENGINE'S HYDRAULIC PUMP TO ENSURE PROPER CLEARANCE.

(6) Install right Lord mount and center shaft with gasket (item 7, inset B) using same type bolts (item 13, inset A), and washers as used on left Lord mount center shaft. Torque 450 to 500 inch-pounds (50.84 to 56.49 N·m) and safety using 0.041 safety wire.

NOTE: On TPE331-10U-501 engines, it may be necessary to remove the tube assembly from the Negative Torque Sensor Checkout Solenoid to beta switch housing, in order to facilitate the right Lord mount installation.

(7) For purposes of servicing as a parting agent, apply Liquid Shim (Scotch Weld EC-1751 B/A or Devcon F, stock no. 10610), to the faying surface of the engine mount frame assembly where it will be attached to both side Lord mounts. Apply wax (Vista, Simonize paste, or equivalent) release agent to mating fay surface of side Lord mounts.

(8) On SA226 series T(B) aircraft only, install rear engine mount, as shown in Figure 202. Tighten nuts on mount bolts (items 2 and 3) securely (no specified torque), and install cotter pins into castellated nuts. See Illustrated Parts Catalog, Section 71-22-00 for engine mount part numbers.

C. Removal – Engine Mount Assembly (Truss) (Figure 201)

(1) Remove engine from aircraft, see Engine Removal Procedures, Section 71-00-00.

(2) With engine on hoist and away from aircraft, remove engine mount truss support assembly by removing the retaining bolts (items 11 and 12, inset B), securing the forward section of the support assembly to the bottom Lord mount (item 10, inset B).

(3) Remove the retaining bolts, securing the support assembly (item 4) to the upper engine mount truss assembly (item 3, inset C). Remove support assembly.

(4) Lower engine onto shipping container, remove hoist sling.

(5) Remove upper engine mount truss assembly by removing the retaining bolts (item 15, inset A) securing it to the engine Lord mounts, remove truss assembly.

D. Installation –- Engine Mount Assembly (Truss) (Figure 201)

(1) Install engine mount assembly (item 3, inset A) by securing it to the left and right Lord mounts using four bolts (item 15, inset A), two washers under head of each bolt, and one washer and nut per each bolt. Torque bolts from head side of bolts 230 to 250 inch-pounds (25.99 to 28.25 N·m) (eight places, four per each side mount).

(2) Install engine mount support assembly (item 4, inset C), to engine mount assembly.

NOTE: Engine mount support assemblies (item 4) are matched by serial number with their respective engine mounts (item 3). Separate replacement of these items may result in alignment problems.

(3) Attach the support assembly (item 4, inset B) to the bottom Lord mount (item 10, inset B) by installing the longer bolts (item 11, inset B) through the forward holes of the bottom Lord mount and then through the support assembly (item 4, inset B). Install clip, washer, and nut on each forward bolt.

(4) Install the shorter bolts (item 12, inset B), and washer through the aft holes of the bottom Lord mount and engine mount support assembly. Install washer and nut on each aft bolt. Torque all four nuts 160 to 180 inch-pounds (18.08 to 20.34 N·m).

(5) Attach the opposite two ends of the engine mount support assembly (item 4, inset C) to the engine mount assembly using two bolts, four washers (two under head of each bolt), two washers (one with each nut) and nuts. On -3 engines, torque nuts 100 to 140 inch-pounds (11.30 to 15.82 N·m). On -10 engines, torque nuts 160 to 180 inch-pounds (18.08 to 20.34 N·m).

NOTE: Install shim (27-62078-S1), if required, between mount support assembly (item 4, inset C) and ears of mount assembly (item 3, inset C) to reduce any gap of 0.005 maximum.

(6) On -3 engines, tighten and torque all center shaft (items 1, inset A and Item 6, inset B) nuts 450 to 500 inch-pounds (50.84 to 56.49 N·m). Install keys. On -10 engines, tighten and torque side Lord mount center shaft nuts 800 to 1,000 inch-pounds (90.39 to 112.98 N·m). Tighten and torque bottom Lord mount center shaft nut 450 to 500 inch-pounds (50.84 to 56.49 N·m). Install cotter keys.

(7) Reference note in Lord Mount Installation procedures, **1.**, B, (6); if tube assembly was removed to facilitate installation, reinstall tube assembly.

2. Inspection/Check – Engine Mounts

A. Inspection/Check – Lord Mounts

NOTE: Inspect engine mount isolator assembly part numbers; LM-827-SA2, SA7, SA10, and SA11 at each engine removal IAW Service Letter 226-SL-034 and LORD, LM827-71-001.

CAUTION: AT ANY INSPECTION PERIOD WHEN CLEANING THE ENGINE, PROTECT MOUNTS FROM CONTAMINATION FROM CLEANING FLUIDS AND SOLVENTS. CLEAN ALL EXCESS CLEANING FLUID FROM MOUNTINGS. DURING ENGINE MAINTENANCE, PROTECT THE MOUNT FROM EXCESSIVE OIL EXPOSURE.

(1) Visually check metal for cracks or damage which indicate failure. Replace component as required.

(2) Check safetying of the mounting bolts of each engine mount. If safety is broken, torque bolts 450 to 500 inch-pounds (50.84 to 56.49 N·m) and safety.

CAUTION: IF LORD MOUNT-TO-ENGINE GEAR CASE BOLTS MEASURE LESS THAN 1.093 INCHES GRIP AND THREAD LENGTH, REPLACE WITH LM821-124 BOLTS.

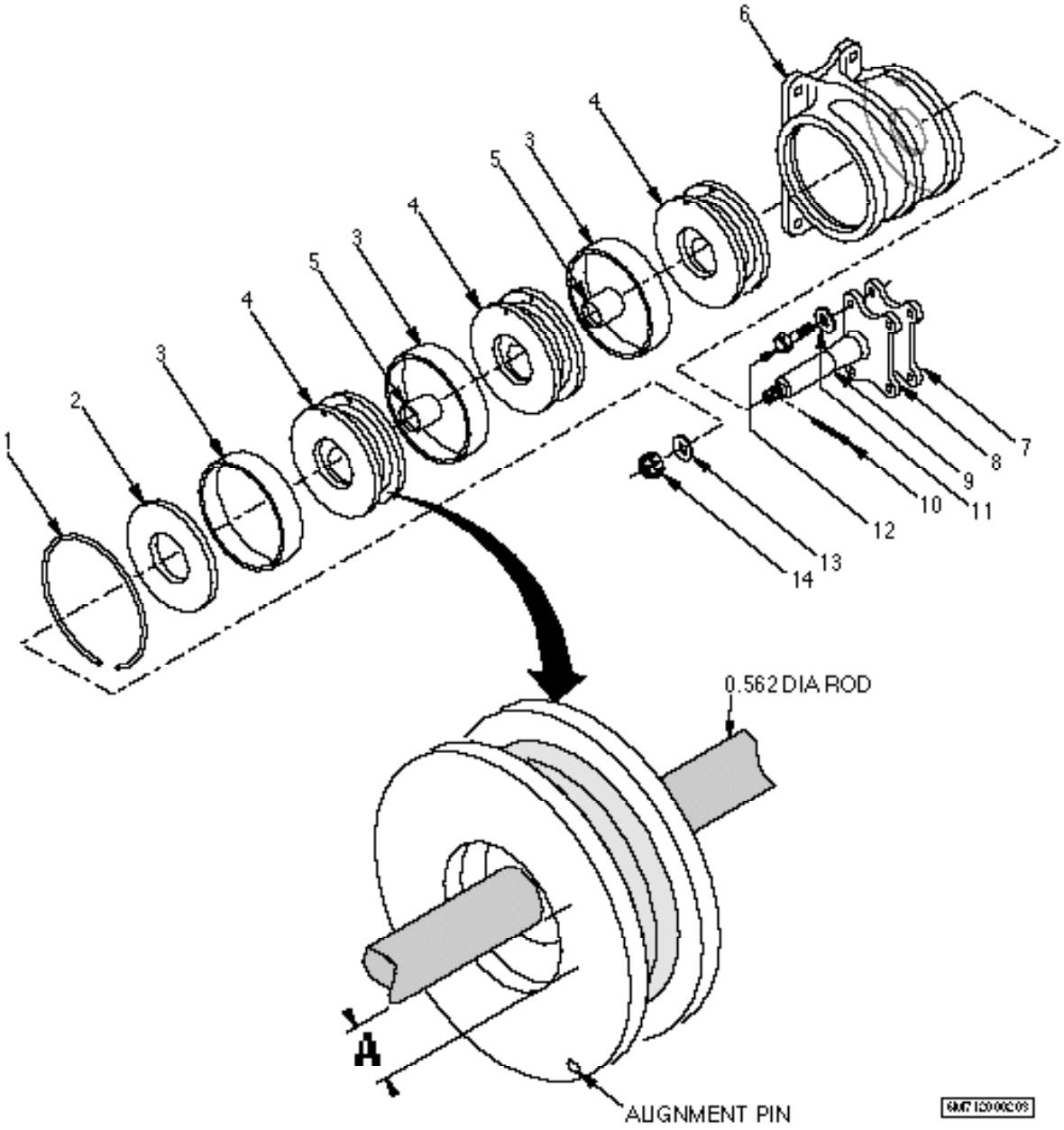
B. Disassembly Lord Mount (Reference Figure 203)

NOTE: The following Lord mount disassembly, cleaning, and inspection procedure is to be accomplished with regular scheduled engine change:

(1) Remove bolts (item 12), washers (item 11), and gasket (item 7), from mount shaft.

(2) Slide housing (item 6) with mounting off of mount shaft.

(3) Grip flat side of mount shaft (item 9) in padded vise; then remove cotter pin (item 10), nut (item 14), washer (item 13) from threaded end of mount shaft.

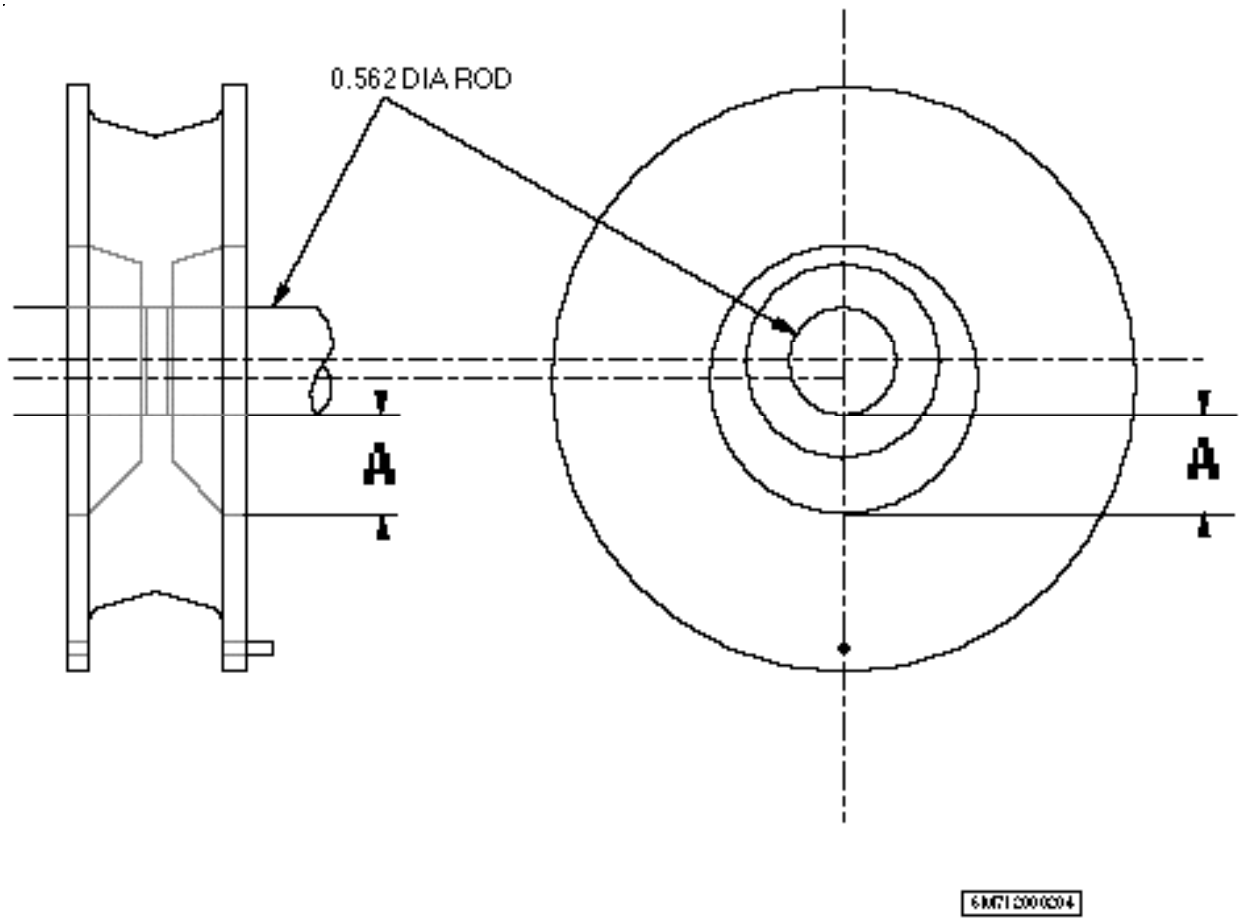


Lord Mount Buildup
Figure 203

(4) Apply approximately 600 pounds load to plate (item 2) using an arbor press and loading through a 2.25 inch diameter rod. While applying this load, use commercially available retaining ring pliers (Waldes No. 5 with 90° tips) to remove ring (item 13). Release load gradually.

(5) Remove plate (item 2), sandwich mounting (item 4), outer spacers (item 3), and inner spacers (item 5) from housing.

(6) Spread split outer spacers (item 3) to disengage them from sandwich mounting (item 4).



Checking Shear Set of Bonded Sandwich mounting
Figure 204

C. Inspection/Repair/Replacement

(1) Inspect individual component parts of Lord mounts as indicated in Table 201. Replace unsatisfactory parts as necessary.

PARTS	INSPECTION
Unbonded metal parts	Cracks: Using Magnaglo or Zyglo, as applicable. Replace component if cracks exist. Excessive scratches or nicks. Replace component as necessary. Wear of cadium plating on steel parts and replace if necessary per QQ-P-416 (a) or equivalent to .0003/.0005 thickness.
Bonded sandwich Mounting (Figure 203, item 4)	Metal-to rubber bond separation. Replace if separation has progressed around a circumference of 180° or has reached a depth of 0.10 inches. Permanent shear set (failure of rubber to return to its original shape when load is removed). Replace if dimension A, (The distance between the large hole and the rod, in line with the alignment pin) is less than 0.41 inches measured from hole toward pin (see Figure 203, and 204).
Housing (Figure 203, Item 6)	Cracks: Using Zyglo. Replace component if cracks exist. Excessive nicks, scratches, and gouges particularly in flange fillet radii or snap ring groove. Replace component if evident. Excessive galling or wear in 0.313/0.313 diameter holes in flange. Replace if greater than 0.315 inches.

Lord Mount Component Inspection
Table 201
(Continue)

PARTS	INSPECTION
Mount shaft (Figure 203, item 8)	Cracks: Using Magnaflux. Replace component if cracks are evident. (Check 0.05 inch fillet radius area at root of shaft stem closely.) Pilot diameter on forward face of mount shaft should be 1.496/1.498 inches. Replace component if less than 1.493 inches. Stem of mount shaft should have a Min. diameter of 0.555 inches on -3 engine assembly, and 0.644 inches diameter on -10 engine assembly. Galling or wear on threads of mount shaft. Replace component if evident.
Snap Ring (Figure 203, item 1	Cracks: Using Magnaflux. Replace if evident. Excessive distortion. Replace if evident.
Plate (Figure 203, item 2)	Cracks: Using Magnaflux. Replace if evident. Distortion: Replace if more than 0.020 inches out of flat.
Spacers (Figure 203, items 3 & 5)	Excessive galling or distortion. Replace if evident.

Lord Mount Component Inspection
Table 201
(End)

D. Reassembly Lord Mount (Figure 203)

(1) Install three outer spacers (item 3) over sandwich mounting (item 4).

(2) Insert spacers (item 3) and sandwich mountings (item 4) into housing (item 6) with an inner spacer (item 5) between each pair of sandwich mountings (item 4). Insert plate (item 2) in housing

(3) Apply load to plate (item 2) using arbor press loading with use of a 2.25 inch diameter rod. Continue load application, approximately 600 to 1,000 inch-pounds (67.79 to 112.98 N·m) until sandwich mountings (item 4) are sufficiently compressed and until internal snap ring groove of housing is visible.

NOTE: Excessive load or use of smaller diameter rod could result in permanent distortion to end plate (item 2). Use commercial retaining ring pliers to install snap ring (item 1).

(4) Install snap ring.

CAUTION: CHECK TO ENSURE THAT THE SNAP RING IS FULLY SEATED AND TIGHT WITHIN HOUSING BY MEASURING INNER DIAMETER OF INSTALLED RING AT SEVERAL LOCATIONS. INNER DIAMETER SHOULD BE 2.860 INCHES MINIMUM IF RING DOES NOT FULLY SEAT, INSERT 2.75 INCH WASHER INSIDE OF SNAP RING, AND USE SCREWDRIVER BETWEEN GAPS TO WEDGE SNAP RING OUT INTO HOUSING GROOVE, OR TAP RING CAREFULLY TO SEAT.

(5) Insert washer (item 9) over stem of mount shaft (item 8) with washer's chamfered side toward shoulder of shaft.

(6) Insert mount shaft (item 8), with bolts (item 12), and washers (item 11), in place, through inner member of bonded mountings.

(7) Install washer (item 13) and nut (item 14). Tighten nut (item 14) and torque 450 to 500 inch-pounds (50.84 to 56.49 N·m); maintain flat face of housing (item 6) parallel to bolt pattern centerline in mount shaft (item 8) with plus or minus 30 minutes.

NOTE: Twist in excess of the above amount will cause engine installation difficulties.

(8) Install cotter pin (item 10).

(9) Affix gasket (item 7) to back face of flange of mount shaft (item 8).

E. Inspection/Check – Engine Mount Support Assembly Truss

(1) Visually inspect lower engine mount truss assembly at regularly scheduled intervals during inspection program and after hard landing or extreme turbulence.

(2) Remove lower aft cowling, see Section 71-10-00.

(3) Using number 10 power or higher magnification glass and high intensity light, inspect weld and adjacent truss tube for signs of cracking. Discovery of cracks is cause for repair or replacement of mount truss.

3. Cleaning/Painting – Engine Mounts

A. Cleaning

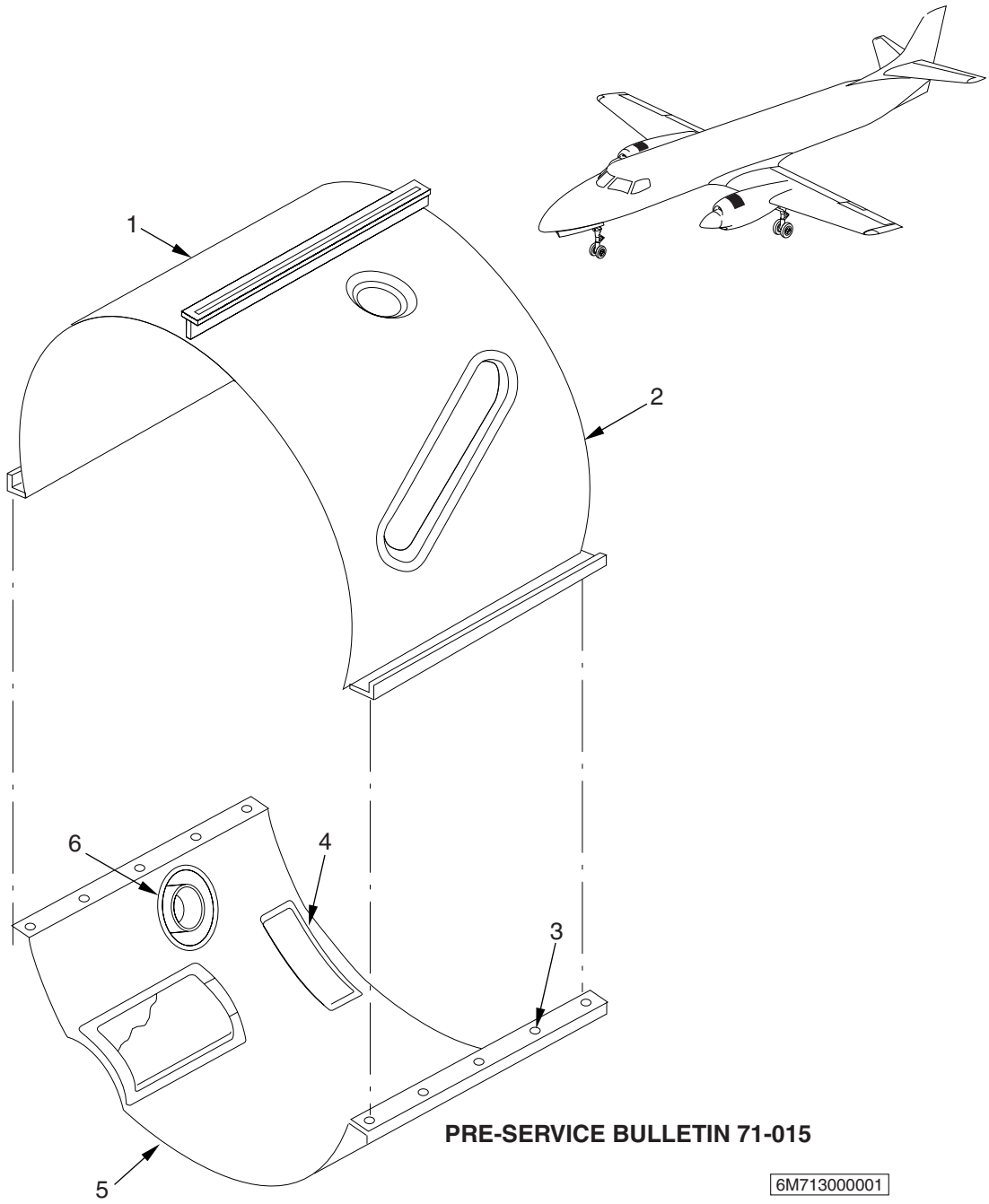
(1) Wipe oil and dirt from bonded rubber parts using a clean, dry cloth.

CAUTION: DO NOT USE CLEANING SOLVENT ON BONDED RUBBER PARTS.

(2) Wash unbonded metal parts in dry cleaning solvent, Federal Specification PS-661 or equivalent.

MAINTENANCE MANUAL

FIRESEALS – DESCRIPTION AND OPERATION



Combuster Heat Shield
Figure 1

FIRESEALS – DESCRIPTION AND OPERATION

1. General

A. On SA226 Series aircraft prior to effectivities T360, AT423, and TC361, a three section stainless steel heat shroud assembly (se Figure 1) is installed around the combustion section of the engine from the engine furnished fireseal to the seal assembly of the aspirator (see Figure 3). M7 Aerospace Service Bulletin 226-71-015 details the optional removal of the heat shroud assemblies, rework of the aspirator, revision of the bleed air system, and installation of vented ignitor plug heat shield. Each section of the heat shroud assembly mates to the other with Camloc fasteners. All sections of the assembly attach with Camloc fasteners to retainers riveted to the engine fireshield at the forward edge, and when attached together form a friction fit around the seal assembly at the rear. Two sections, the upper left and right shroud join at the top of the engine and cover the upper half of the combustion section. The third section, lower shroud, covers the lower half of the combustion section. The shroud assembly is shaped and fitted to accommodate accessories and ignitor circuitry. The shroud assembly is vented through the aspirator to the exhaust pipe. When the engine is operating, the low pressure created by the exhausted air withdraws the hot air within the shroud overboard to prevent high temperature buildup in the nacelle.

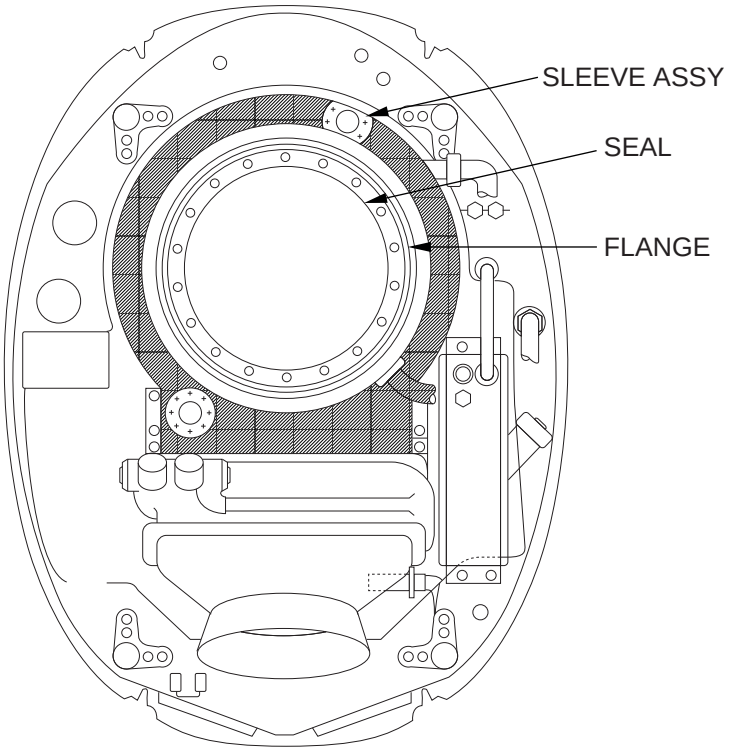
2. Ignitor

A. Plug Heat Shields

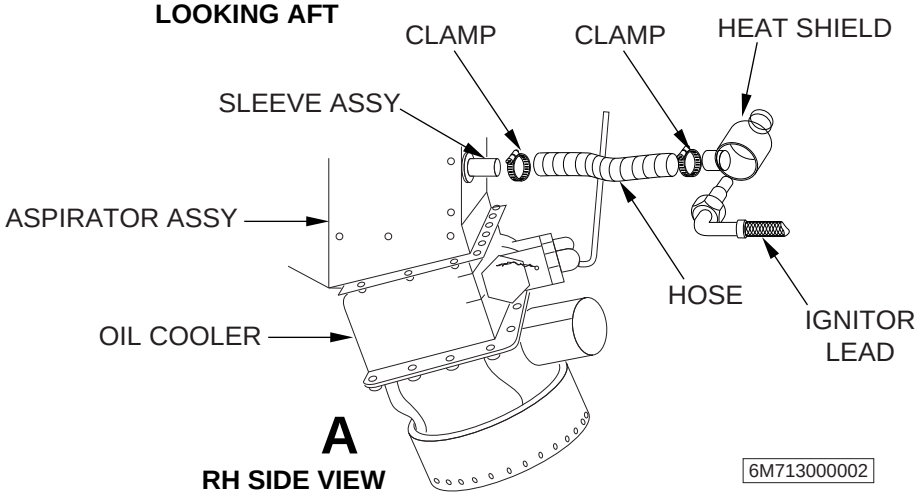
(1) An ignitor plug heat shield is installed on the bottom ignitor plug and on the ignitor plug located at the 11 o'clock position on TPE331-3U-303/304 engines prior to SA226 series aircraft T360, AT423, TC361 and those aircraft not having complied with M7 Aerospace Service bulletin 226-71-015 (see Figure 4, item 9). On TPE331-10U-501 engines prior to T360 and those aircraft not having complied with M7 Aerospace Service Bulletin 226-71-015, an ignitor plug heat shield is installed on both bottom ignitor plugs (see Figure 5, items 11 and 12). On SA226 series aircraft T360-up, AT423-up, TC361-up and those aircraft having complied with M7 Aerospace Service Bulletin 226-71-015, a vented ignitor plug heat shield is installed on both bottom ignitor plugs of both the TPE331-3U-303/304 and TPE331-10U-501 engines. On the -3 engines, the heat shields are vented to both lower sides of the aspirator. On -10 engines, the heat shields are vented to the right of the oil cooler air scoop.

3. Aspirator (Figure 2)

A. The aspirator assembly is mounted to the forward portion of the nacelle firewall. It encircles the exhaust stack and extends downward to a point; whereas, its flanged duct slopes forward to provide the mounting surface for the oil cooler. The upper cylindrical portion of the assembly extends forward slightly past the forward end of the engine tail cone. A circular flange around the engine provides a mounting surface for the aspirator cover and seal assembly. Air entering the oil cooler air scoop passes through the oil cooler, is ducted upwards and around the tail cone, then exits through the exhaust stack. Low pressure, created by the exhausted air when the engine is operating, draws hot air generated by the engine accessories overboard through the exhaust stack, thus maintaining nacelle temperature within desired limits. On aircraft not using heat shrouds and on aircraft having complied with M7 Aerospace Service Bulletin 226-71-015, the aspirator vents the combustion section of the engine through two one inch holes located at the top exterior of the aspirator. On aircraft using -3 engines and having complied with M7 Aerospace Service Bulletin 226-71-015, the aspirator vents the ignitor plug head shields through two sleeve assemblies, each consisting of a one inch opening located at both lower exterior sides of the aspirator.



ASPIRATOR ASSEMBLY
LOOKING AFT



Aspirator
Figure 2

1. Removal/Installation – Fireseals

A. Removal Lower Shroud Assembly

(1) Ensure all electrical power to the aircraft is OFF.

(2) Remove side cowling, see Section 71-10-00.

(3) Remove the lower aft cowling, see Section 71-10-00.

(4) Remove starter-generator, see Section 24-00-00.

(5) Disconnect the lower ignitor plug lead.

(6) Loosen Camloc fasteners.

(7) Remove shroud with care to avoid bending or crimping.

B. Left And Right Upper Shrouds

(1) Ensure all electrical power to aircraft is OFF.

(2) Remove side cowling, see Section 71-10-00.

(3) Disconnect upper ignitor plug lead.

(4) Loosen Camloc fasteners.

(5) Remove shrouds carefully to avoid bending or crimping.

C. Installation Lower Shroud Assembly

NOTE: Ensure ignitor plug shield fits easily through access hole in lower shroud assembly before installing shroud.

For ease of installation, the combustion case overboard drain line may be removed. See Section 71-70-00, Engine Drain Block Installation procedures for drain line location.

(1) Install lower shroud assembly inserting the aft portion of the assembly between the oil cooler and heat exchanger assembly fireseal. Ensure correct alignment and fit. Ensure that the front of the shroud assembly aligns at the front of the heat shroud retainer flange.

NOTE: A 1/4 inch clearance must be maintained between the lower shroud assembly and the oil tank. On new installations, remove approximately 10/100 inches, if necessary, from aft portion of lower cover assembly in order to obtain correct alignment and fit.

(2) Install Camloc fastener plate on sides of lower shroud assembly using five screws for each plate.

NOTE: For ease of installation, install Camloc fasteners by starting in the middle of the lower shroud assembly and work upwards.

D. Left And Right Upper Shroud Assemblies

(1) Install left and right shroud assemblies by installing Camloc fasteners.

NOTE: For ease of installation, first install fasteners at the shroud assembly sides. Ensure ignitor plug shield fits easily through access hole in shroud assembly before installing. On new installation, trim sides of shroud assembly as required for flush fit with front of engine flange shroud retainers, for contact with fireseal, and for clearance between shroud assembly and cover over heat exchanger (aspirator) assembly access area.

(2) Install combustion case overboard drain, if removed, to facilitate heat shroud installation.

(3) Connect ignitor plug lead.

ATTACH FITTINGS – DESCRIPTION AND OPERATION

1. General

A. The powerplant attach fittings used on SA226 series aircraft consist of engine control cable brackets for mounting of power, fuel shutoff, and RPM control cables, engine shock mount mounting brackets, forward nose cowling mounting brackets, and inlet valve mounting brackets.

B. The Engine Control Cable Brackets Include

(1) A bracket for mounting of the power lever and fuel shutoff control cables which are located below, and attached to, the flow divider and RPM control bracket.

(2) A bracket for mounting of a bellcrank assembly for the operation of the fuel shutoff and feathering pump.

(3) The fuel shutoff and feathering pump bellcrank assembly.

(4) A bracket, bolted to the flange immediately forward of the fuel flow divider at the bottom center of the compressor section, for mounting of the RPM control cable.

C. Engine Mount Mounting Brackets Include

(1) A left and right bracket for mounting of the engine left and right shock (Lord) mounts.

(2) A left and right bracket for mounting the left and right shock mount bracket [see step 1.C.(1)] to the forward nose cowling aft frame.

(3) A bracket for mounting the engine bottom shock (Lord) mount to the forward nose cowling aft frame.

(4) A lower left and right bracket for securing the forward nose cowling aft frame to the engine reduction gear case section.

NOTE: On earlier model aircraft (T201-275, T277-291, AT001-069, and TC201-275), two brackets, located at the upper left of the nose cowling section between the forward and center cowling frames, are used for mounting the intake inlet anti-ice valve.

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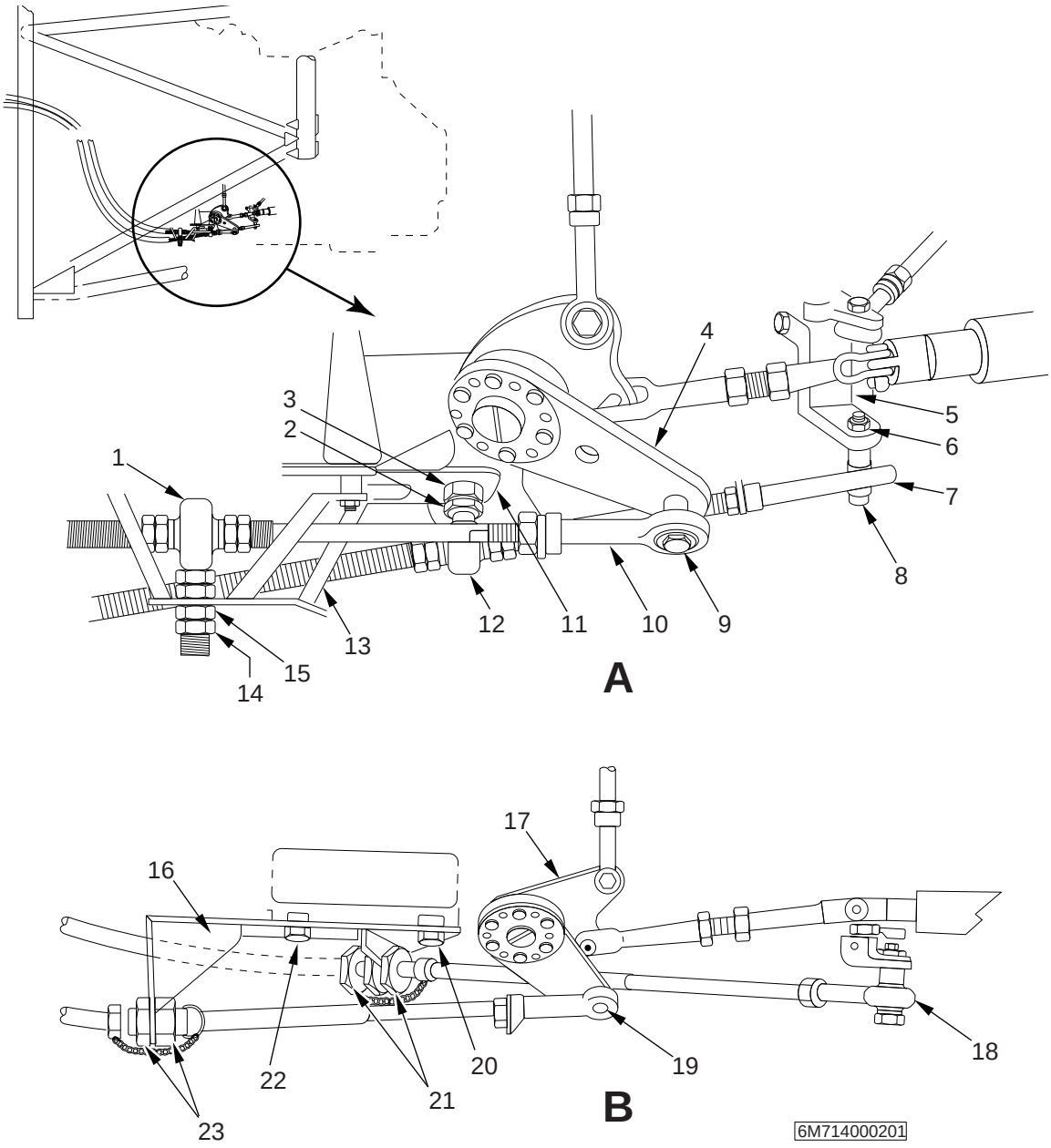


Fuel Shutoff Control Cable Installation Figure 201

ATTACH FITTINGS – MAINTENANCE PRACTICES

1. Removal/Installation – Attach Fittings

A. Removal – Power Lever And Fuel Shutoff Control Cable Bracket (T201-383, AT001-069, TC201-384). (Reference Figure 201, Inset A)

(1) Remove the right cowling, see Section 71-10-00. Remove the starter-generator, see Removal procedures, Chapter 24.

(2) Disconnect rod-end (item 10) from bellcrank assembly (item 4) by removing retaining bolt, (item 9) washer, and nut.

(3) Remove Pal nut (item 14) and jam nut (item 15), and free the fuel shutoff cable swivel bearing (item 1) from bracket (item 13).

(4) Disconnect the rod-end (item 7) of the power control cable from the propeller pitch control arm (item 5) by removing the retaining bolt, (item 8) washer, and nut (item 6).

(5) Remove the Pal nut (item 2) and jam nut (item 3), and free the swivel bearing (item 12) from the bracket leg (item 11).

(6) Remove retaining bolt, washer, and spacer, attaching clamps, and remove bracket (item 13).

B. Removal – Power Lever and Fuel Shutoff Control Cable Bracket (T384-999, AT070-999, TC385-999) (Reference Figure 201, Inset B)

(1) Disconnect rod-end (item 19) of fuel shutoff control cable from bellcrank (item 17) by removing retaining bolt, washer, and nut.

(2) Disconnect rod-end (item 18) of power control cable from power lever arm by removing retaining bolt, washer, and nut.

(3) Remove rod-ends (items 18 and 19) from both control cables.

(4) Remove nuts (item 21) from both control cables and remove cables from bracket (item 16).

(5) Remove retaining bolts (item 20) from bracket (item 16) and remove bracket.

C. Installation – Attach Fittings (T201-383, AT001-069, TC201-384). (Figure 201, Inset A)

(1) Install bracket (item 11) by installing retaining bolt washer, and spacer. Install attaching clamps and secure tube assembly.

(2) Install swivel bearing (item 12) of power control cable by installing Pal nut and jam nut securing it through leg of bracket (item 11) and bracket (item 13).

(3) Install rod-end (item 7) of the power control cable to the propeller pitch control arm by installing retaining bolt, washer, and nut.

(4) Install swivel bearing (item 1) for fuel shutoff cable to bracket by installing Pal nut and jam nut.

(5) Connect rod-end (item 10) to bellcrank by installing bolt, washer, and nut.

(6) Accomplish general rigging requirements and safetying of fuel shutoff and power controls, see Rigging procedures, Chapter 76.

D. Installation – Attach Fittings (T384-999, AT070-999, TC385-999). (Reference Figure 201, Inset B)

(1) Place bracket (item 16) into place and install retaining bolts.

(2) Install both control cables into bracket (item 16) and install nuts (item 21) securing cables to bracket.

(3) Install rod-ends (items 18 and 19) onto both control cables.

(4) Connect rod-end (item 18) of power control cable to bellcrank and install retaining bolt, washer, and nut.

(5) Connect rod-end (item 19) of fuel shutoff control cable to bellcrank and install retaining bolt, washer, and nut.

(6) Accomplish general rigging requirements and safetying of fuel shutoff and power controls, see Rigging procedures, Chapter 76.

(a) Install starter-generator, see Installation Procedures, Chapter 24.

(b) Install right and lower cowling, see Installation procedures.

E. Removal – Fuel Shutoff Control/Feathering Pump Bellcrank and Mount Bracket (T-201-383, AT001-069, TC201-384) (Figure 202).

(1) Remove the right and lower cowling, see Removal procedures, Section 71-10-00. Remove the starter-generator, see Removal procedures, Chapter 24.

(a) Disconnect the rod-end (item 5) of the fuel shutoff control cable (item 1) from the bellcrank assembly (item 2) by removing the retaining bolt, washer, and nut.

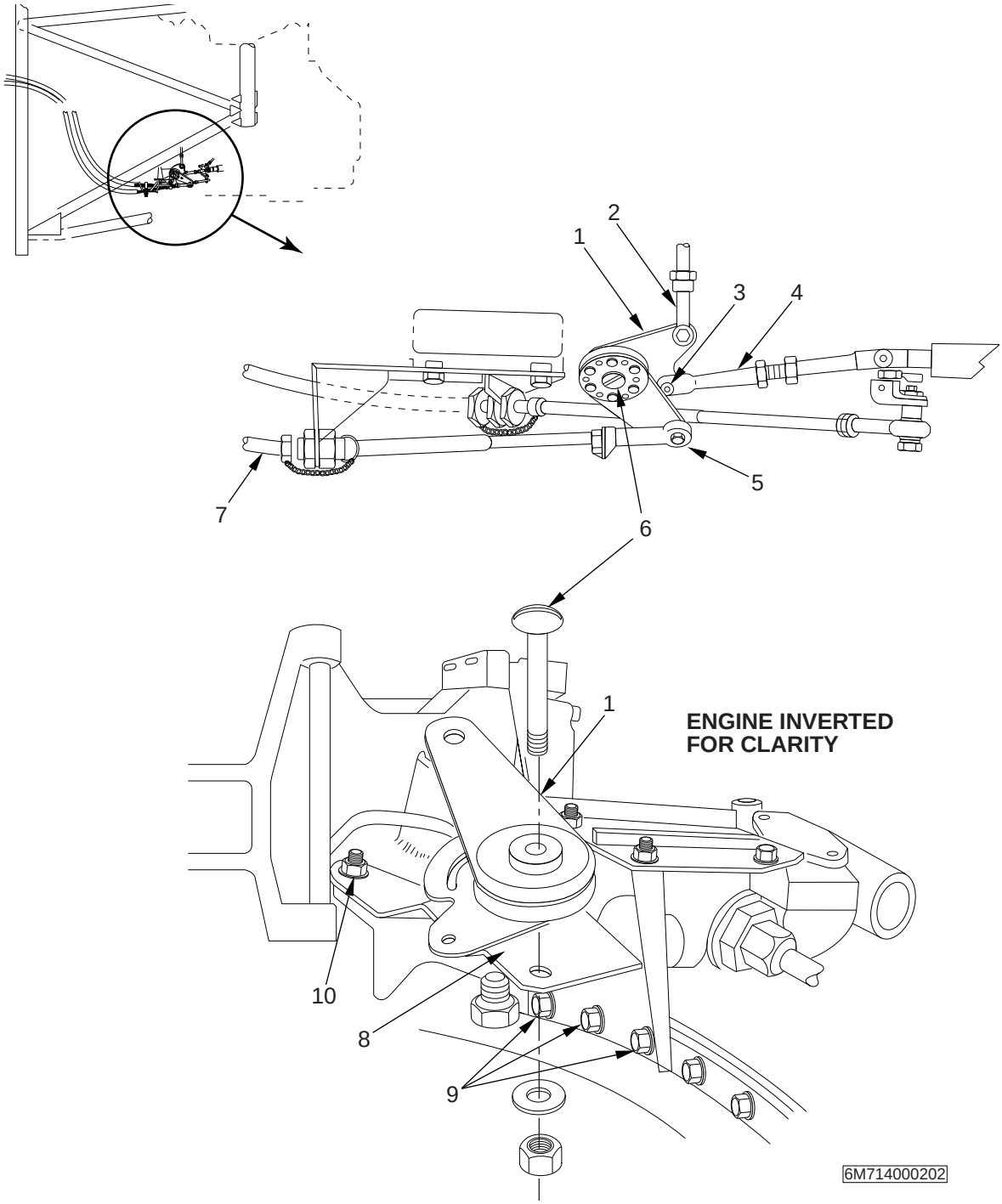
(b) Disconnect the rod-end (item 3) of the manual fuel shutoff linkage from the bellcrank assembly (item 2) by removing the bolt, washer, and nut.

(c) Remove the clevis pin (item 4) of the propeller feathering valve linkage from the bellcrank assembly and disconnect the linkage.

(d) Remove the bellcrank assembly mount bolt (item 6) and remove bellcrank (item 2).

(e) Remove the Pal nut (item 9) and jam nut (item 10) and free the swivel bearing (item 7) of the power control cable from the leg of bracket (item 11).

(f) Remove the three flange bolts (item 12) and the retaining bolt (item 8) and remove the bellcrank assembly mount bracket (item 11).



Propeller Feathering Mechanism
Figure 202

F. Removal – Fuel Shutoff Control/Feathering Pump Bellcrank and Mount Bracket (T384-999, AT070-999, TC385-999). (Reference Figure 202)

(1) Disconnect the rod-end (item 5) of the fuel shutoff control cable by removing the retaining bolt, washer, and nut.

(2) Disconnect the rod-end (item 2) of the manual fuel shutoff linkage from the bellcrank assembly by removing the bolt, washer, and nut.

(3) Remove the Clevis pin (item 3) of the propeller feathering valve linkage from the bellcrank assembly and disconnect the linkage.

(4) Remove the bellcrank assembly mount bolt (item 6) and remove the bellcrank assembly (item 1).

(5) Remove the three flange bolts (item 9) and retaining bolt (item 10) and remove bellcrank assembly mount bracket (item 8).

G. Installation – Attach Fittings (T201-383, AT001-069, TC201-384) (Reference Figure 203).

(1) Install bracket (item 11) by installing the three flange bolts (item 12) and retaining bolt (item 8).

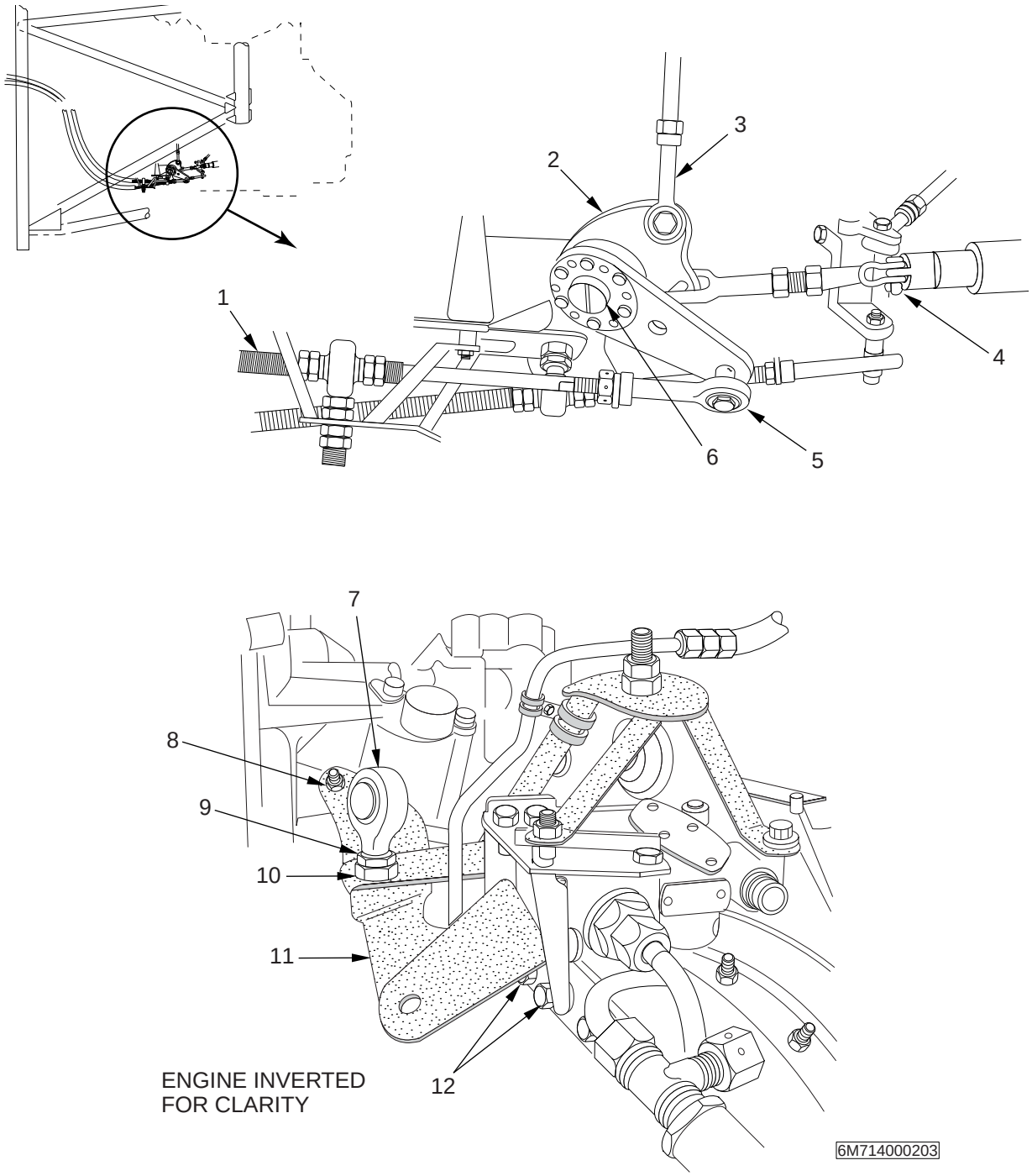
NOTE: Install bracket with spacers between it and the existing AiResearch bracket.

(2) Install bellcrank onto bracket by installing with bolt (item 6) and nut.

(3) Connect propeller feathering linkage to the bellcrank assembly and install Clevis pin (item 4), washer, and cotter key.

(4) Connect the rod-end bearing of the manual fuel shutoff linkage (item 3) to the bellcrank assembly and install bolt, washer, and nut.

(5) Connect the rod-end of the fuel shutoff control cable (item 5) and install bolt, washer, and nut.



Propeller Feathering Bellcrank
Bracket installatiuon
Figure 203

H. Installation – Attach Fittings (T384-999, AT070-999, TC385-999) (Reference Figure 203).

(1) Install bracket (item 8) by installing the three flange bolts (item 9) and retaining bolt (item 10).

NOTE: Install bracket with spacers between it and the existing AiResearch bracket.

(2) Install bellcrank onto bracket by installing bolt, (item 6) washer, and nut (item 7).

(3) Connect propeller feathering linkages (item 4) to the bellcrank assembly and install Clevis pin (item 3), washer, and cotter key.

(4) Connect the rod-end (item 2) of the manual fuel shutoff linkage to the bellcrank assembly (item 1) and install bolt (item 6), washer, and nut.

(5) Connect the rod-end (item 5) of the fuel shutoff control cable (item 7) and install bolt, washer, and nut.

(a) Actuate the STOP and FEATHER control and check for proper operation. Check rigging of controls and make any adjustment necessary. Safety rod-end nuts, see Rigging procedures, Chapter 76.

(b) Install the starter-generator, see Installation Procedures, Chapter 24.

(c) Install cowling, see Section 71-10-00.

I. Removal – RPM Control Cable Bracket (T201-383, AT001-069, TC201-384) (Reference Figure 204).

(1) Remove the left and lower cowling, see Removal procedures, Section 71-10-00.

(2) Disconnect the rod-end (item 13, inset C) of the RPM control cable from the RPM lever arm of the fuel control by removing the bolt, washers and serrated plate, and nut.

(3) Remove the Pal nut (item 4) and jam nut (item 5, inset A) from the inside of the bracket (item 1) remove the swivel bearing with the RPM control cable from the bracket.

(4) Remove the retaining bolt (item 2, inset B) securing the leg of the bracket at the nose case.

(5) Remove the two retaining bolts (item 3, inset B) securing the two legs of the bracket from the compressor flange.

(6) Remove hose clamps (item 7) to allow removal of the bracket, remove bracket.

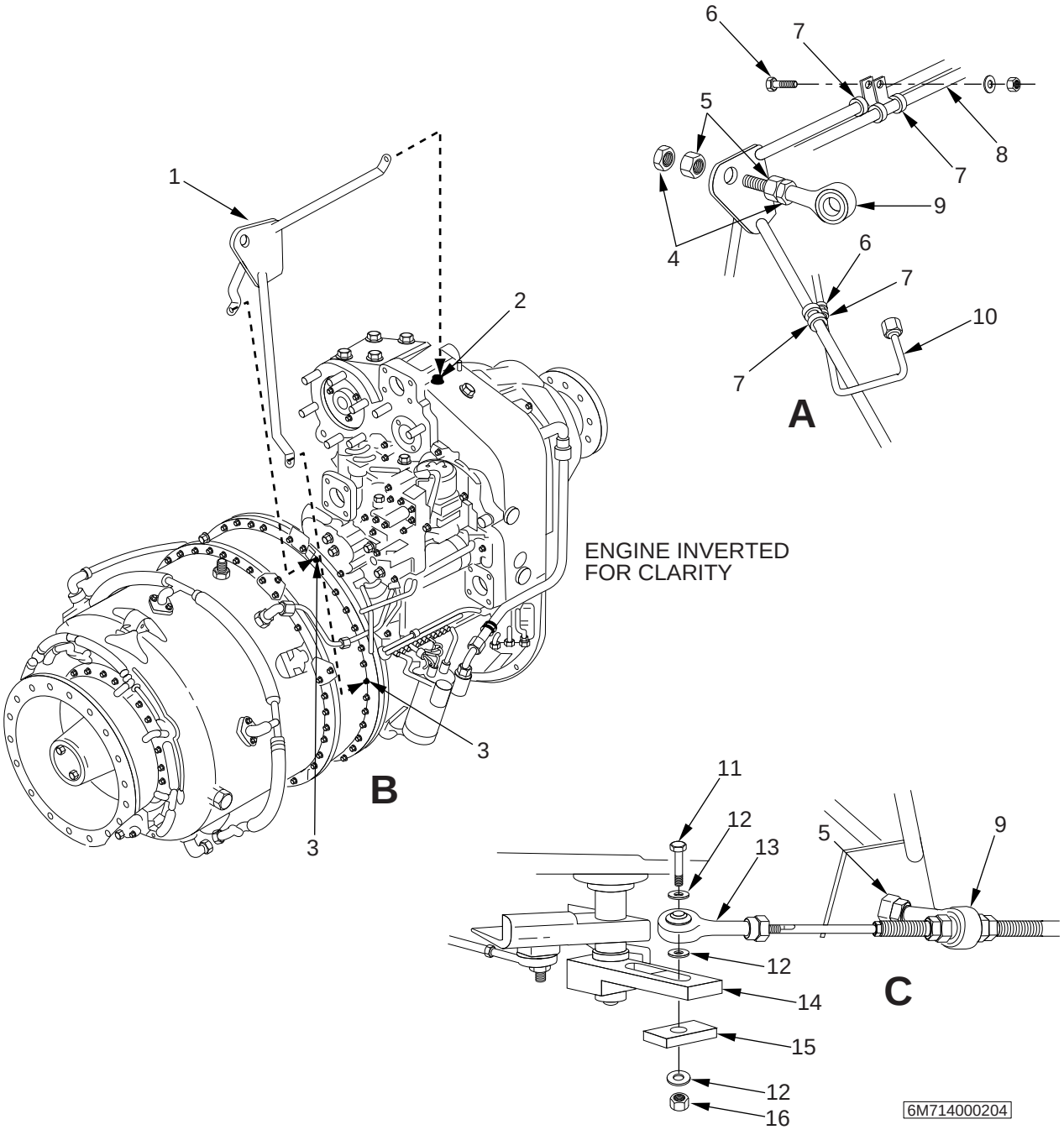
J. Removal – RPM Control Cable Bracket (T384-999, AT070-999, TC385-999) (Reference Figure 204).

(1) Disconnect rod-end bearing (item 9, inset B) of the RPM control cable. Remove nut (item 11), washer (item 7), plate (item 10) and bolt (item 6).

(2) Remove rod-end (item 9, inset B) from the control cable.

(3) Remove nut (item 8, inset B) and withdraw RPM control cable from the bracket assembly (item 4).

(4) Remove retaining bolts (items 1, 3 and 5) securing the three legs of the bracket assembly at the fuel regulator, compressor flange, and at power lever and fuel shutoff control bracket (item 2, inset A). Remove bracket assembly.



RPM Control Cable Bracket Installation
Figure 204

K. Installation – Attach Fittings (T201-393, AT001-069, TC201-384) (Reference Figure 204).

(1) Position bracket assembly (item 1) into place and install the two retaining bolts (item 3) securing two of the legs of the bracket assembly to the compressor flange.

(2) Install retaining bolt (item 2) securing the third leg of the bracket at the nose case.

(3) Install swivel bearing with RPM control cable to bracket assembly (item 1). Secure swivel bearing to the bracket assembly (item 1), with jam nut (item 5) and pal nut (item 4).

L. Installation – Attach Fittings (T384-999, AT070-999, TC385-999) (Reference Figure 205).

(1) Position bracket assembly (item 4) into place and install the retaining bolts securing the three legs of the bracket assembly at the fuel regulator, compressor flange, and power lever and fuel shutoff control bracket (item 2, inset A).

(2) Insert RPM control cable through bracket assembly, (item 1) and install retaining nut (item 8).

(3) Install rod-end (item 9, inset B) onto the control cable.

(a) Connect the rod-end of the RPM control cable onto the lever arm of the fuel control. Center the rod-end onto the lever arm such that the distance from center of mount bolt to end of RPM lever is 1/2 inch. Install serrated plate and align plate with serration on bottom of RPM lever arm of fuel control. Install washer and locknut.

(b) Accomplish general rigging check of power lever control, see Rigging Procedures, Chapter 76.

(c) Install left and lower cowling, see Installation procedures, Chapter 71.

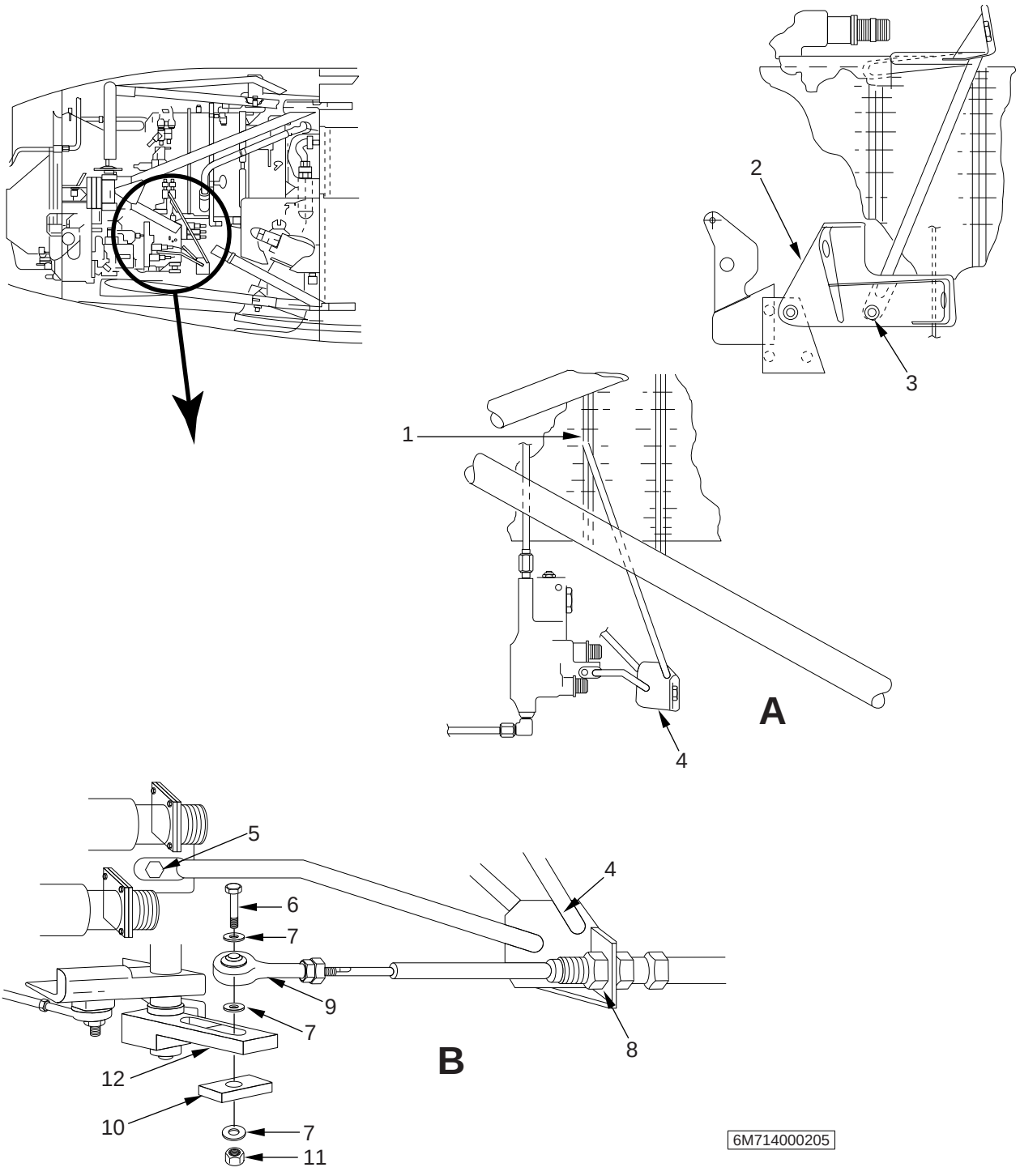
M. Removal – Frame to Side Shock Mount Left and Right Brackets (Reference Figure 206)

NOTE: See Engine Shock (Lord) Mount Removal procedures, Section 71-20-00, for removal of brackets and engine side shock mounts when engine is on hoist.

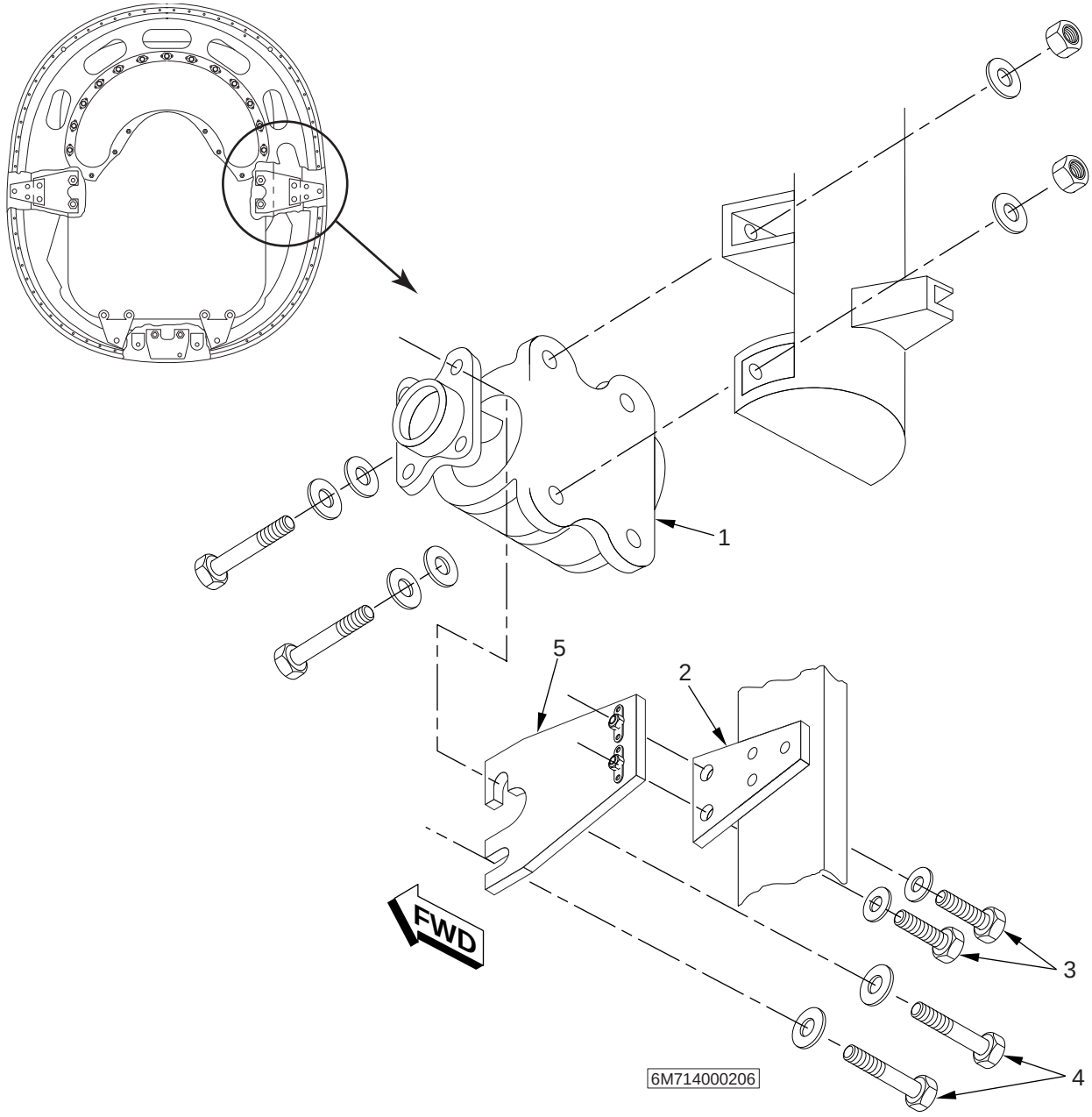
(1) Loosen the two retaining bolts securing bracket (item 5) to engine side shock mount (item 1).

(2) Remove the retaining bolts (item 3) securing bracket (item 5) to frame bracket (item 2).

(3) Pivot bracket (item 5) upward using the engine side shock mount upper bolt as a pivot point until the bracket is free of side shock mount lower bolt. Lift upward until clear of the side shock mount upper bolt, remove brackets.



Attach Fittings
Figure 205



Side mount Brackets
Figure 206

N. Installation – Frame to Side Shock Mount Left and Right Brackets (Reference Figure 206)

NOTE: See Engine Shock (Lord) Mount Installation procedures, Section 71-20-00, for installation of brackets and engine side shock mounts when engine is on hoist.

(1) Hook bracket (item 5) over the loosened upper bolt of the engine side shock mount and pivot the brackets downward over the loosened lower bolt.

NOTE: Install brackets onto shock mount such that the nutplates on the brackets face toward aft of the engine.

(2) Tighten bracket bolts to shock mount and torque bolts 450 to 500 inch-pounds. Safety bolts using 0.041 safety wire.

(3) Install bolts (item 3) with washer under head of each bolt securing brackets (item 5) to frame brackets (item 2).

O. Removal – Frame to Bottom Shock Mount Bracket (Reference Figure 207)

NOTE: See Engine Shock (Lord) Mount Removal procedures, Section 71-20-00, for removal of bracket and engine bottom shock mount when engine is on hoist.

(1) Loosen the two retaining bolts (item 2) securing the bottom shock mount bracket (item 1) to the bottom mount.

(2) Remove the retaining bolts (item 3) securing the bottom shock mount bracket to the cowling frame.

(3) Pivot the bottom shock mount bracket to the right until it clears the shock mount retaining bolt. Then lift the bracket clear of the opposite retaining bolt and remove the bracket

P. Installation – Frame to Bottom Shock Mount Bracket (Reference Figure 207)

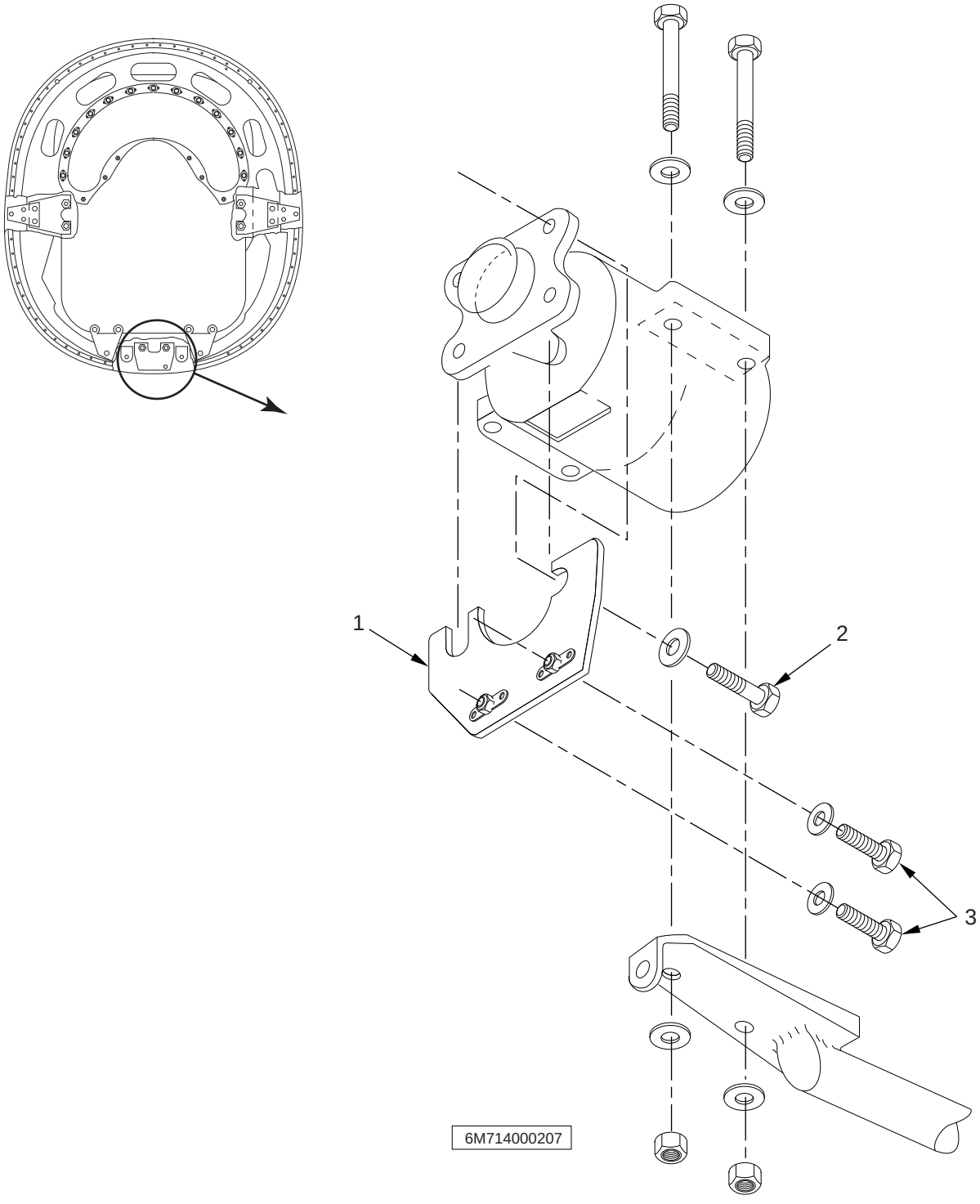
NOTE: See Engine Shock (Lord) Mount Removal procedures, Section 71-20-00, for removal of bracket and engine bottom shock mount when engine is on hoist.

(1) Hook the bottom bracket (item 1) over the loosened retaining bolt of the engine bottom shock mount and pivot the bracket onto the opposite retaining bolt.

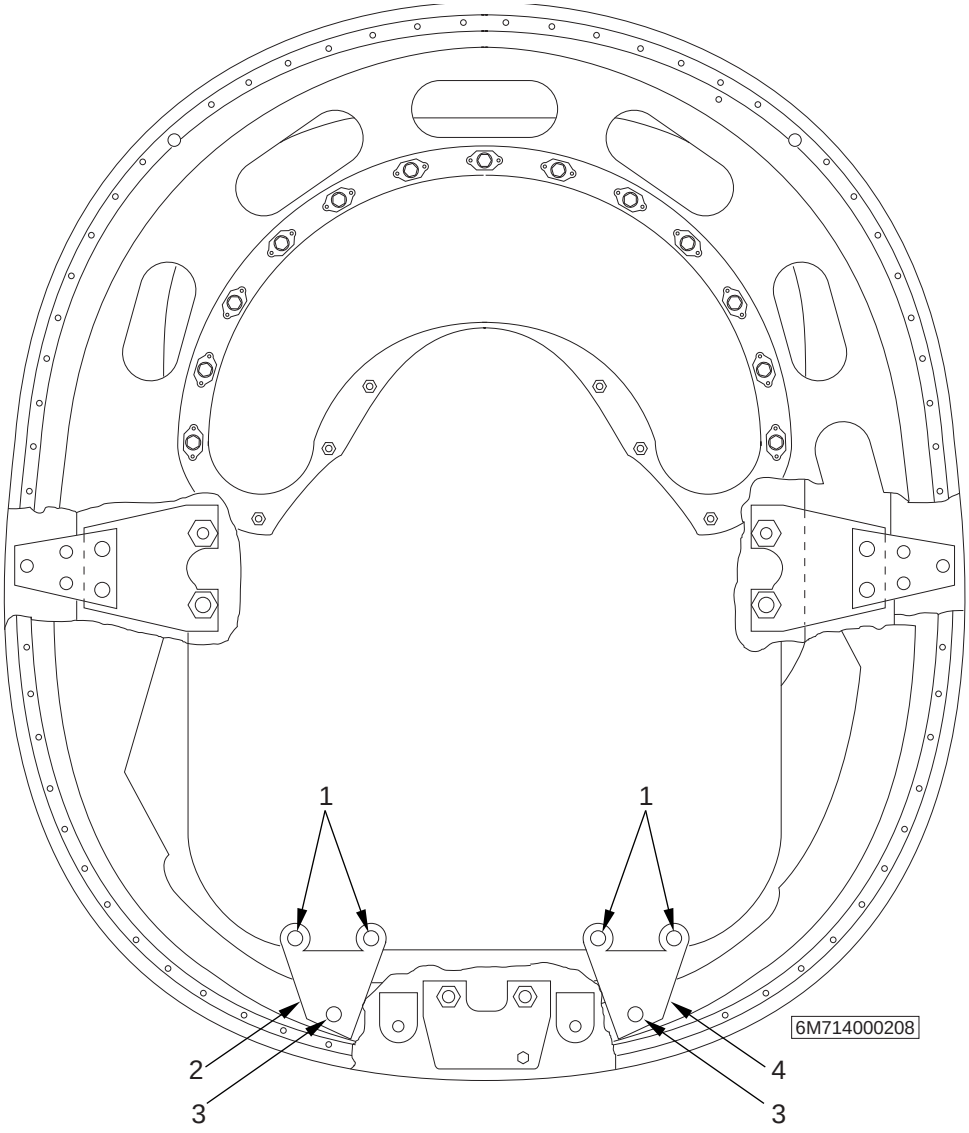
NOTE: Install bracket onto shock mount such that the nutplates on the bracket faces toward aft of the engine.

(2) Tighten bracket bolts to shock mount and torque bolts 450 to 500 inch-pounds. Safety bolts using 0.041 safety wire.

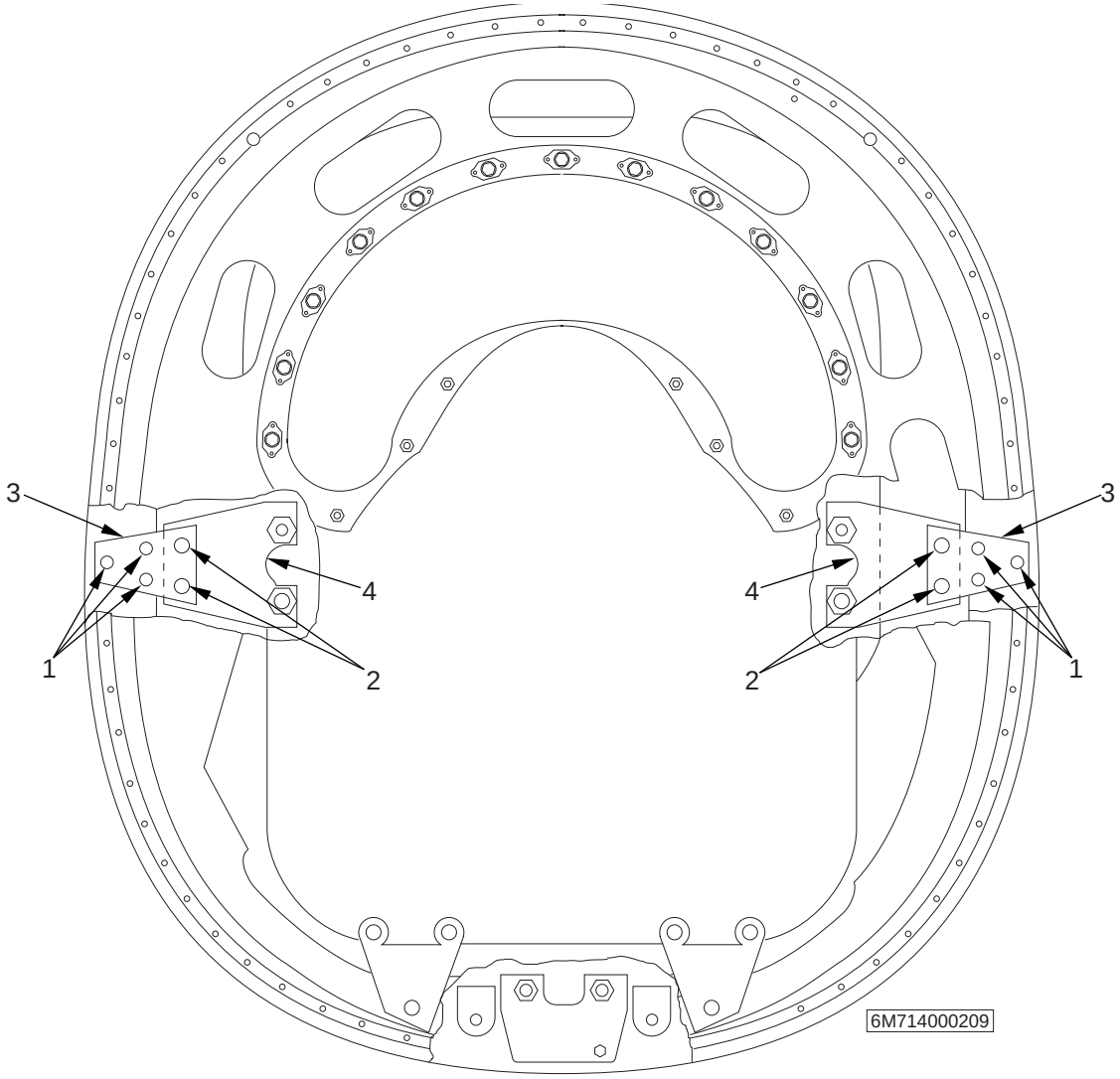
(3) Install the retaining bolt (item 3) securing the bracket to the cowling frame.



Lower Mount Bracket
Figure 207



Forward Cowling Lower Mount Brackets
Figure 208



Forward Cowling Side Mount Brackets
Figure 209

R. Installation – Frame to Reduction Gear Case Left and Right Brackets (Reference Figure 208)

(1) Position left and/or right bracket (items 2 or 4) into place. Install the retaining bolt with washer under head of bolt securing the bracket to the reduction gear case of the engine.

(2) Install the two retaining bolts with a washer under head of each bolt securing the left and/or right bracket to the nose cowling frame assembly.

Q. Removal – Frame to Reduction Gear Case Left and Right Brackets (Reference Figure 208)

(1) Remove lower forward cowling of nose cowling assembly, see Section 71-20-00.

(2) Remove lower cowling panel, see Section 71-10-00.

(3) Remove the two retaining bolts (item 3) securing the left and/or right bracket (items 2 or 4) to the nose cowling frame assembly.

(4) Remove the retaining nut (item 1) securing the left and/or right bracket (items 2 or 4) to the reduction gear case.

(5) Remove brackets.

S. Removal – Left and Right Frame Brackets to Side Shock Mount Brackets (Reference Figure 209)

NOTE: See Engine Shock (Lord) Mount Removal procedures, Section 71-20-00 for removal of engine mounts when engine is on hoist.

(1) Remove retaining bolts (item 2) securing brackets (item 3) to engine side shock mount bracket (item 4).

(2) Drill out the three rivets (item 1) securing the brackets (item 3) to the aft frame of the nose cowling assembly and remove brackets.

T. Installation – Left and Right Frame Brackets to Side Shock Mount Brackets (Reference Figure 209)

NOTE: See Engine Shock (Lord) Mount Removal procedures, Section 71-20-00 for removal of engine mounts when engine is on hoist.

(1) Drill through the three pilot holes of each new bracket (item 3) with No. 30 drill.

(2) Place bracket (item 3) into position and secure to aft frame of nose cowling assembly by installing Cleco fastener onto each bracket to hold it into place.

(3) Install rivets (item 1) (three each MS20470AD44 per bracket) securing the brackets to the aft frame of the nose cowling assembly.

(4) Install retaining bolts (item 2) securing brackets (item 3) to engine side shock mount brackets (item 4).

U. Removal – Anti-Ice Valve Brackets (Reference Figure 206)

NOTE: See M7 Aerospace Service Bulletin 226-30-011 for conversion procedures between TPE331-3U-303G/304G engines using single or dual anti-ice valves. Also see SERV-30 of Maintenance Practices.

(1) Remove anti-ice valve by removing connecting tubing and bolts (item 5) from support (item 1), and remove both the valve and attached bracket (item 4).

(2) Drill out rivets securing bracket (item 3) between cowling frames and remove bracket. (3) Remove retaining bolts (item 6) securing bracket (item 4) to anti-ice valve and remove bracket.

V. Installation – Anti-Ice Valve Brackets (Reference Figure 206)

(1) Position new bracket (item 3) into place and install between engine cowling frames with use of MS20470AD-4 rivets.

(2) Install bracket (item 4) onto anti-ice valve with installation of retaining bolts (item 6).

(3) Install anti-ice valve with bracket (item 4) onto support bracket (item 1) by installing retaining bolts (item 5).

(4) Perform operational Check of anti-ice system as described in Maintenance Manual, Chapter 30.

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ENGINE WIRE HARNESS – DESCRIPTION AND OPERATION

1. General

A. The engine wire harness is a M7 Aerospace manufactured wire bundle designed to fit onto the engine mount truss member. Each lead in the wire harness is numbered and, with use of the aircraft Wiring Manual, can be identified.

B. Because some engines components between TPE331-3U-303G/304G and TPE331-10U-501 engines differ, a different engine wire harness is used on different engines.

C. On SA226 series aircraft using -3 engines, effective T201-278, AT001-056, and TC201-229 a different wire harness is used on the left engine than is used on the right engine. After the above effectivities, the same part number wire harness is used on both the left and right engines. On SA226 series aircraft using -10 engines, both the left and right engines use the same part numbered wire harness.

D. TPE331-3U-303G Electrical Harness (71-50-00)

(1) This section details procedures to remove and/or replace the left and right engine electrical harness assembly.

E. TPE331-10U-501 Electrical Harness (71-50-10)

(1) This section details procedures to remove and/or replace either engine electrical harness assembly.

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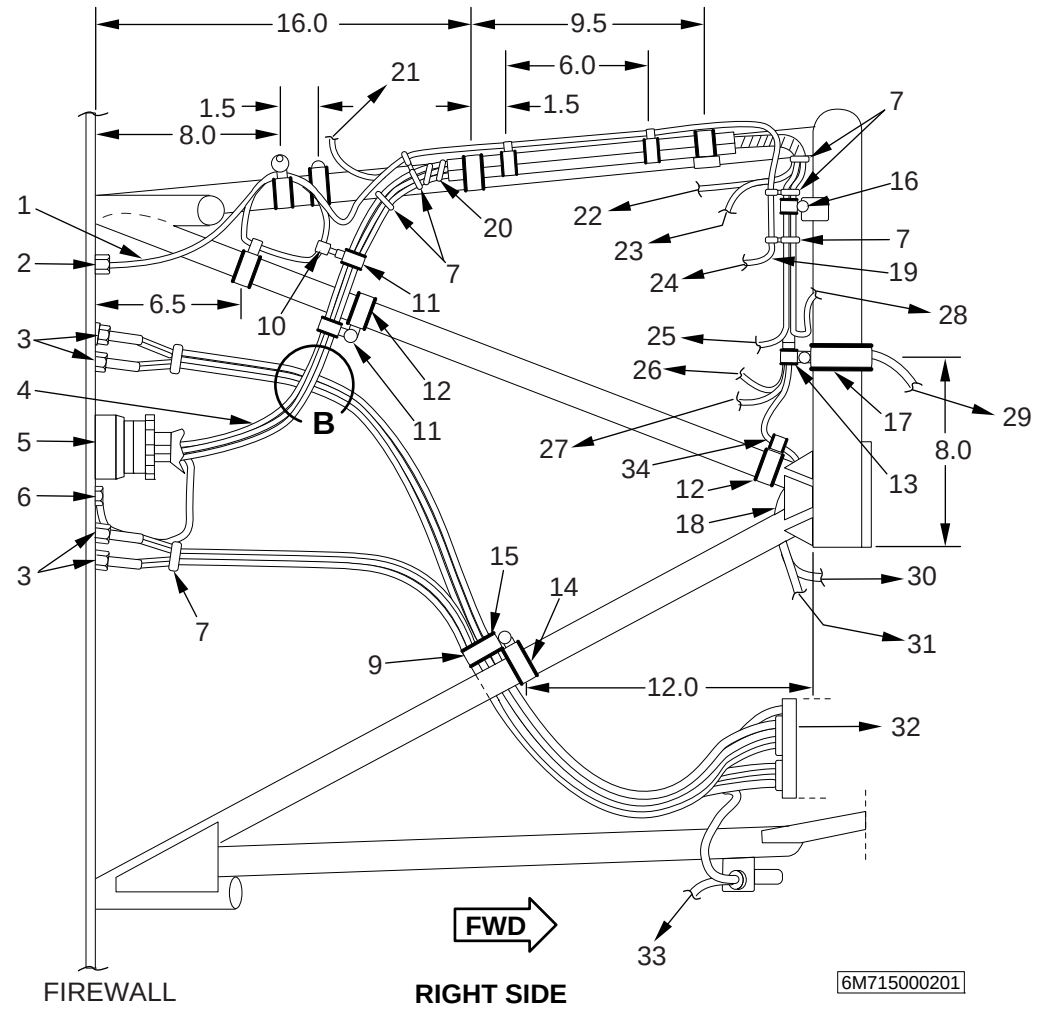


Engine Wire Routing Right Side
(TPE331-3U-303G / 304G)
Figure 201

ENGINE WIRE HARNESS – MAINTENANCE PRACTICES

1. Servicing – Engine Wire Harness

A. Wire Harness Build-Up (TPE331-3U-303G/304G)

(1) Wrap wire harness with asbestos tape at the main lead (Figure 201, item 5) from the firewall where it divides (Y's) into two; pulling back and wrapping each separate lead.

(2) Wrap right half of wire harness with asbestos tape below the ignitor box cannon plug (Figure 201, item 28) approximately 14 inches. (See Table 201, block 4 for wire lead numbers).

(3) Wrap left half of harness with asbestos tape (approximately nine inches) leaving enough distance to enable splicing of the Oil Temp leads (Figure 203, item 5), and the Tach Generator leads (Figure 203, item 6). See Table 201, block 17 for part numbers of Oil Temp leads and block 16 for Tach Generator leads.

(4) Position cover tube (Figure 202, item 7) on the main lead (Figure 201, item 5) approximately two inches aft of the "Y". Secure in place with a piece of asbestos tape at each end. Clamp as illustrated (Figure 202, items 2 and 4, two places) and (Figure 201, items 11 and 12, one place).

NOTE: Pass left half of harness under mount frame (see Figure 202).

2. Removal/Installation – Engine Wire Harness

A. Installation – Right Half Harness

NOTE: All cannon plugs will be installed in accordance with Table 201. Safety all cannon plugs with 0.020 steel wire unless otherwise stated.

(1) Install:

(a) Anti-ice valve cannon plug (Figure 201, item 22). Tighten and safety. (See Table 201, block 1 for leads and connector part numbers).

(b) Beta Pressure Switch cannon plug (Figure 201, item 23) and secure. (See Table 201, block 2 for leads and connector part numbers.)

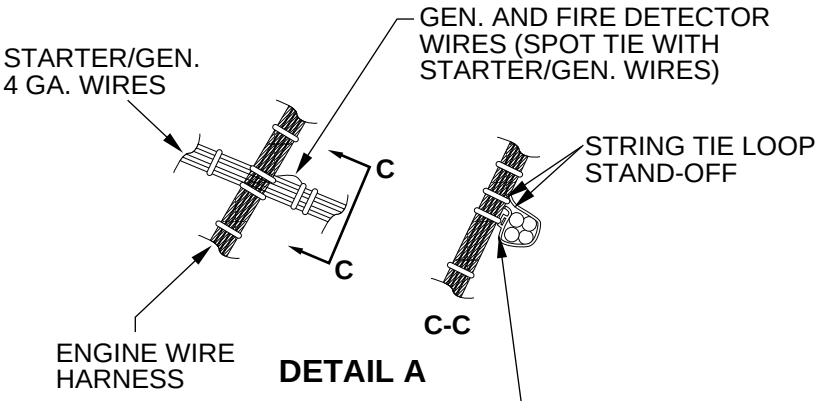
(c) ITT Compensator cannon plug (Figure 201, item 24). Do not tighten or safety until after engine installation. (See Table 201, block 3 for leads and connector.)

(2) Clamp lead (Figure 201, item 16) to engine mount - one place.

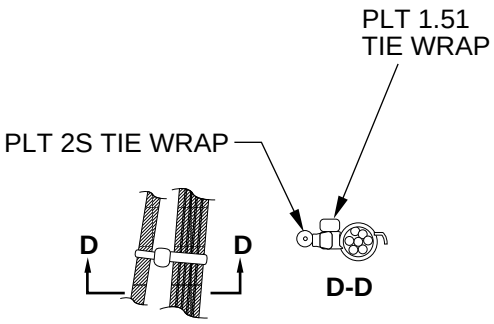
(3) Install plastic cap on ignitor box cannon plug (Figure 201, item 28).

CAUTION: DO NOT CONNECT CANNON PLUG UNTIL JUST PRIOR TO INITIAL ENGINE RUN.

(a) Measure distance required for ground lead (See Table 201, block 4) in order to reach rear mount bolt of ignitor box, cut lead and install terminal (See Table 201, block 4 for terminal number). Remove and re-install the mount bolt installing the ground wire beneath it. (4) Clamp leads where leads "Y" off (Figure 201, item 13 and 17). For another view see Figure 4, items 1 and 2.

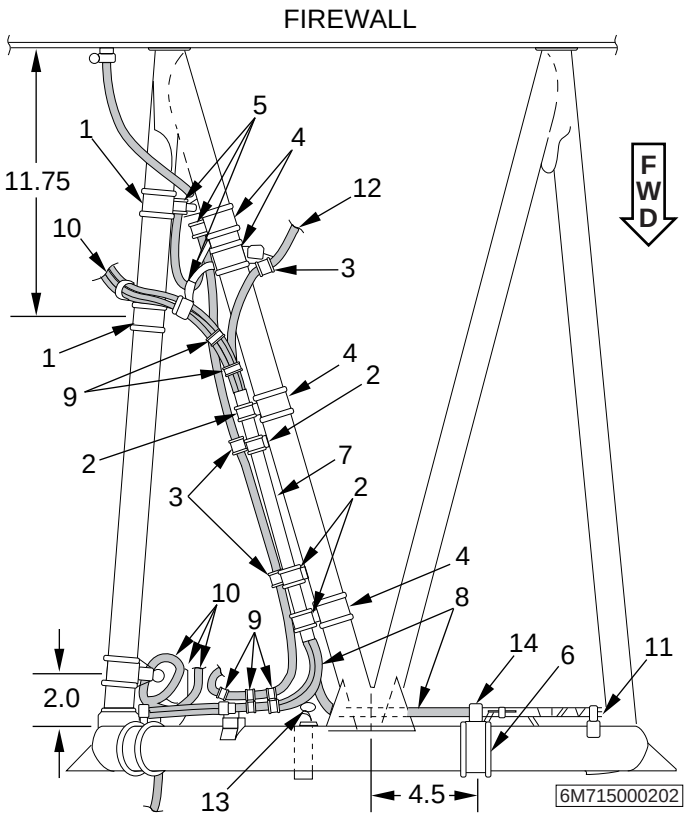


DETAIL A



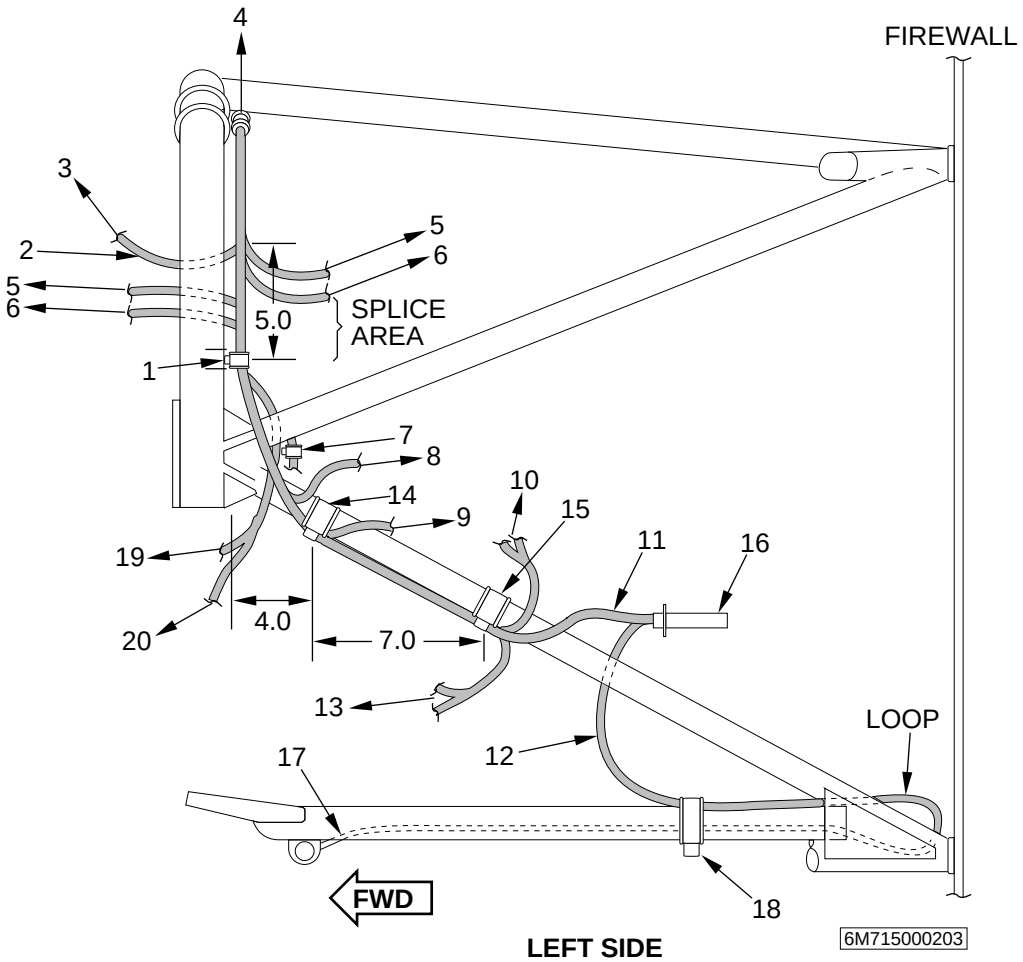
DETAIL B

TYP. TIE WRAP STAND-OFF



LOOKING DOWN

Engine Wire Routing Over Engine
(TPE331-3U-303G / 304G)
Figure 202



Engine Wire Routing Left Side
(TPE331-3U-303G / 304G)
Figure 203

(4) Clamp leads where leads "Y" off (Figure 201, items 13 and 17). For another view, see Figure 204, items 1 and 2.

NOTE: Use AN3-5A bolt. Washer and nut are same as indicated on illustration.

(5) Install:

(a) Negative Torque Switch cannon plug (Figure 201, item 26). Tighten and safety. (See Table 201, block 5 for leads and connector).

(b) Fuel Purge Valve cannon plug (Figure 201, item 25). Tighten and safety. (See Table 201, block 6 for leads and connector).

(c) Fuel Flow Shutoff Valve cannon plug (Figure 201, item 27). Tighten and safety. (See Table 201, block 7 for leads and connector).

(6) Clamp three remaining leads to engine support mount (Figure 201, items 12 and 34). String tie any excess leads above the clamp.

(7) Leave Torque Pressure Switch lead and Torque Transmitter cannon plug lead (Figure 201, item 30) unconnected. They will be connected when Torque Transmitter is installed after engine installation. (See Table 201, block 8 for Torque Pressure Switch and block 9 for Torque Transmitter wire leads and connector part numbers.)

(8) Measure and allow for excess wire (10 to 15 inches) and cut wires to propeller governor (Figure 201, item 31). Install quick disconnect terminals (See Table 201, block 10) to leads from governor. Install leads (see Table 201, block 10) to governor. Insulate and tie wrap the excessive lead close to the governor.

(9) Clamp propeller brush block leads to gear box split case as illustrated (Figure 204, item 2, inset - “Looking Aft”) using existing case nut and washer.

NOTE: Drill out hole in clamp to 1/4 inch.

(10) Install:

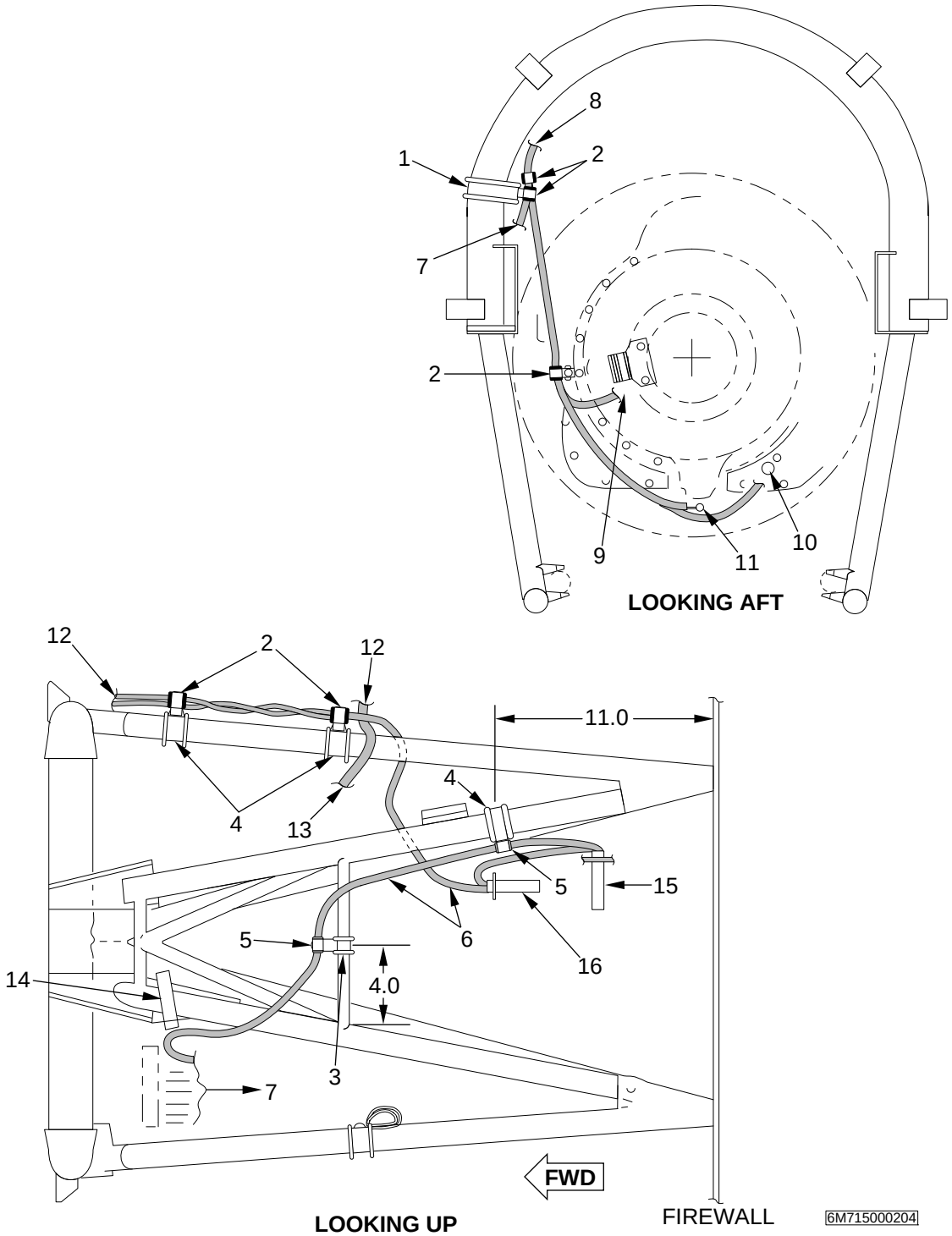
(a) Chip detector cannon plug (Figure 204, item 11). Tighten and safety. (See Table 201, block 11 for wire leads and connector part numbers.)

(b) Oil Vent Valve cannon plug (Figure 204, item 10). Tighten and safety. (See Table 201, block 12 for wire leads and connector part numbers).

(11) Measure Propeller Deice Brush Block leads (Figure 204, item 9) so that they will not chaff cowling when installed, cut and install connector or leads (See Table 201, block 13) to pickups (pickups/terminals are in parenthesis next to wire lead numbers on Table 201, block 13). String tie excess Chip Detector lead and Oil Vent Valve leads to Prop Brush Block leads.

NOTE: On Brush Blocks with connector, install so that the connector faces upward position. Brush Blocks with terminals, install so that the terminal faces downward. Reference M7 Aerospace Service Note 084.

On Brush Blocks utilizing terminals, blue color coded terminals (MS25036-106), No. 6 stud size are to be used on 14 gauge wire, and yellow color coded terminals (MS25036-111), No. 6 stud size, are to be used on 10 gauge wire.



Engine Wire Routing Under Engine
(TPE331-3U-303G / 304G)
Figure 204

B. Installation – Left Half Engine Wire Harness

(1) Measure and cut leads for the 450°F Fire Detector (Figure 202, item 13). Slide nipple over leads. Install terminals (see Table 201, block 14) and attach to detector as shown in Figure 205. Tighten and slide nipple over terminals.

(2) Clamp harness to engine support mount (Figure 202, item 6 and 14).

(3) Cut two 10 inch pieces of 400/461 PVC tubing (size 6) and slide them over the Oil Temperature cannon plug leads (Figure 203, item 5) and the Tach Generator cannon plug leads (Figure 203, item 6). Clamp main harness to engine mount (Figure 203, item 1).

(4) Splice Tach Generator lead wires at the harness (Figure 203, item 6) to Tach Generator cannon plug lead wires (see Table 201, block 16 for lead wires and connector part numbers) and Oil Temperature Bulb lead wires from harness (Figure 203, item 5) to Oil Temperature Bulb cannon plug lead wires (see Table 201, block 17 for lead wires and connector part numbers). Cover with heat shrink (size 8). String tie and tape neatly between clamp (see Figure 203, items 1 to 6).

(5) Tie wrap Tach Generator and Oil Temp leads (Figure 203, items 5, 6, and 7) to oil line from fuel oil heater.

(6) Install unfeather pump cannon plug (Figure 203, item 8). Do not tighten or safety at this time. (Done after calibration). Where applicable, splice ground wire using window splice (see Table 201, block 18 for ground wire and window splice part numbers).

(7) Clamp harness to gear case oil hose clamps, clamping both hoses and harness to mount support just in front of unfeather pump (see Figure 203, item 4). Ensure that both oil lines are tightened at this time.

(8) Clamp harness to gear case pressure hose to support mount just aft of the unfeather pump.

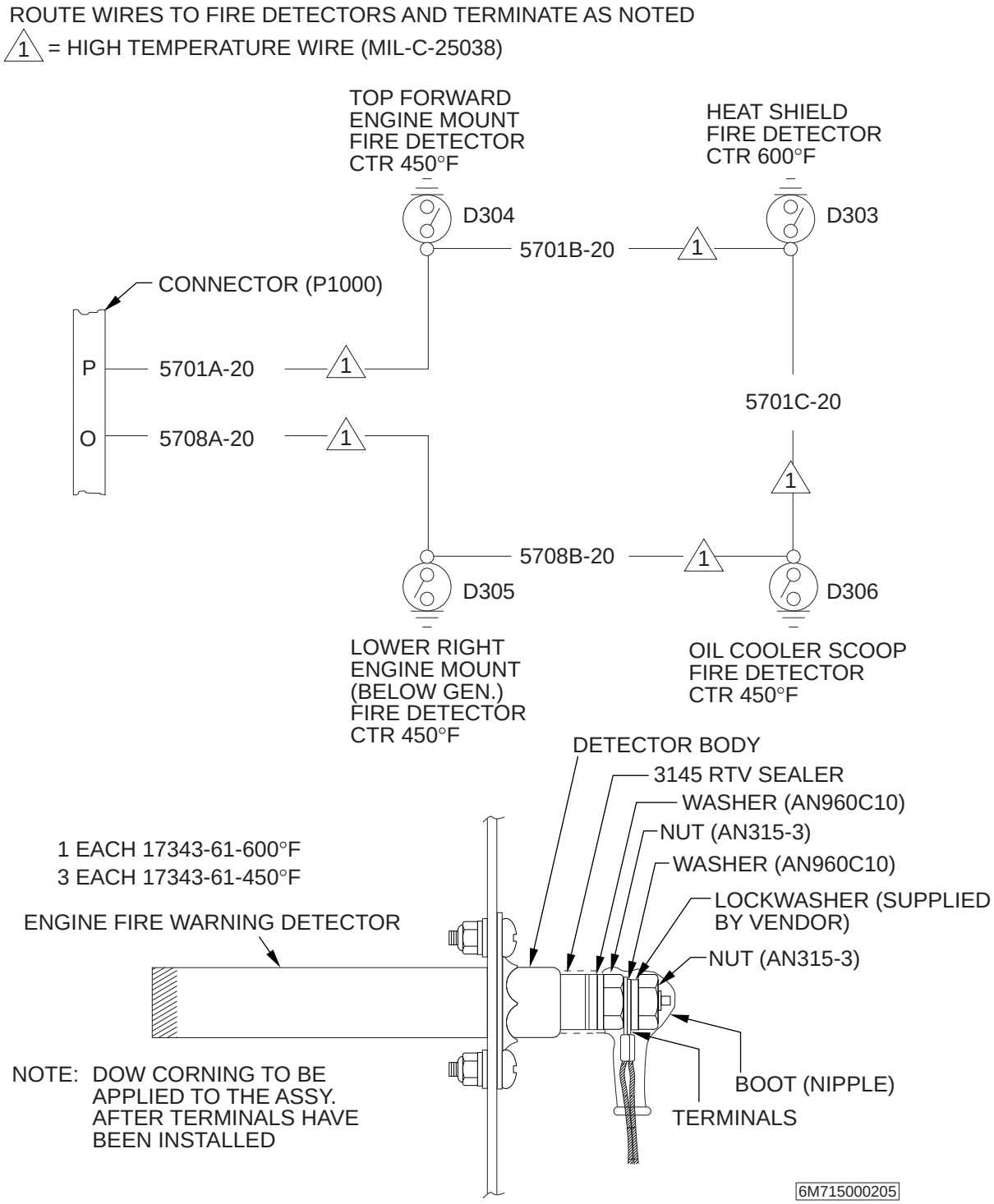
(9) Install:

(a) Connector(s) to Fuel Flow Transmitter lead (Figure 203, item 10). (See Table 201, block 19 for lead(s) and connector(s) part numbers.) Tighten and safety.

(b) One elbow cannon plug to Fuel Flow Pressure Regulator lead (Figure 203, item 13) (also see Table 201, block 20 for lead and connector part numbers) and one cannon plug to Anti-ice Lock-Out Valve lead (Figure 203, item 13) (also see Table 201, block 21 for lead and connector part numbers). Tighten and safety all connectors.

(10) Cut a 14 inch piece of 400/461 PVC tubing (size 6) and slide it over Fire Detector lead (Figure 203, item 12). Route lead over mount and in front of heat shield to 600° Fire Detector (Figure 203, item 16). (See Table 201, block 15 for Fire Detector leads and terminal part numbers).

(11) Slide piece of 400/461 PVC tubing (size 6) over entire length of additional lead (Figure 203, items 12 and 17). Attach terminal to each end - (see Table 201, block 14 for leads and terminal part numbers). Route one end through clamp on engine mount support (Figure 204, items 3 and 5). Also see Figure 202, item 15 for clamp installation location. Attach nipple and install lead end on 450° Fire Detector (Figure 204, item 14). Also see Figure 205, for lead and nipple installation.



Fire Detector Wire Routing
Figure 205

(12) Clamp middle of lead leaving a loop (see Figure 203) at the firewall so that it can be cut and connected to 450° Fire Detector on Oil Cooler upon engine installation (Figure 203, item 18, or Figure 204, items 4 and 5).

(13) Route other end of lead along frame and up forward side of heat shield to 600° Fire Detector.

(14) Remove 600° Fire Detector in order to pass wire leads to detector through hole in heat shield. Measure distance required for wire lead (5701B or 5711B - see Table 201, block 15 for applicable effectivity) to reach detector. Cut and install terminal (see Table 201, block 15, for terminal part number). Slide leads through nipple and attach to 600° Fire Detector as shown in Figure 205. Slide nipple over leads and reinstall detector and tighten. Tie wrap any excess lead to Heim Bearing bracket assembly.

(15) Apply 3145RTV Sealer to all Fire Detectors (see Figure 205).

(16) Reference Maintenance Manual, Section 82-00-00, for optional Alcohol and Water Injection (AWI) installation.

TPE331-3U-303G/304G

Unit Nomenclature	Left Engine Wire Lead	Right Engine Wire Lead	Connector P/N
ANTI-ICE VALVE (Intake Heat)	Left Valve	Left Valve (TPE331-3U-303G Engine)	MS3106K-14S-5S (TPE331-3U-303G Engine)
High Temp Wire (MIL-C-25038)	2269-22 2267A-22 2265-22 2261A-22 2269A-22	2273-22 2271A-22 2263-22 2275-22 2279A-22 2273A-22	 2277-22
	Right Valve (TPE331-3U-303G Engine)	Right Valve (TPE331-3U-303G Engine)	MS3106K-14S-5S
1	2269A-22 2267A-22 2263-22 2261A-22 2261-22 2269B-22 2267-22N	2273A-22 2271A-22 2277-22 2279A-22 2279-22 2273B-22 2271-22N	
	Right Valve (TPE331-3U-304G Engine) 2269-22 2261-22 2265-22 2267-22N 2269A-22	Right Valve (TPE331-3U-304G Engine) 2271-22 2273-22 2279-22 2275-22 2269A-22	MS3106K-14S-5S
BETA PRES-SURE SWITCH 2	6625-22 6647-22	6640-22 6627-22	MS3116F8-3S or PT06E-8-3S

Table 201

TPE331-3U-303G/304G

Unit Nomenclature	Left Engine Wire Lead	Right Engine Wire Lead	Connector P/N
ITT COM-PENSATOR 3	Alumel & Chromel Lead 3114-22N 3105-22	Alumel & Chromel Lead 3110-22 3116-22N	SG318E14S52S
IGNITOR BOX 4	2101-16 2103-22N	2105-16 2107-16N	MS3106A10SL-3S MS3106A10SL-3S MS25036-103
NEGATIVE TORQUE SWITCH (NTS) 5	8701A-22 8702A-22	8703A-22 8704A-22	MS3106R-6S-3S
FUEL PURGE VALVE (EPA) 6	1625B-22N 1624B-22	1631B-22N 1630B-22	MS3108E-10SL-3S
ENGINE FUEL SHUTOFF VALVE 7	1623-16 1625-16N	1629-16 1624-16 1631-16N	MS3108B-10SL-3S 1630A-16
AWI TORQUE PRESSURE SWITCH 8	7620AB-20 7621AB-20 NOTE: AT-C/O, TC-Standard, T-N/A.	7619AB-20 7618AB-20 7609-20N	MS3106R-10SL-3S 7608-20N
TORQUE TRANSMITTER 9	2006A-22 2004A-22 2000A-22N	2007A-22 2009A-22 2006-22N	CA3108ER10SL-3S
PROPELLER GOVERNOR 10	2409-22 2410-22	2411-22 2412-22	Quick Disconnect (MS25181-1)
CHIP DETECTOR 11	7302-22	7306-22	MS3108R-8S-1S
OIL VENT VALVE 12	109-16	113-16 111-22N	MS3106A-10SL-3S 115-22N

Table 201

TPE331-3U-303G/304G

Unit Nomenclature	Left Engine Wire Lead	Right Engine Wire Lead	Connector P/N
PROPELLER DEICE adapter 13	2310-14(C) 2312-14(A) NOTE: These leads may also be 10 gauge wire.	2313-14(A) 2311-14(B) 2315-14(C) NOTE: These leads may also be 10 gauge wire.	MS3108E-10S-3S 2314-14(B) with MS3057-4A NOTE: No connector for 10 gauge wire.
450~F FIRE DETECTOR 14	(Below Gen.) 5708A-20 5708B-20 (Oil cooler scoop) 5708B-20 5701C-20 (Top fwd of eng. mount) 5701A-20 5701B-20	(Below Gen.) 5718A-20 5718B-20 (Oil cooler scoop) 5718B-20 5718C-20 (Top fwd of eng. mount) 5711B-20 5711A-20	Terminals Red-color coded #10 stud-size. (MS25036-103)
600~F FIRE DETECTOR 15	5701B-20 5701C-20	5711B-20 5718B-20	Terminal (MS25036-103) Red-color coded #10 stud-size.
TACHOMETER GENERATOR 16	2803-22 2804-22	2809-22 2810-22	SC06P-12-3S
OIL TEMPERATURE BULB 17	1205-22 1206-22N	1208-22 1207-22N	MS3106A-12S-3S
UNFEATHER UNFEATHER PUMP 18	117-16 175-16 (Gnd)	119-16 176-16N (Gnd)	MS3106A-10S-2S

Table 201

TPE331-3U-303G/304G

Unit Nomenclature	Left Engine Wire Lead	Right Engine Wire Lead	Connector P/N
FUEL FLOW TRANSMITTER 19	White-18 Black-18 (Shielded wire)	White-18 Black-18 (Shielded wire)	CA3108ER10SL-4S CA3108ER10SL-4S MS3106R10SL-4S MS3116E-8-3S
FUEL FLOW PRESSURE REGULATOR (Start Pressure regulator) 20	101-16 103-22N	105-16 107-22N	MS3108B1056-3S
ANTI-ICE LOCK OUT VALVE 21	169C-16 169D-22N	144C-16 144D-22N	MS3106R10SL-4S
AWI LINE PRESSURE SWITCH 22	7635A-20N 7634A-20 NOTE: AT-C/O,	7637A-20N 7636A-20 TC-Standard, T-N/A.	MS3106R10SL-3S
AWI TEST VALVE 23	7606A-20 7609A-20N NOTE: AT-C/O,	7607A-20 7608A-20N TC-Standard, T-N/A.	MS3106R10SL-4S
AWI FIRE VALVE 24	7623A-20 7625A-20 NOTE: AT-C/O,	7628A-20 7626A-20 TC-Standard, T-N/A.	MS3106R10SL-4S
PROPELLER SYNCRONIZER ACTUATOR 25	N/A	2413-22 2414-22 2415-22 2416-22	MS3108B-14S-2S with adapter MS3057-6A

Table 201

TPE331-3U-303G/304G

Unit Nomenclature	Left/Right Engine Wire Lead	Connector P/N
ANTI-ICE VALVE (Intake Heat)	Left Valve (TPE331-3U-303G Engine)	MS3106K-14S-5S
High Temp Wire (MIL-C-25038)	2267A-22 2267-22 2263-22 2265-22 2261A-22 2269A-22	
	Right Valve (TPE331-3U-303G Engine)	MS3106K-14S-5S
1	2269A-22 2267A-22 2263-22 2261A-22 2261-22 2269B-22 2267-22N	
	Right Valve (TPE331-3U-304G Engine)	MS3106K-14S-5S
	2269-22 2261-22 2265-22 2267-22N 2269A-22	
BETA PRES-SURE SWITCH 2	6625-22 6647-22	MS3116F8-3S or PT06E-8-3S
ITT COM-PENSATOR 3	Alumel & Chromel Lead 3114-22N 3105-22	SG318E14S52S

Table 202

TPE331-3U-303G/304G

Unit Nomenclature	Left/Right Engine Wire Lead	Connector P/N
IGNITOR BOX 4	2105-16 2107-16N 2107A-22N (Gnd)	MS3106A10SL-3S MS3106A10SL-3S MS25036-103
NEGATIVE TORQUE SWITCH (NTS) 5	8701A-22 8702A-22	MS3106R-6S-3S
FUEL PURGE VALVE (EPA) 6	1631B-22N 1630B-22	MS3108E-10SL-3S
ENGINE FUEL SHUTOFF VALVE 7	1629-16 1630A-16 1631-22N	MS3108B-10SL-3S
AWI TORQUE PRESSURE SWITCH 8	7602AB-20 7621AB-20 7609A-20N NOTE: AT-C/O, TC-Standard, T-N/A.	MS3106R-10SL-3S
TORQUE TRANSMITTER 9	2006A-22 2004A-22 2000A-22N	CA3108ER10SL-3S
PROPELLER GOVERNOR 10	2409-22 2410-22	Quick Disconnect (MS25181-1)
CHIP DETECTOR 11	7302-22	MS3108R-8S-1S
OIL VENT VALVE 12	113-16 115-22N	MS3106A-10SL-3S

Table 202

TPE331-3U-303G/304G

Unit Nomenclature	Left/Right Engine Wire Lead	Connector P/N
PROPELLER DEICE 13	2310-14(C) 2311-14(B) 2312-14(A) NOTE: These leads may also be 10 gauge wire.	MS3108E-10S-3S with adapter MS3057-4A NOTE: No connector for 10 gauge wire.
450~F FIRE DETECTOR 14	(Below Gen.) 5708A-20 5708B-20 (Oil cooler scoop) 5708B-20 5701C-20 (Top fwd of eng. mount) 5701A-20 5701B-20	Terminals Red-color coded #10 stud-size. (MS25036-103)
600~F FIRE DETECTOR 15	5701B-20 5701C-20 Red-color	Terminal (MS25036-103) coded #10 stud-size.
TACHOMETER GENERATOR 16	2803-22 2804-22	SC06P-12-3S
OIL TEMPERATURE BULB 17	Wire Lead 1205-22 1206-22N	MS3106A-12S-3S

Table 202

TPE331-3U-303G/304G

Unit Nomenclature	Left/Right Engine Wire Lead	Connector P/N
UNFEATHER PUMP 18	119-16 175-16N (Gnd)	MS3106A-10S-2S MS25181-1
FUEL FLOW TRANSMITTER 19	1914C-22S 1913C-22S (Each of the above lead numbers contain a black and white shielded wire).	CA3108ER10SL-4S CA3108ER10SL-4S MS3106R10SL-4S MS3116E-8-3S
FUEL FLOW PRESSURE REGULATOR (Start Pressure regulator) 20	106-16 107-22N	MS3108B1056-3S

Unit Nomenclature	Left/Right Engine Wire Lead	Connector P/N
ANTI-ICE LOCK OUT VALVE 21	144C-16 144D-22N	MS3106R10SL-4S
AWI LINE PRESSURE SWITCH 22	N/A	MS3106R10SL-3S

NOTE: AT-C/O, TC-Standard, T-N/A.

Table 202

Unit Nomenclature	Left/Right Engine Wire Lead	Connector P/N
AWI TEST VALVE 23	N/A NOTE: AT-C/O, TC-Standard, T-N/A.	MS3106R10SL-4S
AWI FIRE VALVE 24	N/A NOTE: AT-C/O, TC-Standard, T-N/A.	MS3106R10SL-4S
PROPELLER SYNCRONIZER ACTUATOR 25	2413-22 2414-22 2415-22 2416-22 RH Engine Only	MS3108B-14S-2S with adapter MS3057-6A

Table 202

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ENGINE WIRE HARNESS – MAINTENANCE PRACTICES

1. Servicing – Engine Wire Harness

A. Wire Harness Build-Up (TPE331-10U-501)

(1) Wrap wire harness (Figure 201, item 29) with asbestos tape from fire wall connector (Figure 201, item 3) pulling back each separate lead. Wrap main lead approximately 10 inches past the 450° fire detector leads (Figure 201, item 23). (2) Wrap propeller leads approximately 20 inches with insulated tape.

NOTE: All cannon plugs will be installed in accordance with Table 201 and applicable engine wire routing figure. Safety all cannon plugs with 0.020 steel wire unless otherwise stated.

2. Removal/Installation – Engine Wire Harness

A. Right Half Harness Installation

(1) Route propeller leads (Figure 201, item 19) behind engine mount. Route main lead over mount at top of engine.

(2) Install:

(a) Clamp (Figure 201, item 18) to main lead below area where propeller leads “Y” off.

(b) Clamp (Figure 201, item 22) to main lead.

(c) Clamp (Figure 203, item 3) to propeller lead using existing case nut and washer.

NOTE: Drill out hole in clamp to 1/4 inch to facilitate installation.

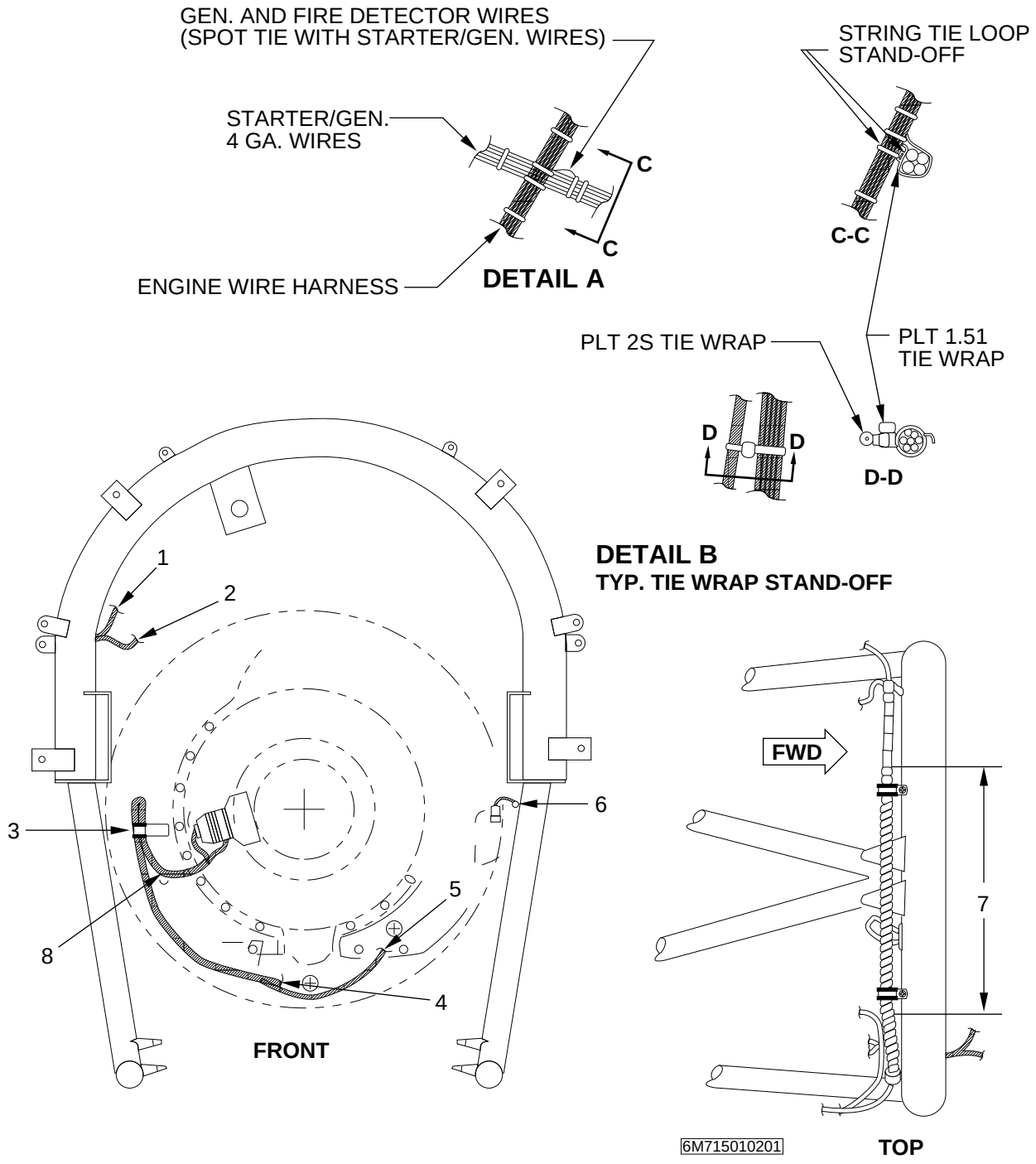
(3) Ensure cap assembly (Figure 201, item 17a) is installed onto Primary Only Valve Solenoid cannon plug (Figure 201, item 17). Install Primary Only Valve Solenoid cannon plug (Figure 201, item 17) and NTS Checkout Valve cannon plug (Figure 201, item 16). See Table 201, block 26 for Primary Only Valve cannon plug part number and Table 201, block 8 for NTS Checkout Valve cannon plug part number. Safety both cannon plugs.

NOTE: Leads should be on outboard of engine mount.

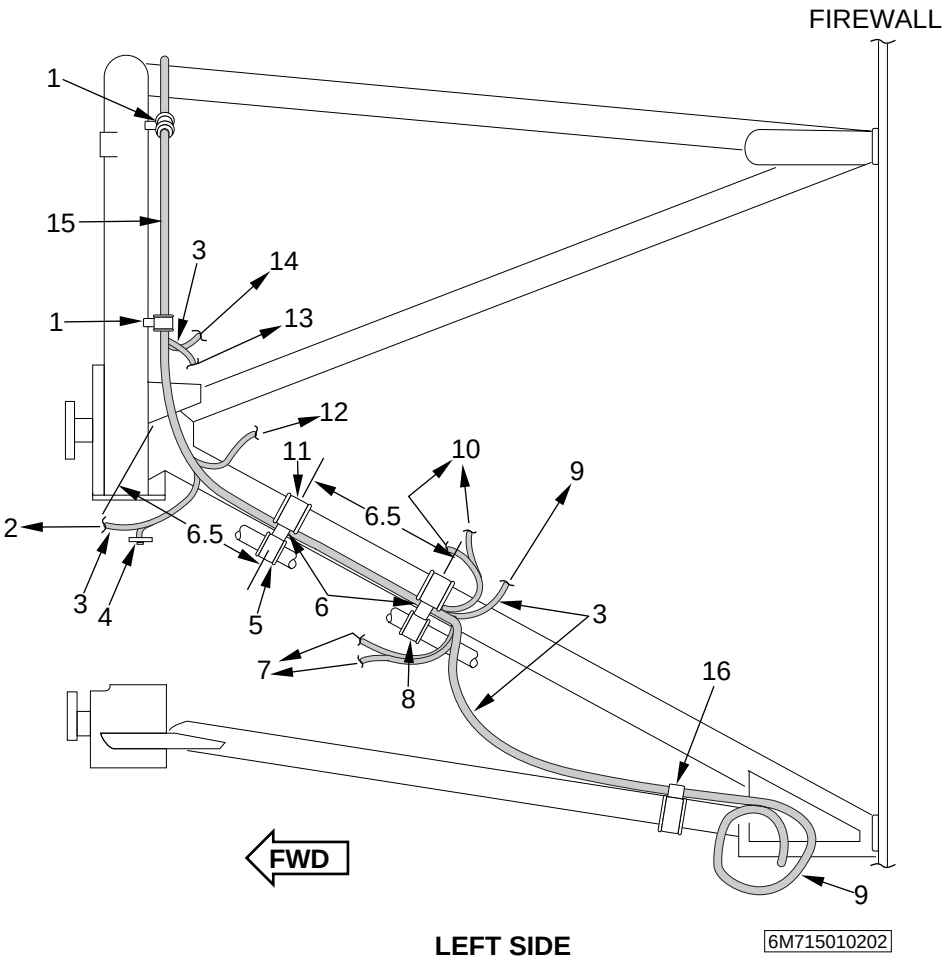
(4) Measure and allow for excess wire (10 to 15 inches) and cut wires to propeller governor (Figure 201, item 15).

(a) Propeller Governor Monopole: Peel outer insulation from Propeller Governor Monopole lead (See Table 201, block 13) to expose approximately three inches of the black and white leads. Install quick disconnect terminals. Connect white lead to white lead from governor and connect black lead to black lead from governor. Insulate and tie wrap excessive lead close to the governor.

(b) Propeller Synchronizer: Install quick disconnect terminals (See Table 201, block 14) to propeller governor. Connect 2407-22 lead to the long black lead from the governor and 2409-22 lead to the short black lead from the governor. Insulate and tie wrap excessive lead close to the governor.



Engine Wire Harness
(TPE331-10U-501)
Figure 201



Engine Wire Routing
(TPE331-10U-501)
Figure 202

(5) Install:

(a) Fuel shutoff cannon plug (Figure 201, item 11). Tighten and safety cannon plug. (See Table 201, block 10 for lead and connector part numbers).

(b) Fuel Manifold Purge Kit cannon plug (Figure 201, item 26). Tighten and safety cannon plug. (See Table 201, block 9 for lead and connector part numbers.)

(c) NTS Pressure Switch cannon plug (Figure 201, item 27). Tighten cannon plug and safety. (See Table 201, block 7 for lead and connector part numbers.)

(d) Anti-ice Valve cannon plug (Figure 201, item 28). Tighten and safety cannon plug. (See Table 201, block 1 for lead and connector part numbers).

(e) Beta Switch cannon plug (Figure 201, item 25). Tighten cannon plug and safety. (See Table 201, block 2 for lead and connector part numbers.)

(f) TT2 Sensor cannon plug (Figure 203, item 2). Tighten and safety cannon plug. (See Table 201, block 29 for lead and connector part numbers).

(6) Install plastic cap on Ignitor Box cannon plug (Figure 201, item 21). (See Table 201, block 6 for lead and connector part numbers).

CAUTION: DO NOT CONNECT CANNON PLUG UNTIL JUST PRIOR TO INITIAL ENGINE RUN.

(7) Install 450° fire detector leads (Figure 201, item 23) in accordance with Table 201, block 18 and Figure 204.

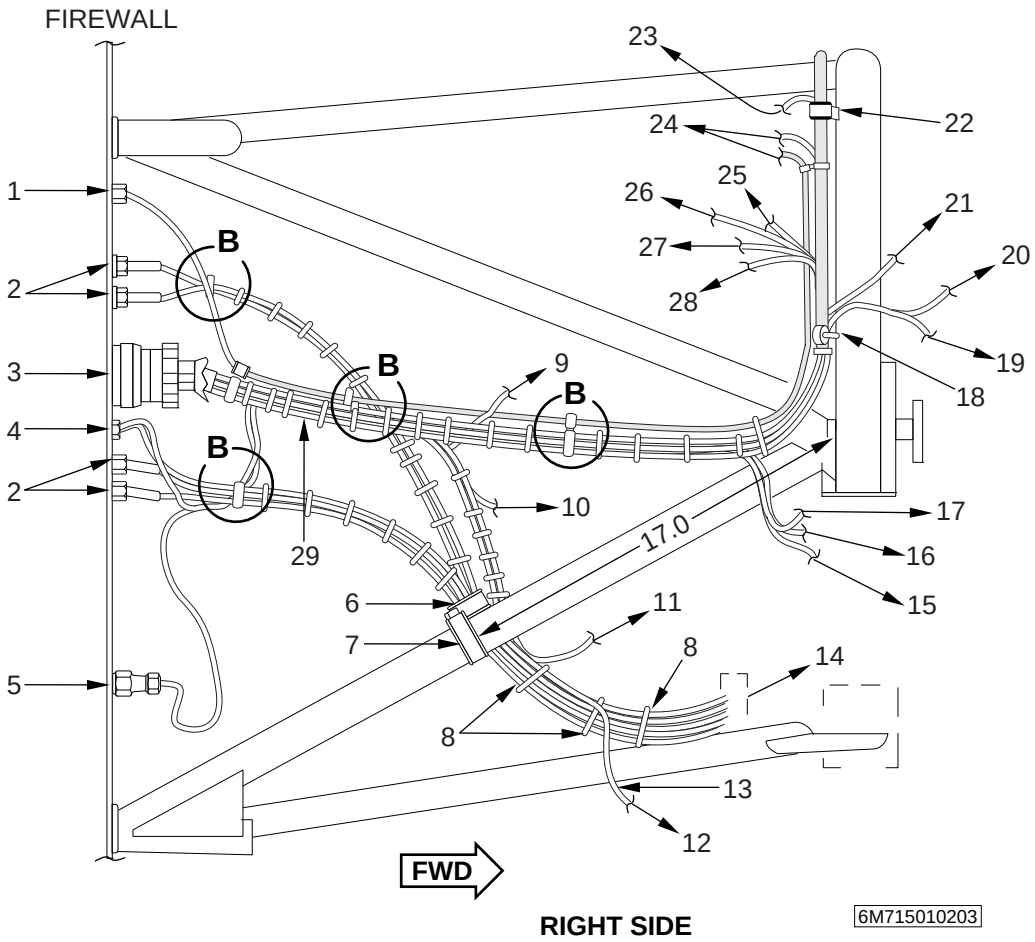
(8) Install:

(a) Chip Detector cannon plug (Figure 203, item 5). Tighten and safety cannon plug. (See Table 201, block 15 for lead and connector part numbers.)

(b) Oil Vent Valve cannon plug (Figure 203, item 6). Tighten and safety cannon plug. (See Table 201, block 16 for lead and connector part numbers).

(9) Measure Propeller Deice Brush Block leads (Figure 203, item 9) so that they will not chafe when cowling is installed, cut wire and install terminals to leads (see Table 201, block 17). Connect to pick-up on Brush Block (letter in parenthesis on Block 17 of Table 201 indicates which lead connects to which pick-up).

(10) Measure Propeller Deice Brush Block Speed Pick-Up (Magnetic Pick-Up) leads (Figure 203, item 8) allowing for 10 to 12 inches f excess wire. Cut wires and install terminals. Connect wires; insulate, and tie wrap excess wire to Brush Block main leads. (See Table 201, block 14 for lead and connector part numbers.)



Engine Wire Routing
(TPE331-10U-501)
Figure 203

B. Left Half Harness Installation

(1) Install clamp (Figure 202, item 1) securing main lead to top forward of engine mount.

(2) Cut two 10 inch pieces of 400/461 PVC tubing (Size 6) and slide them over the Oil Temperature cannon plug leads (Figure 202, item 14) and the Tachometer Generator cannon plug leads (Figure 202, item 13). Clamp main harness to lower forward area of engine mount (Figure 202, item 1).

(3) Splice Tachometer Generator leads and Oil Temperature Bulb leads at the harness splice area (Figure 202, item 15) to the Tachometer Generator Cannon plug lead wires (Figure 202, item 13) and Oil Temperature Bulb lead wires (Figure 202, item 14). See Table 201, block 20 for Tachometer Generator lead and connector part numbers, and block 21 for Oil Temperature Bulb lead and connector part numbers. Cover with heat shrink (size 8). String tie and tape neatly between clamps (Figure 202, item 1).

(4) Tie wrap Tachometer Generator and Oil Temperature leads (Figure 202, items 13 and 14) to oil line from fuel oil heater.

(5) Install unfeather pump cannon plug (Figure 202, item 12). Do not tighten or safety at this time (done after calibration). Where applicable, splice ground wire using window splice (see Table 201, block 22 for lead and connector part numbers.)

(6) Install Bypass Limiter cannon plug (Figure 202, item 2). Tighten and safety cannon plug. (See Table 201, block 27 for lead and connector part numbers).

(7) Clamp harness to gear case oil hose clamps, clamping both hoses and harness to mount support just in front of unfeather pump. (See Figure 202, item 11.) Ensure that both oil lines are tightened at this time.

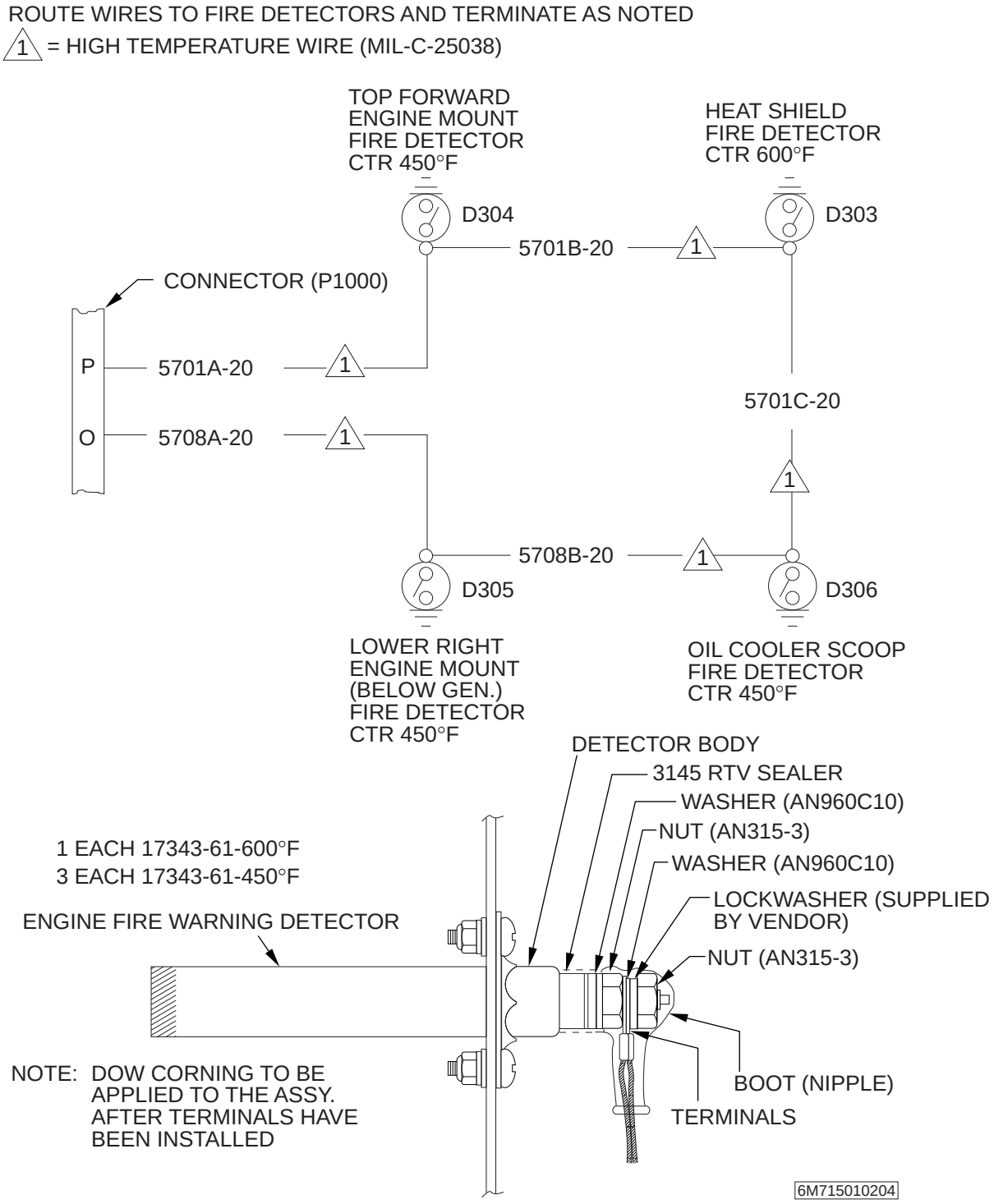
(8) Clamp harness to gear case pressure hose clamp to support mount just aft of the unfeather pump (Figure 202, item 8).

(9) Install:

(a) Connector(s) to Fuel Flow Transmitter lead (Figure 202, item 10). (See Table 201, block 23 for lead(s) and connector(s) part numbers). Tighten and safety connector(s).

(b) One elbow cannon plug to Fuel Flow Pressure Regulator lead (Figure 2, item 7) (see Table 201, block 24 for lead and connector part numbers) and one cannon plug to Anti-Ice Lock-Out Valve lead (Figure 202, item 7) (see Table 201, block 25 for lead and connector part numbers). Tighten and safety all connectors.

(10) Route 600° fire detector lead (Figure 202, item 9) over mount and in front of heat shield to 600° fire detector.



Fire Detector Wire Routing
Figure 204

(11) Remove 600° fire detector in order to pass wire leads to detector through hole in heat shield. Measure distance required for wire lead (see Table 201, block 19 for lead and connector part number) to reach detector. Cut wire and install terminal. Slide leads through nipple and attach to 600° fire detector as shown in Figure 204. Slide nipple over leads and reinstall detector and tighten. Tie wrap any excess lead to Heim Bearing bracket assembly.

(12) Apply 3145RTV Sealer to all Fire Detectors (see Figure 204). (13) Route 450° fire detector lead (Figure 202, item 9) (see Table 201, block 18 for lead and connector part numbers) over engine mount to inside of lower mount support. Route lead along support leaving a little excess wire. Clamp at rear of mount (Figure 202, item 16). Loop and tie excess wire lead. (Excess lead loop will be used upon installation of 450° fire detector at the oil cooler after engine installation.)

(14) Reference Maintenance Manual, Section 82-00-00 for optional Alcohol and Water Injection (AWI) installation.

ENGINE WIRING HARNESS – MAINTENANCE PRACTICES
TPE331-10U-501

LEFT AND RIGHT ENGINE WIRE BUNDLE

UNIT NOMENCLATURE	WIRE LEAD PART NUMBER	TERMINAL OR CONNECTOR
ANTI-ICE VALVE (INTAKE HEAT) High Temp Wire (MIL-C-25038) 1	2269-22 2261-22 2265-22 2267-22N 2269A-22	MS3106K-14S-5S
BETA PRESSURE SWITCH 2	6625-22 6647-22	MS31168F-3S or PT06E-8-3S
ET COMPENSATOR 3	Alumel & Chromel Lead 3114-22N 3105-22	SG3108E14S52S
AWI ET COMPENSATOR 4	7627AC-22 76D9B-22N	SG3106E10SL-56S
THERMOCOUPLE LEAD BETWEEN ET & AWI COMPENSATORS 5	264-61046 (Galileo Elector Optics)	SG3108E14-52S
IGNITOR BOX 6	2105-16 2107-16N 2107A-22N (Gnd)	MS3106A10SL-3S MS25036-103
NEGATIVE TORQUE SWITCH (NTS) 7	8707B-22 8701A-22 8702A-22	MS3106R-10SL-3S

Table 201

LEFT AND RIGHT ENGINE WIRE BUNDLE

UNIT NOMENCLATURE	WIRE LEAD PART NUMBER	TERMINAL OR CONNECTOR
NTS CHECKOUT VALVE 8	8707A-22 8707B-22	MS3106R10SL-3S
FUEL PURGE VALVE (EPA) 9	1631B-22N 1630B-22	MS3106R-10SL-3S
ENGINE FUEL SHUTOFF VALVE 10	1629-16 1630A-16 1631-22N	MS3108R-10SL-3S
AWI TORQUE PRESSURE SWITCH 11	7620AB-20 7609A-20N	MS3106R-10SL-3S
TORQUE TRANSDUCER 12	2006A-22 2004A-22 2000A-22N 2010A-22	CA3108ER10SL-3S
PROPELLER GOVERNOR MONOPOLE 13	SRL11A-22S (Contains white and black lead) (White to White and black to black)	RA18-D Wristlock
PROPELLER SYNCHROPHASER (SPEED PICK-UP) Magnetic pick-up to brush blocks	2411-22S 2413-22S	RA18-D Wristlock
(PROP GOVERNOR) 14	2407-22 (Lo black 2409-22 (Short black)	32446 Knife-Disconnect

Table 201
(Continued)

LEFT AND RIGHT ENGINE WIRE BUNDLE

UNIT NOMENCLATURE	WIRE LEAD PART NUMBER	TERMINAL OR CONNECTOR
CHIP DETECTOR 15	7302-22	MS3108R-8S-1S
OIL VENT VALVE 16	113-16 115-22N	MS3106R10SL-3S
PROPELLER DEICER 17	2317-10 (C) 2315-10 (B) 2313-10 (A)	MS3057-4A
450~F FIRE DETECTOR 18	(Below generator) 5708A-20 5708B-20 (Oil cooler scoop) 5708B-20 5701C-20 (Top forward of engine mount) 5701A-20 5701B-20	Terminals - MS25036-103 Red-color coded #10 stud size.
600~F FIRE DETECTOR 19	5701B-20 5701C-20	Terminals - MS25036-103 Red-color ceded #10 stud size.
TACHOMETER GENERATOR 20	2803-22 2804-22	SC06P-12-3S
OIL TEMPERATURE BULB 21	1205-22 1206-22N	MS3106A-12S-3S
UNFEATHER PUMP 22	119-16 175-16N (Gnd)	MS3106A-10S-2S MS25181-1

Table 201
(Continued)

LEFT AND RIGHT ENGINE WIRE BUNDLE

UNIT NOMENCLATURE	WIRE LEAD PART NUMBER	TERMINAL OR CONNECTOR
FUEL FLOW TRANSMITTER 23	1914C-22S 1913C-22S (Each of the above leads contain a black and white shielded wire)	MS3106R10SL-4S MS3116E-8-3S
FUEL ENRICHMENT START SOLENOID VALVE (SPR) 24	SRL19B-20 SRL19C-20N	MS3106R10SL-3S
ANTI-ICE LOCKOUT VALVE 25	144C-16 144D-22N	MS3106R10SL-4S
PRIMARY ONLY SOLENOID VALVE 26	SRL5C-22N SRL6B-22	MS3106R10SL-3S
LIMITER FUEL BYPASS VALVE 27	SRL50A-22(C) SRL51A-22(A)	MS3116F8-4S
RUDDER BIAS TORQUE PRESSURE SWITCH 28	7609-22N 7621AB-22	MS3106R10SL-3S
TT2 SENSOR 29	SRL15A-22S (Contains white and black lead)	MS3108R12S-3S
AWI MANIFOLD PRESSURE SIWTCH 30	7627AC-22 7627AB-22	MS3106E10SL-3S

Table 201
(Continued)

AIR INTAKES – DESCRIPTION AND OPERATION

1. General

A. The engine air intake is an integral function of the nose cowling assembly. The engine inlets are anti-iced with engine compressor bleed air. A lower cowling panel on the engine nacelle serves as an intake for the oil cooler.

B. For detailed maintenance practices concerning the SA226 series aircraft air intakes, see Section 71-10-00, Engine Cowlings, and Section 30-20-00, Air Intakes Anti-Ice.

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ENGINE DRAIN – DESCRIPTION AND OPERATION

1. General

A. Five overboard drains and vents are incorporated in the fuel and lubrication systems. The overboard outlets are grouped in a drain block attached to the cowling flange at the lower right of the firewall. Some hose drain a single point and others by means of tee fittings drain two or more points. Three of the drains are associated with the fuel system; one from the flow divider, one from the fuel control and engine water drain, and one from the two plenum drains. Two drains from the lubrication system provide for the hydraulic pump seal and unfeathering pump, and vent for the oil tank. The engine gear case is vented to the oil tank. The plumbing for the drains is a composite of tube assemblies and hoses, the latter being clamped at particular locations to ensure correct routing and minimize wear.

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ENGINE DRAIN – MAINTENANCE PRACTICES

1. Removal/Installation – Engine Drain

A. Removal – Engine Drain

(1) Remove lower cowling panel, see Section 71-10-00.

(2) Remove the drain hose to be replaced by loosening the hose clamp or end fitting and remove the drain hose.

(3) Cap all open connections.

(4) For removal of the drain block assembly remove all drain hoses and remove assembly from cowling area.

B. Installation – Engine Drain

(1) Ensure drain block support assembly is secure at the cowling flange at the lower right of the firewall.

(2) Install hose connecting cowling drain tube assemblies together, and secure with clamps. Install tube assembly into drain block assembly.

(3) Install union on unfeather pump drain hose. Install tube assembly to union; then, install tube assembly into drain block assembly.

(4) Install union onto engine case water drain hose. Install tube assembly onto union, then, install tube assembly into drain block assembly.

(5) Install union onto fuel control and pump case drain hose. Install tube assembly onto union, then install tube assembly into drain block assembly.

(6) Install union onto combustion case overboard drain hose. Install tube assembly onto union; then, install tube assembly into drain block assembly.

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MAINTENANCE MANUAL

CHAPTER 72 – ENGINE

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72-CONTENTS	-	1	NOV 01/85	72-00-00	DO	2	NOV 01/85

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MAINTENANCE MANUAL

CHAPTER 72 – ENGINE

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CHAPTER 72 – ENGINE

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STANDARD PRACTICE – ENGINE – DESCRIPTION AND OPERATION

1. General

A. This chapter concerns the primary divisions of the TPE331 engine to include the reduction gear and shaft section, compressor section, combustion section, turbine section, and power management system. Accessories and components of associated systems (fuel, oil, air, ignition electrical, and temperature indicating) required for engine operation are mounted on the reduction gear section and along the engine body.. For information pertaining to these primary divisions, refer to the following listed Garrett-AiResearch Maintenance Manuals.

B. For information pertaining to engine systems not included in the referenced manuals listed below, refer to applicable chapters within this Maintenance Manual.

C. For information pertaining to Fairchild components installed on the TPE331 engine, refer to Fairchild Powerplant Build-Up Manuals (dash 3 and dash 10).

Maintenance Manual: TPE331-3U-303G
Engine Part Number: 894040-1/2
Applicable Aircraft: Merlin III model SA226-T
Merlin IV model SA226-AT
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valves.

Maintenance Manual: TPE331-3UW-303G
Engine Part Number: 895880-1/-2
Applicable Aircraft: Metro model SA226-TC
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valves.
2. Alcohol/Water Injection System.

Maintenance Manual: TPE331-3U-304G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III model SA226-T
Merlin IV model SA226-AT
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valve.

*See Fairchild Aircraft Service Bulletin 30-011 for conversion procedures between TPE331-3U-303G and TPE331-3U-304G engines.

STANDARD PRACTICE – ENGINE – DESCRIPTION AND OPERATION

Maintenance Manual: TPE331-3UW-304G
Engine Part Number: 3102550-1
Applicable Aircraft: Metro model SA226-TC
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valve.
2. Alcohol/Water Injection System.

Maintenance Manual: TPE331-10U-501G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III (B) model SA226-T(B)

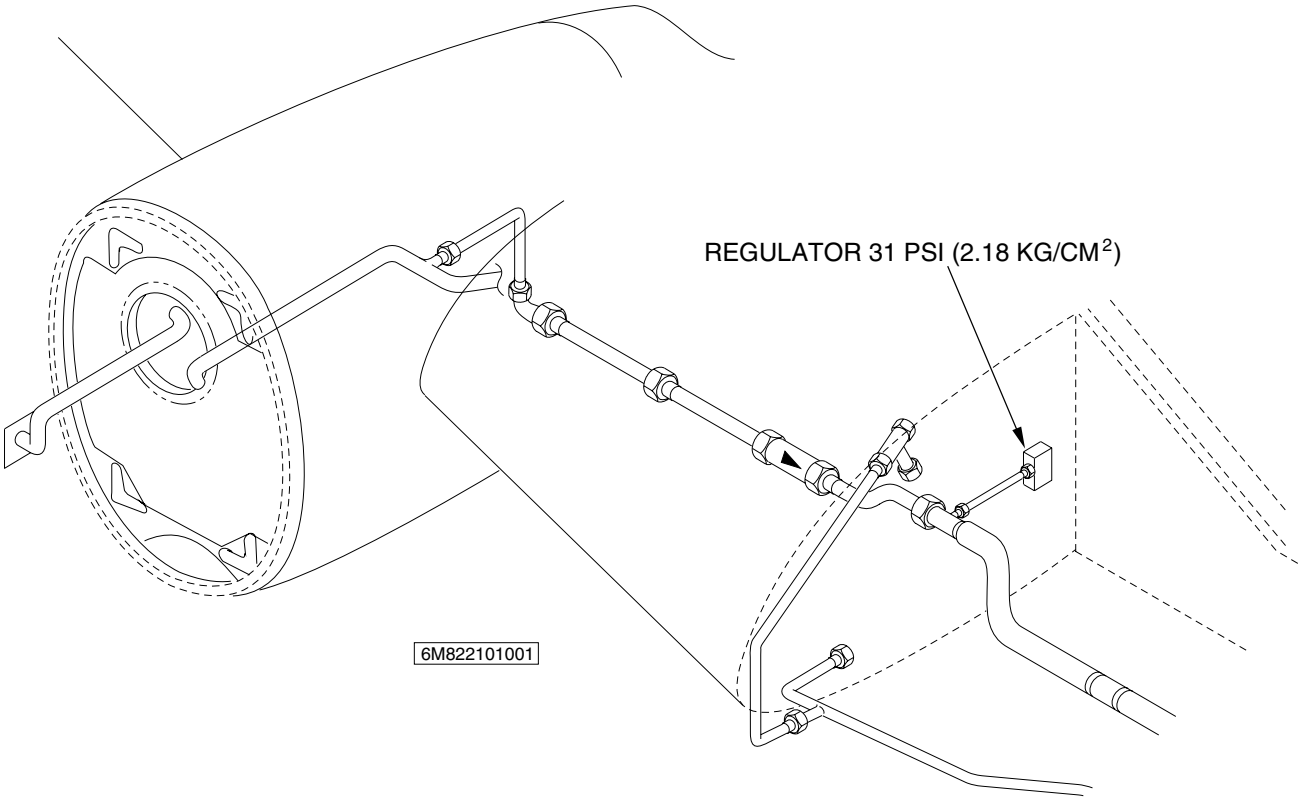
Maintenance Manual: TPE331-10UW-501G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III (B) model SA226-T(B)
Identifying Characteristic:
*1. Alcohol/Water Injection System.

*See Fairchild Aircraft Service Bulletin 30-011 for conversion procedures between TPE331-3U-303G and TPE331-3U-304G engines.

AIR PRESSURE REGULATOR – DESCRIPTION AND OPERATION

1. General

A. The air pressure regulator is used to reduce bleed air pressure to 31.5 ± 0.5 psi for use in the AWI system. The regulator is located under the right hand forward wing-to-fuselage fairing. Air pressure supplied by the regulator is also used to pressurize the hydraulic reservoir.



Air Pressure Regulator
Figure 1

AIR PRESSURE REGULATOR – FAULT ISOLATION

1. Troubleshooting – Air Pressure Regulator

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Awi system will not arm (as indicated by no air pressure in tank, arming valve operation has been checked and is satisfactory)	Blocked supply line	Remove supply line to regulator and check for air pressure with engine operating.
	Defective regulator	Replace regulator.

AIR PRESSURE REGULATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Air Pressure Regulator

A. Removal – Air Pressure regulator

(1) Gain access to the regulator by removing the right hand longitudinal section center section access panel.

(2) Remove lines connected to the pressure regulator.

(3) Remove screws securing the regulator to the bracket.

(4) Remove the regulator.

(5) Remove the fittings from the old regulator and store in a safe place.

B. Installation – Air Pressure regulator

(1) Install the fittings taken from the old regulator into the new regulator.

(2) Position the new regulator within the wing.

(3) Secure the regulator to the bracket with screws.

(4) Connect the lines to the regulator.

(5) Install access panel.

(6) Operational check the AWI system IAW the Airplane Flight Manual instructions.

2. Inspection/Check – Air Pressure Regulator

A. Inspect the regulator for security of mounting, physical damage, and security of the connections to the regulator.

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MAINTENANCE MANUAL

CHAPTER 73 – ENGINE FUEL

LIST OF EFFECTIVE PAGES

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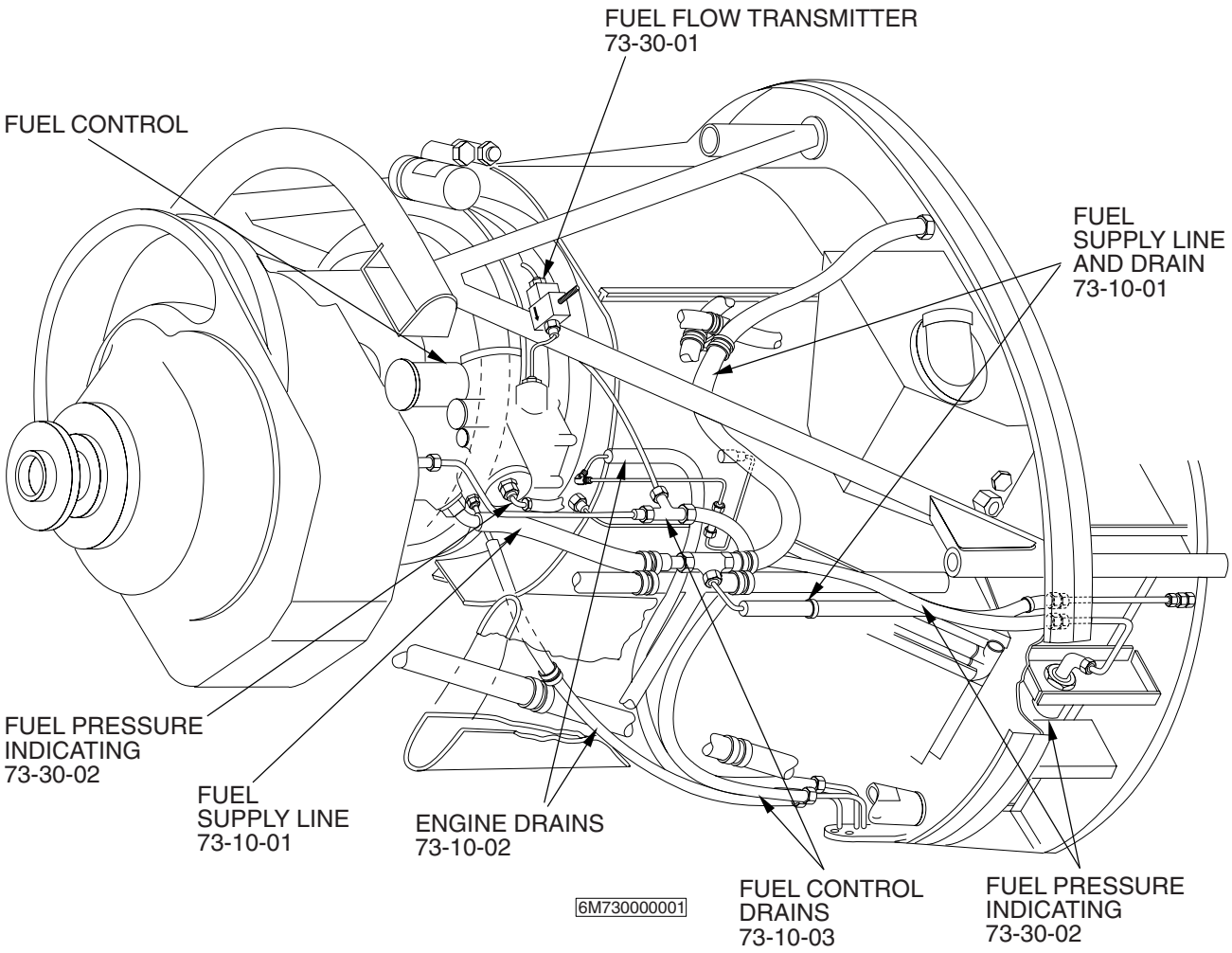
ENGINE FUEL AND CONTROL – DESCRIPTION AND OPERATION

1. General

A. The fuel system forward of the engine firewall consists of the fuel supply line and drain, engine and flow divider drains, fuel control drains, and fuel flow and fuel indicating (see Figure 1).

B. Coverage is provided for the components added to the basic engine by M7 Aerospace. The Honeywell Maintenance Manual provides information relative to the actual engine components.

C. Fuel system components located aft of the firewall area are covered in Chapter 28 of this Maintenance Manual.



Engine Fuel and Control Components
Figure 1

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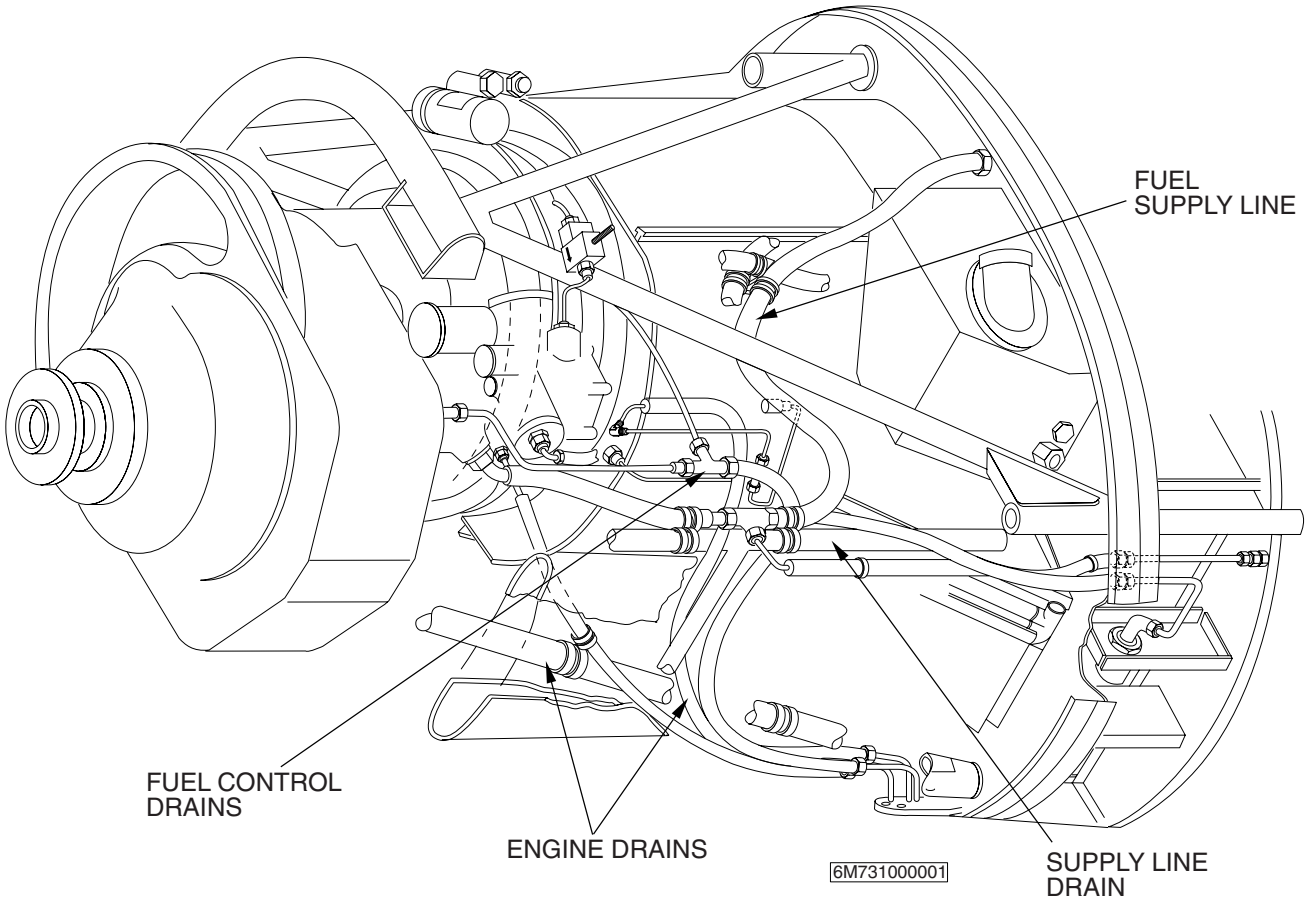
FUEL DISTRIBUTION – DESCRIPTION AND OPERATION

1. General

A. The fuel distribution system forward of the engine firewall (see Figure 1) consists of the fuel supply line, the fuel supply line drain, the engine drains and the fuel control drains. The distribution system also includes the EPA fuel purge kit, installed on all aircraft in order to comply with EPA fuel dumping regulations.



Fuel Distribution
Figure 1

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FUEL SUPPLY LINE AND DRAIN – DESCRIPTION AND OPERATION

1. General

A. The fuel supply line runs from the engine firewall forward to the engine driven fuel pump. A fuel supply line drain hose is joined to the fuel line by a tee connection. The drain valve is located in the wheel well aft of the transmitter access bay.

FUEL SUPPLY LINE AND DRAIN – MAINTENANCE PRACTICES

1. Inspection/Check – Fuel Supply Line and Drain

A. Inspection

(1) Inspect fuel lines for kinks, cracks or other obvious damage.

CAUTION: IT IS POSSIBLE TO REVERSE THE FUEL, OIL AND AWI CONNECTIONS AT THE FIREWALL. MAINTENANCE PERSONNEL SHOULD BE ADVISED TO VERIFY THE PROPER CONNECTIONS FOLLOWING AN ENGINE CHANGE OR MAINTENANCE ACTIVITY INVOLVING REMOVAL OF THE LINES.

EARLIER AIRCRAFT WERE DELIVERED WITH A LARGER DRAIN VALVE (P/N 23020060). CURRENT AIRCRAFT ARE DELIVERED WITH A SMALLER DRAIN VALVE (P/N 140B). EITHER DRAIN MAY BE USED ON AIRCRAFT WITH STANDARD MAIN TIRES. AIRCRAFT EQUIPPED WITH LOW PRESSURE TIRES MUST USE THE SMALLER DRAIN VLAVE (P/N 140B).

(2) Inspect all connection points for evidence of fuel leakage.

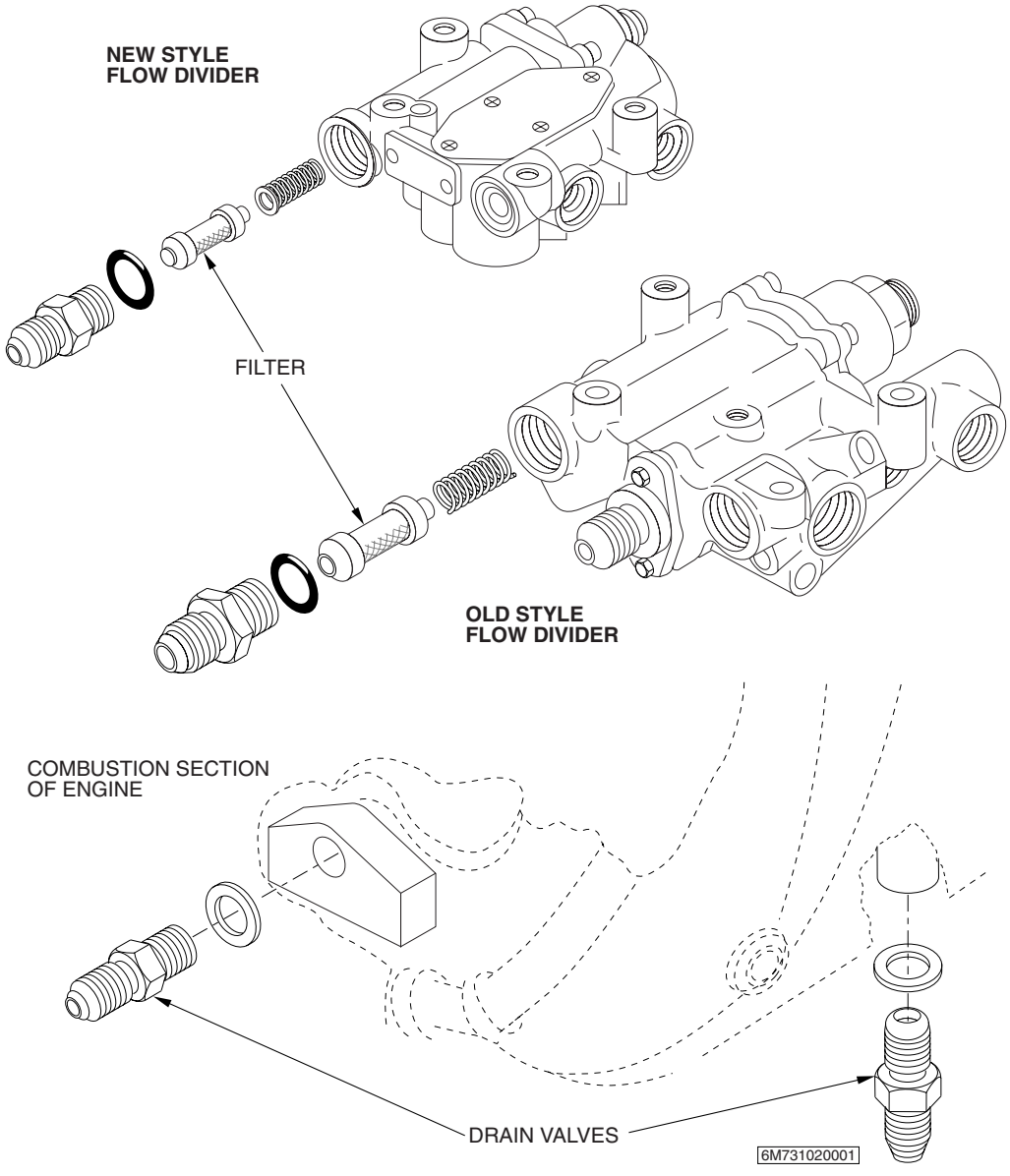
(3) Inspect threaded parts for damage.

(4) Inspect clamps for worn or missing cushions.

ENGINE AND FLOW DIVIDER DRAINS – DESCRIPTION AND OPERATION

1. General

A. Three drains, two in the combustion section and one in the compressor section, are installed on each engine. All three drains terminate at the drain block on the lower right hand side of the engine. The flow divider drain is also routed to the drain block on earlier and unmodified aircraft. During engine shutdown, unburned fuel in the system was dumped overboard. An Environmental Protection Agency regulation required that aircraft no longer dump fuel during normal operations. Most earlier aircraft and all current production aircraft are delivered with a fuel purge kit that causes all remaining fuel to be forced into the engine and burned during shutdown.



Engine and Flow Divider Drains
Figure 1

ENGINE AND FLOW DIVIDER DRAINS – MAINTENANCE PRACTICES

1. Servicing – Engine and Flow Divider Drains

A. Service engine and flow divider drains as follows:

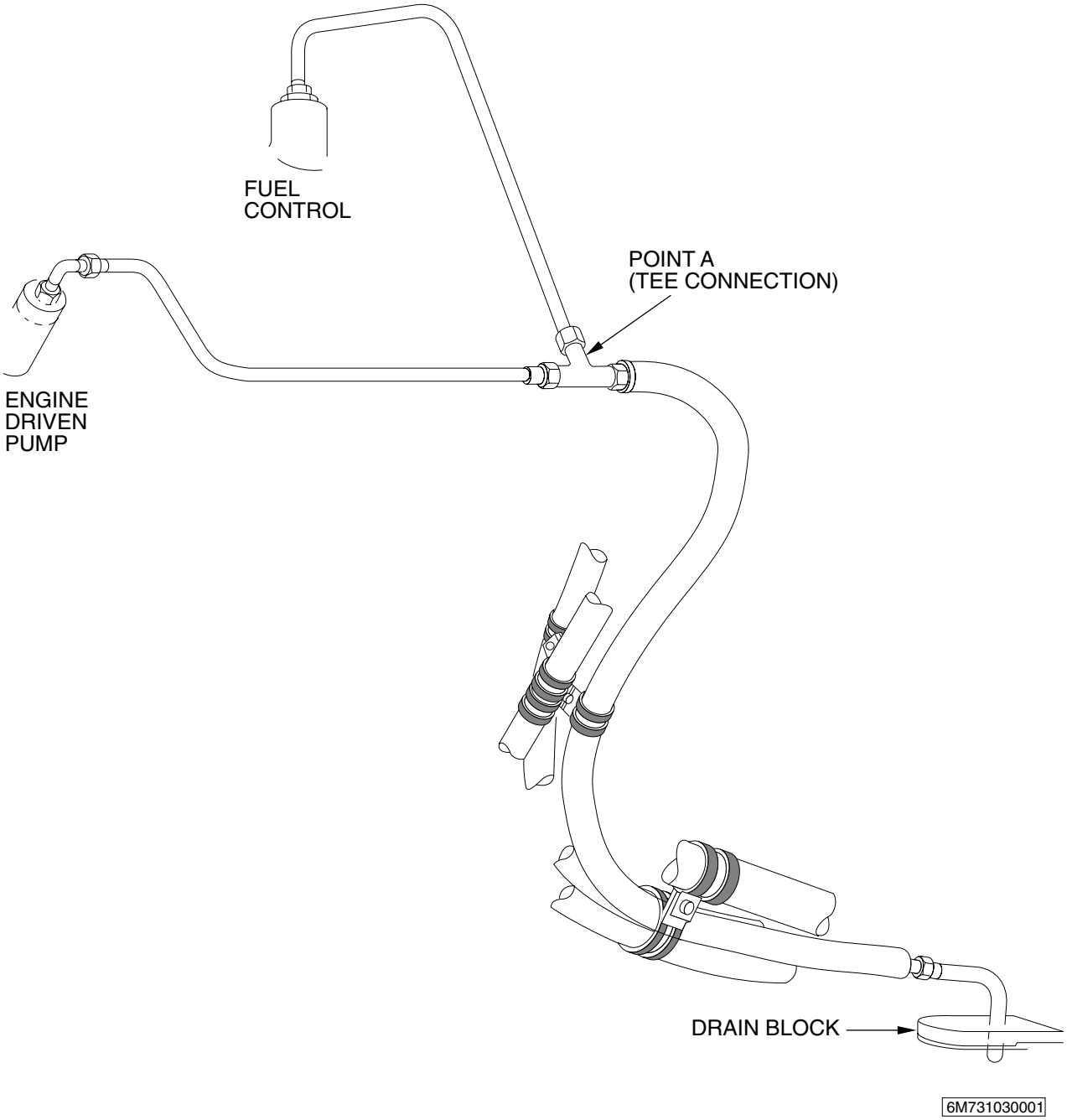
(1) Remove, inspect, and clean the flow divider filter. The filter should be cleaned ultrasonically if proper equipment is available. As an alternate the filter may be dipped in solvent (Shell 360 or equivalent) and dried with filtered compressed air. Complete cleaning procedures are outlined in the Honeywell Maintenance Manual, Section 73-10-07.

(2) Remove, inspect, and clean the two combustion section drain valves (reference Figure 1). The valves are normally open and should close with 2.0 to 5.0 psi air pressure. Make sure to coat the threads of the valve with high temperature anti-seize (Fel-Pro C5-A or equivalent) before installation. Torque drain valves to 60 to 65 inch-pounds (6.78 to 7.34 N·m).

FUEL CONTROL DRAINS – DESCRIPTION AND OPERATION

1. General

A. The two fuel control drains, engine driven fuel pump and fuel control unit, are joined together by a tee connection and terminate at the drain block.



Fuel Control Drains
Figure 1

FUEL CONTROL DRAINS – MAINTENANCE PRACTICES

1. Inspection/Check – Fuel Control Drains

A. Inspection – Fuel Control Drains

(1) Inspect drain hoses and lines for kinks, cracks or other obvious damage.

(2) Inspect all connection points for evidence of fuel leakage.

NOTE: Maximum allowable leakage for fuel control is 1.0 cc/min (approximately 960 drops per hour) and for engine driven fuel pump, 0.5 cc/hr (approximately eight drops per hour). Before removing a fuel control unit because of leakage, separate the fuel control and pump drains at point A in Figure 1. Operate the engine and observe which unit is leaking. Only the defective unit need be replaced.

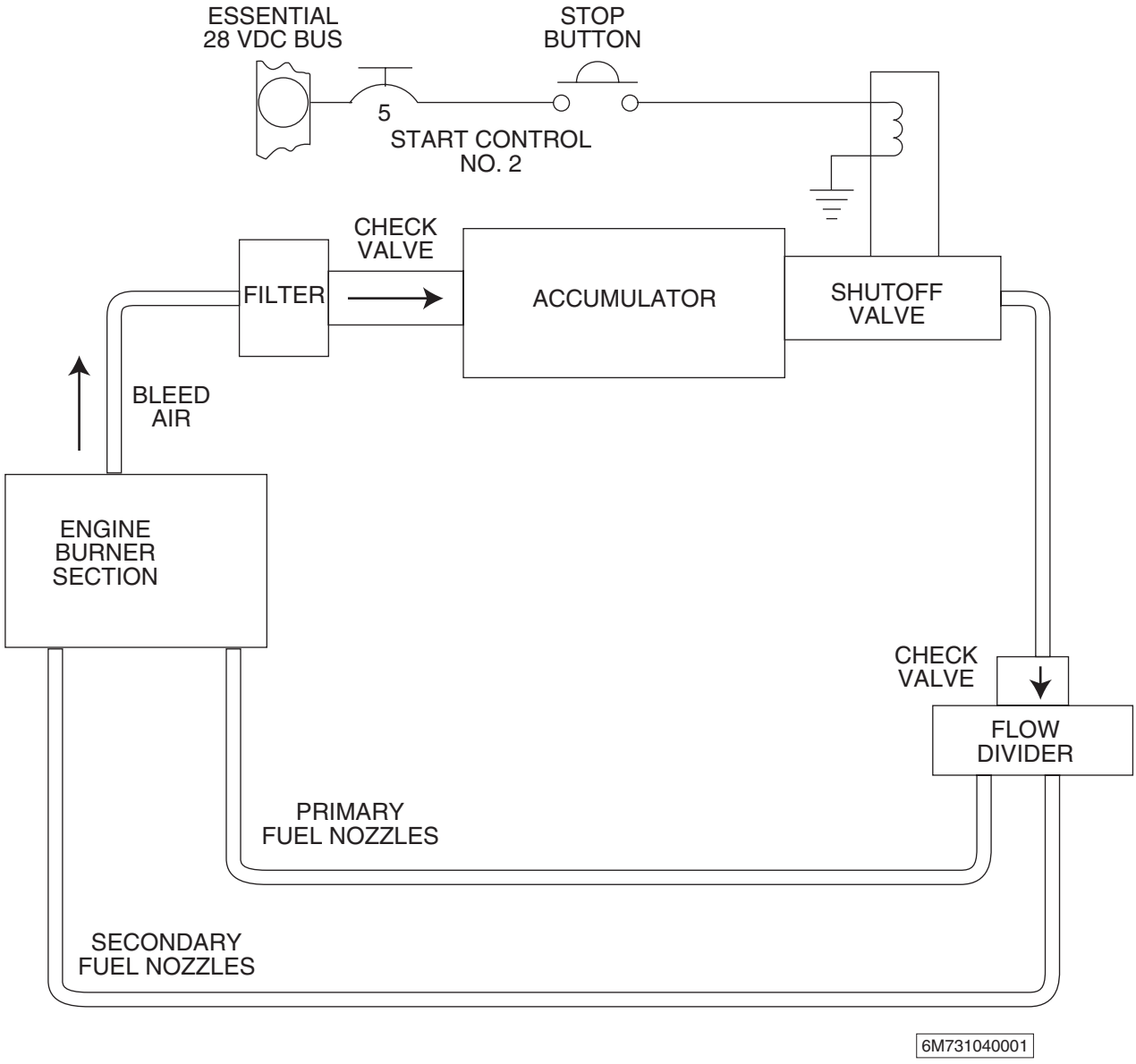
(3) Inspect threaded parts for damage.

(4) Inspect clamps for worn or missing cushions.

EPA FUEL PURGE KIT – DESCRIPTION AND OPERATION

1. General

A. The fuel purge system was developed to comply with EPA fuel venting requirements which prohibit overboard draining of fuel. This purge system causes the residual fuel in the fuel mainfolds to be burned during shutdown, thereby eliminating the need for drain fuel storage.



EPA Fuel Purge System
Figure 1

B. As shown in Figure 1, bleed air is extracted from a pad on the right hand side of the engine for use in the fuel purge system. A filter, check valve, accumulator and solenoid operated shutoff valve are installed as an assembly on the right hand side of the engine mount.

CAUTION: THE ACCUMULATOR STORES APPROXIMATELY 150 PSI OF AIR PRESSURE IF NOT DISCHARGED. THIS CAN PRESENT A HAZARD TO PERSONNEL PERFORMING MAINTENANCE TO THE ENGINE.

(1) The accumulator stores air pressure that is trapped by a check valve at one end and a normally closed shutoff valve at the opposite end.

CAUTION: BEFORE STOPPING AN ENGINE, MAKE SURE THAT THE ENGINE HAS BEEN OPERATED AT 96 TO 100% TO CHARGE THE ACCUMULATOR. WHEN STOPPING THE ENGINE, HOLD THE STOP BUTTON DEPRESSED FOR AT LEAST FIVE SECONDS. THE ENGINE SHOULD ACCELERATE FIVE TO SEVEN PERCENT WHEN THE STOP BUTTON IS PUSHED. SMOKE FROM THE TAIL PIPE OR FAILURE OF THE ENGINE TO ACCELERATE INDICATES THE ACCUMULATOR WAS NOT PROPERLY CHARGED OR THE SYSTEM IS MALFUNCTIONING. FAILURE OF THE SYSTEM TO PROPERLY PURGE THE FUEL SYSTEM WILL NOT NECESSARILY DAMAGE THE ENGINE BUT MAY CAUSE QUICKER PLUGGING OF THE FUEL NOZZLES AND SHOULD BE AVOIDED.

WHEN STOPPING THE ENGINE USING THE STOP AND FEATHER KNOB, THE ACCUMULATOR WILL NOT BE DISCHARGED. IF ON THE NEXT START CYCLE THE ENGINE TEMPERATURE BEGINS TO NEAR RED LINE, ACTUATION OF THE STOP BUTTON WILL DISCHARGE THE EPA KIT; THIS WILL CAUSE AN EVEN GREATER RISE IN ENGINE TEMPERATURE. BECAUSE OF THIS POSSIBILITY, AFTER A SHUTDOWN USING THE STOP AND FEATHER CONTROL AND AFTER THE ENGINE HAS COOLED SUFFICIENTLY TO PREVENT COMBUSTION, ACTUATE THE STOP BUTTON FOR A MINIMUM OF FIVE SECONDS BEFORE BEGINNING ANOTHER START CYCLE. IN-FLIGHT SHUTDOWN PROCEDURES ARE CONTAINED IN SECTION VI OF THE FLIGHT MANUAL AND AIRESEARCH SIL P331-80.

(2) The shutoff valve is actuated to the open position by depressing the stop button in the cockpit. With the shutoff valve open, the stored air pressure is supplied to the drain port of the engine fuel flow divider. Inside the flow divider, the air pressure moves the primary and secondary drain valves to a position that blocks the fuel supply to the engine and supplies air pressure to the primary and secondary fuel nozzles.

(3) All remaining fuel in the lines, manifolds and nozzles will be pushed into the engine and burned. A check valve installed upstream of the flow divider prevents reverse flow should the purge system malfunction.

EPA FUEL PURGE KIT – FAULT ISOLATION

1. Troubleshooting – EPA Fuel Purge Kit

PROBLEM:

EPA KIT WILL NOT CHARGE

PREREQUISITES:

None.

START:

Test check valve at accumulator for pressure. Is check valve holding pressure?	**NO** ►	Replace check valve.

END

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EPA FUEL PURGE KIT – FAULT ISOLATION

PROBLEM:

EPA KIT WILL NOT DISCHARGE

PREREQUISITES:

None.

START:

Test continuity at electrical connector and solenoid for 28 VDC. Does electrical connector have 28 VDC?

↓ YES

Replace solenoid valve.

END

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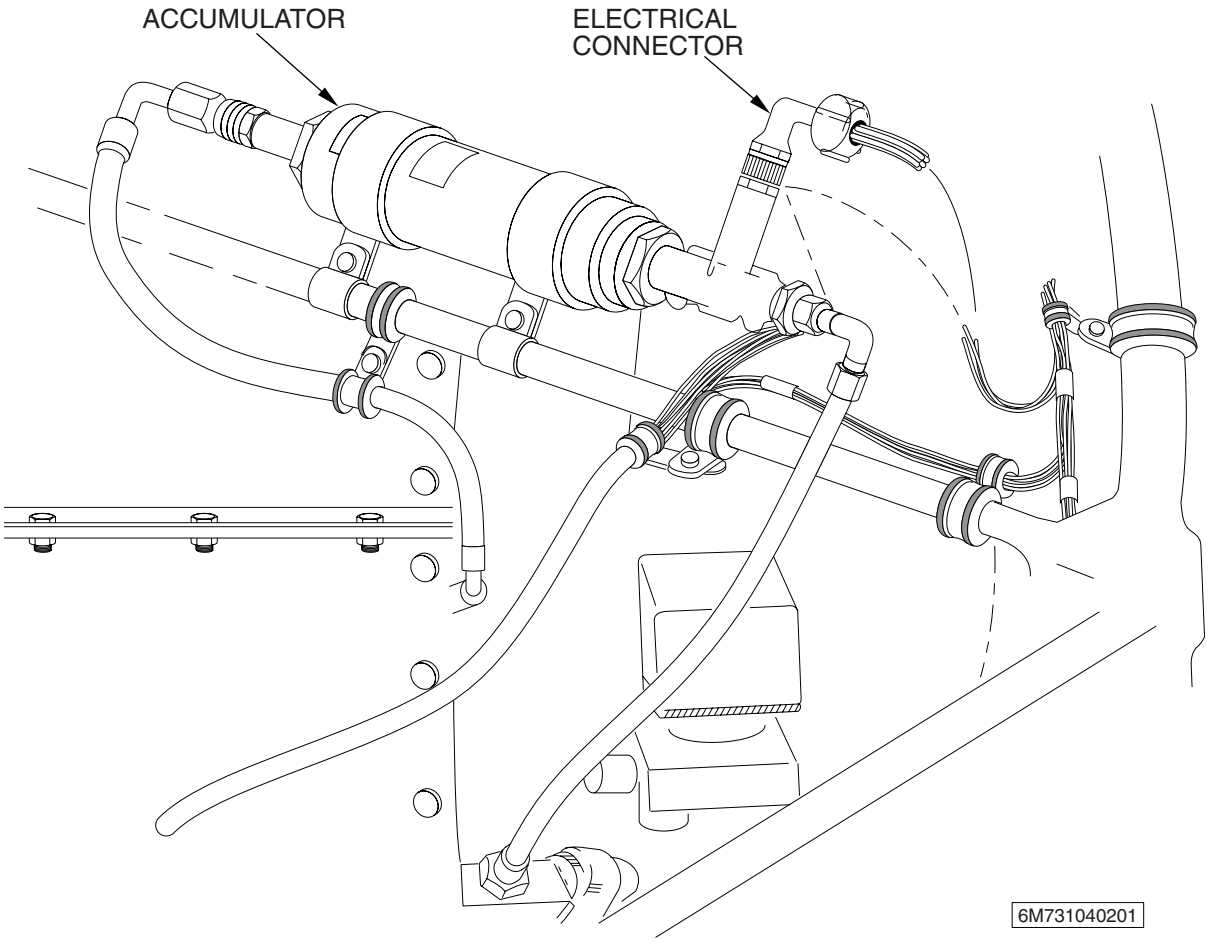
1. Servicing – EPA Fuel Purge Kit

A. Service EPA Fuel Purge Kit as follows:

(1) Remove, clean and inspect the fuel purge system filter. The filter should be cleaned ultrasonically if proper equipment is available. As an alternate the filter may be back flushed using solvent (PD-680 or equivalent). Complete cleaning procedure are outlined in the Honeywell Maintenance Manual, section 73-10-13.



Accumulator
Figure 201

2. Removal/Installation – EPA Fuel Purge Kit

CAUTION: THE AIR ACCUMULATOR AND ADJACENT TUBING MAY BE CHARGED WITH AIR APPROXIMATELY 90 TO 130 PSI AND MAY PRESENT A HAZARD DURING MAINTENANCE OPERATION.

A. Removal – EPA Fuel Purge Kit

(1) Disconnect solenoid valve electrical plug.

(2) Disconnect B nuts at both ends of accumulator.

(3) Disconnect two clamps securing accumulator and remove unit.

(4) Clamp unit in a vice and slowly unscrew solenoid valve from accumulator (to slowly relieve any pressure that may be in accumulator.

B. Installation – EPA Fuel Purge Kit

(1) Install solenoid valve to accumulator.

(2) Secure accumulator to truss with clamps.

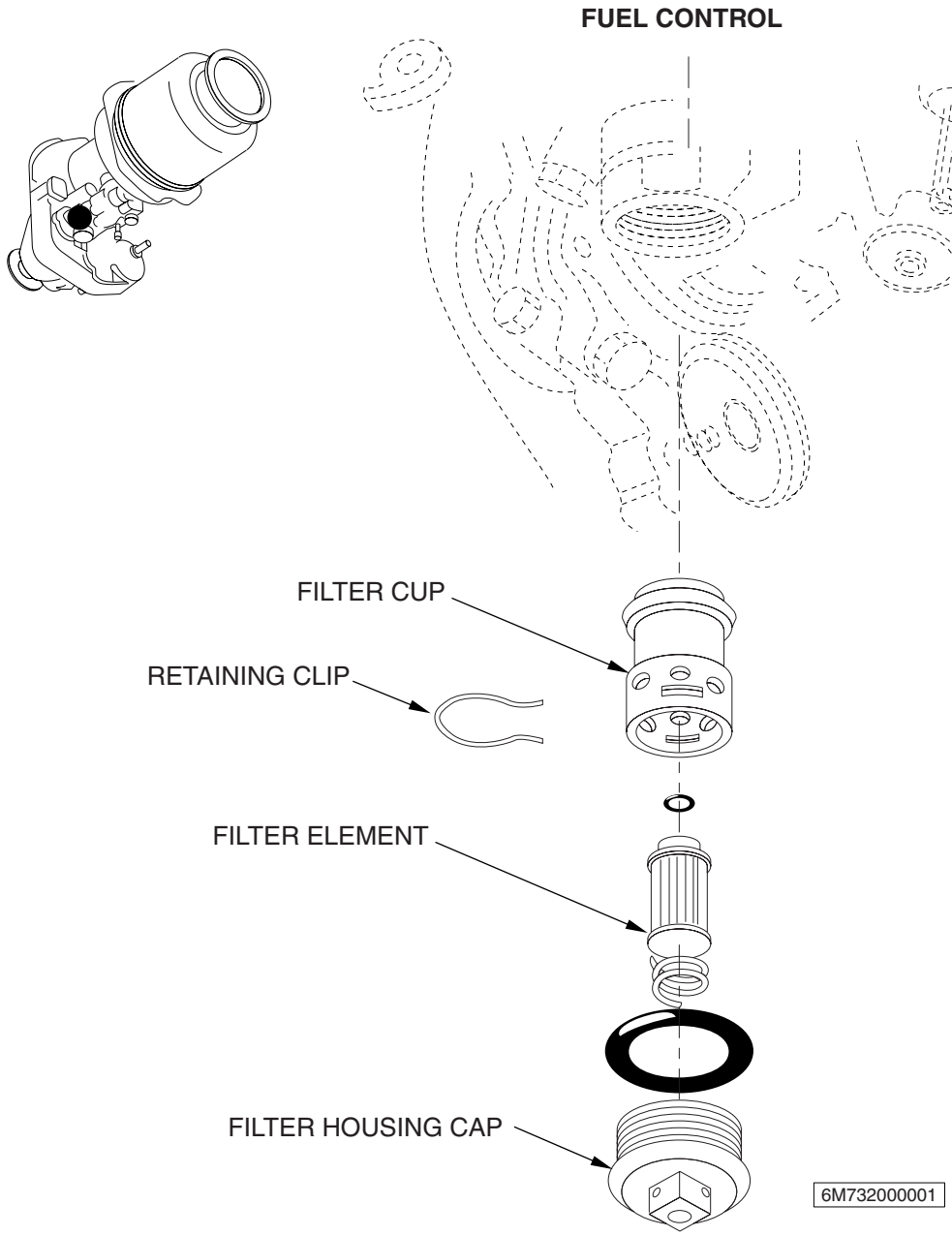
(3) Connect and tighten B-nuts at both ends of accumulator.

(4) Connect solenoid valve electrical plug.

FUEL CONTROL – DESCRIPTION AND OPERATION

1. General

A. Each engine-furnished control assembly is composed of an engine-driven fuel pump and a fuel control unit. They are stacked together as one unit. The fuel control unit meters the fuel supplied by the fuel pump to the engine combustion chamber.



Fuel Control Filter
Figure 1

1. Inspection/Check – Fuel Control

NOTE: The following outlines the basic maintenance practices and interval for fuel control maintenance. For more detailed fuel control maintenance, refer to the Honeywell Maintenance Manual.

A. After initial engine run, remove, clean, and inspect the fuel control filter as follows:

(1) Cut safety wire and unscrew filter housing cap.

(2) Withdraw retaining clip and remove the filter cup.

(3) Remove and clean filter. The recommended method and testing is detailed in the Honeywell Maintenance Manual, section 73-21-06.

(4) Install filter in cup and secure in the housing cap using the retaining clip.

(5) Coat the threads of the cap with anti-seize (liqui-Moly Grade NV, Lockrey Co., Southampton, New York or equivalent) and install filter assembly. Tighten filter cap to 15 to 20 inch-pounds (1.69 to 2.26 N·m) and safety wire. Check filter assembly for leaks after the next engine run.

B. Pre Garrett Turbine Engine Company (GTEC) SB73-0131; After initial operation:

(1) Check torque of fasteners securing support (triangular shaped piece of metal at rear of fuel control) to engine and fuel control. The fastener at the fuel control should be torqued to 60 inch-pounds (6.78 N·m). The two fasteners securing the support should be torqued to 45 to 55 inch-pounds (5.08 to 6.21 N·m).

C. Limited inspection:

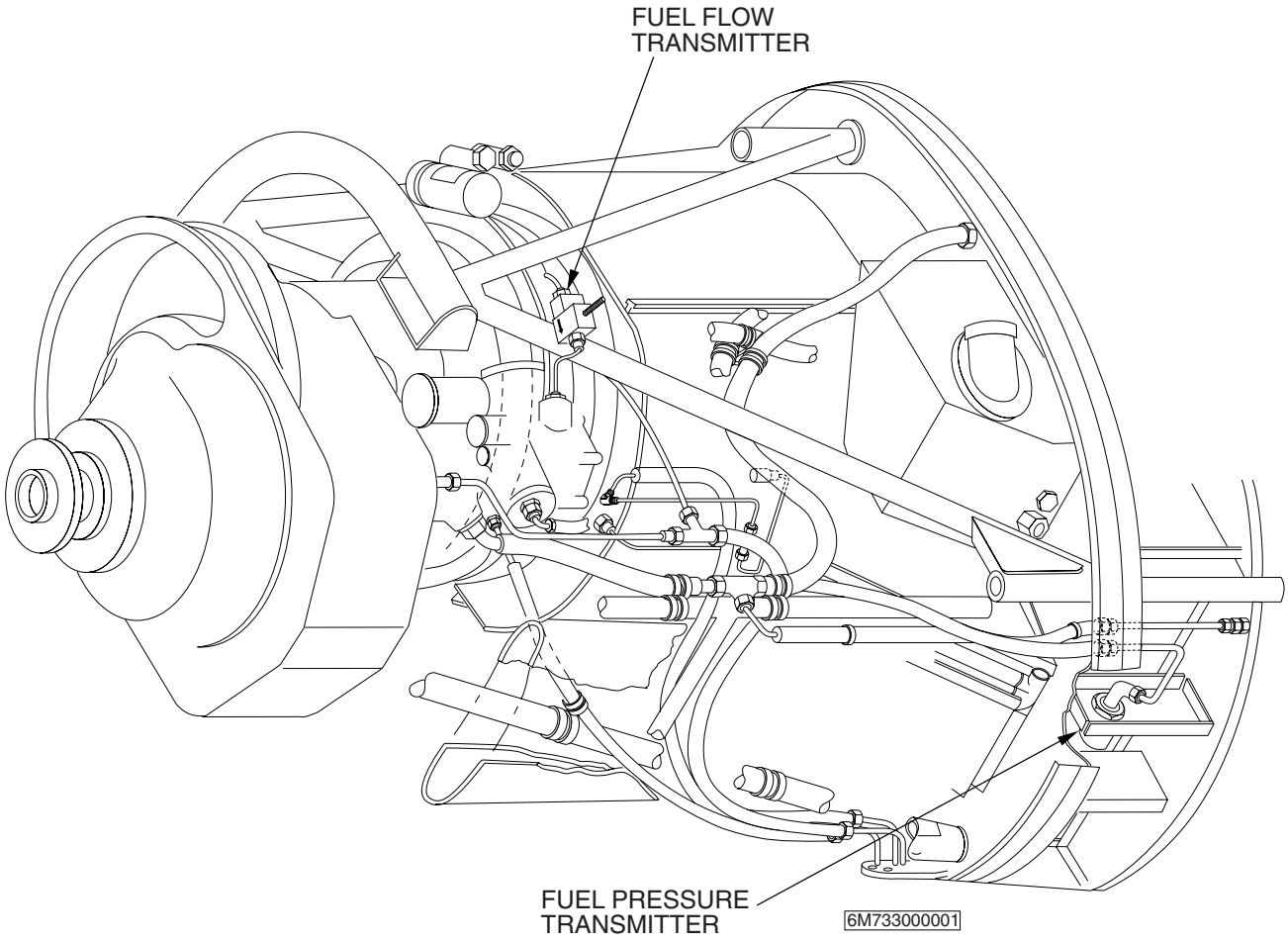
(1) Remove, clean, inspect and lubricate fuel control drive coupling. The effectivity of this inspection is limited. Refer to GTEC SB72-0104 and 73-0051 for specific instructions.

(2) Inspect fuel control drive backlash. Effectivity of this inspection is limited. Refer to GTEC SB72-0095 for specific instructions.

FUELINDICATING – DESCRIPTION AND OPERATION

1. General

A. Engine fuel indicating consists of fuel pressure indicating and fuel flow indicating. Each system uses indicators to display information to the pilot. Both systems are independent of each other and of the other aircraft systems. There are no fuel pressure or fuel flow warning lights.



Indicating Components
Figure 1

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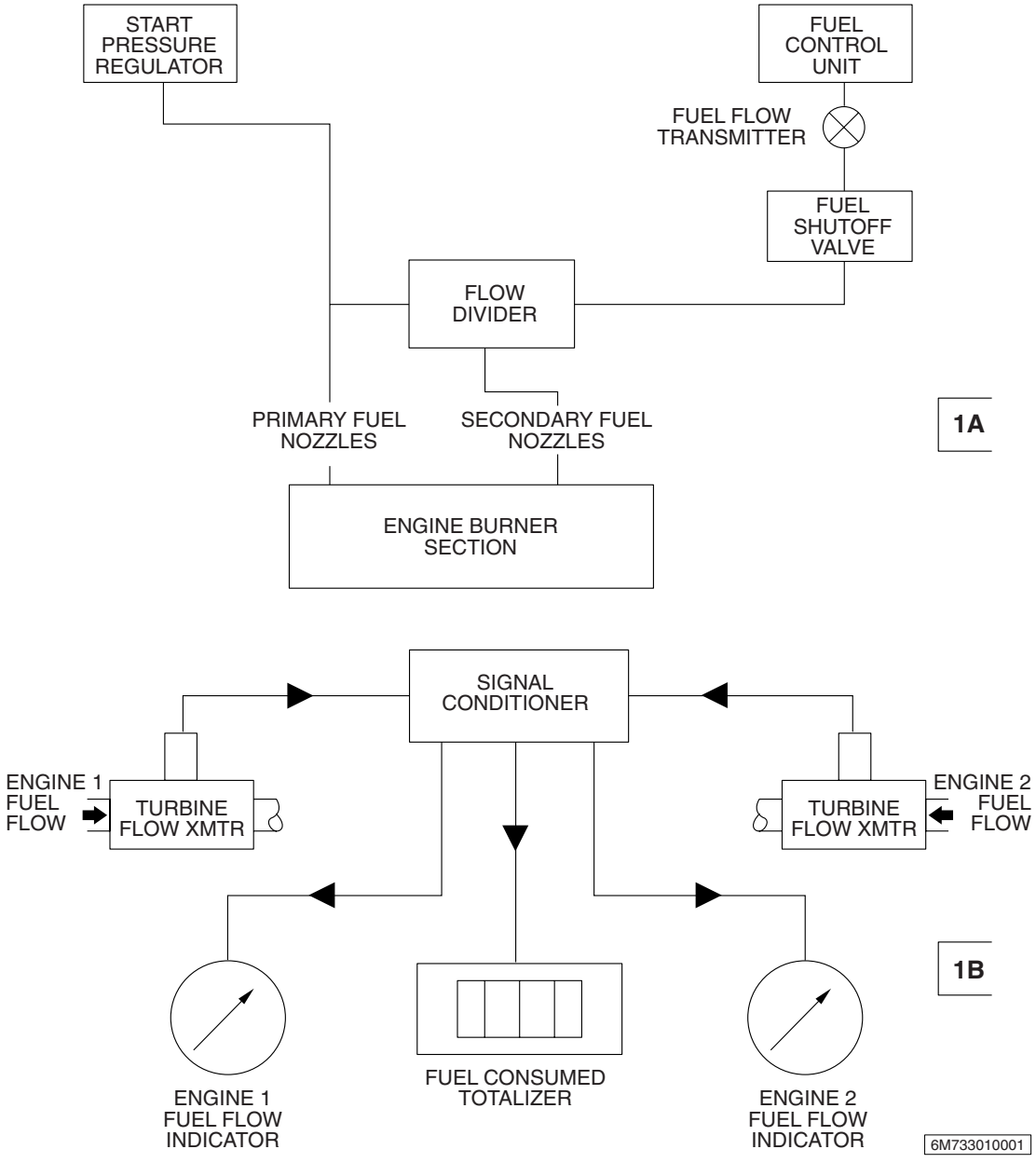
FUEL FLOW INDICATING – DESCRIPTION AND OPERATION

1. General

A. Fuel flow indicating consists of a transmitter installed on each engine, a signal conditioner installed in the left side console, two fuel flow indicators, and a fuel totalizer indicator. The system displays the amount of fuel in pounds per hour that each engine is consuming and the total amount fuel in pounds that has been consumed by both engines. Fuel flow indicating and totalizing is not part of the fuel quantity system nor is there any mechanical or electrical commonality to the two systems. As shown in Figure 1A, the fuel flow system will not display start pressure regulator (SPR) fuel flow.



Fuel Flow Indicating
Figure 1

FUEL FLOW INDICATING – FAULT ISOLATION

1. Troubleshooting – Fuel Flow Indicating

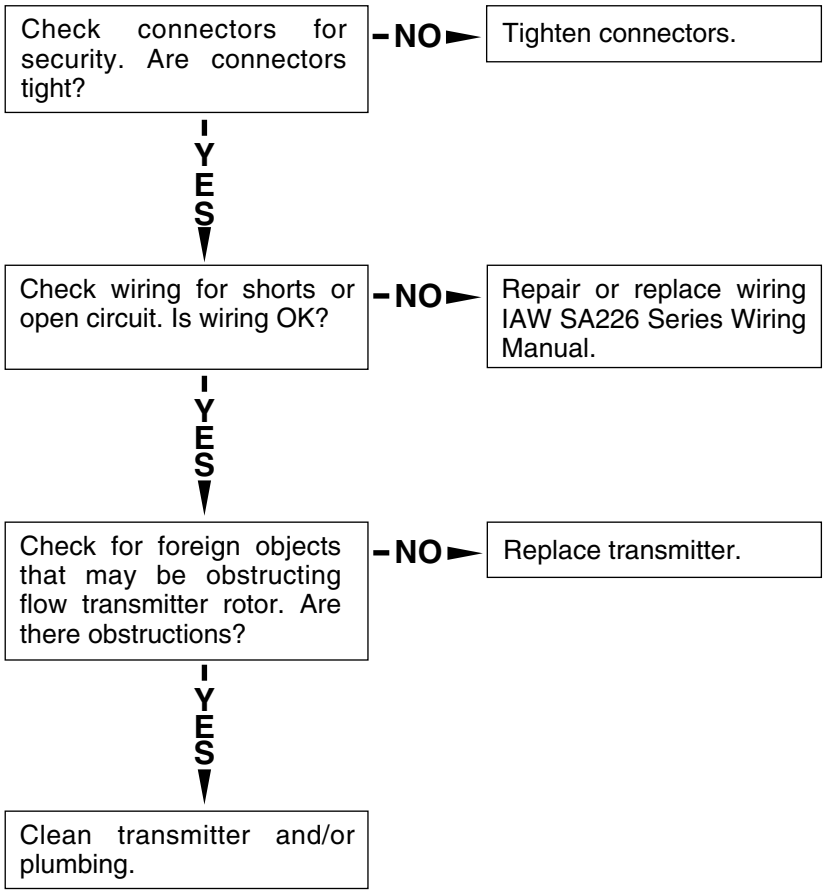
PROBLEM:

ERRATIC INDICATION

PREREQUISITES:

Both engines operating.

START:



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END

FUEL FLOW INDICATING – FAULT ISOLATION

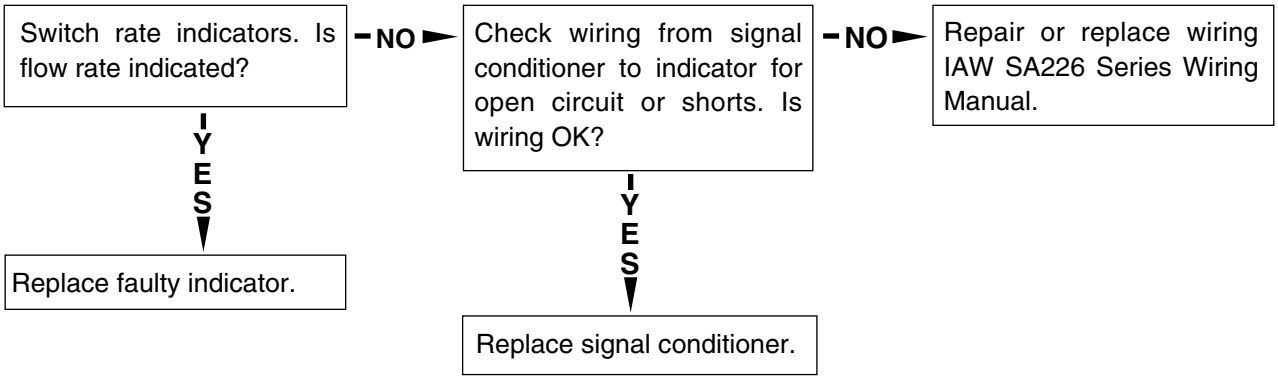
PROBLEM:

NO RATE INDICATION, TOTALIZER NORMAL

PREREQUISITES:

Both engines operating.

START:



END

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MAINTENANCE MANUAL

FUEL FLOW INDICATING – FAULT ISOLATION

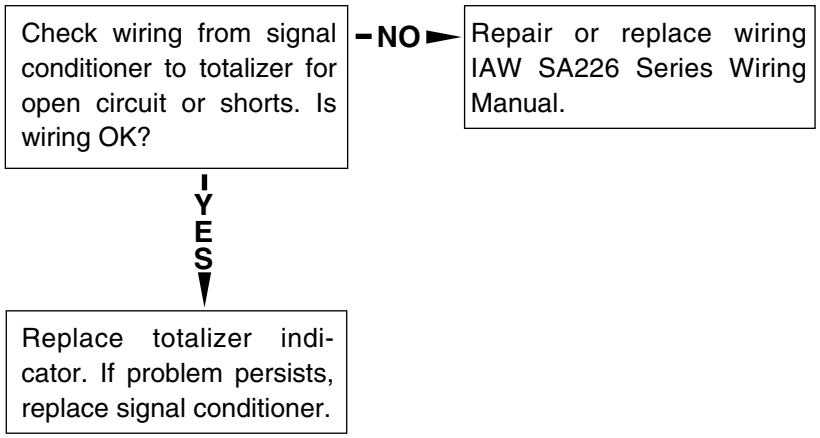
PROBLEM:

TOTALIZER INOPERATIVE, RATE INDICATION NORMAL

PREREQUISITES:

Both engines operating.

START:



END

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FUEL FLOW INDICATING – FAULT ISOLATION

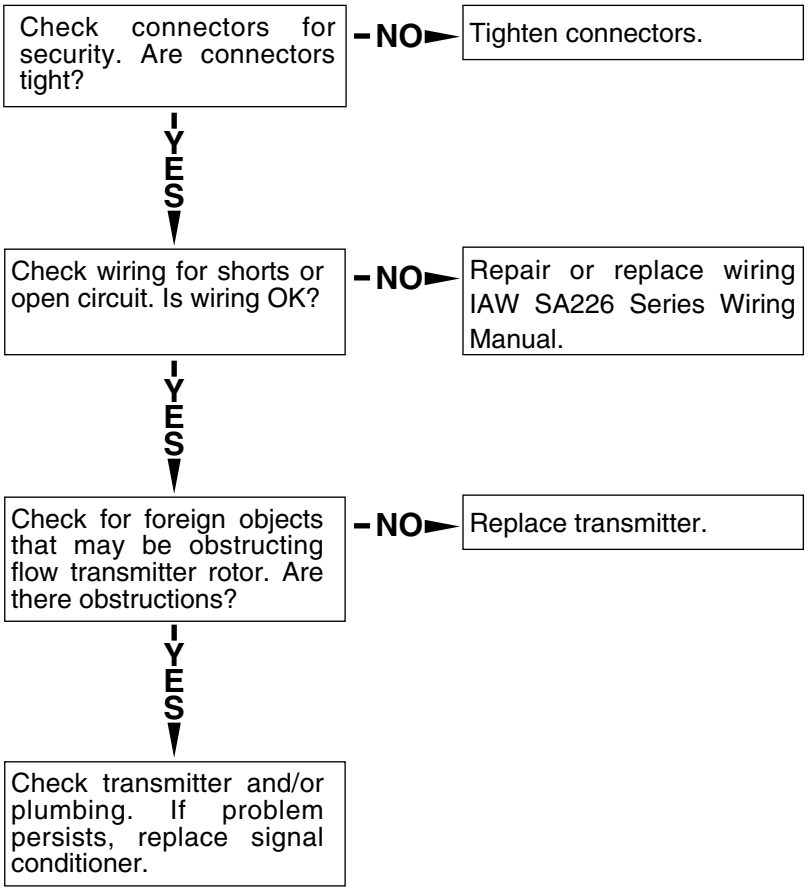
PROBLEM:

NO RATE OR TOTALIZER INDICATION

PREREQUISITES:

Both engines operating.

START:



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FUEL FLOW INDICATING – MAINTENANCE PRACTICES

1. Removal/Installation – Fuel Flow Indicating

A. Removal – Fuel Flow Transmitter

(1) Remove left side cowling.

(2) Disconnect fuel flow transmitter electrical connector(s).

NOTE: When B-nut is removed from bottom connection of transmitter it will slide down the tube and out of sight and is difficult to retrieve when reconnecting transmitter. Wrap tape around tube two to three inches below connection before loosening nut; this will prevent nut from sliding out of reach.

(3) Disconnect B-nuts at top and bottom of transmitter and remove from line.

(4) Cap fuel lines to prevent contamination.

B. Installation – Fuel Flow Transmitter

(1) Remove caps from fuel lines.

(2) Connect B–nuts to top and bottom of transmitter and secure to line.

(3) Connect fuel flow transmitter electrical connector(s).

(4) Install cowling.

C. Removal – Fuel Flow Indicator

(1) Remove post light from instrument panel.

(2) Loosen holding screw (lower right one) securing indicator to instrument panel.

(3) Pull indicator straight out of instrument panel.

(4) Disconnect electrical connector from indictor.

D. Installation – Fuel Flow Indicator

(1) Connect electrical connector to indicator.

(2) Insert indicator into instrument panel.

(3) Secure indicator to instrument panel with holding screw.

(4) Secure post light to instrument panel.

E. Removal – Fuel Totalizer

(1) Remove two post lights from instrument panel.

(2) Remove two holding screws securing totalizer to instrument panel.

(3) Pull totalizer straight out of instrument panel.

(4) Disconnect electrical connector from totalizer.

F. Installation – Fuel Totalizer

(1) Connect electrical connector to totalizer.

(2) Insert totalizer into instrument panel.

(3) Secure totalizer to instrument panel with holding screws.

(4) Secure post lights to instrument panel.

G. Removal – Signal Conditioner

(1) Remove center panel on left side console (pilot side).

NOTE: Signal conditioner is attached to backside of panel.

(2) Remove safety wire from electrical connector(s).

(3) Disconnect electrical connector(s) from signal conditioner.

(4) Remove screws securing signal conditioner to panel. Remove signal conditioner.

H. Installation – Signal Conditioner

NOTE: Replacement for fuel flow computer (signal conditioner) P/N PC-423 is PC-620 (SC10-0071A). Replacement for computer P/N PC-620 is PC-620 (SC10-0134).

(1) Position signal conditioner to panel.

(2) Secure signal conditioner to panel with screws.

(3) Connect electrical connector(s) to signal conditioner.

(4) Install safety wire to electrical connector.

(5) Install panel to console.

2. Cleaning/Painting – Fuel Flow Indicating

A. Cleaning – Fuel Flow Transmitter

CAUTION: DO NOT USE AIR HOSE TO SPIN ROTOR. EXCESSIVE PRESSURE WILL CAUSE TRANSMITTER DAMAGE.

(1) Blow lightly into the transmitter housing and observe rotor movement.

NOTE: If transmitter is serviceable, the rotor will spin freely for several seconds.

(2) Flush transmitter with isopropyl alcohol taking care not to immerse coil assembly.

(3) If rotor does not spin freely, repeat step **2.**, A., (1).

(4) If rotor still does not spin freely, repeat step **2.**, A., (2).

(5) If malfunction is not corrected by flushing, replace transmitter.

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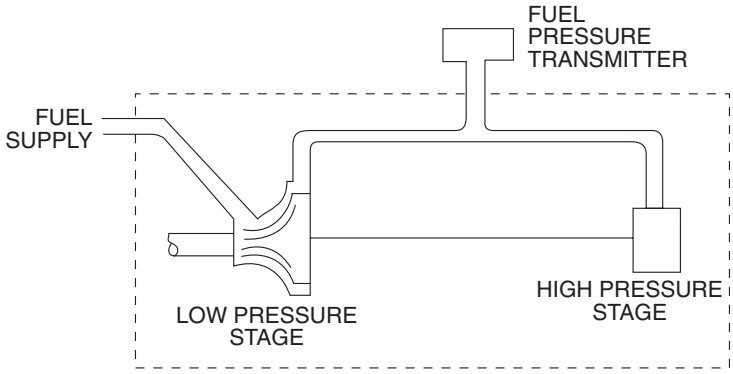
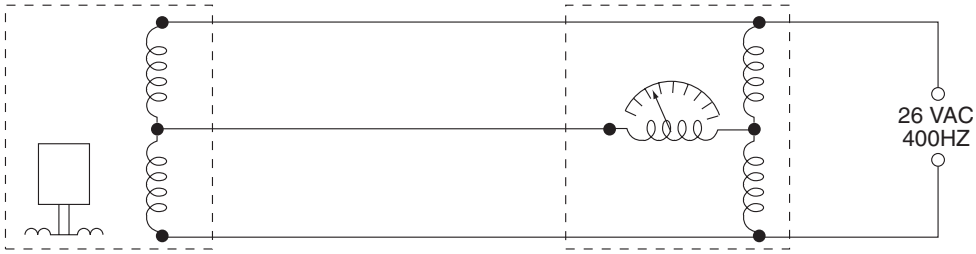
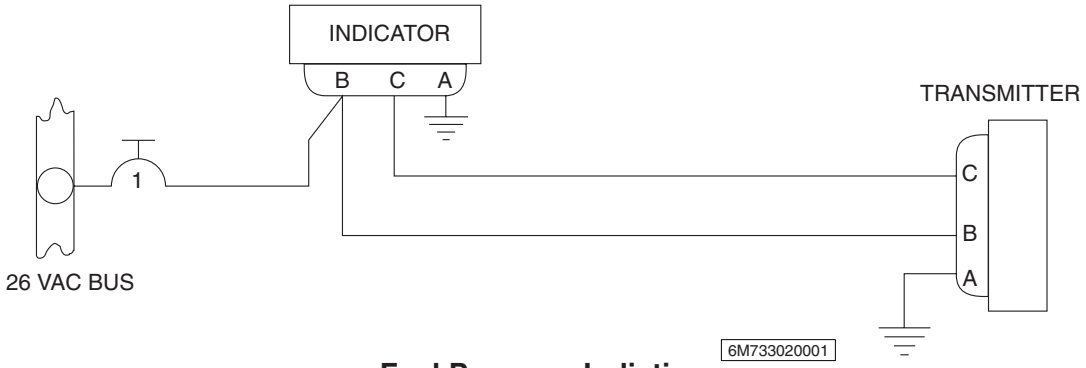
FUEL PRESSURE INDICATING – DESCRIPTION AND OPERATION

1. General

A. Fuel pressure indicating consists of a transmitter installed in the bay immediately behind the engine firewall and an indicator installed in the cockpit. As shown in Figure 1A, the fuel pressure transmitter senses the pressure within the two stages of the engine driven fuel pump. Before starting the engine, the fuel pressure indicator displays boost pump pressure. After the engine is running, the pressure displayed on the indicator is the sum of the boost pump pressure and the engine driven pump interstage pressure. If the boost pump is turned OFF with the engine operating, the fuel pressure indicator still displays the engine driven pump interstage pressure.







Fuel Pressure Indicting
Figure 1

FUEL PRESSURE INDICATING – FAULT ISOLATION

1. Troubleshooting – Fuel Pressure Indicating

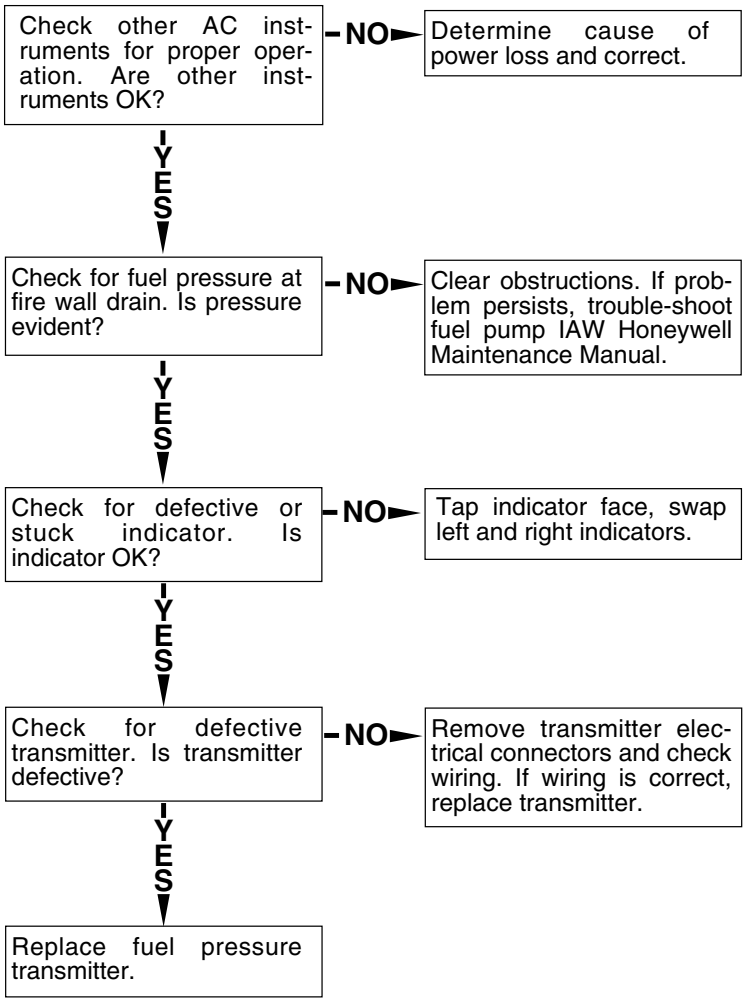
PROBLEM:

NO RESPONSE ON FUEL PRESSURE INDICATOR

PREREQUISITES:

Both engines operating.

START:



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END

FUEL PRESSURE INDICATING – MAINTENANCE PRACTICES

1. Removal/Installation – Fuel Pressure Indicating

A. Removal – Fuel Pressure Transmitter

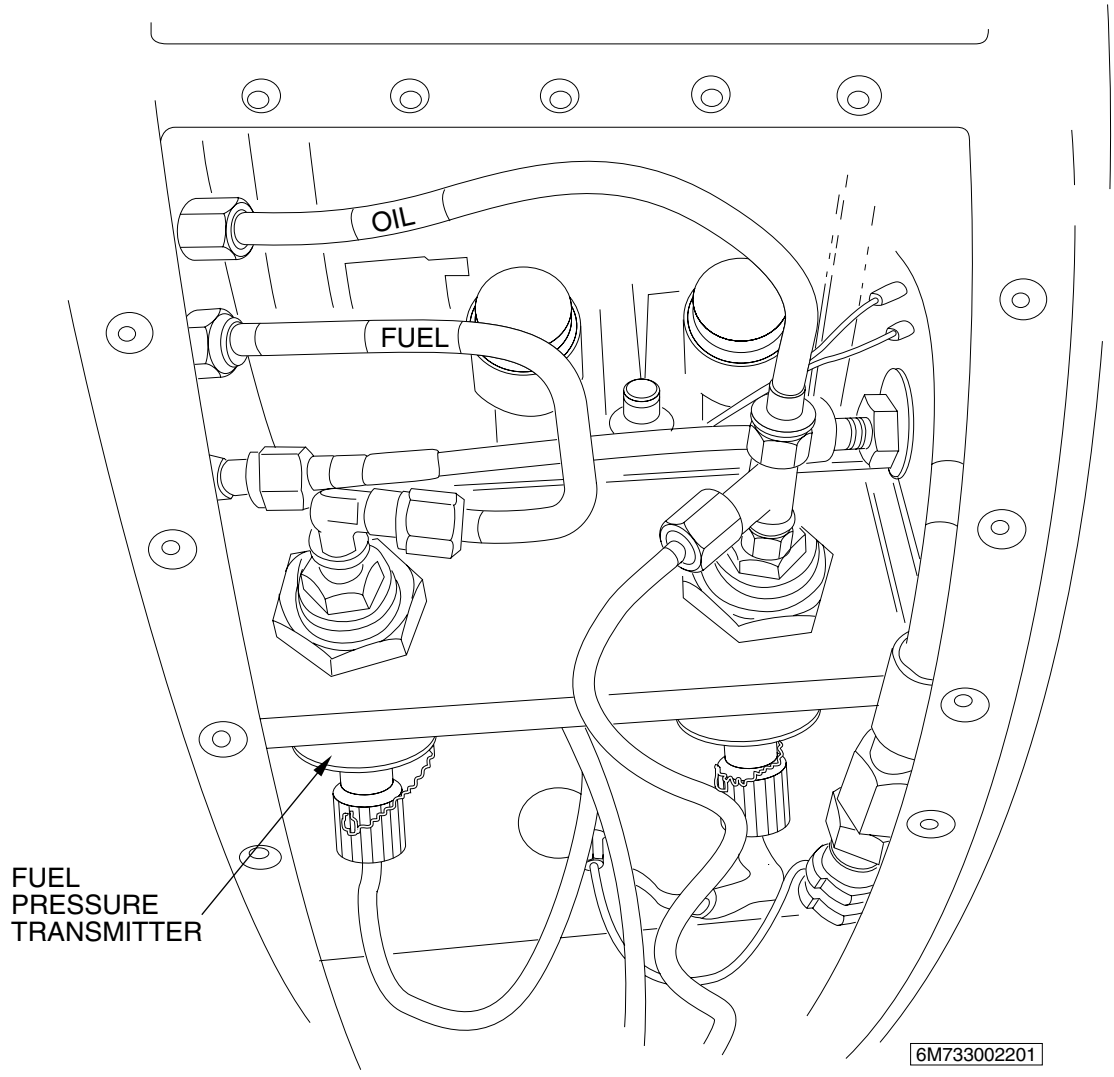
(1) Ensure that battery switch is OFF.

(2) Remove cowling to transmitter access bay.

(3) Disconnect fuel pressure transmitter electrical connector

(4) Disconnect tube assembly connector nut (P/N AN924-4D), and O-Ring (P/N AN6290-4) from transmitter.

(5) Disconnect retaining nut (P/N AN924-12D) and washer (P/N AN901-12A) securing transmitter to bracket and lift out.



Fuel Pressure Transmitter
Figure 201

B. Installation – Fuel Pressure Transmitter

(1) Position transmitter to bracket and secure with washer and nut.

(2) Connect tube assembly connector nut and O-ring to transmitter.

(3) Connect electrical connector to transmitter.

(4) Install cowling.

C. Removal – Fuel Pressure Indicator

(1) Remove post light from instrument panel.

(2) Loosen holding screw (lower right one) securing indicator to instrument panel.

(3) Pull indicator straight out.

(4) Disconnect electrical connector from indicator.

D. Installation – Fuel Pressure Indicator

(1) Connect electrical connector to indicator.

(2) Position indicator to instrument panel.

(3) Secure indicator to panel with holding screw.

(4) Install post light to instrument panel.

(5) Perform functional check of indicator.

2. Inspection/Check – Fuel Pressure Indicating

A. Inspect all components of the fuel pressure system for security, leakage and condition.

MAINTENANCE MANUAL

CHAPTER 74 – IGNITION

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74-CONTENTS	-	1	APR 01/05	74-00-00	MP	201	APR 01/04
74-CONTENTS	-	2	APR 01/04	74-00-00	MP	202	APR 01/04

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MAINTENANCE MANUAL

CHAPTER 74 – IGNITION

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IGNITION – DESCRIPTION AND OPERATION

1. General

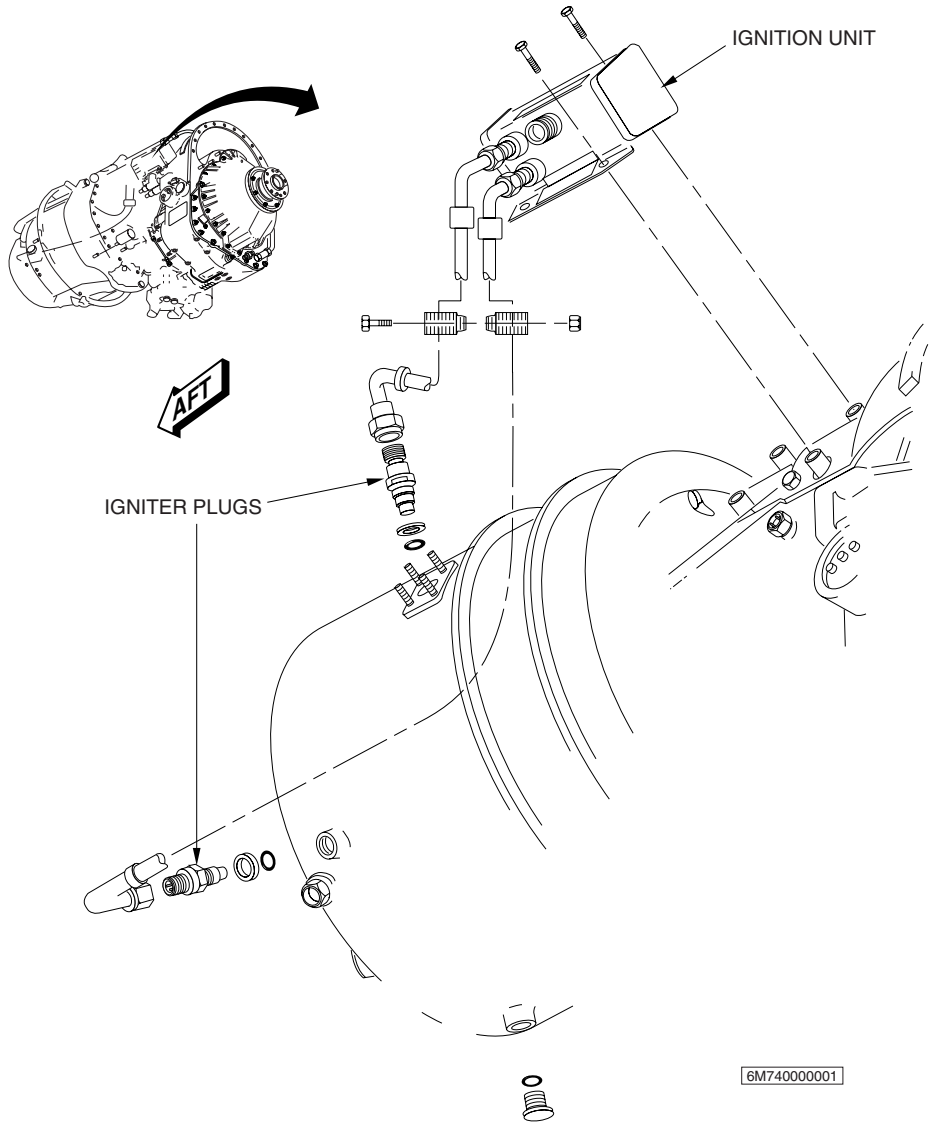
A. The ignition system operates on 24 volts DC power. It consists of an ignition unit, ignition leads, igniter plugs, and speed switch (see Figures 1 and 2).

2. Ignition Unit

A. The ignition unit, mounted on the upper right side of the input housing assembly is a radio noise suppressed, high voltage, capacitor discharge, intermittent sparking type unit. The ignition unit utilizes a DC input to provide approximately 18,000 volts output to the igniter plugs.

3. Ignition Lead

A. The ignition leads are shielded high tension flexible leads connecting the ignition unit and igniter plugs.



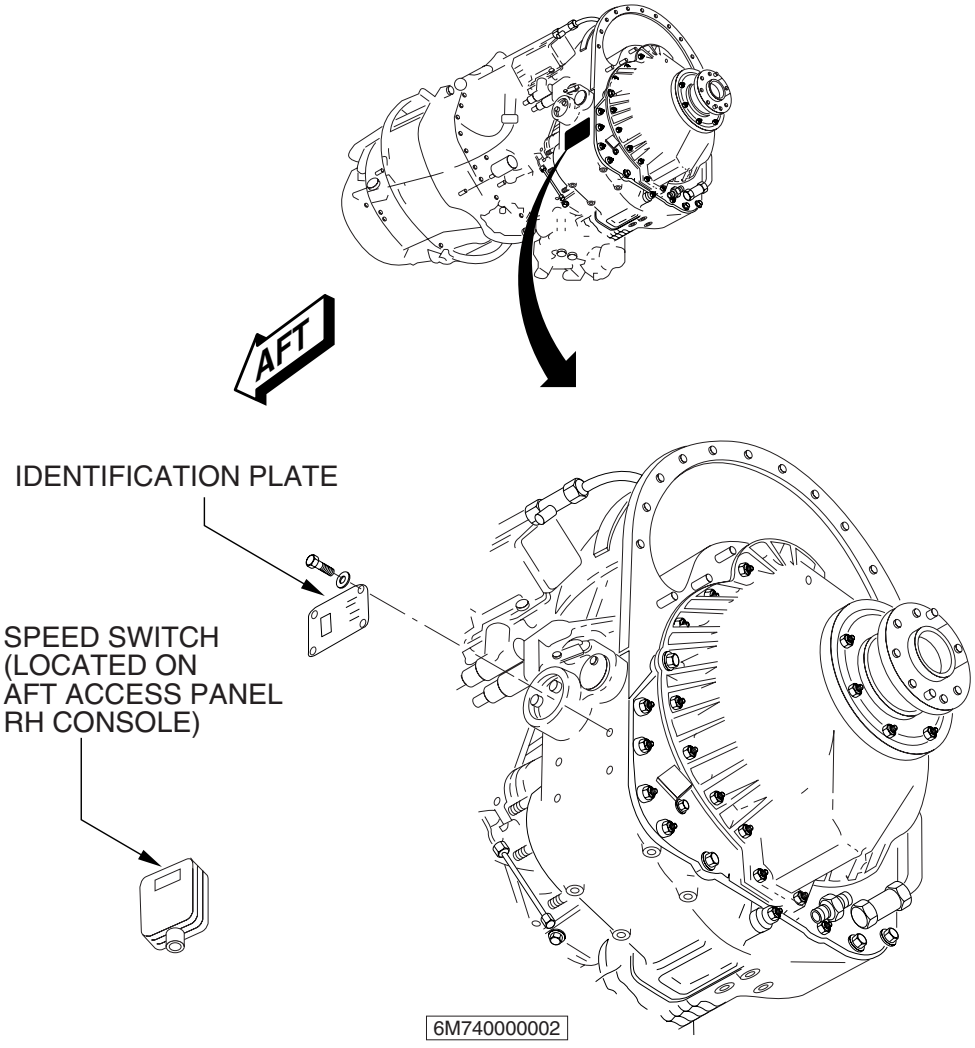
Ignition Components
Figure 1

4. Igniter Plugs

A. The igniter plugs are the air gap type with center electrode and ground electrode. They are mounted in the plenum assembly at approximately the four and eleven o'clock positions.

5. Speed Switch

A. The speed switch consists of a dual relay activated by integral solid state circuitry. Control signals received from the tachometer-generator pass through a signal conditioner which is integral to the speed switch. This determines the open or closed condition of the two relay sections of the dual relay which in turn, determines the operational condition of the ignition system, the fuel-on portion of the fuel solenoid valve, and the starter-generator. The normally open light off relay is activated to close at approximately ten percent engine speed and apply electrical power to the ignition system and the fuel-on portion of the fuel solenoid valve. The normally closed starter cutout relay is activated to open at approximately 50 percent speed to cut power from the aircraft start relay.



Nameplate and Speed Switch
Figure 2

IGNITION – MAINTENANCE PRACTICES

1. Removal/Installation – Ignition

A. Removal – Ignition Unit

(1) Remove ignition leads from ignition unit.

(2) Remove bolts securing ignition unit and remove ignition unit from mounting bosses.

B. Installation – Ignition Unit

(1) Apply a coat of thread lubricant to bolts used to secure ignition unit to engine; apply a coat of general purpose grease under head of bolts and to flanges of ignition unit and mounting bosses.

(2) Secure ignition unit to engine with bolts. Torque bolts 25 inch-pounds (2.82 N·m).

(3) Connect ignition leads to ignition unit.

C. Removal – Igniter Plugs

(1) Disconnect ignition leads from igniter plugs.

(2) Remove igniter plugs from engine.

D. Installation – Igniter Plugs

(1) Install washers and new gaskets onto igniter plug.

(2) install igniter plugs. Torque plugs 200 to 250 inch-pounds (22.60 to 28.25 N·m).

(3) Feed safety wire into igniter plug lead B nut and connect igniter plug leads. Torque plug leads 70 inch-pounds (7.91 N·m).

(4) Complete safety wire twist and feed thru #50 hole pre-drilled in heat shield.

E. Removal – Speed switch

(1) Ensure that all electrical power to the aircraft is OFF.

(2) Remove the copilot's seat IAW Chapter 25.

(3) Remove the aft access panel from the RH console.

(4) Disconnect the electrical plug from the speed switch to be removed.

(5) Remove the speed switch mounting bolts and remove the speed switch.

F. Installation – Speed switch

(1) Place the speed switch on the bracket and install the mounting bolts.

(2) Connect and safety the electrical plug.

(3) Install access panel.

(4) Replace the copilot's seat IAW Chapter 25.

2. Adjustment/Test – Ignition

NOTE: When the ignition system is suspected of malfunction as evidenced by improper engine start, perform the following tests.

A. Perform ignition adjustment/test as follows:

(1) Test ignition leads for continuity.

(2) Connect igniter plugs to a satisfactory ignition unit. Apply 28 volts DC to ignition unit and check plugs for spark rate of two sparks per second. Replace plugs which fail test procedures.

(3) Connect two satisfactory igniter plugs to ignition unit, apply 28 volts DC current to unit and check for spark rate of two sparks per second.

(4) Replace ignition unit which fails test.

3. Inspection/Check – Ignition

NOTE: Any time that maintenance is performed on any part of the engine ignition system, inspect the remaining components for serviceability and take corrective action as required.

A. Perform ignition inspection/check as follows:

(1) Visually inspect igniter leads for cracks, burned or frayed covering, and other damage. Replace damaged or defective leads.

(2) Visually inspect ceramic parts of igniter plugs for cracks or chipping. Inspect electrodes for excessive burning or damage. Electrodes shall protrude 0.040 inch maximum. Replace defective or damaged plugs.

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CHAPTER 75 – AIR

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AIR – DESCRIPTION AND OPERATION

1. General

A. This chapter concerns those components of the TPE331 turboprop engine used to eliminate and prevent the formation of ice within the engine air inlet system by engine heated bleed air. For information, description and operation, and maintenance practices pertaining to this system, refer to the following listed Garrett-AiResearch Maintenance Manual, Chapter 75.

B. For information pertaining to elimination and prevention of ice formation to the powerplant cowling and its anti-icing system components, refer to Chapter 30 in the Maintenance Manual.

Maintenance Manual: TPE331-3U-303G
Engine Part Number: 894040-1/2
Applicable Aircraft: Merlin III model SA226-T
Merlin IV model SA226-AT
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valves.

Maintenance Manual: TPE331-3UW-303G
Engine Part Number: 895880-1/-2
Applicable Aircraft: Metro model SA226-TC
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valves.
2. Alcohol/Water Injection System.

Maintenance Manual: TPE331-3U-304G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III model SA226-T
Merlin IV model SA226-AT
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valve.

Maintenance Manual: TPE331-3UW-304G
Engine Part Number: 3102550-1
Applicable Aircraft: Metro model SA226-TC
Identifying Characteristic:
*1. Engine inlet anti-ice system with dual anti-ice valve.
2. Alcohol/Water Injection System.

*See Fairchild Aircraft Service Bulletin 30-011 for conversion procedures between TPE331-3U-303G and TPE331-3U-304G engines.

Maintenance Manual: TPE331-10U-501G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III (B) model SA226-T(B)

Maintenance Manual: TPE331-10UW-501G
Engine Part Number: 3102550-1
Applicable Aircraft: Merlin III (B) model SA226-T(B)
Identifying Characteristic:
*1. Alcohol/Water Injection System.

*See Fairchild Aircraft Service Bulletin 30-011 for conversion procedures between TPE331-3U-303G and TPE331-3U-304G engines.

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CHAPTER 76 – ENGINE CONTROLS

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76-00-00	DO	4	OCT 01/04
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▮ 76-10-10	MP	203	APR 15/11
76-10-10	MP	204	OCT 01/04
76-10-10	MP	205	OCT 01/04
▮ 76-10-10	MP	206	APR 15/11
76-10-10	MP	207	OCT 01/04
76-10-20	DO	1	OCT 01/04
76-10-20	MP	201	OCT 01/04
76-10-20	MP	202	OCT 01/04
76-10-20	MP	203	OCT 01/04
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76-10-20	MP	205	OCT 01/04
76-10-20	MP	206	OCT 01/04
76-10-20	MP	207	OCT 01/04
76-10-20	MP	208	OCT 01/04
76-10-20	MP	209	OCT 01/04
76-20-10	DO	1	OCT 01/04
76-20-10	MP	201	OCT 01/04
76-20-10	MP	202	OCT 01/04
76-20-10	MP	203	OCT 01/04
76-20-10	MP	204	OCT 01/04
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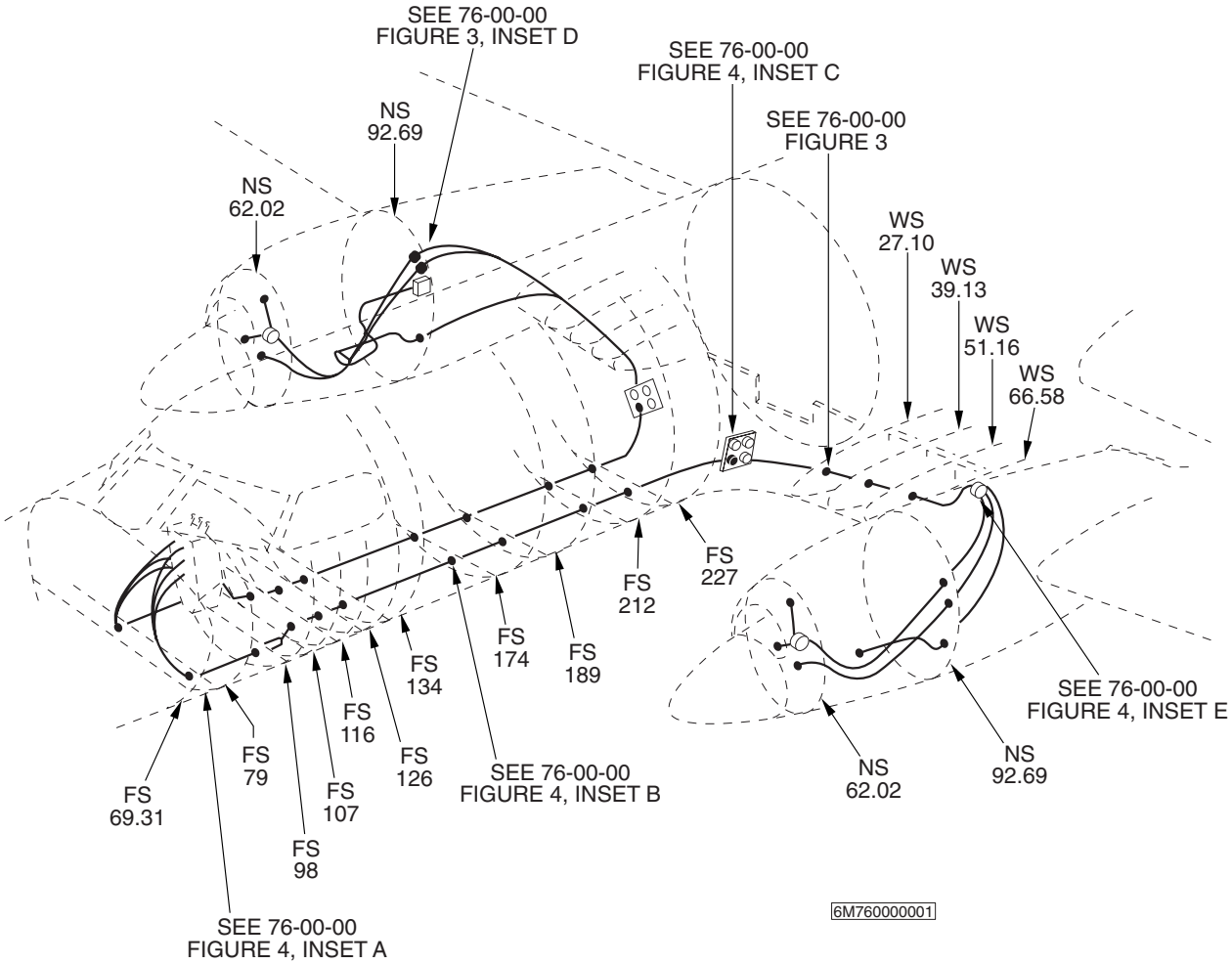
ENGINE POWER, RPM, AND EMERGENCY SHUTDOWN CONTROL CABLES – DESCRIPTION AND OPERATION

1. General

A. Each control cable used for engine power, RPM, and emergency shutdown is a forced transmission device made up of a flex cable within a flex casing containing a nylon polyethylene liner jacket. Each cable is used with rod-end assemblies that attach to the controls at the cockpit pedestal and in the engine compartment where they attach to engine control units. The cables require few anchor points enroute and are relatively unaffected by bends provided that minimum bend radii are not exceeded.

B. SA226 series aircraft T201-248, AT001-023, and TC201-211 were manufactured with a one section cable for each engine control.

C. SA226 series aircraft T249-383, AT024-069, and TC212-384 were manufactured with a two section cable for each engine control utilizing a quick disconnect to join the sections. When installed, the quick disconnect for each control is located in the wing leading edge just outboard of the wing root rib. The control sections are not interchangeable with any other control section, except of like part numbers.



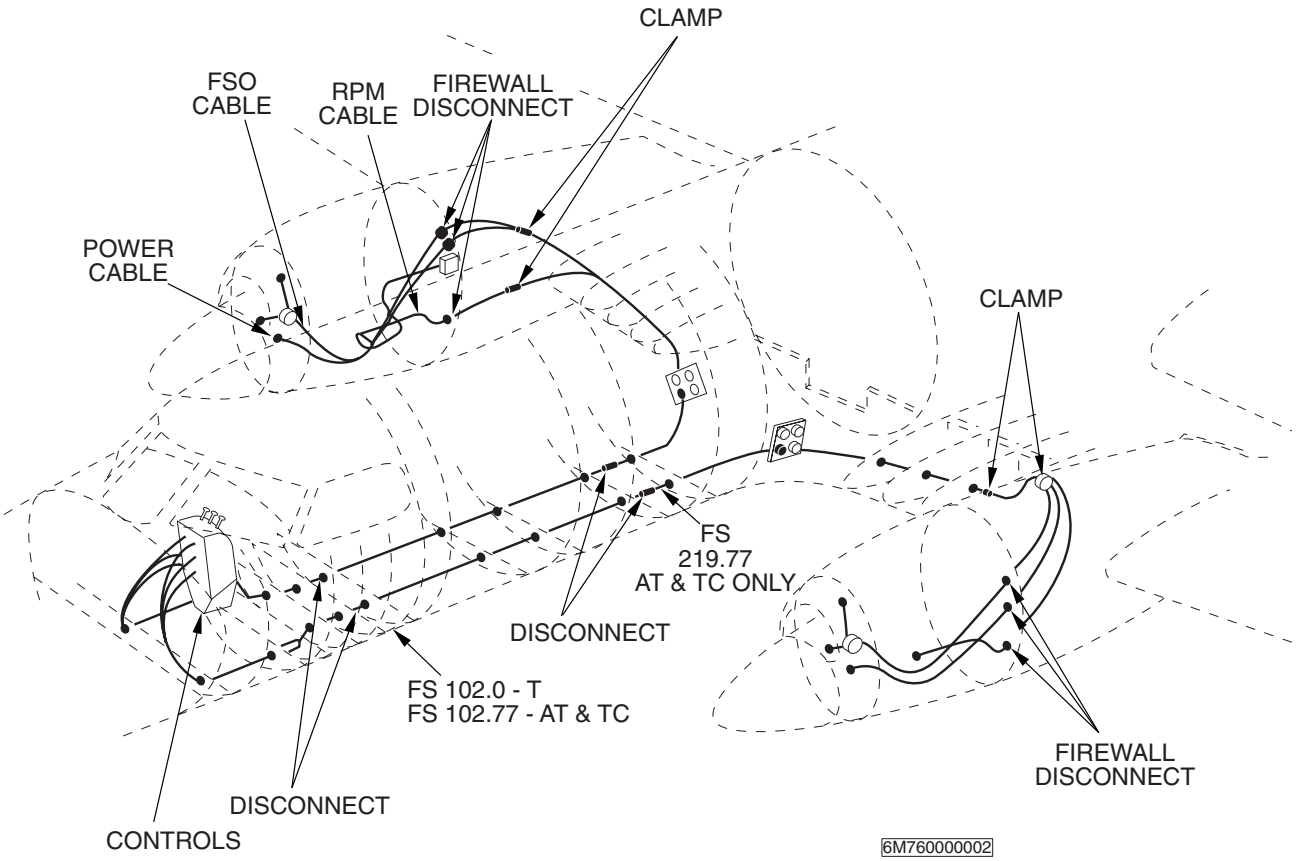
Engine Controls
Figure 1

ENGINE POWER, RPM, AND EMERGENCY SHUTDOWN CONTROL CABLES – DESCRIPTION AND OPERATION

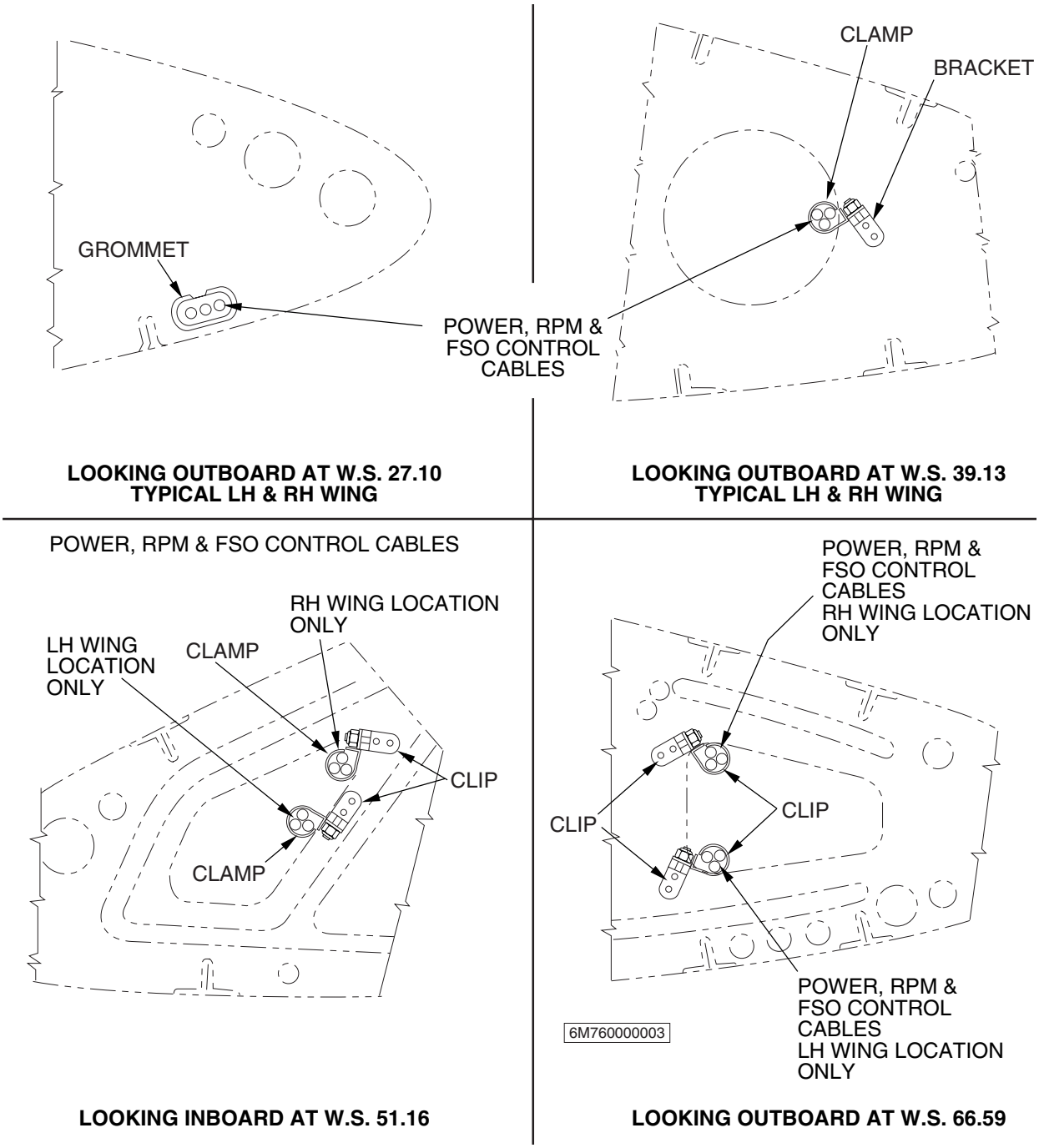
D. On SA226 series aircraft T384-999, three control cable segments connect between each pedestal control to the engine unit. The pedestal cable extends from the pedestal control and connects to the wing fuselage cable at F.S. 102.0, located just below the pilot and copilot floorboards. From there, the wing/fuselage cable is routed through the fuselage into and through the wing, into the engine nacelle, and connects through the back of the engine firewall at N.S. 92.69. The engine cable segment extends from the front of the firewall routed through and connected to an engine mounted bracket, and then connects to the engine unit.

E. On SA226 series aircraft, AT070-999, and TC385-999, because of the longer fuselage, a fourth cable (fuselage cable) connects from the pedestal cable at F.S. 102.77 beneath the pilot and copilot floorboard, then connects to the wing cable segment at F.S. 219.77, see Figure 2.

F. Each end of the cable conduit has a threaded rod for anchoring at terminal points. Enroute support serves only to confine the cable to the desired routing path and prevent shifting. Because all moving parts are confined within the conduit, no anti-fouling clearance is necessary, and the cable requires no lubrication or maintenance. The action element has threaded rods for connection and adjustment at both the controlling and controlled units. Because the cable is flexible and has few anchor points, the cable can be moved or shifted to allow access to, or installation of, other components without disturbing the rigging.



Engine Controls
Figure 2



LOOKING OUTBOARD AT W.S. 27.10 TYPICAL LH & RH WING

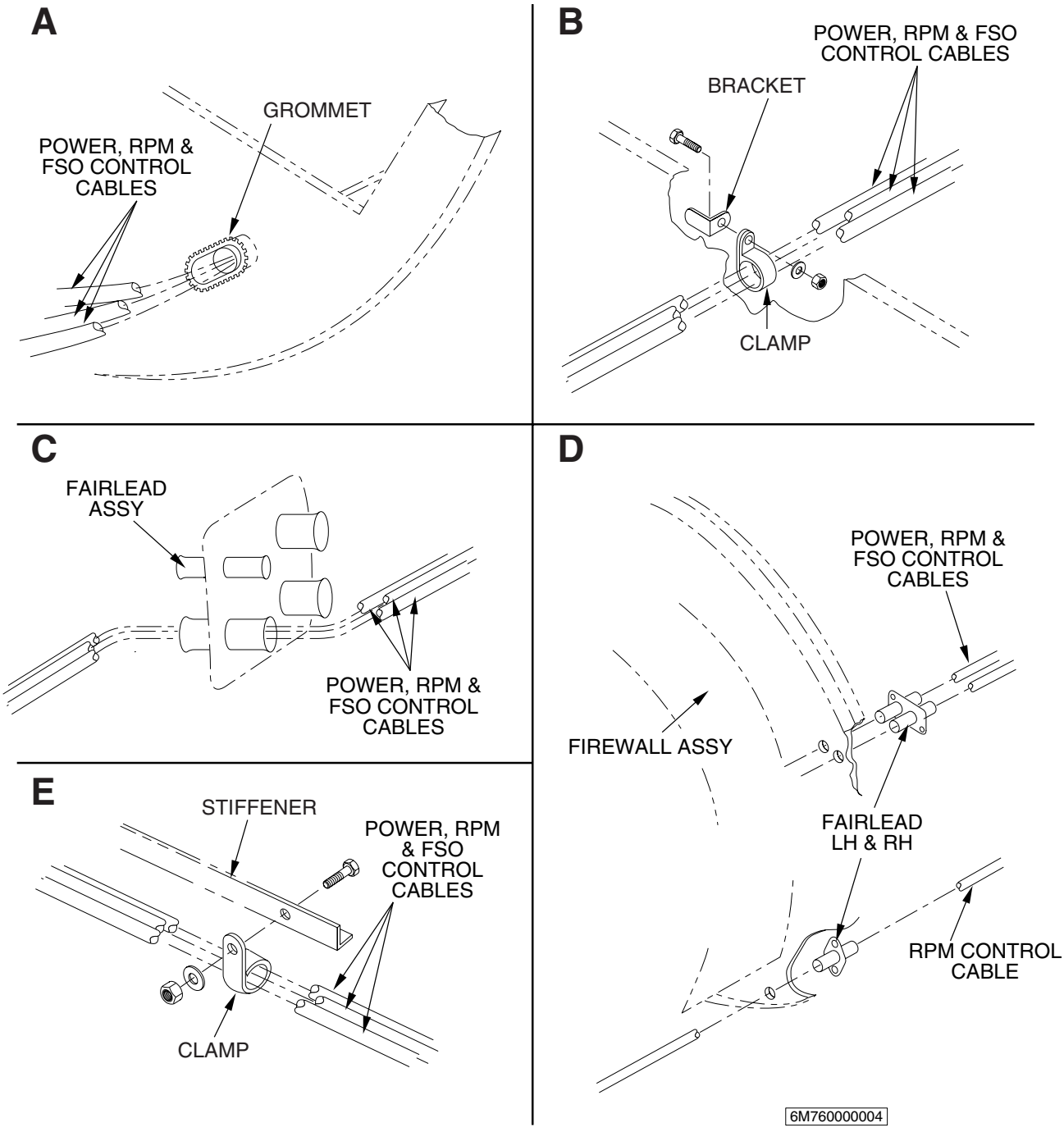
LOOKING OUTBOARD AT W.S. 39.13 TYPICAL LH & RH WING

LOOKING INBOARD AT W.S. 51.16

LOOKING OUTBOARD AT W.S. 66.59

SEE FIGURE 1 FOR W.S. LOCATIONS

Engine Control Attach Fittings
Figure 3



SEE FIGURE 1 FOR INSET LOCATIONS

Engine Control Attach Fittings
Figure 4

ENGINE POWER, RPM, AND EMERGENCY SHUTDOWN CONTROL CABLES – MAINTENANCE PRACTICES

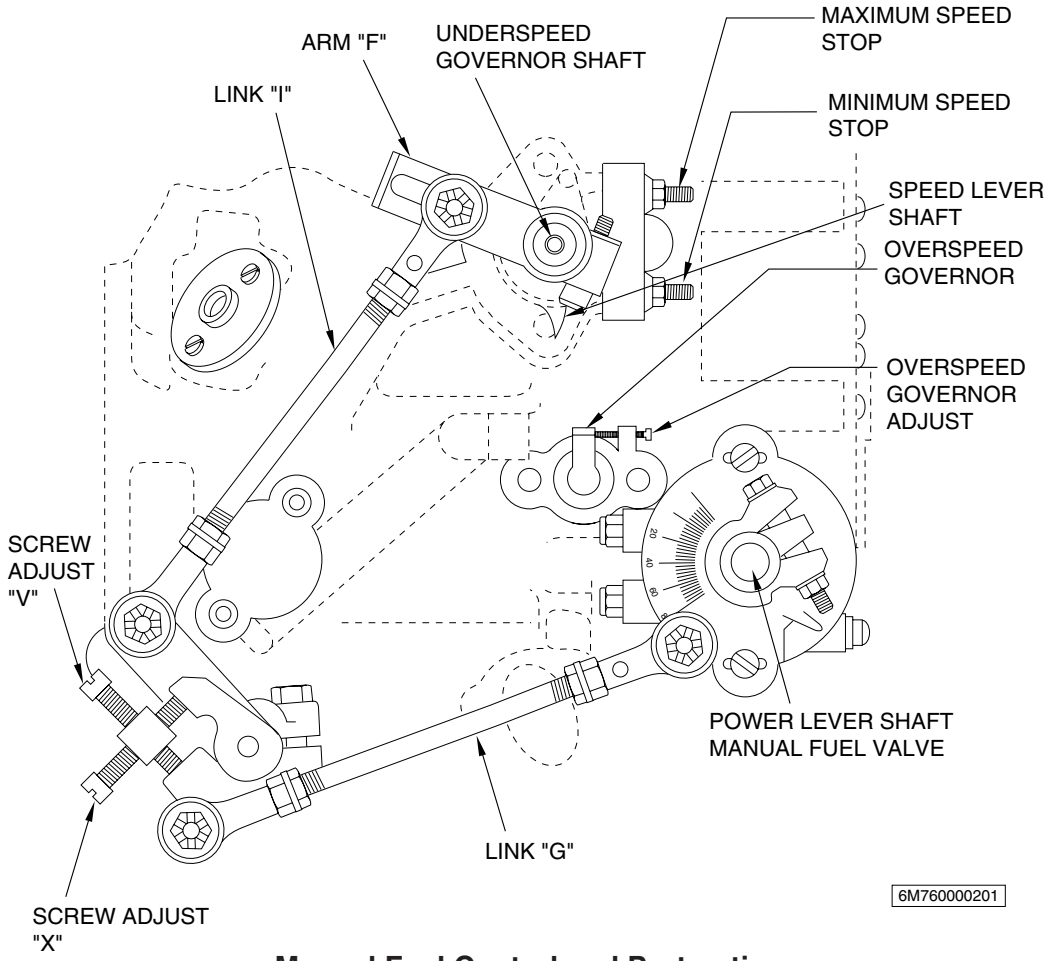
1. Adjustment/Test – Engine Power, RPM, and Emergency Shutdown Control Cables

A. Engine rigging Check

(1) The following practices include the engine rigging checkout and rigging check after control cables are installed for the Honeywell TPE331-3U-303G.304G and TPE331-10U-501G engines. For procedures concerning propeller rigging check, see Maintenance Manual, Chapter 61. For procedures involving more detailed rigging adjustment and test procedures, see PowerPlant Maintenance Manual, Chapter 71 and/or see the Honeywell Engine Maintenance Manual.

B. Engine Rigging Check After cable Installation

(1) Place the cockpit power lever in the maximum power position. Move the cockpit RPM (speed) lever to the high RPM position, and observe that the underspeed fuel governor shaft is making positive contact with the adjustable maximum RPM stop (see Figure 201). The propeller shaft lever should be approximately 1/8 inch from the maximum speed stop.

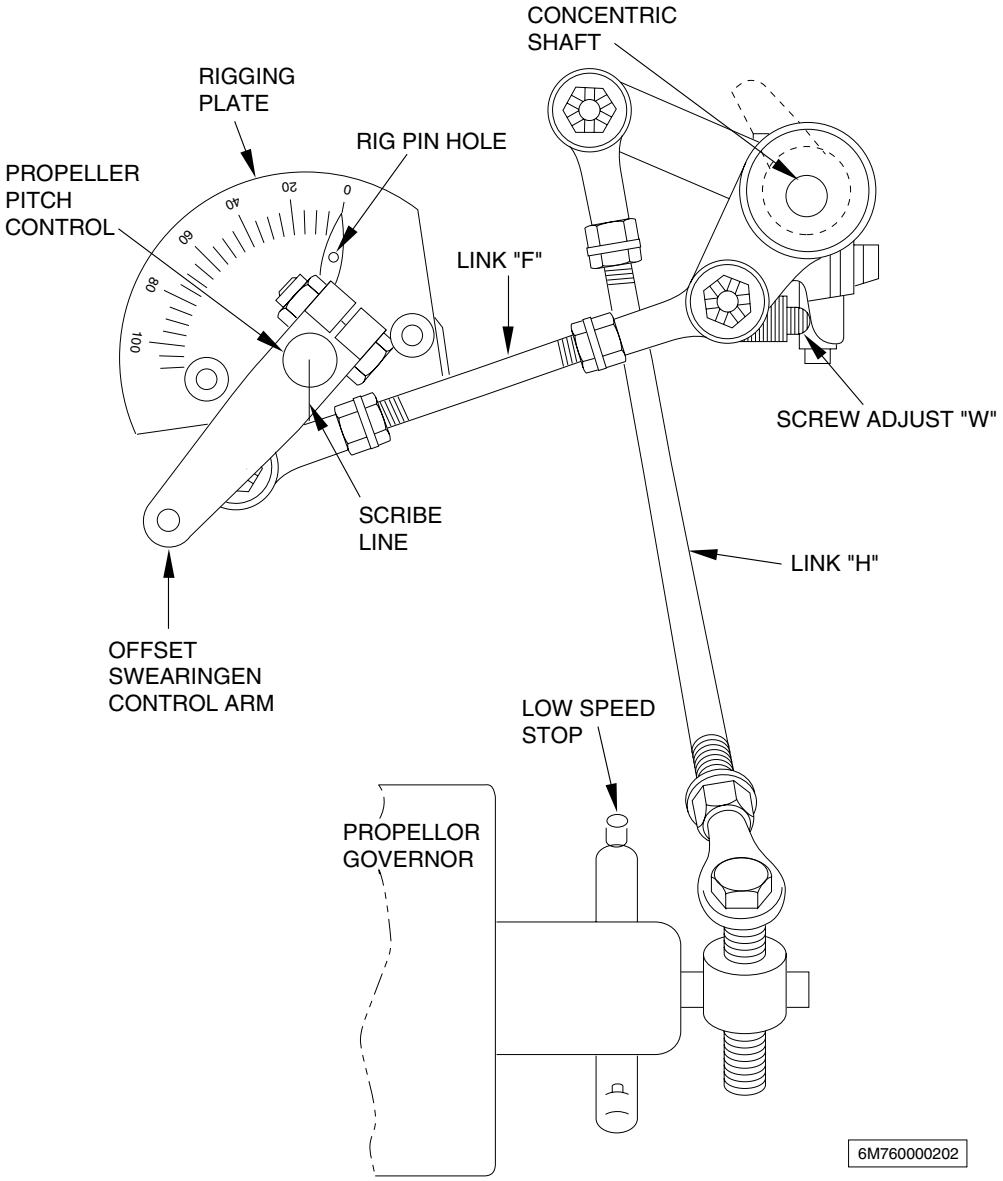


Manual Fuel Control and Protraction
Figure 201

(2) Move the cockpit power lever to full reverse and the cockpit RPM lever to low position. Observe that the propeller governor shaft is firmly contacting the maximum speed stop.

(3) With the cockpit engine RPM lever at the low position, move the power lever until the propeller pitch control pointer is between 35° and 38°. The set screw on the upper part of the concentric shaft (screw W on Figure 202) should just make contact with the spring tab. Return the power lever to the maximum position.

(4) With the cockpit RPM lever in the low position, observe that both the propeller governor and the underspeed fuel governor shafts are contacting their adjustable minimum stops.



Rigging Check
Figure 202

CAUTION: DO NOT ATTEMPT TO ADJUST THE STOPS OR THE ENGINE LINKAGE AT THIS TIME. HOWEVER IF THE AIRCRAFT RIGGING, LINKAGE, OR STOPS IN THE COCKPIT QUADRANT PREVENT REALIZATION OF THE DESCRIBED CONDITIONS, CORRECT THE PROBLEM PRIOR TO CONTINUING. SEE HONEYWELL MAINTENANCE MANUAL.

(5) Move the cockpit power lever to the full reverse position, and verify that the rigging pin can be inserted through the hole in the arm of the propeller pitch control. Make sure that the pointer of the propeller pitch control is at zero degrees and that the pointer on the manual fuel valve is within three degrees of the minimum stop.

(6) Advance the cockpit power lever to the maximum power position, and verify that the pointers on the propeller pitch control is at 100°, minus one degree, or plus three degrees and the manual fuel valve pointer is within three degrees of its maximum stop.

(7) Move the cockpit power lever forward then back rapidly to the flight idle position, and make sure that the pointer on the manual fuel valve and propeller pitch control are at 40° on their respective protractors. Make sure that the propeller pitch control can be pinned in this position.

NOTE: Power lever adjustments at flight idle are considered critical. Careful attention should be directed to adjust the aircraft power lever control such that the 40° protractor positions are exactly aligned when the cockpit power lever is placed against the flight idle stop on he control.

(8) Slowly pull the engine stop and feather knobs toward the feather position and observe that the main fuel shutoff valve arm moves 45° before actuation of the propeller feather valve. Continue to pull the engine stop and feather knob to its fullest out position. At this point the feather valve should be at maximum travel.

NOTE: See Propeller Rigging check in Chapter 61 of the M7 Aerospace Maintenance Manual.

(9) If reverse stop block assembly at pedestal has been adjusted, safety IAW MS33540. Verify all jam nuts are safetied before replacing access panels, floorboards, carpets, and seats.

C. Cable Installation Adjustment

(1) Rotate the pitch control shaft and the manual fuel valve shafts to the flight idle position (clockwise). Place the cockpit power lever in flight idle position and connect the power lever control cable to the arm of the propeller pitch control on the engine. Adjust control to make contact with the minimum and maximum stops when operated from full reverse position to maximum power position.

(2) Place the cockpit RPM lever in the low position. Connect the RPM control cable to the underspeed governor on the engine and adjust cable until the propeller governor and the underspeed governor shafts are contacting their respective adjustable minimum stops. When attaching the cable with synchronizer (RH engine only, T201-275, T277-291, AT001-069, AT071-999, and TC201-999) ensure that the rod end assembly and actuator are centered before attaching (see installation procedure, Figure 204, inset A, 76-10-20 in this chapter).

(3) Operate powerplant controls to ensure smooth and positive movement during all modes of engine operation.

2. Inspection/Check – Engine Power, RPM, and Emergency Shutdown Control Cables

A. Inspection/Check – Engine Rigging Checkout

(1) The propellers and powerplants for the SA226 series, T, AT, and TC aircraft are supplied prerigged. The following procedures are intended to be used as an engine rigging checkout procedure. If the checkout shows any discrepancy, see Chapter 71 of the Honeywell or M7 Aerospace Maintenance Manuals for more detailed rigging instructions, testing, and/or component replacement.

(a) Rotate the propeller pitch control linkage rapidly to the flight idle position (counterclockwise) and verify that the pointer on the manual fuel valve and propeller pitch control are at 40°, plus or minus 1/4° on their respective protractors and verify that the propeller pitch control can be pinned at this position.

(b) Rotate the propeller pitch control linkage to the full reverse (clockwise) position and verify that the rigging pin can be inserted and that the pointer on the propeller pitch control is at zero degrees, and that the pointer on the manual fuel valve is within three degrees of the minimum stop. Rotate the propeller pitch control linkage to the maximum power (counterclockwise) position, and verify that the pointer on the propeller pitch control is at 100° (plus three degrees minus one degree) and that the manual fuel valve pointer is within three degrees of its maximum stop.

(c) Verify that the propeller pitch control arm is installed on the propeller pitch control shaft with the scribe line on the arm lined up with the pointer on the control shaft, as shown in Figure 202.

(d) Rotate the propeller pitch control pointer to 37°. Check that the adjustable stop screw on the upper end of the concentric shaft touches the stop.

(e) If required, reorient the fuel shutoff arm as follows:

1 With the scribe line on the shaft pointing straight up, reposition the arm so that the arrow on the arm points to A of the word AUTO printed on the valve body.

2 Install the link between the arm and the feather control system bellcrank.

3 Adjust the link so that when the bellcrank is rotated toward the feather position it will move the arm of the fuel solenoid valve 45° before initial movement of the feathering valve.

4 Safety all nuts, and lockwire as necessary.

POWER LEVER CONTROL CABLE – DESCRIPTION AND OPERATION

1. General

A. Engine power output is controlled by the power levers through control cables to engine mounted units. The power levers, one for each engine, are located next to each other on the cockpit pedestal.

B. The power levers are connected to the propeller pitch control unit by means of the flexible control cables which route from the pedestal, through the fuselage, through the wing, then from the engine nacelle firewall to the engine propeller pitch control unit. The power lever establishes the engine power level at a prescribed engine RPM.

C. Each cable conduit contains threaded rod-ends which provide for control length adjustment at the cockpit lever and the engine compartment. The control conduit is anchored to a bracket immediately forward of the cockpit pedestal and the threaded rod cable end is connected to the respective control lever through a rod-end with a clevis. The power control cable incorporates a coil spring and retainer which requires additional pressure rearward in order to move the control into the reverse pitch range.

D. At the engine end of the power cable conduit, the conduit is anchored to an engine bracket installed below the fuel flow divider. A rod-end on the end of the cable is bolted to the arm of the propeller pitch control unit shaft.

E. The power levers, located in the left upper section of the cockpit pedestal, are moved forward or aft and the pedestal quadrant is marked from front to rear to indicate the positions HIGH, FLT IDLE, GRN IDLE, and REVERSE. A flight control surface locking lever, located between the power levers on the pedestal, incorporates a stop which prevents the power lever from being advanced beyond the flight idle position when the control surface locking lever is in the locked position.

F. The power lever also incorporates a spring loaded guard which engages a gate on the quadrant at the flight idle position to prevent movement of the lever into ground operating (beta) range while in flight. By raising the guard, the lever can be moved into the ground idle or reverse positions. When entering the reverse range, additional rearward pressure is necessary to overcome the coil spring tension on the control cable.

G. Rods attached to the lower end of each power lever move forward and contact the landing gear warning horn switch when the lever is moved at or behind the flight idle gate.

POWER LEVER CONTROL CABLE – MAINTENANCE PRACTICES

1. Removal/Installation – Power Lever Control Cable

CAUTION: WHEN TIGHTENING OR LOOSENING CONTROL CABLE RETAINING NUTS OR JAM NUTS, USE A WRENCH TO HOLD CABLE HOUSING AND PREVENT TWISTING CABLE.

A. Removal – Engine Power Control Cable

(1) Remove necessary engine cowling from appropriate engine IAW Chapter 71.

(2) Remove seats, carpets IAW Chapter 25, and necessary floorboards IAW Chapter 53 to gain access to the control cable segment for those cables within the aircraft fuselage.

NOTE: Depending on model series, cable disconnects are located at different points within the aircraft, see Figures 1 and 2, 76-00-00.

(3) To gain access to wing cable (from engine firewall through wing to fuselage disconnect) segment, remove the access panels from underside of wing between the fuselage and nacelle. Remove leading edge portion of fuselage to wing fairing.

(4) Loosen the applicable retaining clamps along the cable route in the fuselage and the wing, see Figures 3 and 4, Section 76-00-00.

(5) Disconnect the action elements of the power control cable as follows:

(a) For all SA226 aircraft, refer to Figure 201 to disconnect power control cable at quadrant assembly. Remove items 2-15.

(b) For SA226 aircraft T201-383, AT001-069, AT071-999, and TC201-384, refer to Figure 202, inset A to disconnect power control cable from engine. Remove items 2-9.

(c) For SA226 aircraft T384-999, AT070, and TC385-999, refer to Figure 202, inset B to disconnect power control cables (item 13) from engine. Remove items 14-18.

(d) For SA226 aircraft T249-383, AT024-069, and TC212-384 (two piece segments). also disconnect at the quick disconnect between wing station 27 and 39 in each leading edge.

(e) For SA226 aircraft T384-999 (three piece segments), disconnect at the quick disconnect at F.S. 102.0 (below pilot and copilot floorboard) and at the engine firewall at N.S. 92.69.

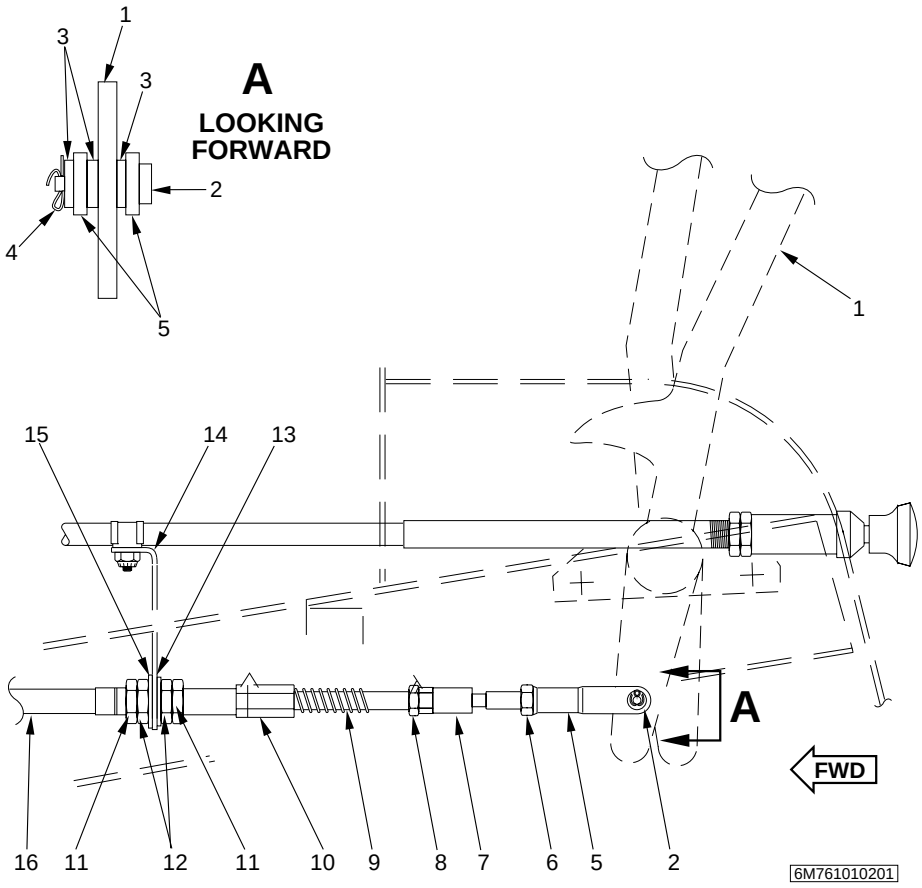
(f) For SA226 aircraft AT070-999 and TC384-999 (four piece segments), disconnect at quick disconnect at F.S. 102.77 (below pilot and copilot floorboard). at F.S. 219.77 where the fuselage mates with the wing, and at the engine firewall at N.S. 92.69.

(6) Loosen the appropriate split fairlead at the nacelle firewall (single and two segment cables only).

(7) Remove the sealant from the fairlead at the fuselage.

(8) Attach a spool of safety wire to the cable being removed at the opposite end from which the cable is to be withdrawn. The wire will serve as a guide during cable installation or replacement.

1. RPM LEVER
2. PIN
3. WASHER
4. COTTER PIN
5. CLEVIS ROD END
6. JAM NUT
7. BALL JOINT
8. JAM NUT
9. SPRING
10. REVERSE STOP BLOCK ASSY
11. JAM NUT
12. RETAINING NUT
13. WASHER
14. CABLE MOUNT
15. BUSHING
16. POWER CONTROL CABLE



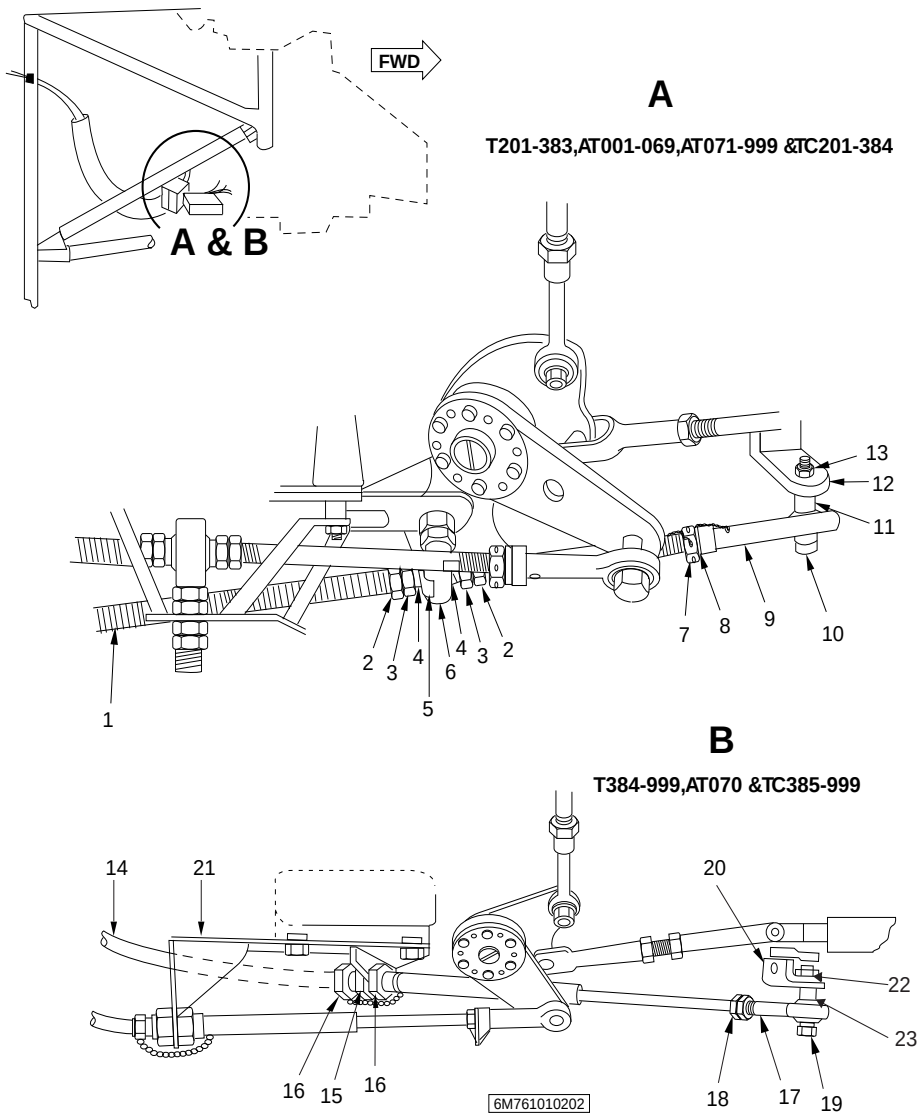
Power Control Cable Pedestal Connection
Figure 201

(9) Remove the cable through the inside of the aircraft drawing it through the retaining clamps and, at the same time, draw the lead wire along the route:

(a) For two piece cables, withdraw the wing/nacelle portion from the engine end. Remove the fuselage portion by first pulling the wing end toward the fuselage until it clears the wing root rib and then pulling the cable from the fuselage at the wing end.

(b) For multiple piece cables, withdraw pedestal and fuselage cable segment through inside of aircraft, and at the same time, draw the guide wire along the route.

(c) For wing cables, withdraw the cable segment from the engine firewall end, withdrawing it through the loosening retaining clamp(s).



Power Control Cable Engine Connection
Figure 202

(10) If cable is to be reinstalled, avoid coiling it when removed. Gather cable into a loose figure eight, no smaller than one foot by three feet, and secured at the center cross over.

(11) Detach guidewire.

B. Installation – Engine Power Control Cable

CAUTION: DO NOT MIX CABLE SEGMENTS MANUFACTURED BY DIFFERENT VENDORS. CONNECTORS VARY FROM ONE MANUFACTURER TO ANOTHER. ANY ATTEMPT TO INTERMIX BRANDS MAY DAMAGE CABLE SEGMENTS.

WHEN TIGHTENING OR LOOSENING CONTROL CABLE RETAINING NUTS OR JAM NUTS, USE A WRENCH TO HOLD CABLE HOUSING AND PREVENT TWISTING OF CABLE.

(1) Unpack new cable carefully, loop by loop. Straighten it by forming a hump near one end and walking it out the entire length of cable.

(2) Check the action element for smooth and easy movement. If any unusual load or binding is noted, reject cable.

(3) If the action element moves satisfactorily, attach at the guidewire (spool of safety wire) to end of cable and draw through cable route.

(4) Avoid twisting the cable. Check the action element at the convenient intervals during installation and before securing the cable at its terminus to ensure no binding has occurred. When installing two piece cables, route cables from cockpit and nacelle ends to the leading edge area between wing stations 27 and 39.

(5) Install pedestal end of power cable as follows:

(a) Refer to Figure 201.

(b) Install forward jam nut (Figure 201, item 11), and retaining nut (item 12).

(c) Feed cable through pedestal mount, (item 14). Secure with retaining nut (item 12), and jam nut (item 11) on aft side of mount.

(d) Install reverse stop block assembly (item 10), and spring (item 9). Ensure that stop block is completely screwed onto end of cable. Do not safety until rigging check has been conducted.

(e) Trim length of spring if required (1.80 inches minimum length), so that the power lever will spring out of beta mode towards ground idle when it is released from full reverse.

(f) Install jam nut (item 8), and swivel fitting (item 7). Safety IAW MS33540.

(g) Install clevis (item 5) and secure with pin (item 2).

NOTE: Perform a functional check of the control cable by moving power lever through full travel before making other cable connections in fuselage, wing, or engine nacelle. No binding or ratcheting is acceptable.

(6) Install intermediate connections of two-piece, or multiple-piece cable as follows:

(a) Mate the bayonet couplings.

CAUTION: WHEN TIGHTENING OR LOOSENING CONTROL CABLE RETAINING NUTS OR JAM NUTS, USE A WRENCH TO HOLD CABLE HOUSING AND PREVENT TWISTING CABLE.

WHEN SECURING CABLES AND SEALING FAIRLEADS, VERIFY THAT POWER CONTROL LEVER CONTINUES TO MOVE FREELY.

(b) Slide outer housing elements together, and tighten retaining nut.

(c) Tighten jam nut and safety IAW MS33540

(d) Move power lever and verify that cable movement is free at engine end of joined cables.

(e) Secure applicable clamps.

(f) Install and secure the split fairlead at the firewall on applicable aircraft. Seal fairlead with PR812 high temperature sealant (or equivalent).

(g) Seal cables at fuselage fairlead with PR1422B-2 sealant (or equivalent).

(h) Install tie-wraps and/or chaff covering around cables at applicable areas where cables group together.

(7) Install engine end of power control cable for aircraft T201-383, AT001-069, AT071-999, and TC201-384 as follows:

(a) Refer to Figure 202A.

(b) Apply locktite 242 (Locktite Corp. – Newington, Conn, 06111) onto threads of portion of the power lever control cable (item 1) that slides over the bayonet, then threads into housing of each control cable assembly.

(c) Tighten snug by using 3/8 inch open-end wrench on flat surface of control cable portion which slides over bayonet, and use 1/2 inch open-end wrench on flat surface of each cable housing.

(d) Install Pal nut (item 2), jam nut (item 3), spacer (item 4), anti-rotation stop (item 5) onto control cable (item 1) and install cable through Heim bearing rod-end (item 6). Install spacer (item 4, jam nut (item 3), Pal nut (item 2) onto control cable on opposite side of Heim bearing rod-end (item 6).

NOTE: Leave Pal nut and jam nut on each side of Heim bearing rod-end loose until rigging is accomplished (see Rigging Procedures, Section 76-00-00).

(e) Install jam nut (item 7), rod-end lockwasher (item 8) onto control cable bayonet.

NOTE: On power lever cable (item 1) install lockwasher (item 8) so that it faces downward for safety purposes.

(f) Install rod-end (item 9) onto bayonet of control cable.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

Ensure tab lockwasher (item 8) mates with groove in rod-end (item 9).

(g) Safety rod-end (item 9) to jam nut (item 7) with MS33540 safety wire.

CAUTION: USE A NEW LOCKNUT (ITEM 13) EACH TIME THE LOCKNUT IS REMOVED.

(h) Install bolt (item 10) upwards through rod-end (item 9), spacer (item 11) into propeller power lever arm (item 12). Secure bolt (item 10) with washer and locknut (item 13).

(i) Operate power lever control from the cockpit and accomplish rigging adjustment (see Section 76-00-00). Be sure to safety reverse stop block assembly IAW MS33540 after rigging adjustment.

(8) Install engine end of multiple-piece power control cable, (T384-999, AT070, and TC385-999) as follows:

(a) Refer to Figure 202B.

(b) Install the engine power cable segment (item 14) connecting it to the wing power cable segment at the firewall.

NOTE: It may be necessary to advance the cable forward from the cockpit pedestal power control. Pull power lever to full reverse so that the power cable will advance and protrude beyond the engine firewall to facilitate connection.

(c) Connect the power control cable segment at the engine firewall. Safety the firewall jam nut and engine power control cable nut with MS33540 safety wire.

(d) Install power control cable through bracket assembly (item 21).

(e) Install flanged bushing (item 15).

NOTE: Flanged bushing is used on Controlex cables only.

(f) Safety wire jam nut (item 16) with MS33540 safety wire.

(g) Install jam nut (item 18) onto control cable.

(h) Install rod-end (item 17) onto control cable.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

Ensure tab on lockwasher (item 18) mates with groove in rod-end (item 17).

CAUTION: USE A NEW LOCKNUT EACH TIME THE LOCKNUT IS REMOVED.

(j) Install bolt (item 19) upwards through rod-end (item 17); install spacer (item 23) onto bolt (item 19), install to propeller pitch control arm (item 20). Install washer and locknut (item 22) .

(k) Operate power lever control from the cockpit and accomplish rigging adjustments, see Section 76-00-00. After rigging adjustment, be sure to safety reverse stop block assembly IAW MS 33540.

(l) Install floorboards and carpets IAW Chapter 53, and seats IAW Chapter 25.

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RPM (SPEED) CONTROL CABLE – DESCRIPTION AND OPERATION

1. General

A. Engine operation output is controlled by the RPM (speed) levers via control cables to engine mounted units. The RPM levers, one for each engine, are located next to each other on the cockpit control pedestal beside the power levers.

B. The RPM levers connect to the engine speed lever shaft by flexible control cables that are routed from the pedestal, through the fuselage, through the wing, into the engine nacelle firewall, and to the speed lever shaft of the engine fuel control unit. A mechanical linkage connects the speed lever shaft (underspeed governor) of the fuel control to the propeller governor speed control.

C. The position of the RPM lever determines the preload on the inner governor speeder spring, which sets the engine speed control lever during propeller governing mode.

D. Each control conduit is anchored to a bracket immediately forward of the cockpit pedestal, and the threaded rod cable end is connected to the respective RPM control lever through a rod-end with a clevis.

E. At the engine of the RPM cable conduit, the conduit is anchored to an engine bracket attached to the rear compressor section. A rod-end on the end of the cable is bolted to the lever arm on the fuel control speed shaft of each engine. On SA226 series models T201-275, T277-291, and all At and TC aircraft, an electric synchronizer assists in synchronizing the propellers in the climb and cruise power settings (RH engine only). The RPM cable on the RH engine of these models, is bolted to the underspeed governor speed lever shaft through a rod-end assembly designed to accommodate the actuator of the propeller synchronizer. See Figure 202, inset A.

F. The RPM levers are mounted in the right upper section of the cockpit pedestal and are moved forward or aft. The pedestal quadrant is marked from to rear to indicate the positions HIGH and LOW.

RPM (SPEED) CONTROL CABLE – MAINTENANCE PRACTICES

1. Removal/Installation – RPM (Speed) Control Cable

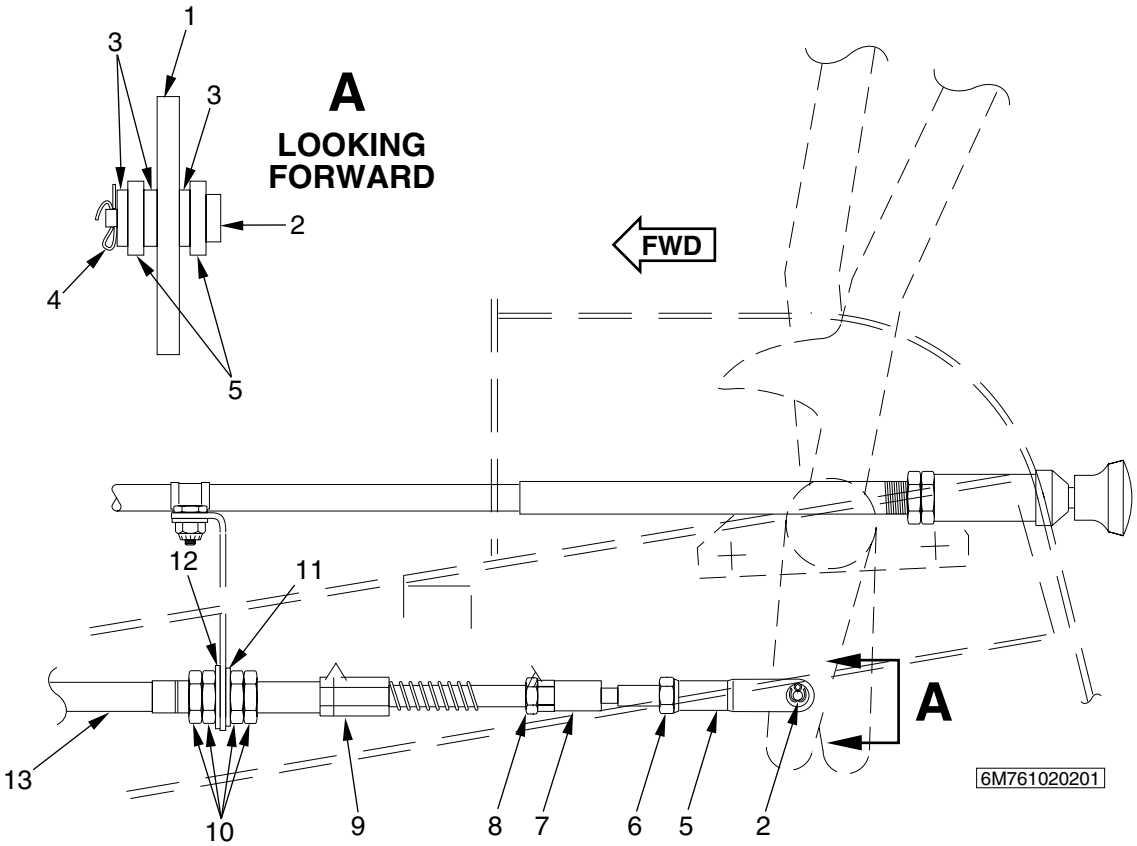
A. Removal – RPM (Speed) Control cable

(1) Remove necessary engine cowling from appropriate engine IAW Chapter 71.

(2) Remove seats, carpet IAW Chapter 25, and necessary floorboards IAW Chapter 53 to gain access to control cable segment for those cables within the fuselage.

NOTE: Depending on model series, cable disconnects are located at different points within the aircraft, see Figures 1 and 2, Section 76-00-00.

1. RPM LEVER
2. PIN
3. WASHER
4. COTTER PIN
5. CLEVIS ROD END
6. JAM NUT
7. BALL JOINT
8. JAM NUT
9. SEAL NUT
10. NUT
11. WASHER
12. BUSHING
13. RPM CONTROL CABLE



RPM Control Cable Quadrant Assembly
Figure 201

(3) To gain access to wing cable (from engine firewall through wing to fuselage disconnect segment, remove the access panels from underside of wing between the fuselage and nacelle. Remove leading edge portion of fuselage to wing fairing.

(4) Loosen the applicable retaining clamps long the cable route in the fuselage and the wing, see Figures 3 and 4, Section 76-00-00.

(5) Disconnect the action element of the cable and cable conduit at each end.

NOTE: On SA226 series T201-999, AT001-999, and TC201-999, see Figure 201. To disconnect RPM CONTROL CABLE (item 13) from quadrant assembly, remove items (2-12).

On SA226 series T201-275, T277-291, AT001-069, AT071-999, and TC201-384, RH ENGINE ONLY, see Figure 203, inset A. To disconnect RPM CONTROL CABLE (item 6) from engine, remove items 3-5 and 11-14.

On SA226 series T276, T292-383, RH ENGINE, or T201-383, AT001-069, AT071-999, and TC201-384, LH ENGINE, see Figure 203, inset B. To disconnect RPM CONTROL CABLE (item 6) from engine, remove items 3-5 and 13-16.

On SA226 series T384-999, AT070, and TC385-999, LH ENGINE, or T384-999, RH ENGINE, see Figure 302, inset C. To disconnect RPM CONTROL CABLE (item 19) from engine, remove items 21-25.

On SA226 series T249-383, AT024-069, AT071-999, and TC212-384, (two piece segments), also disconnect at quick disconnect between wing stations 27 and 39 in each leading edge.

On SA226 series T384-999 (three piece segments), disconnect at quick disconnect at F.S. 102.0 (below pilot and copilot floorboard) and at engine firewall at N.S. 92.69.

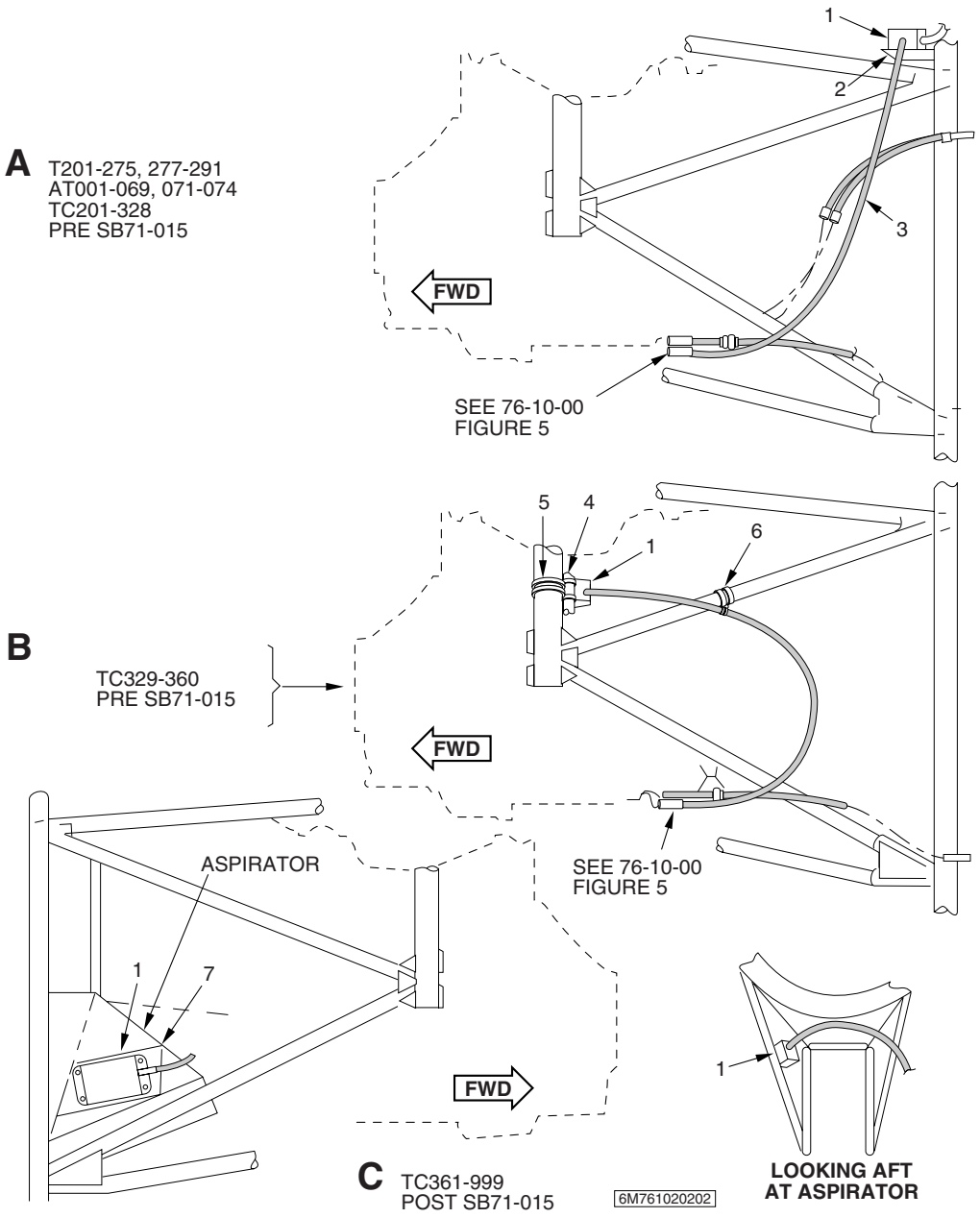
On SA226 series AT070-999 and TC384-999 (four piece segments), disconnect at quick disconnect at F.S. 102.77 (below pilot and copilot floorboard), at F.S. 219.77 where the fuselage mates with the wing, and at engine firewall at N.S. 92.69.

(6) Loosen the appropriate split fairlead at the nacelle firewall (single and two segment cables only).

(7) Remove the sealant from the fairlead at the fuselage.

(8) Attach a spool of safety wire to the cable being removed at the opposite end from which the cable is to be withdrawn. The wire will serve as a guide during cable installation or replacement.

(9) Remove the cable through the inside of the aircraft drawing it through the retaining clamp and, at the same time, draw the lead wire along the route. For two piece cables, withdraw the wing/nacelle portion from the engine end. Remove the fuselage portion by first pulling the wing end toward the fuselage until it clears the wing root rib and then pulling the cable from the fuselage at the wing end. For multiple piece cables, withdraw pedestal and fuselage cable segment through inside of aircraft, and at the same time, draw the guide wire along the route.

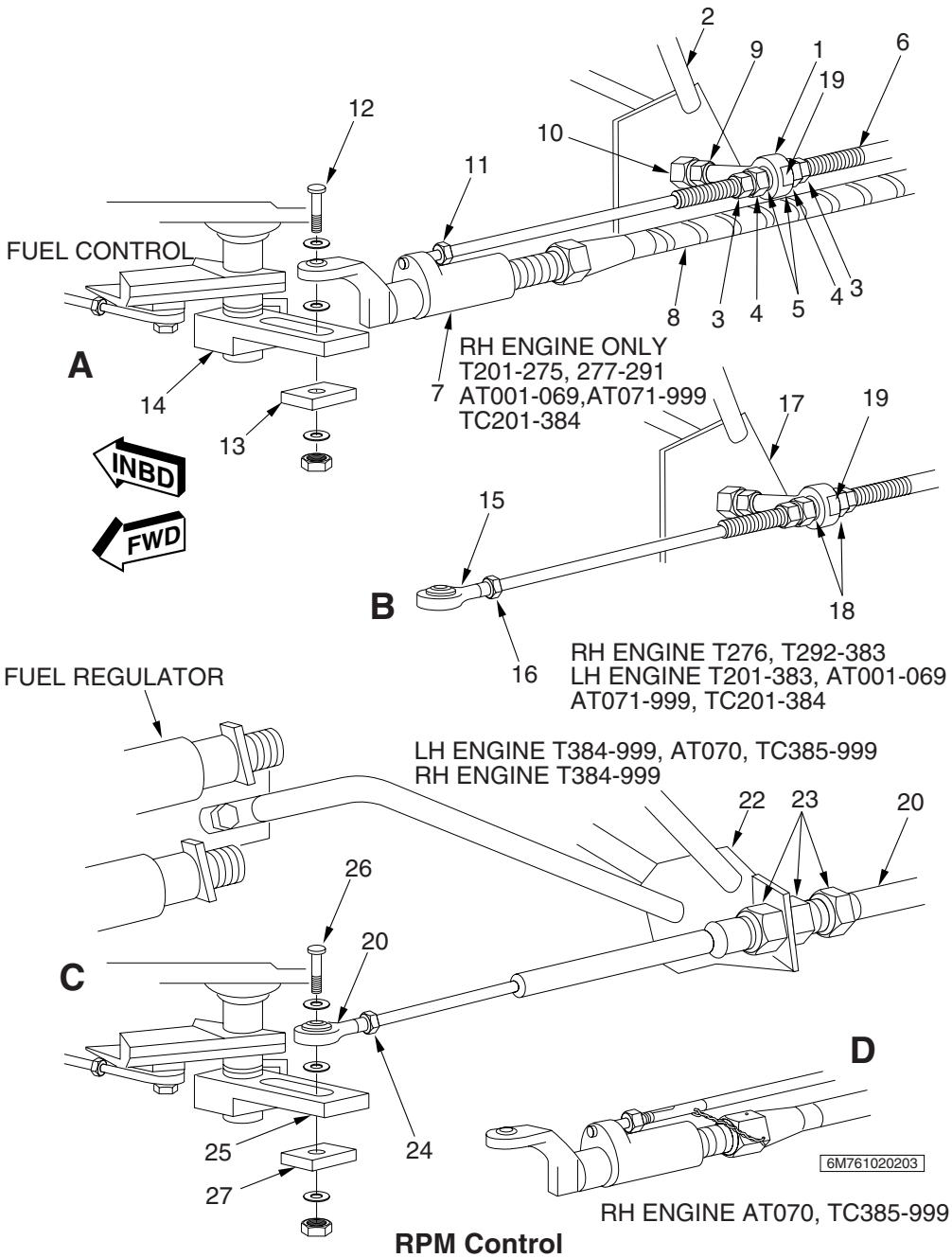


Engine Propeller Synchronizer and RPM Control, RH Engine Only
Figure 202

(10) For wing cables, withdraw the cable segment from the engine firewall end, withdrawing it through the loosened retaining clamp(s).

(11) Avoid coiling the cable on removal. Gather cable into a loose figure eight, no smaller than one foot by three feet and secure at the center crossover.

(12) Detach guidewire.



RPM Control
Figure 203

B. Installation – RPM (Speed) Control Cable

(1) Unpack new cable carefully, loop by loop, and straighten it by forming a hump near one end and walking it out the entire length of cable.

(2) Check the action element for smooth and easy movement. If any unusual load or binding is noted, reject cable for use.

(3) If the action element moves satisfactorily, attach the guidewire (spool of safety wire) to end of cable and draw through cable route.

(4) Avoid twisting the cable. Check the action element at the convenient intervals during installation and before securing the cable at its terminus to ensure no binding has incurred. When installing two piece cables, route cables from cockpit and nacelle ends to the leading edge area between wing stations 27 and 39.

(5) Secure applicable clamps.

(6) Install and secure the split fairlead at the firewall on applicable aircraft. Seal fairlead with PR812 high temperature sealant (or equivalent).

(7) Seal cables at fuselage fairlead with PR1442B-2 sealant (or equivalent).

(8) install tie-wraps and/or chaff covering around cables at applicable areas where cables group together.

(9) See Figure 201, connect RPM CONTROL CABLE (item 13) to quadrant assembly in installing items 2-12.

(10) Install RPM control cable and rod-end as follows:

NOTE: Certain SA226 series aircraft have incorporated a propeller Synchronizer Actuator, mounted on the RH engine only. See Figure 202 for production installation and respective applicable effectivities of the synchronizer actuator and mounting bracket. For installation instructions of propeller synchronizer actuator to the RH side of the aspirator as shown in Figure 202, inset C, refer to Fairchild Aircraft Service Bulletin 71-015, Heat Shield Removal.

See IPC Chapter 76 for RPM control and for Power Control installation part numbers.

C. Installation – Heim Bearing Rod-End, LH & RH Engine. (T201-383, AT001-069, AT071-999, and TC201-384, see Figure 203, insets A and B)

(1) Remove Heim bearing rod-end (Figure 203, item 1) from bracket assembly (item 2) in order to install it onto RPM lever control cable (item 6) and to avoid kinking of the control cable during installation.

(2) Install Pal nut (Figure 203, item 3), jam nut (item 4), spacer (item 5), and anti-rotation stop (item 19) onto control cable (item 6)

NOTE: A Pal nut, jam nut, and spacer is installed on both sides of Heim bearing rod-end typically.

(3) Install Heim bearing rod-end (Figure 203, item 1) onto the RPM lever control cable (item 6), then install spacer (item 5), nut (item 4) and Pal nut (item 3).

NOTE: Leave Pal nuts and jam nuts on each side of Heim bearing rod-end loose until rigging is accomplished (see Rigging Procedures, Section 76-00-00).

(4) Install Heim bearing rod-end and control cable to bracket assembly (Figure 203, item 2).

NOTE: Identical nuts and Pal nuts (two each) to those used to secure the Heim bearing rod-end to the control cable are also used to secure the Heim bearing rod-end to the bracket assembly.

(5) Apply locktite 242 (Locktite Corp. – Newington, Conn, 06111) onto threads of portion of the RPM lever control cable (Figure 203, item 6) that slides over the bayonet, and threads into housing of RPM lever cable assembly.

NOTE: Tighten snug by using 3/8 inch open-end wrench on flat surface of RPM lever cable portion which slides over bayonet, and use 1/2 inch open-end wrench on flat surface of cable housing.

D. Installation – Propeller Synchronizer Actuator Rod-End, RH ENGINE ONLY, (T201-275, T277-291, AT001-999, and TC201-999, see Figure 203, inset A, item 7, and inset D)

(1) Adjust propeller synchronizer shaft rod-end (Figure 203, item 7) so that the actuator of the rod end extends .25 inches as shown in illustration.

NOTE: The rod-end can be extended by connecting the synchronizer shaft housing (Figure 203, item 8) to the splines within the rod-end, then by turning the shaft housing at its opposite end. After adjustment is made, disconnect the shaft housing to the rod-end splines so as not to disturb the rod-end setting until final installation of shaft is completed (see step (6) under this section).

(2) With synchronizer shaft housing disconnected, adjust Propeller Synchronizer Actuator (Figure 202, item 1) by turning shaft within actuator fully clockwise until shaft stops turning, then shaft counterclockwise 27 graduations (audible clicks).

(3) Propeller Synchronizer Actuator Installation

(a) At top of aspirator – see Figure 202, inset A – Install Propeller Synchronizer Actuator (Figure 202, item 1) to top of aspirator just forward of firewall and secure it to mount (item 2) using four each screws and washers. See IPC, Chapter 76 for item part numbers).

(b) At engine truss member – See Figure 202, inset B – Install Propeller Synchronizer Actuator (Figure 202, item 1) onto engine truss member as shown in illustration (Figure 202, inset B). 3.75 inches from top of LH engine mount. Install using bracket assembly, clamps, screws, and washers (item 4). See IPC, Chapter 76 for item part numbers.

(c) At RH side of aspirator – See Figure 202, inset C – Install Propeller Synchronizer Actuator onto doubler attached to side of aspirator with four each screws and washers. See IPC, Chapter 76 for item part numbers.

NOTE: See Fairchild Aircraft Service Bulletin 71-015 for installation procedures of doubler (Figure 202, item 7) to side of aspirator.

(4) Install jam nut (Figure 203, item 11) onto RPM lever cable bayonet, then install Propeller Synchronizer rod-end (item 7) onto bayonet allowing for approximately three threads of bayonet extending beyond rod-end. Safety jam nut (item 11) with .025 safety wire as shown in figure 203, inset D.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

(5) Install rod-end (Figure 203, item 7) onto fuel control RPM lever arm (item 14). Install bolt, three washers and nuts (item 12), and plate (item 13) as shown in illustration.

NOTE: When installing mounting bolt (item 12), center the rod-end (item 7) onto the RPM lever arm (item 14) such that the distance from center of mounting bolt (item 12) to end of RPM arm (item 14) is .50 inches as shown in illustration. Ensure that the serrations of the plate (item 13 align with the serrations on the bottom of the RPM lever arm (item 14).

CAUTION: A NEW LOCKNUT FOR BOLT (ITEM 12) SHALL BE USED EACH TIME THE NUT IS REMOVED.

(6) Propeller Synchronizer Actuator Cable Shaft Installation.

(a) Install Propeller Synchronizer Actuator Shaft (Figure 203, item 8) by installing into Synchronizer Actuator first.

NOTE: Avoid disturbing adjustment of actuator while connecting shaft, (reference step (1) of this section.

(b) Install opposite end of shaft to Synchronizer Actuator rod-end.

(c) Safety both fittings at each end of Synchronizer Actuator shaft housing with 0.025 safety wire, see Figure 203, inset D, showing safetying of fitting to RPM control cable.

(7) Install applicable clamps as shown in Figure 202, item 3 and/or item 6, see IPC, Chapter 76 for item part numbers.

(8) Connect connector and adapter (cannon plug) to Propeller Synchronizer Actuator and safety connector with 0.025 safety wire.

E. Installation – RPM Lever Rod-End on RH Engines (T276, T292-383) and LH Engines (T201-383, AT001-069, AT071-999, and TC201-384), see Figure 203, inset B, item 15.

(1) Install jam nut (item 16) and rod-end locking washer onto RPM lever control bayonet. Install rod-end (item 15) onto bayonet.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

(2) Safety rod-end (Figure 203, item 15) to locking washer with 0.025 safety wire.

(3) Install rod-end onto fuel control RPM lever (item 25). Install bolt (item 26, three washers and nuts, and plate (item 27) as shown in Figure 203, inset B.

CAUTION: IF LOCKNUT FOR BOLT (ITEM 26) IS REMOVED, REPLACE WITH NEW LOCKNUT. SEE IPC, CHAPTER 76 FOR ITEM PART NUMBER.

(4) Operate RPM lever control from the cockpit and accomplish general rigging requirements, see Engine Rigging Check, Section 76-00-00.

F. Installation – RPM Control Cable and Control Cable Rod-End on LH Engines T384-999, AT070, and TC385-999) and RH Engines T284-999).

NOTE: The above effectivities reflect latest installation of multiple piece control cables and attaching bracket. For installation procedures of the Propeller Synchronizer Actuator, see Figure 203, inset D. For RH engines AT070 and TC385-999, refer to Installation – Propeller Synchronizer Actuator Rod-End, RH Engine Only, T201-275, T277-291, AT001-999, and TC201-999.

(1) Install the RPM control cable segment (item 19) connecting it to the wing RPM cable segment at the engine firewall.

NOTE: It may be necessary to advance the RPM cable forward from the cockpit pedestal RPM control. Remove the clevis pin attaching the RPM control cable to the RPM Lever. Push the cable forward so that the cable will advance and protrude beyond the engine firewall to, facilitate connection.

(2) Connect the RPM control cable segment at the engine firewall. Safety the firewall jam nut and engine RPM control cable nut with MS33540 safety wire.

(3) Install engine RPM control cable through bracket assembly (item 20).

(4) Safety jam nuts (item 21) with MS33540 safety wire.

(5) Install rod-end (item 13) onto control cable.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

Ensure tab on lockwasher (item 22) mates with grooves in rod-end.

(6) Safety rod-end to lockwasher with MS33540 safety wire.

(7) Install rod-end (item 18) onto fuel control RPM lever arm (item 23). Install bolt, three washers and nuts (item 24), and plate (item 25) as shown in illustration.

NOTE: When installing mounting bolt, center the rod-end onto the RPM lever arm (item 23) such that the distance from center of mounting bolt (item 24) to end of RPM lever arm (item 23) is .50 inches as shown in illustration. Ensure that the serrations of the plate (item 25) align with the serrations on the bottom of the RPM lever arm (item 23).

CAUTION: A NEW LOCKNUT FOR BOLT (ITEM 18) SHALL BE USED EACH TIME THE LOCKNUT IS REMOVED.

(8) Install RPM cable to RPM lever. Operate RPM lever control from the cockpit and accomplish general rigging requirement. See Maintenance manual, Section 76-00-00, Rigging Check.

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EMERGENCY SHUTDOWN ENGINE FUEL SHUTOFF (FSO) CONTROL CABLES – DESCRIPTION AND OPERATION

1. General

A. Emergency shutdown is provided through two knobs located on the pedestal and marked "L & R – PULL". Each cable conduit contains threaded rod-ends which provide for proper position at the pedestal control knob and at the engine. At the pedestal, the threaded end of the cable is anchored to the face of the pedestal, see Figure 201. At the engine, it is anchored through a bracket mounted aft of the fuel control unit. The pedestal control knob is threaded onto the action element rod-end. At the engine, the action element rod-end is bolted to a bellcrank. Linkage extends from the bellcrank to the manual override lever of the fuel solenoid control valve and the propeller feathering valve. The bellcrank is designed so the fuel is shut off before the propeller feather valve is opened.

B. When the knobs are pulled, the connecting cable rotates the bellcrank, which, through mechanical linkage, closes the manual fuel shutoff of the fuel solenoid valve, and opens the propeller feathering valve. As the propeller feather valve opens, oil in the propeller dome transfers into the engine case, and allows the feathering spring to drive the blades into feather. The actions are sequenced so that the fuel supply to the engine is interrupted before the propeller is feathered. The manual fuel shutoff overrides the normal electric control of the fuel flow. When the stop and feather knob is returned to the normal position, the spring loaded propeller feathering valve closes and the manual shutoff is armed. Arming the manual fuel shutoff does not start the fuel flow to the engine. It rearms the overriden electric control, and the start button must be depressed to activate the normal electrically controlled flow.

EMERGENCY SHUTDOWN ENGINE FUEL SHUTOFF (FSO) CONTROL CABLES – MAINTENANCE PRACTICES

1. Removal/Installation – Engine Fuel Shutoff (FSO) Control Cables

A. Removal – Engine Fuel Shutoff (FSO) Control Cables

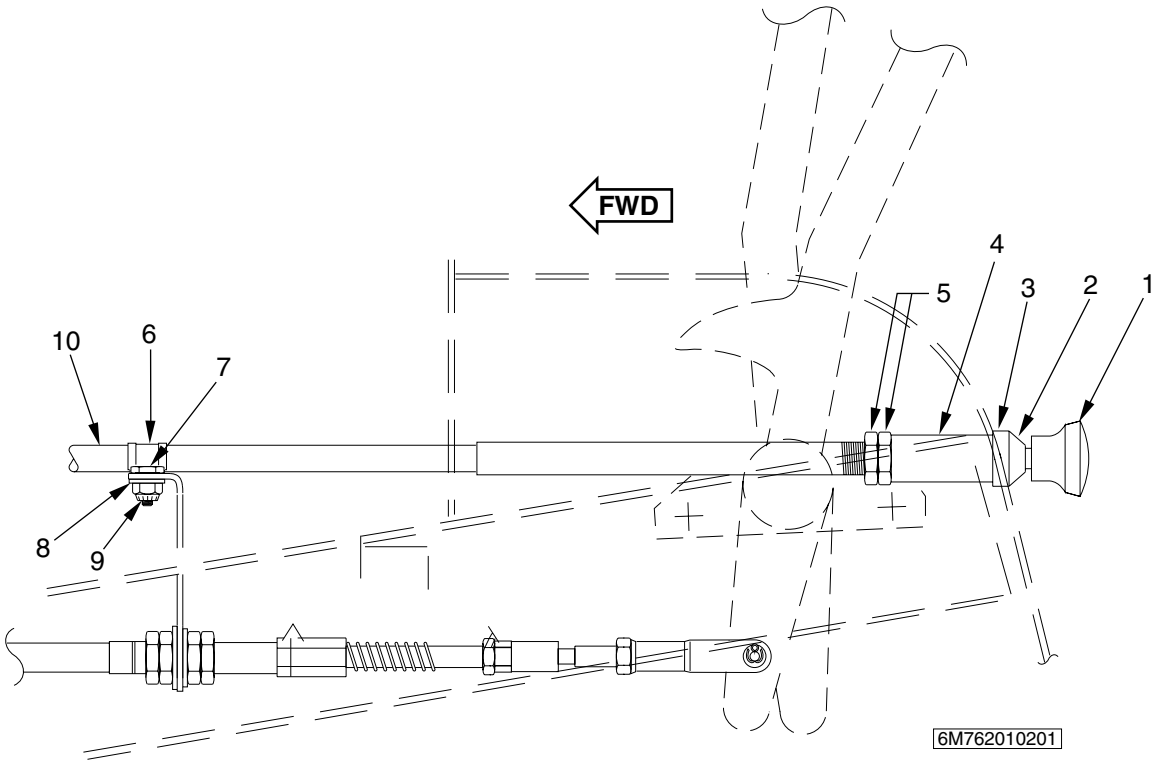
(1) Remove necessary engine cowling from appropriate engine, see Chapter 71.

(2) Remove seats, carpet IAW Chapter 25, and necessary floorboards IAW Chapter 53 to gain access to control cable segment for those cables within the fuselage.

NOTE: Depending on model series, cable disconnects are located at different points within the aircraft, see Figures 1 and 2, Section 76-00-00.

(3) To gain access to wing cable (from engine firewall through wing to fuselage disconnect) segment, remove the access panels from underside of wing between the fuselage and nacelle. Remove leading edge portion of fuselage to wing fairing.

1. KNOB, FUEL SHUTOFF
2. NUT
3. WASHER
4. BUSHING
5. NUT
6. CLAMP
7. BOLT
8. WASHER
9. NUT
10. FUEL SHUTOFF CABLE



Fuel Shutoff Cable Quadrant Assembly
Figure 201

(4) Loosen the applicable retaining clamps along the cable route in the fuselage and the wing, see Figures 3 and 4, Section 76-00-00.

(5) Disconnect the action element of the cable and cable conduit at each end.

NOTE: On SA226 series T201-999, AT001-999, and TC201-999, see Figure 201. To disconnect FUEL SHUTOFF CABLE (item 10) from quadrant assembly, remove items 1-9.

On SA226 series T201-383, AT001-069, AT071-999, and TC201-384, see Figure 202, inset A. To disconnect FUEL SHUTOFF CABLE (item 1) form engine, remove items 2-9.

On SA226 series T384-000, AT070, and TC385-999, see Figure 202, inset B. To disconnect FUEL SHUTOFF CABLE (item 12) from engine, remove items 14-18.

On SA226 series T249-383, AT024-069, and TC212-384 (two piece segments), also disconnect at quick disconnect between wing stations 27 and 39 in each leading edge.

On SA226 series T384-999 (three piece segments), disconnect at quick disconnect at F.S. 102.0 (below pilot and copilot floorboard) and at engine firewall at N.S, 92.69.

On SA226 series AT070-999 and TC384-999 (four piece segments), disconnect at quick disconnect at F.S. 102.77 (below pilot and copilot floorboard), at F.S. 219.77 where the fuselage mates with the wing, and at engine firewall at N.S. 92.69.

(6) Loosen the appropriate split fairlead at the nacelle firewall (single and two segment cables only).

(7) Remove the sealant from the fairlead at the fuselage.

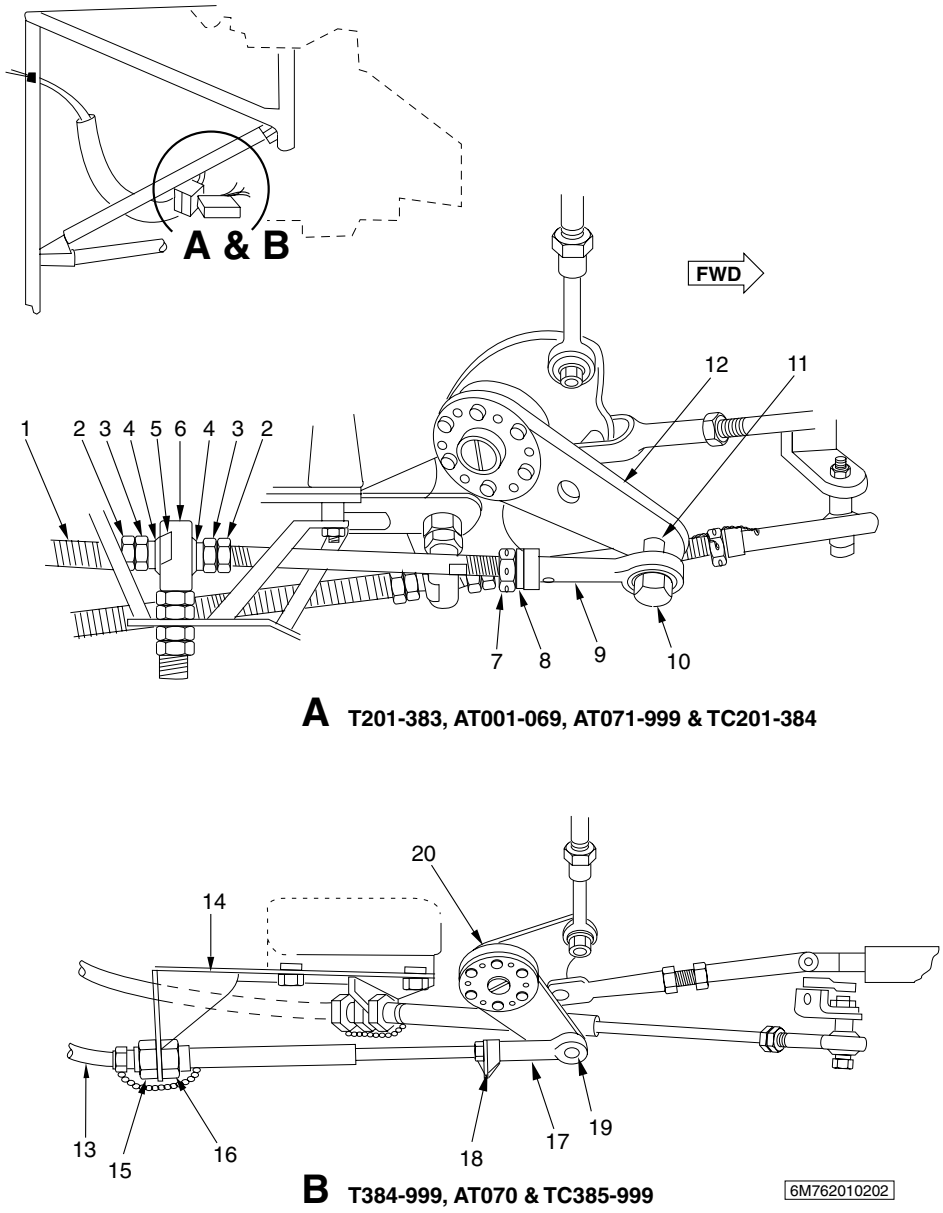
(8) Attach a spool of safety wire to the cable being removed at the opposite end from which the cable is to be withdrawn. The wire will serve as a guide during cable installation or replacement.

(9) Remove the cable through the inside of the aircraft drawing it through the retaining clamp and, at the same time, draw the lead wire along the route. For two piece cables, withdraw the wing/nacelle portion from the engine end. Remove the fuselage portion by first pulling the wing end toward the fuselage until it clears the wing root rib and then pulling the cable from the fuselage at the wing end. For multiple piece cables, withdraw pedestal and fuselage cable segment through inside of aircraft, and at the same time, draw the guide wire along the route.

(10) For wing cables, withdraw the cable segment from the engine firewall end, withdrawing it through the loosened retaining clamp(s).

(11) Avoid coiling the cable on removal. Gather cable into a loose figure eight, no smaller than one foot by three feet and secure at the center crossover.

(12) Detach guidewire.



Fuel Shutoff Control Cable Engine Assembly
Figure 202

B. Installation – Engine Fuel Shutoff (FSO) Control Cable

(1) Unpack new cable carefully, loop by loop, and straighten it by forming a hump near one end and walking it out the length of the cable.

(2) Check the action element for smooth and easy movement. If any unusual load or binding is noted, reject cable for use.

(3) If the action element moves satisfactorily, attach the guide wire (spool of safety wire) to end of cable and draw through cable route.

(4) Avoid twisting the cable. Check the action element at convenient intervals during installation and before securing the cable at its terminus to ensure no binding has incurred. When installing two piece cables, route cables from cockpit and nacelle ends to the leading edge area between wing stations 27 and 39.

(5) Secure applicable clamps.

(6) Install and secure the split fairlead at the firewall on applicable aircraft. Seal fairlead with PR812 high temperature sealant (or equivalent).

(7) Seal cables at fuselage fairlead with PR1442B-2 sealant (or equivalent).

(8) Install tie-wraps and/or chaff covering around cables at applicable areas where cables group together.

(9) See Figure 201, connect FUEL SHUTOFF CABLE (item 10) to quadrant assembly in installing items 1-9

(10) Install FSO control cable and rod-end as follows:

C. Installation – T201-383, AT001-069, AT071-999, and TC201-384, se Figure 202

(1) Apply Locktite 242 (Locktite Corp. – Newington, Conn 06111) onto threads of portion of FSO control cable (item 1) which slides over bayonent, then threads into housing of each control cable assembly.

(2) Tighten snug by using 3/8 inch open-end wrench on flat surface of control cable portion which slides over bayonet, and use 1/2 inch open-end wrench on flat surface of each cable housing.

(3) Install Pal nut (item 2), jam nut (item 3), spacer (item 4) and anti-rotation stop (item 5) onto control cable (item 1) and install cable through Heim bearing rod-end (item 6). Install spacer (item 4), jam nut (item 3), Pal nut (item 2) onto control able on opposite side of Heim bearing rod-end (item 6).

NOTE: Leave Pal nuts and jam nuts on each side of Heim bearing rod-end loose until rigging is accomplished (see Rigging Procedures, Section 76-00-00).

(4) Install jam nut (item 6), then rod-end lockwasher (item 7) onto control cable bayonet.

NOTE: On FSO cable (item 1) install lockwasher (item 7) so that it faces downward for safety purposes.

(5) Reinstall rod-end (item 9) onto bayonet of control cable.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

Ensure tab lockwasher (item 8) mates with groove in rod-end (item 9).

(6) Safety rod-end to lockwasher with MS33540 safety wires.

CAUTION: ANYTIME LOCKNUT FOR BOLT (ITEM 10) IS REMOVED, REPLACE IT WITH NEW LOCKNUT.

(7) Install bolt (item 10) upwards through rod-end (item 9). Install spacer (item 11) onto bolt, install bolt through bellcrank (item 12). Secure with washer and locknut.

(8) Operate FSO control from the cockpit and accomplish general rigging adjustment (see Section 76-00-00).

D. Installation – Engine Fuel Shutoff (FSO) Control Cable T384-999, AT070, and TC285-999, Multiple Piece Control Cables, see Figure 202

(1) Install the engine FSO cable segment (item 13) connecting it to the wing FSO cable segment at the firewall.

NOTE: It may be necessary to advance the cable forward from the cockpit pedestal FSO control. Remove the knob from the pedestal FSO control, and with a rod, push the FSO control cable forward so that the cable will advance and protrude beyond the engine firewall to facilitate connection.

(2) Connect the FSO control cable segment at the engine firewall. Safety the firewall jam nut and engine FSO control cable nut with MS33540 safety wire.

(3) Install FSO control cable through bracket assembly (item 14).

(4) Install flanged bushing (item 15).

NOTE: Flanged bushing is used on Controlex cables only.

(5) Safety jam nut (item 16) with MS33540 safety wire.

(6) Install rod-end (item 17) onto control cable.

NOTE: Apply Moly-Kote dry lube to bayonet prior to rod-end installation.

Ensure tab lockwasher (item 18) mates with groove in rod-end (item 17).

(7) Safety rod-end to lockwasher with MS33540 safety wire.

CAUTION: ANYTIME LOCKNUT FOR BOLT (ITEM 10) IS REMOVED, REPLACE IT WITH NEW LOCKNUT.

(8) Install bolt (item 19) upwards through rod-end; install spacer onto bolt. Install bolt through bellcrank (item 20). Secure bolt (item 19) with washer and locknut.

(9) Operate FSO control from the cockpit and accomplish general rigging adjustment (see Section 76-00-00).

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MAINTENANCE MANUAL

CHAPTER 77 – ENGINE INDICATING

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ENGINE INDICATING – DESCRIPTION AND OPERATION

1. General

A. The engine indicating system used on the SA226 aircraft, provides the pilot with visual indication of engine performance throughout the full range of operation.

(1) Three electrically operated instruments, located in the center instrument panel, provide indication of engine speed, temperature, and torque for each engine.

(2) A chip detector warning light in the annunciator panel illuminates should magnetic particles be detected in the engine reduction gear case.

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ENGINE POWER INDICATING – DESCRIPTION AND OPERATION

1. General

A. The engine power indication used on SA226 aircraft consists of engine speed and torque.

2. Engine Speed

A. Engine speed indication is provided by a tachometer generator mounted on each engine, that transmits an electrical signal, which operates the indicator synchronous motor producing the RPM indication in percent of RPM.

3. Engine Torque

A. The engine torque internal sensing system provides compensated oil pressure to the transmitter which translates pressure to electrical current indicated in foot-pounds on the torque indicator. The indicator is calibrated from 0 to 29 with each scale graduation representing 100 FT.-LBS.

B. On Merlin IIIB's T276, 292-999 a torque transducer replaces, the transmitter and torque is indicated in percent of torque on the indicator.

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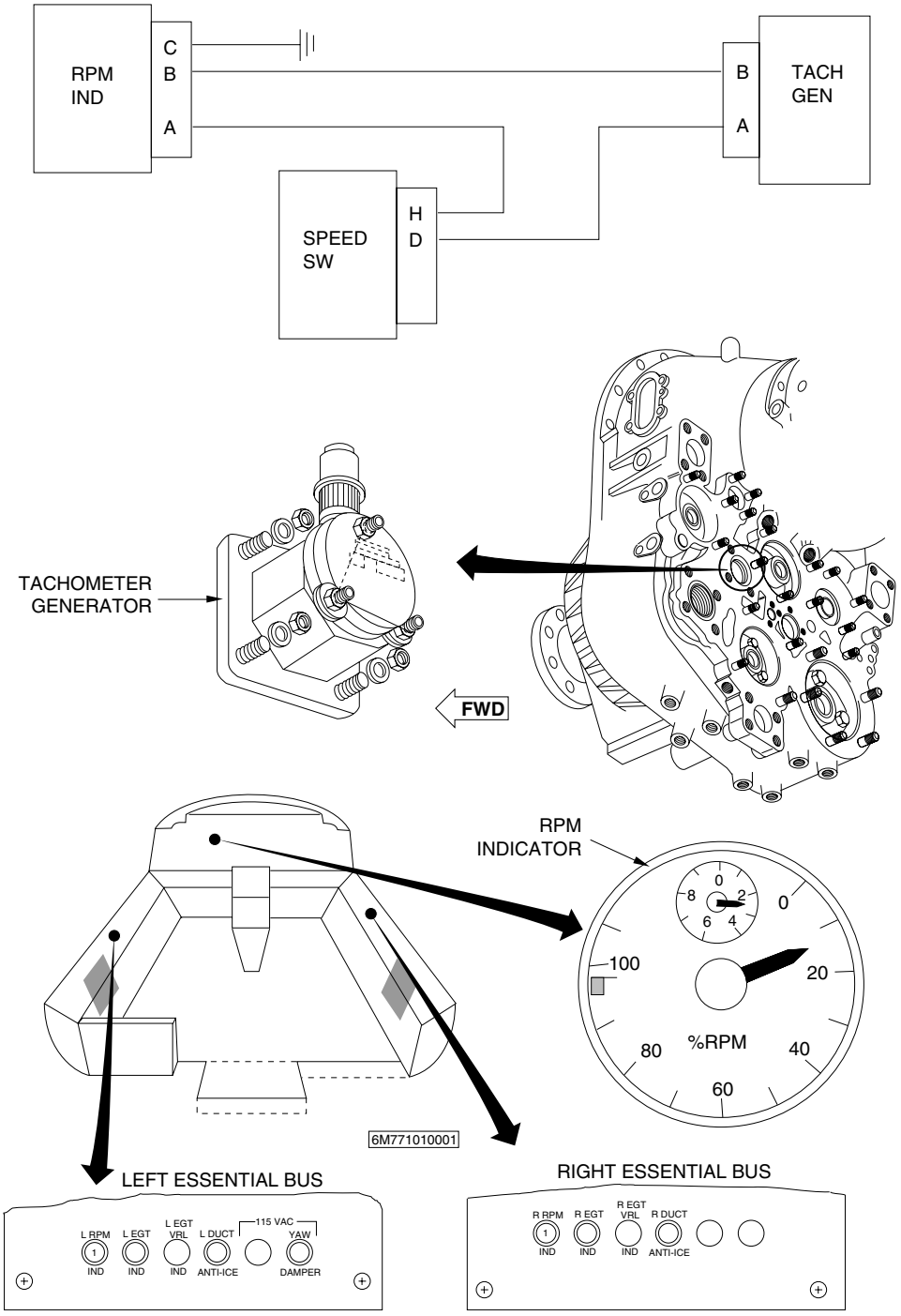
TACHOMETER INDICATING – DESCRIPTION AND OPERATION

1. General

A. Tachometer indicating consists of tachometer generators and revolutions per minute (RPM) indicators.



Tachometer Indicating
Figure 1

TACHOMETER INDICATING – DESCRIPTION AND OPERATION

(1) Tachometer generators, mounted on the upper center mount pad of each engine reduction gear case, are single stage, two pole, three phase AC generators.

(a) The output power varies IAW engine speed with maximum generator output is 21.0 ± 0.5 VAC at 42,000 revolutions per minute.

(2) RPM indicators, mounted on the center instrument panel are powered by 28 VDC from respective left and right essential buses.

(a) The indicators contain a synchronous motor that runs in synchronism with the tachometer generator's variable AC output signal.

(b) The RPM indicators are calibrated from zero (0) to 100% and internally lighted by 5.0 VDC.

1. Removal/Installation – Tachometer Generator

A. Removal – Tachometer Generator (Figure 201)

(1) Ensure electrical power is OFF.

(2) Remove engine cowling IAW instructions in Chapter 71.

(3) Remove starter generator IAW instructions in Chapter 24.

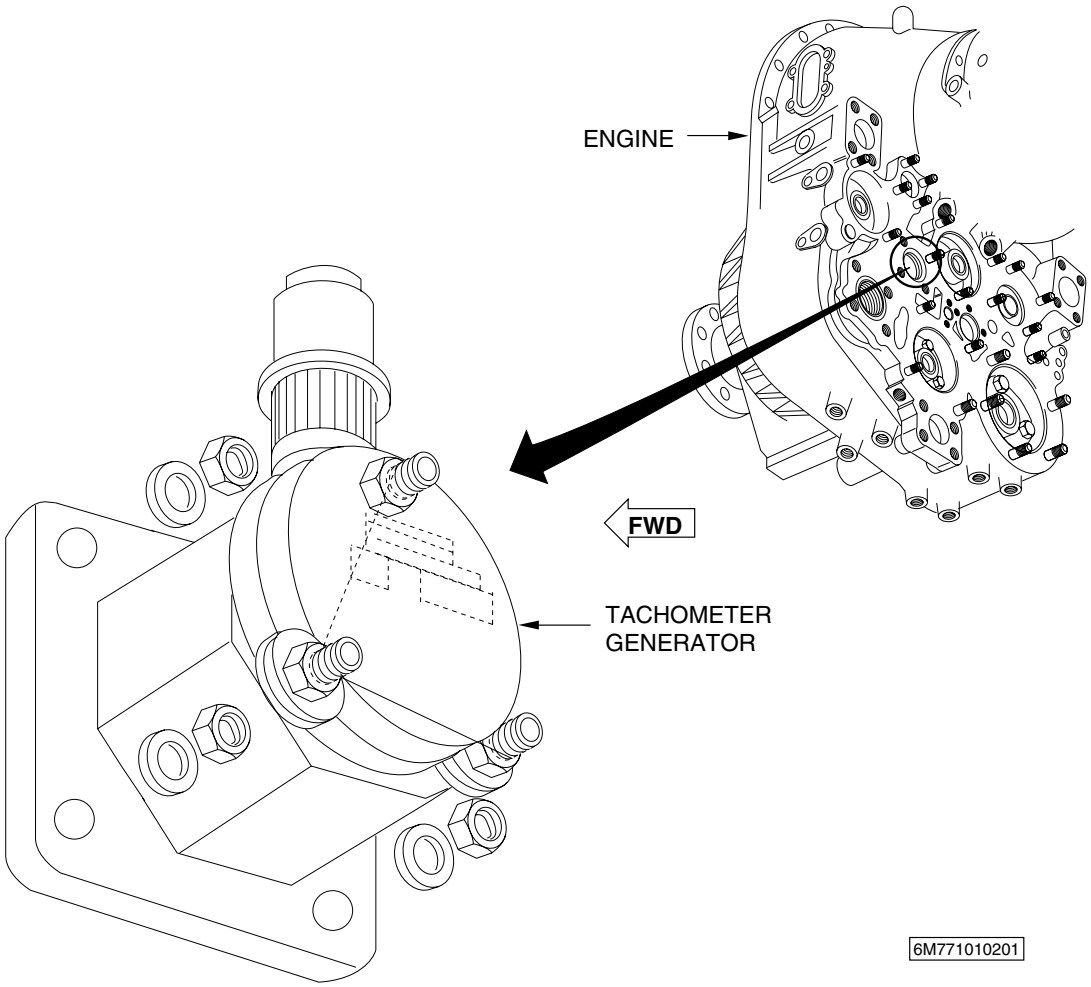
(4) Remove hydraulic pump IAW instructions in Chapter 29.

(5) Disconnect electrical connector on tachometer generator.

(6) Remove nuts and washers securing tachometer generator.

(7) Remove tachometer generator.

(8) Cap mounting pad to prevent contamination of reduction gear case.



Tachometer Generator
Figure 201

B. Installation – Tachometer Generator (Figure 201)

(1) Ensure electrical power is OFF.

(2) Select one mount hole on tachometer generator and burnish to facilitate electrical grounding.

(3) Remove cap from mounting pad.

(4) Lubricate tachometer generator spline gear with a film of grease (MIL-G-21164).

(5) Position tachometer generator on mounting pad studs. Check for proper engagement of drive shaft.

(6) Secure tachometer generator to reduction gear case with nuts and washers. Torque retaining nuts 50 inch-pounds (5.65 N·m).

(7) Connect electrical connector to tachometer generator and safety.

(8) Install hydraulic pump IAW instructions in Chapter 29.

(9) Install starter generator IAW instructions in Chapter 24.

(10) Install engine cowling IAW instructions in Chapter 71.

(11) Turn battery switches ON.

(12) Perform operational check of tachometer indicating system.

C. Removal – RPM Indicator (Figure 202)

NOTE: The following procedures apply to front mounted instruments.

(1) Ensure electrical power is OFF.

(2) Remove post light caps.

(3) Loosen indicator retaining clamp screw (lower right screw).

(4) Withdraw indicator from instrument panel.

(5) Disconnect electrical connector from indicator.

D. Installation – RPM Indicator (Figure 202)

(1) Connect electrical connector to indicator.

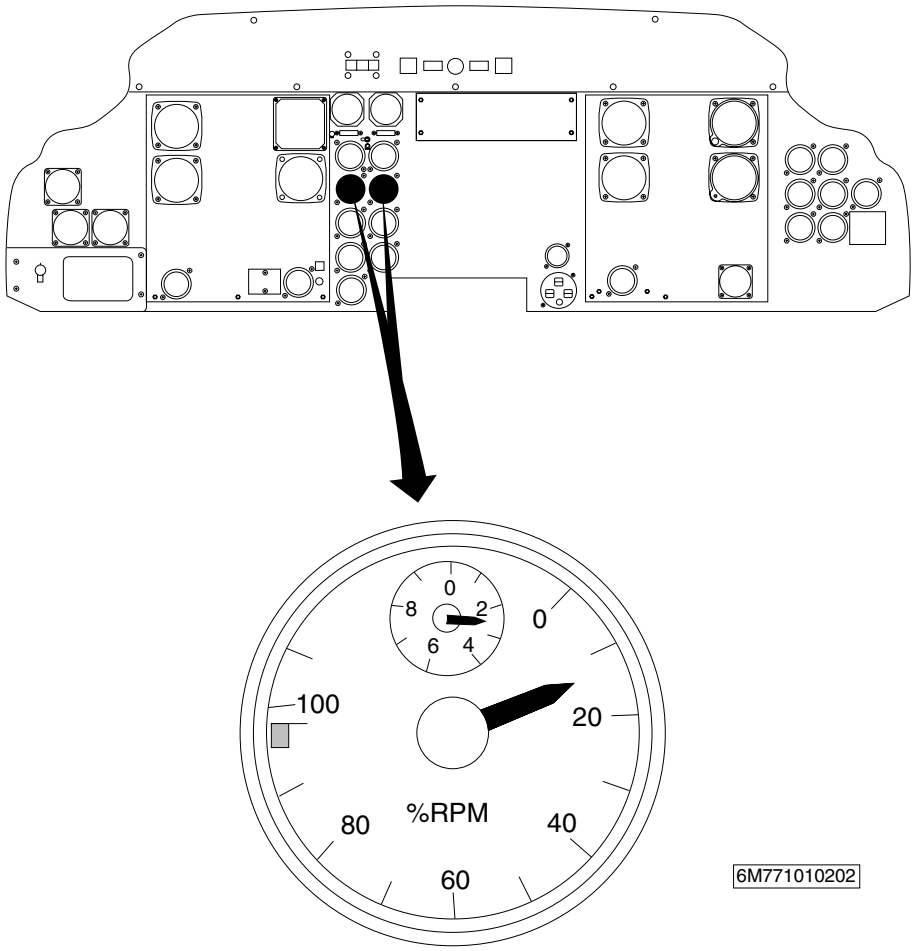
(2) Position indicator in instrument panel.

(3) Tighten indicator retaining clamp screw (lower right screw).

(4) Install post light caps.

(5) Turn battery switches ON.

(6) Perform operational check of tachometer system.



RPM Indicator
Figure 202

2. Inspection/Check – Tachometer Generator

A. Inspect tachometer generator as follows:

(1) Inspect tachometer generator for evidence of leakage and security of mounting.

(2) Check electrical connection for security and safety.

(3) Inspect RPM indicator (tachometer) for security of mounting and general condition.

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NEGATIVE TORQUE SENSING (NTS) – DESCRIPTION AND OPERATION

1. General

A. A negative torque test circuit is installed on some SA226 aircraft. The system provides ground check out of the negative torque system.

(1) An unfeather and negative torque test switch on the forward left console activates the unfeather pump, which supplies oil pressure to the NTS pressure switch. The pressure switch sends an electrical signal to illuminate the annunciator mounted NTS light.

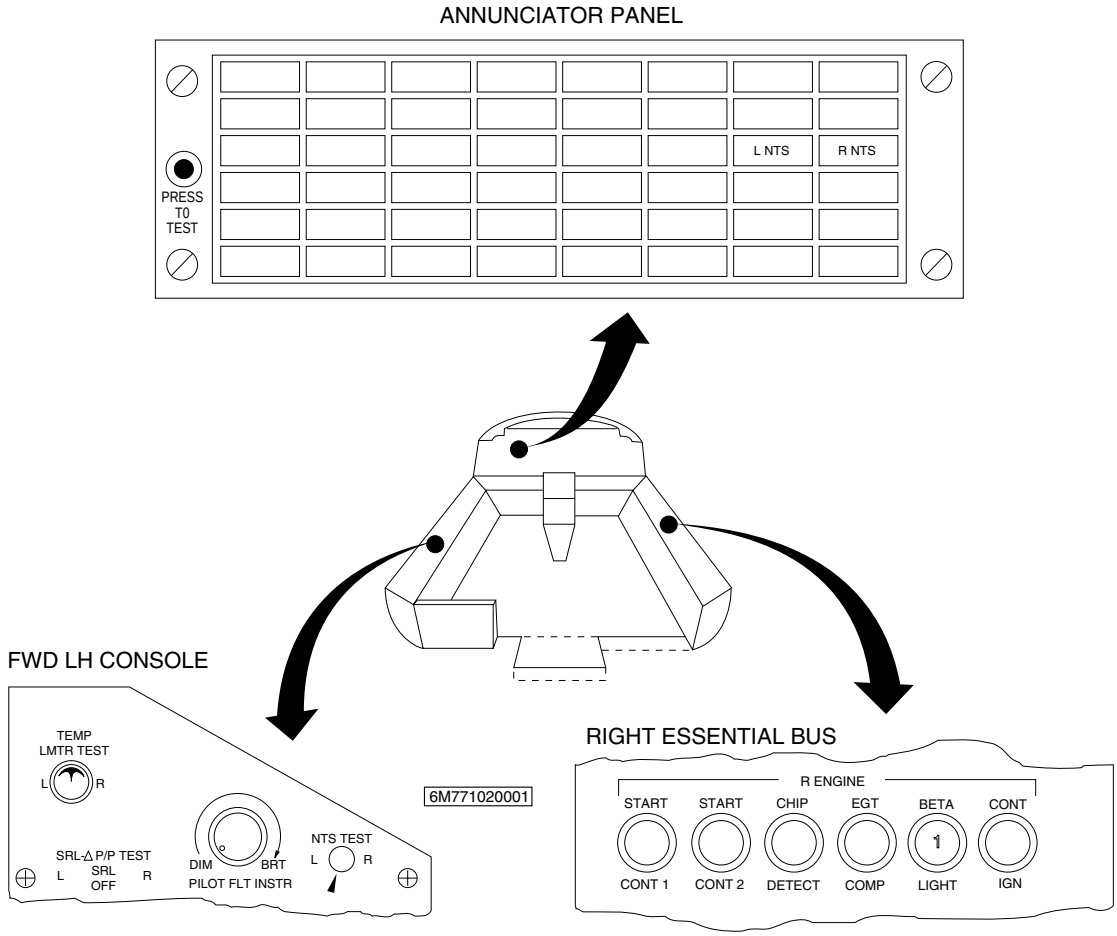
(2) The NTS light is used to indicate the triggering and resetting of the negative torque sensing system during a ground start.

(3) Refer to the applicable aircraft flight manual for operational check procedures.

(4) On aircraft models AT001-024, 026-033, and TC201-212, the NTS indicator lights are installed on the left side console.

(5) System power is from the left or right 28 VDC essential bus through the one amp beta circuit breaker.

(6) For additional information, refer to the applicable AiResearch Maintenance Manual.



Negative Torque Sensing (NTS)
Figure 1

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TORQUE TRANSMITTER INDICATING – DESCRIPTION AND OPERATION

1. General

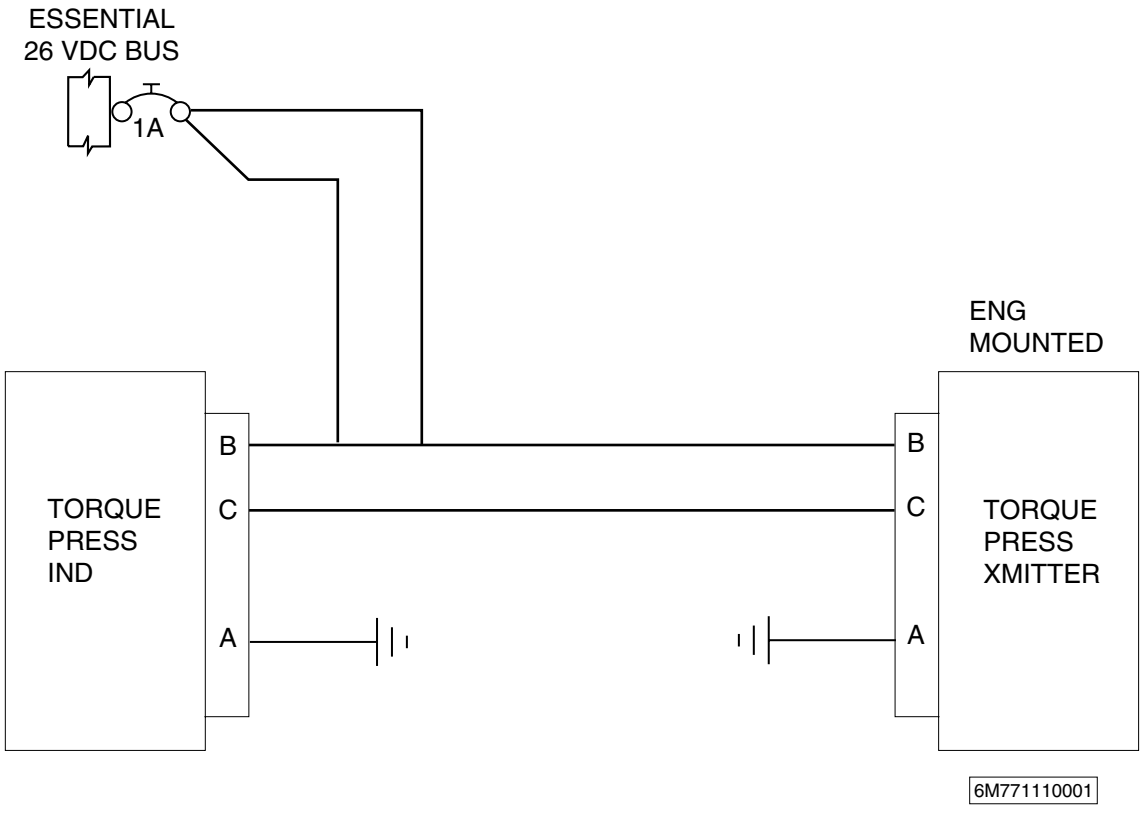
A. The torque pressure transmitter is the sensing unit of the variable reluctance type torque pressure indication system used on the SA226 aircraft.

(1) The transmitter is a factory sealed unit that converts oil pressure to an electrical signal, which varies according to applied pressure. The torque pressure transmitter is connected to both the torque pressure output port and to the case negative pressure port. The differential pressure is sent to the torque indicator as an electrical signal, and is indicated in foot-pounds.

(2) System power is from the left or right 26 VAC bus through a one amp circuit breaker.

(3) On pre-Service Bulletin 77-001/2 aircraft T201-237, AT001-011, TC201-207, the torque transmitter is bracket mounted in the lower forward section of each nacelle aft of the firewall. Hoses and tubing connect the transmitter to the torque pressure and case pressure ports located on the right rear reduction gear case.

(4) Aircraft T238-275, 277-291, AT012-999, TC208-999, and post Service Bulletin 77-001/2 have the torque pressure transmitter mounted to the right rear reduction gear case.



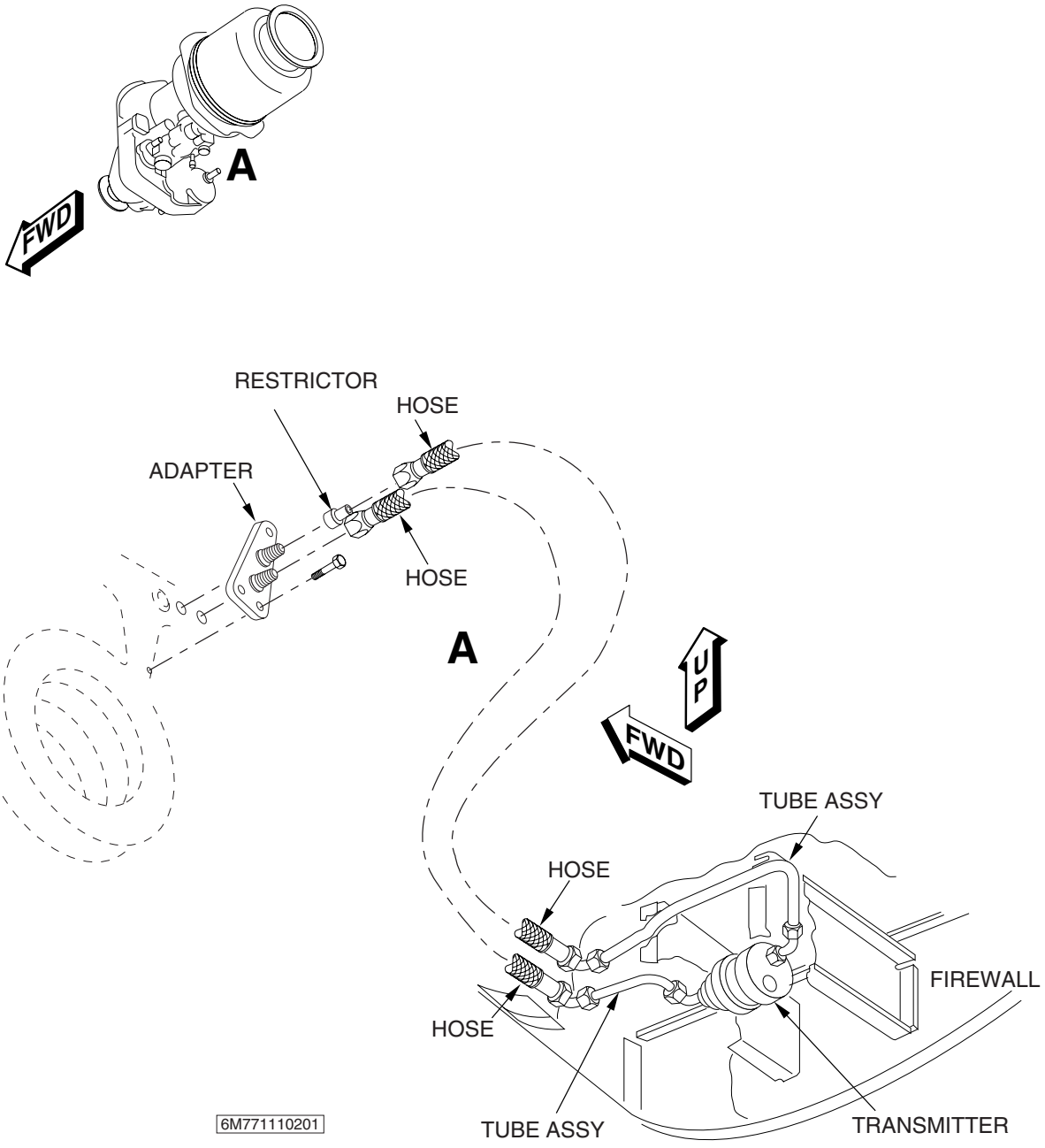
Torque Transmitter Indicating
T201-237, AT001-011, TC201-207
Figure 1

TORQUE TRANSMITTER INDICATING – MAINTENANCE PRACTICES

1. Removal/Installation – Torque Transmitter Indicating

A. Removal – Torque transmitter

NOTE: The following applies to aircraft models T201-237, AT001-011, TC201-207, and pre SB 77-001/2.



Torque Transmitter Indicating
T201-237, AT001-011, TC201-207
Figure 201

(1) Ensure electrical power is OFF.

(2) Remove access panel forward of main landing gear doors.

(3) Disconnect electrical connector from transmitter.

(4) Disconnect tube assembly from transmitter fittings.

(5) Remove elbow fittings.

(6) Cap tubes to prevent contamination.

(7) Remove retaining nut and washer securing transmitter.

(8) Remove transmitter from mounting bracket.

B. Installation – Torque transmitter

(1) Ensure electrical power is OFF.

(2) Position torque transmitter in mounting bracket and secure with washer and nut.

(3) Remove caps from tubes.

(4) Install elbow fittings at each end of transmitter.

(5) Connect tube assemblies to elbow fittings.

(6) Connect electrical connector and safety.

(7) Install access panel.

(8) Turn electrical power ON.

(9) Perform operational check on torque transmitter.

C. Removal – Torque Transmitter

NOTE: The following applies to aircraft models T238-275, 277-291, AT012-999, TC208-999, and post Service Bulletin 77-110/2.

(1) Ensure electrical power is OFF.

(2) Remove the engine right side cowling.

(3) Disconnect the electrical connector from transmitter.

(4) Remove the transmitter from adapter block tube assembly.

(5) Remove the union fitting from transmitter.

(6) Remove the three adapter block mount screws, and remove the adapter and transmitter from the engine.

(7) Remove the plug assembly (P/N 865840-2 from the adapter block.

(8) Remove the bolt assembly (P/N 3101355-1 from the adapter block and transmitter

(9) Withdraw the transmitter from the adapter block.

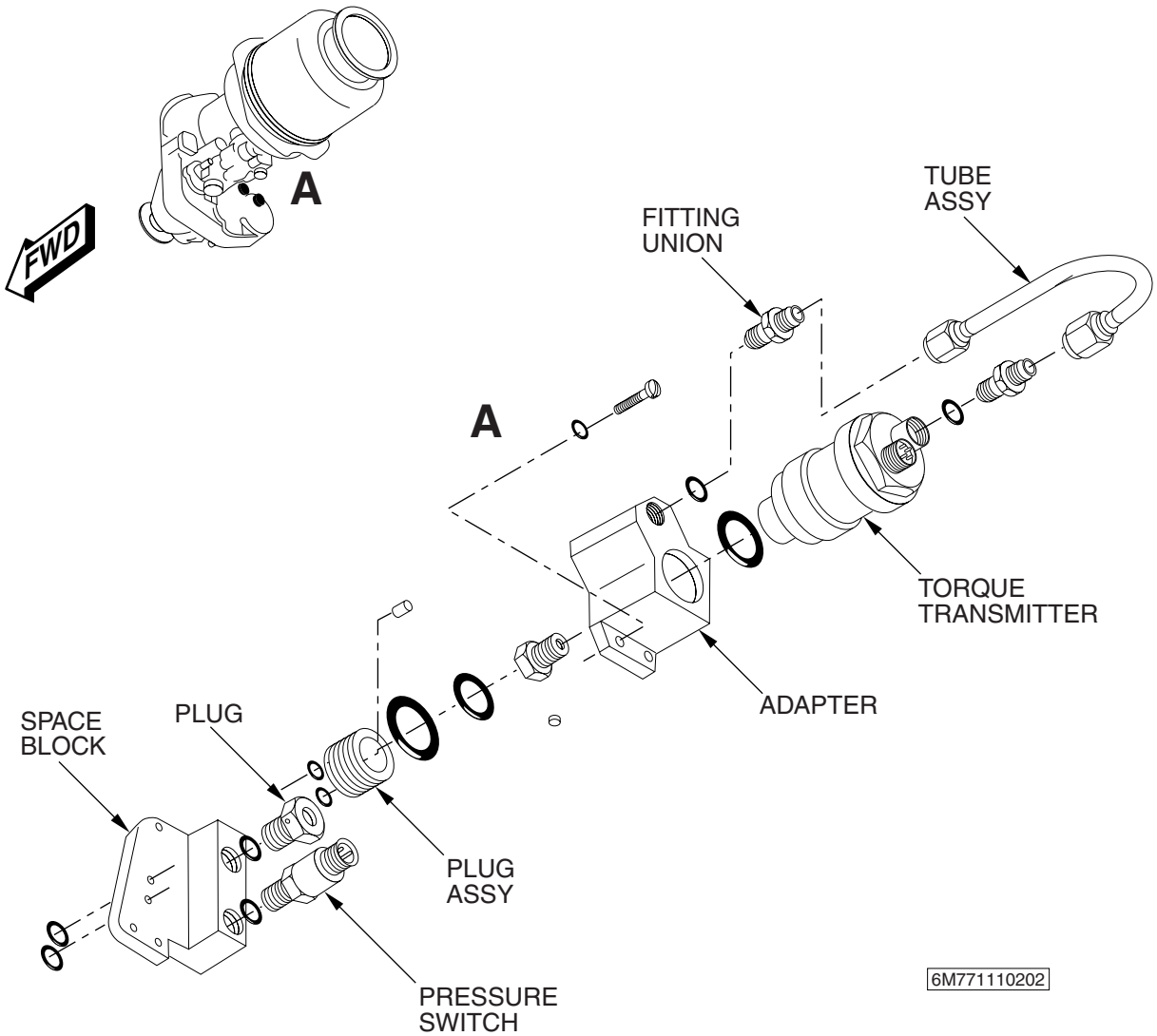
D. Installation – Torque Transmitter

(1) Ensure electrical power is OFF.

(2) Install O-ring (P/N S8316AD12) onto transmitter and position transmitter into the adapter block.

(3) Secure transducer to the adapter block with the bolt assembly (P/N 3101355-1). Torque bolt 30 to 40 inch-pounds (3.39 to 4.52 N·m).

(4) Install O-ring (P/N S9412 -024 and S9412-023) onto the plug assembly (P/N 865840-2, and install the plug assembly into the adapter, aligning pins on the plug to mate with recessed holes in adapter.



Torque Transmitter Indicating
T238-275, 277-291, AT012-999, TC208-999
Figure 202

(5) Install two O-rings (P/N MS29561-010) into recesses of the (P/N 865840-2) plug assembly.

(6) Install transmitter assembly to engine using three washers (P/N AN960C-10) and three screws (P/N MS21279-10), and torque 25 to 30 inch-pounds (2.82 to 3.39 N·m).

(7) Install the union fitting and O-ring (P/N 59413-554) into the transmitter. Torque union 40 to 50 inch-pounds (4.52 to 5.65 N·m).

(8) Install the tube assembly (P/N 3101343-1) to the transmitter and adapter block.

(9) Install the electrical connector, and safety.

(10) Install the engine right side cowling.

(11) Turn electrical power ON.

(12) Perform operational check on the torque transmitter.

E. Removal – Torque Indicator

(1) Ensure electrical power is OFF.

(2) Remove post light cap.

(3) Loosen indicator retaining clamp screw (lower right screw).

(4) Withdraw indicator from instrument panel.

(5) Disconnect the electrical connector from the indicator.

F. Installation – Torque Indicator

(1) Ensure electrical power is OFF.

(2) Connect the electrical connector to the indicator.

(3) Position the indicator in the instrument panel.

(4) Tighten indicator retaining clamp screw (lower right screw).

(5) Install indicator post light cap.

(6) Turn electrical power on.

(7) Perform operational check on indicator.

2. Adjustment/Test – Torque Transmitter

A. Materials and Equipment Required

(1) Dead weight or other pressure tester capable of supplying 0-80 psi pressure.

(2) 26 VAC electrical power.

(3) Zero adjusting tool, Edison P/N 41486.

(4) Scale span adjusting tool, Edison P/N 42015.

B. Adjustment/Test

NOTE: If there is reason to believe the torque transmitter system is not functioning properly, the following test may be performed to determine accuracy.

Calibrate the torque transmitter and indicator together as one unit. The aircraft wiring and power may be used.

(1) Remove the transmitter IAW this chapter.

(2) Connect a dead weight or equivalent tester to the transmitter pressure port.

(3) Connect the electrical connector to the transmitter and energize system power.

(4) Apply pressure to the transmitter, testing at each exact pressure value in Table 201. Before reading lightly tap both the transmitter and indicator. All indicator readings should be IAW Table 201 plus or minus 50 foot-pounds.

(5) If calibration is necessary, remove the plug cap from the transmitter.

(6) Set pressure to zero psi.

(7) Using the zero adjusting tool, turn the armature assembly inside the transmitter until the indicator reads approximately correct.

(8) Set pressure to 60 psi.

(9) Using both the zero adjusting tool and the scale span adjusting tool, hold the zero adjusting tool stationary while turning the scale span adjusting tool to give indicator reading within tolerance after tapping both the transmitter and indicator.

(10) Reduce the pressure to zero psi.

(11) Hold the scale span adjusting tool while turning the zero adjusting tool to give indicator reading within tolerance after tapping both the transmitter and indicator.

(12) Repeat steps 6 through 11 until both readings are within tolerance with tools removed before reading indicator.

(13) Check readings at all test values in Table 201.

(14) Install plug cap and safety.

(15) Disconnect pressure tester.

(16) Install the transmitter per installation instructions in this chapter.

NOTE: If transmitter leaks or cannot be calibrated within tolerance, it must be scrapped since no repairs are possible.

TORQUE PRESSURE APPLIED TO TRANSMITTER	TORQUE INDICATOR READING (FT. LBS.)
5 PSI	0 FT. LBS.
10 PSI	0 FT. LBS.
15 PSI	232 FT. LBS.
20 PSI	462 FT. LBS.
25 PSI	694 FT. LBS.
30 PSI	924 FT. LBS.
35 PSI	1155 FT. LBS.
40 PSI	1385 FT. LBS.
45 PSI	1618 FT. LBS.
50 PSI	1850 FT. LBS.
55 PSI	2080 FT. LBS.
60 PSI	2310 FT. LBS.

Torque Indicator Readings
Table 201

3. Inspection/Check – Torque Transmitter Indicating

A. Inspect torque transmitter as follows:

(1) Inspect torque transmitter for security, general condition, and evidence of leakage.

(2) Inspect torque indicators for security of mounting and general condition.

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TORQUE TRANSDUCER INDICATING SYSTEM – DESCRIPTION AND OPERATION

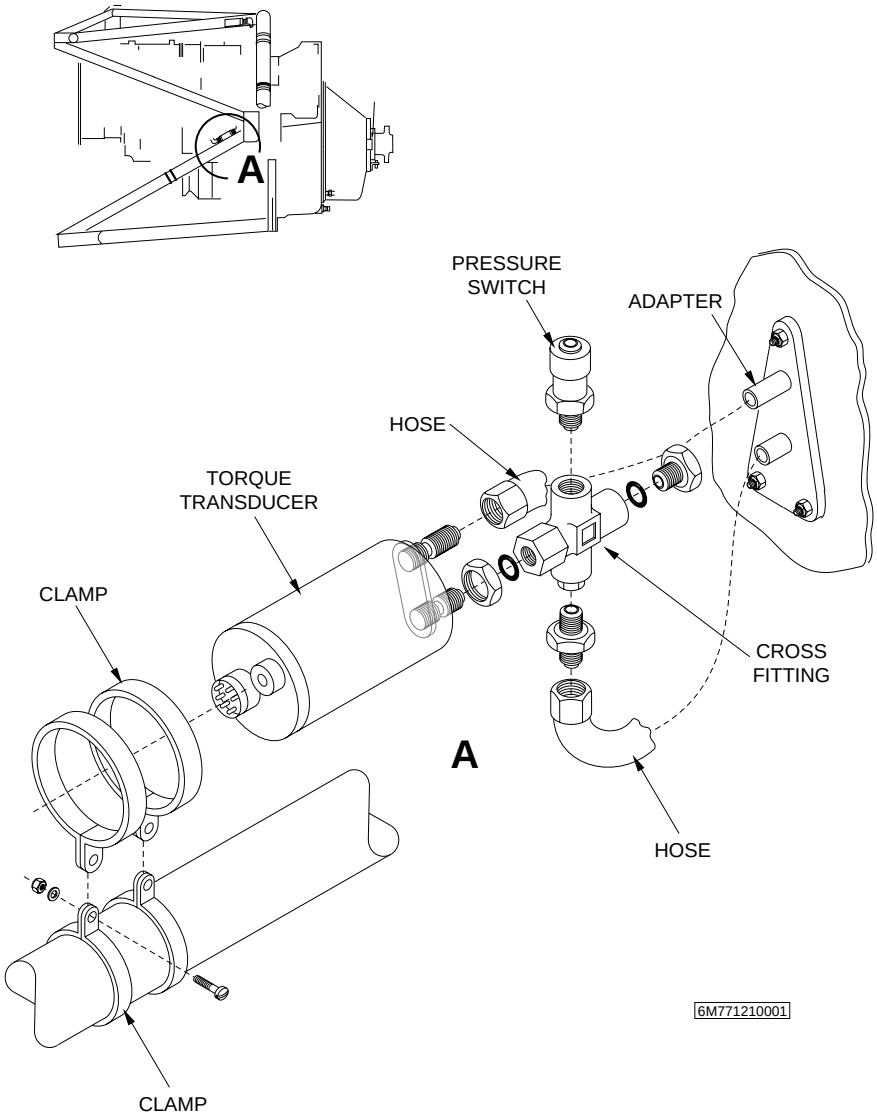
1. General

A. Torque Transducer Indicating

(1) A torque transducer is attached to the lower outboard truss mount of each engine. Hoses connect the transducer to the pressure and case negative pressure ports at the right rear reduction gear case.

(a) The transducer is calibrated to compensate the raw torque pressure and case negative pressure differential to an accurate electrical signal that is indicated in percent of torque on the center instrument panel mounted torque indicator.

(b) System power is supplied from the left or right 26 VAC bus through a one amp circuit breaker.



Torque Transducer
Figure 1

1. Removal/Installation – Torque Transducer Indicating

A. Removal – Torque Transducer

(1) Ensure electrical power is OFF.

(2) Remove engine right side cowling.

(3) Disconnect electrical connectors from transducer and pressure switch.

(4) Remove hoses from transducer and cross fitting.

(5) Cap openings to prevent contamination.

(6) Loosen retaining clamp bolts, and remove transducer.

(7) Remove cross fitting and pressure switch from transducer.

(8) Remove transducer.

B. Installation – Torque Transducer

(1) Ensure electrical power is OFF.

(2) Install cross fitting and pressure switch on transducer torque pressure connection. Use new O-ring (P/N MS28778-3).

(3) Position transducer in mounting clamps, and tighten mounting clamp bolts.

(4) Connect case pressure hose to transducer.

(5) Connect torque pressure hose to cross fitting reducer.

(6) Connect electrical connections to transducer and pressure switch.

(7) Turn electrical power ON.

(8) Perform operational check on the transducer IAW instructions in Adjustment/Test section of this chapter.

(9) Install engine right side cowling.

C. Removal – Torque Indicator

(1) Ensure electrical power is OFF.

(2) Remove post light cap.

(3) Loosen indicator retaining clamp screw (lower right screw).

(4) Withdraw indicator from instrument panel.

(5) Disconnect the electrical connector from indicator.

D. Installation – Torque Indicator

(1) Ensure electrical power is OFF.

(2) Connect the electrical connector to the indicator.

(3) Position the indicator in the instrument panel.

(4) Tighten the indicator retaining clamp screw (lower right screw).

(5) Install indicator post light cap.

(6) Turn electrical power ON.

(7) Perform operational check on the indicator.

3. Adjustment/Test – Torque Transducer Indicating

A. Adjustment – Torque Transducer

NOTE: A dead weight tester with suitable piston and weights to cover 0-80 PSI (0 to 6.624kg/cm^2) range or equivalent is required to complete the adjustment/test.

Calibrate the torque transducer and indicator while installed on the aircraft.

(1) Connect a GPU to the aircraft.

(2) Turn battery switches ON.

(3) Connect all system components to aircraft wiring.

(4) Engage torque indicator circuit breaker.

(5) Connect dead weight tester to transducer high pressure port, bleeding all air possible.

(6) Open the transducer low pressure port to atmosphere.

(7) Remove adjustment access plug.

(8) Obtain AiResearch Form DSC-8022 for the engine being calibrated.

(a) Plot pressure Vs. percent torque at 3,000 inch-pounds and 35,650 inch-pounds (100%).

(b) Connect these two points with a straight line to pick off intermediate values.

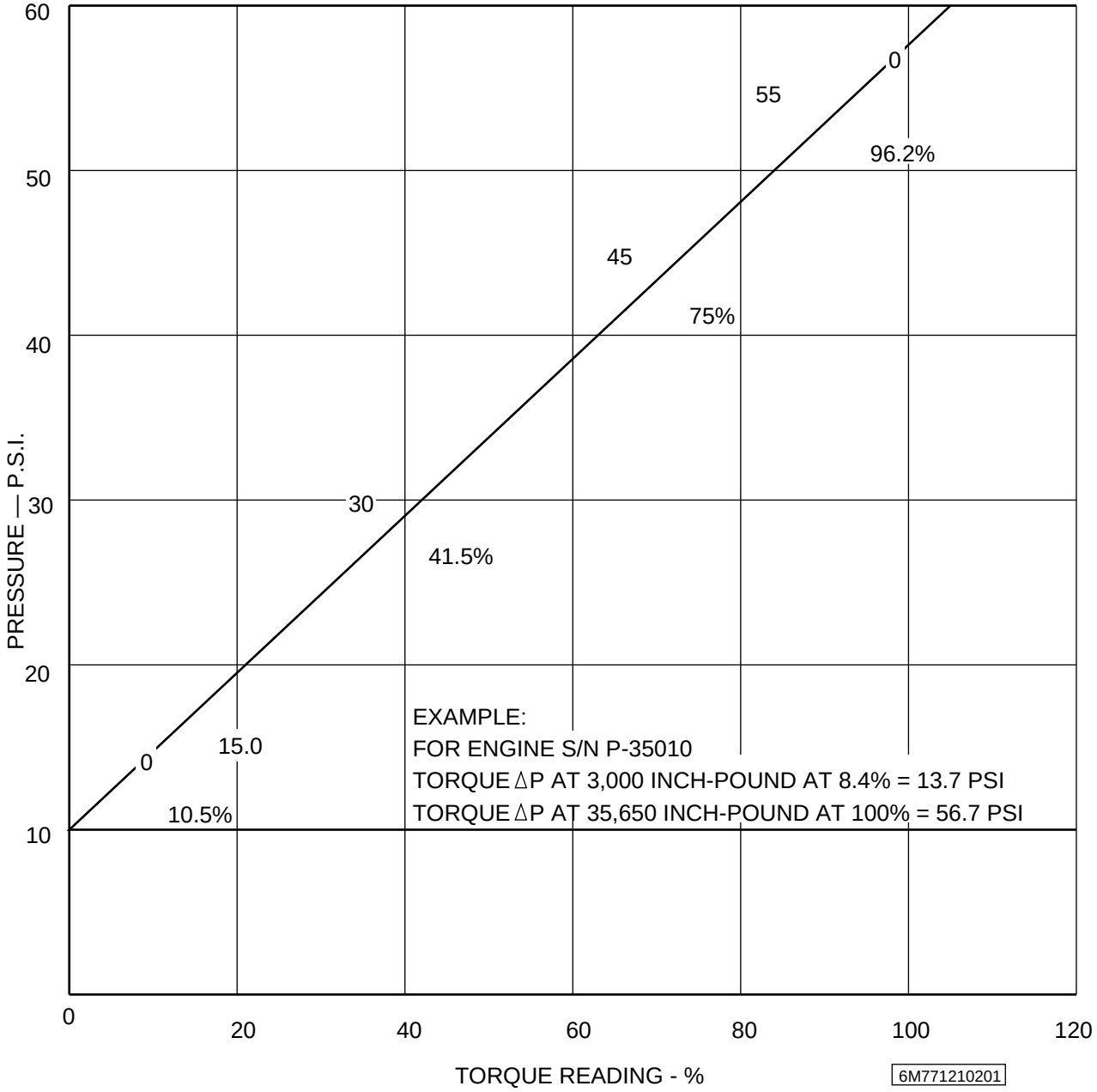
(9) Select weights to apply pressure equivalent to a value of torque as near 100 percent as possible. With this pressure applied, adjust screw marked "Z" to give a torque reading corresponding to the pressure.

NOTE: Adjustment screws are reverse sensing to the markings on the indicator. Clockwise rotation of the adjustment screws increase torque indications, and counterclockwise rotation decreases torque indications.

(10) Select weight to apply a pressure equivalent to a value of torque as near 8.4% as possible. With this pressure applied, adjust screw marked "R" to give a torque reading corresponding to the pressure.

(11) Recheck step 6 and readjust if necessary. Check at least two intermediate points.

(12) Install adjustment plug, and safety.



Torque Transducer Adjustment Chart Figure 201

(13) Connect all transducer lines.

NOTE: Figure 201 examples are as follows:

56.7 PSI = 100%
A convenient weight gives 55 PSI; from the plot this equals 96.2% set IAW step 6.

13.7 PSI = 8.4%
A convenient weight gives 15 PSI; from the plot this equals 10.5% set IAW step 7.

Recheck 96.2%.
Check 30 PSI = 41.5% and 45 PSI = 75%.

3. Servicing – Torque Transducer Indicating

A. Inspect torque transducer as follows:

(1) Inspect torque transducer for security of mounting, general condition, and evidence of leakage.

(2) Inspect torque indicators for security of mounting and general condition.

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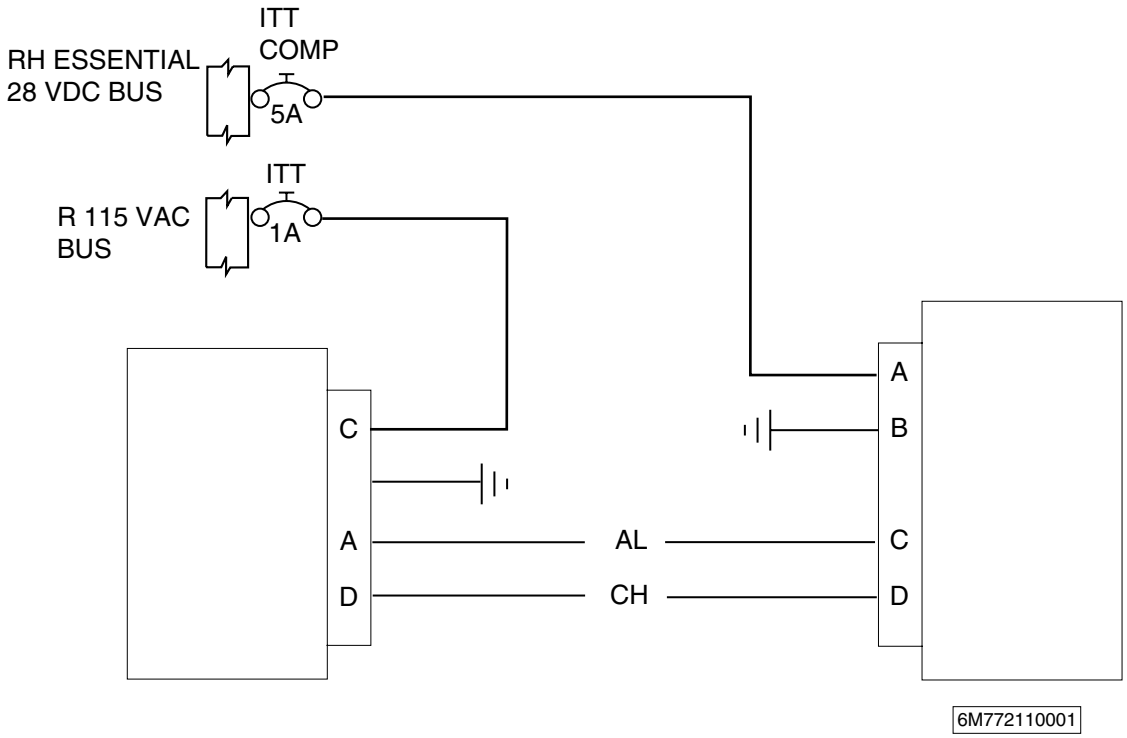
INTERSTAGE TURBINE TEMPERATURE INDICATING – DESCRIPTION AND OPERATION

1. General

A. The interstage turbine temperature (ITT) indicating system consists of an engine temperature indicator for each engine installed in the instrument panel center section, an engine furnished thermocouple harness, and compensating resistor.

(1) The indicators are powered by the left and right 115 VAC bus through a one amp circuit breaker. The indicator lights operate on 28 VDC from this left or right essential bus through a five amp center instrument light circuit breaker.

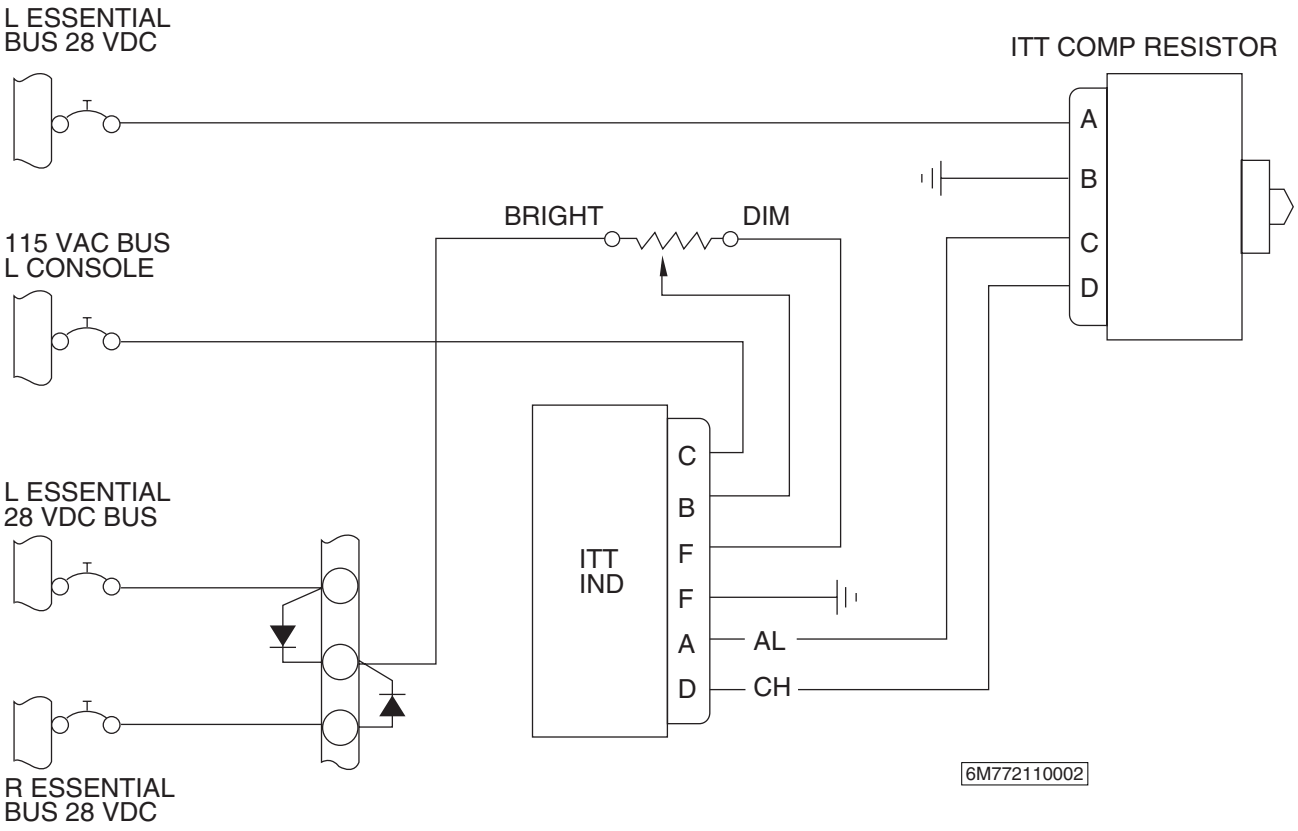
(2) The ITT compensating resistor is powered from the left and right 28 VDC essential bus through a five amp circuit breaker. The indicator receives an electrical signal from the thermocouple harness and indicates engine temperature in degrees celsius. With 24-28 VDC supplied to the compensating resistor, a precise millivoltage provides the corrected ITT reading so that indicated temperature reaches the temperature limit when the engine reaches its horsepower limit.



ITT Indicating
Figure 1

INTERSTAGE TURBINE TEMPERATURE INDICATING – DESCRIPTION AND OPERATION

(3) Aircraft that complied with Service Bulletin 77-003 and T279, 282-291; AT062-999, and TC232. 234-999 use an ITT indicator that incorporates a digital read out as well as a needle indication. A dimming control rheostat for the digital read out is located above the indicators in the instrument panel.



ITT Temperature Schematic
Figure 2

INTERSTAGE TURBINE TEMPERATURE INDICATING – MAINTENANCE PRACTICES

1. Removal/Installation – Interstage Turbine Temperature (ITT) Indicating

A. Removal – ITT Indicator

(1) Ensure electrical power is OFF.

(2) Loosen indicator retaining clamp screw (lower right screw).

(3) Withdraw indicator from instrument panel.

(4) Disconnect electrical connector from indicator.

B. Installation – ITT Indicator

(1) Ensure electrical power is OFF.

(2) Connect electrical connector to indicator.

(3) Position indicator in instrument panel.

(3) Tighten indicator retaining clamp screw (lower right screw).

(5) Turn electrical power ON.

(6) Perform operational check on engine temperature indicating system.

2. Inspection/Check – Interstage Turbine Temperature Indicating

A. Inspect interstage turbine temperature (ITT) indicating system as follows:

(1) Inspect ITT indicator for security of mounting and general condition.

(2) Inspect torque indicators for security of mounting and general condition.

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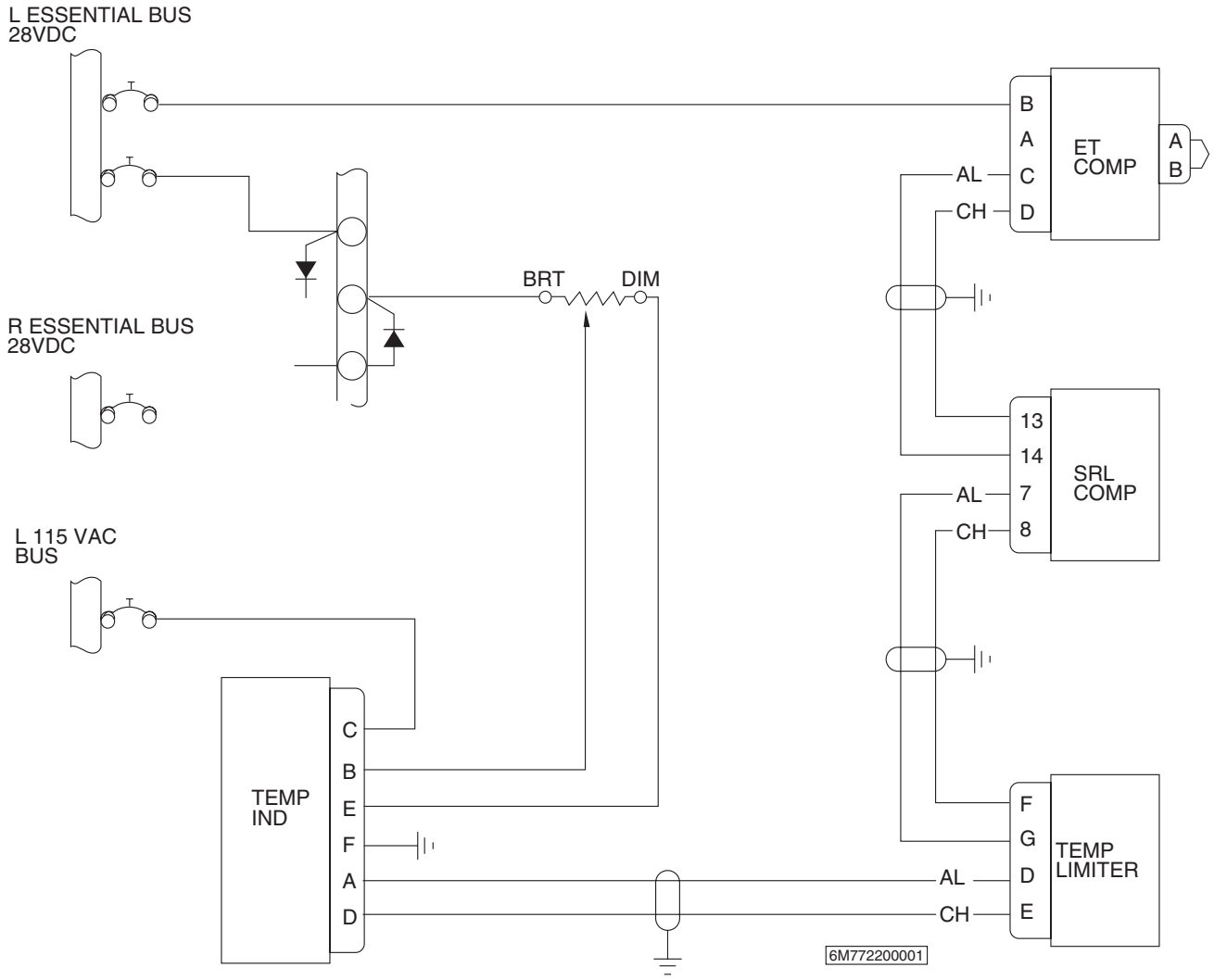
EXHAUST GAS TEMPERATURE INDICATING – DESCRIPTION AND OPERATION

1. General

A. Exhaust gas temperature (EGT) indicating uses a single red line system that operates whenever the engine speed is above 80 percent revolutions per minute (RPM) and the controller is turned on.

(1) The controller receives several input signals from the engine. The input signals are: A temperature signal from the exhaust thermocouple; A pressure ratio from the delta P/P transducer; An engine speed signal from a monopole within the propeller governor; and A temperature signal from the engine inlet temperature sensor.



Exhaust Gas Temperature Indicating
Figure 1

(2) When the system is on and engine speed is above 80 percent RPM, these variable input signals to the controller result in a conditioned EGT signal to the indicating system. The controller output signals also go to the monitoring and temperature limiting systems.

(3) The center instrument panel mounted engine temperature (ET) indicator will indicate the compensated temperature with the controller off, or with the engine speed below 80 percent RPM. With the compensator circuit breaker disengaged, the raw EGT will be indicated.

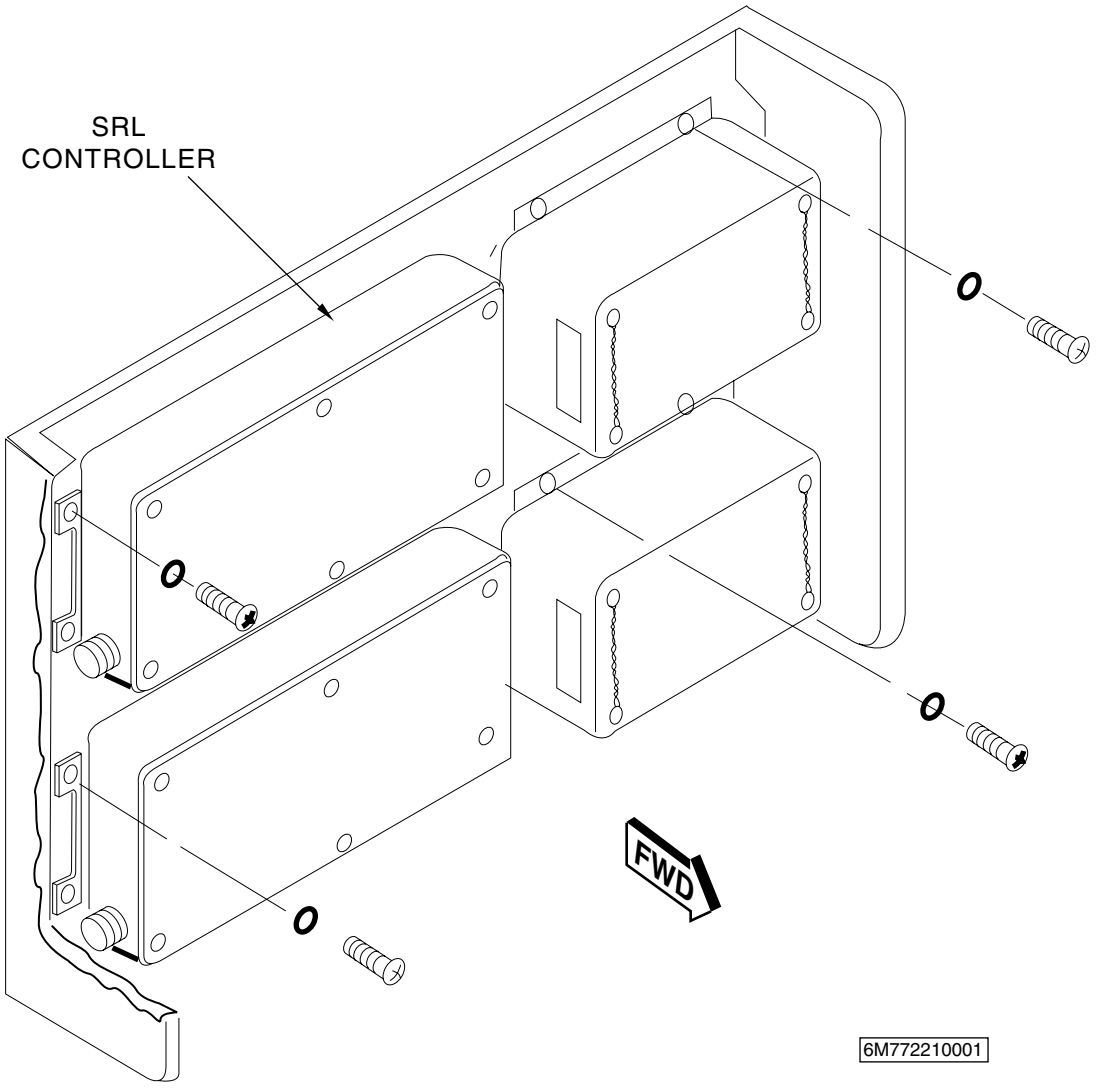
(4) The temperature limiter is wired to the controller internal 90 percent speed switch. With the controller on, engine speed above 90 percent and the conditioned temperature at 650 degrees celsius, the temperature limiter sends a signal to the engine fuel bypass valve through the interface assembly which illuminates the fuel bypass indicator lights on the center instrument panel above the temperature indicator.

SINGLE RED LINE CONTROLLER – DESCRIPTION AND OPERATION

1. General

A. A single red line (SRL) controller for each engine is mounted in the nose section below the right inverter.

(1) The SRL function of the controller is to furnish a conditioned exhaust gas temperature signal to the indicating and controlling circuitry, allowing the engine to be operated to a fixed value of 650°C under cruise conditions from 96 to 100% RPM.



Single Red Line (SRL) Controller
Figure 1

SINGLE RED LINE CONTROLLER – DESCRIPTION AND OPERATION

(2) The SRL controller input signals are: 28VDC from the left or right essential bus through one amp circuit breakers; engine inlet air temperature; exhaust gas temperature from the thermocouple harness through the temperature compensator; air pressure density from the delta P/P transducer; and engine speed from the propeller governor monopole. The controller operates when the system is on and engine speed is above 80% RPM. Using the variable input signals, the controller furnishes a conditioned exhaust gas temperature to the indicating, controlling, and monitoring circuits.

(3) The indicating circuit provides continuous indication of engine temperature regardless of controller or compensator operation.

(4) The temperature limiter controls the maximum engine temperature by sending a signal to the fuel bypass valve and illuminates the instrument panel mounted fuel bypass light when the system is on and engine speed is above 90% RPM.

(5) The engine temperature monitoring system illuminates the annunciator panel SRL OFF light when the conditioned exhaust gas temperature is within 15°C of the compensated exhaust gas temperature. This indication warns the pilot that the SRL system is malfunctioning. If in flight, refer to AFM for SRL OFF operation, if on, the ground system must be repaired prior to flight.

SINGLE RED LINE CONTROLLER – MAINTENANCE PRACTICES

1. Removal/Installation – Single Red Line (SRL) Controller

A. Removal – SRL Controller

(1) Ensure electrical power is OFF.

(2) Remove controller protective cover screen.

(3) Disconnect electrical connector from controller.

(4) Remove bolts and washers securing controller.

(5) Remove controller.

B. Installation – SRL Controller

(1) Ensure electrical power is OFF.

(2) Position controller on mounting bracket and secure with bolts and washers.

(3) Connect electrical connector to controller.

(4) Install controller protective cover screen.

(5) Turn electrical power ON.

(6) Operational check engine temperature indicating system.

2. Inspection/Check – Single Red Line (SRL) Controller

A. Inspect Single Red Line (SRL) Controller as follows:

(1) Inspect SRL controller for general condition of controller and wiring, and security of mounting.

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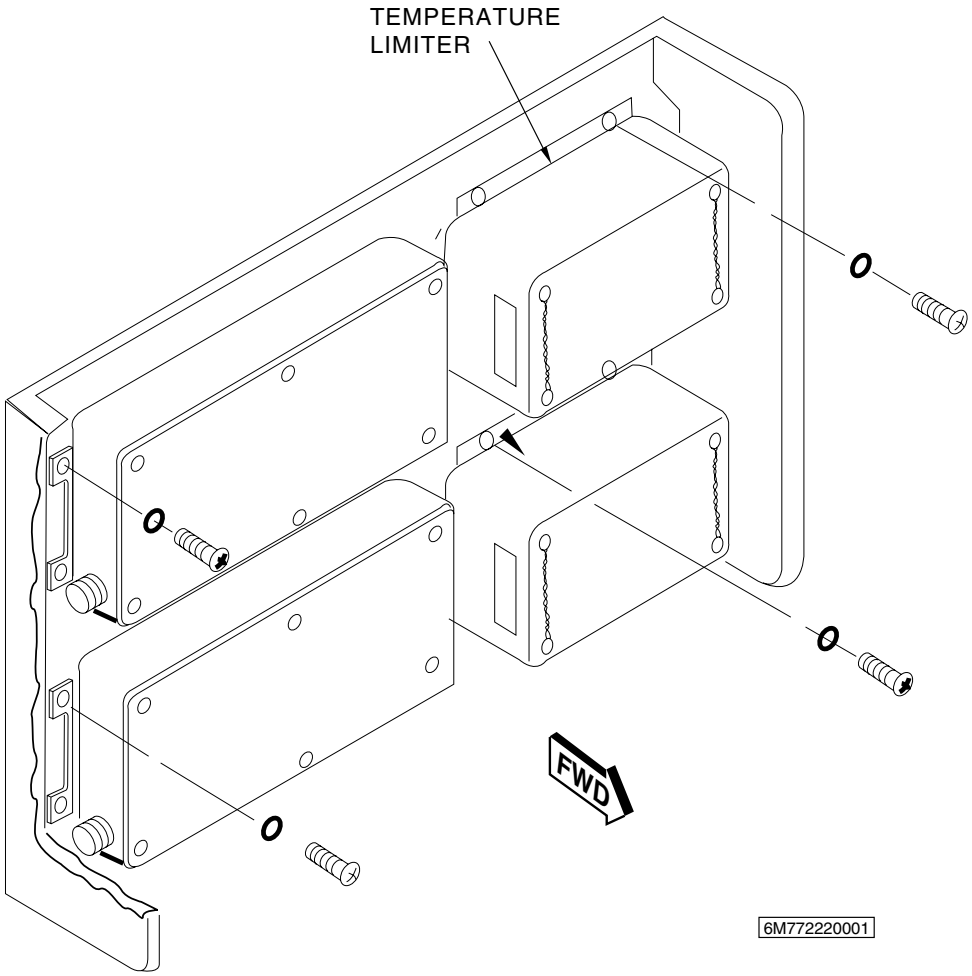
TEMPERATURE LIMITER – DESCRIPTION AND OPERATION

1. General

A. The temperature limiter is located in the aircraft nose section forward of the pressure bulkhead beside the SRL controller.

(1) The temperature limiter receives input signals from the controller and sends signals to the engine temperature indicator and to the SRL/fuel bypass interface assembly. When the maximum engine temperature is reached, the limiter signals the interface assembly, which opens the engine fuel bypass valve, and illuminates the fuel bypass light on the instrument panel using 28VDC power from the nonessential bus.

(2) The temperature limiter circuit also includes a temperature limit test switch on the left side console. The test switch used to ground check the temperature limiter circuit activates the temperature limiter when the SRL system is on and engine speed is above 90% RPM. The limiter then sends a bypass signal to the interface assembly, which in turn opens the fuel bypass valve and illuminates the bypass light. A drop in fuel flow will result when the fuel bypass valve opens.



Temperature Limiter
Figure 1

TEMPERATURE LIMITER – MAINTENANCE PRACTICES

1. Removal/Installation – Temperature Limiter

A. Removal – Temperature Limiter

(1) Ensure electrical power is OFF.

(2) Remove temperature limiter protective cover screen.

(3) Disconnect electrical connector from temperature limiter.

(4) Remove bolts and washers securing temperature limiter.

B. Installation – Temperature Limiter

(1) Ensure electrical power is OFF.

(2) Position temperature limiter on mounting bracket and secure with bolts and washers.

(3) Connect electrical connector to temperature limiter.

(4) Install protective cover screen.

(5) Turn electrical power ON.

(6) Operational check engine temperature indicating system.

2. Inspection/Check – Temperature Limiter

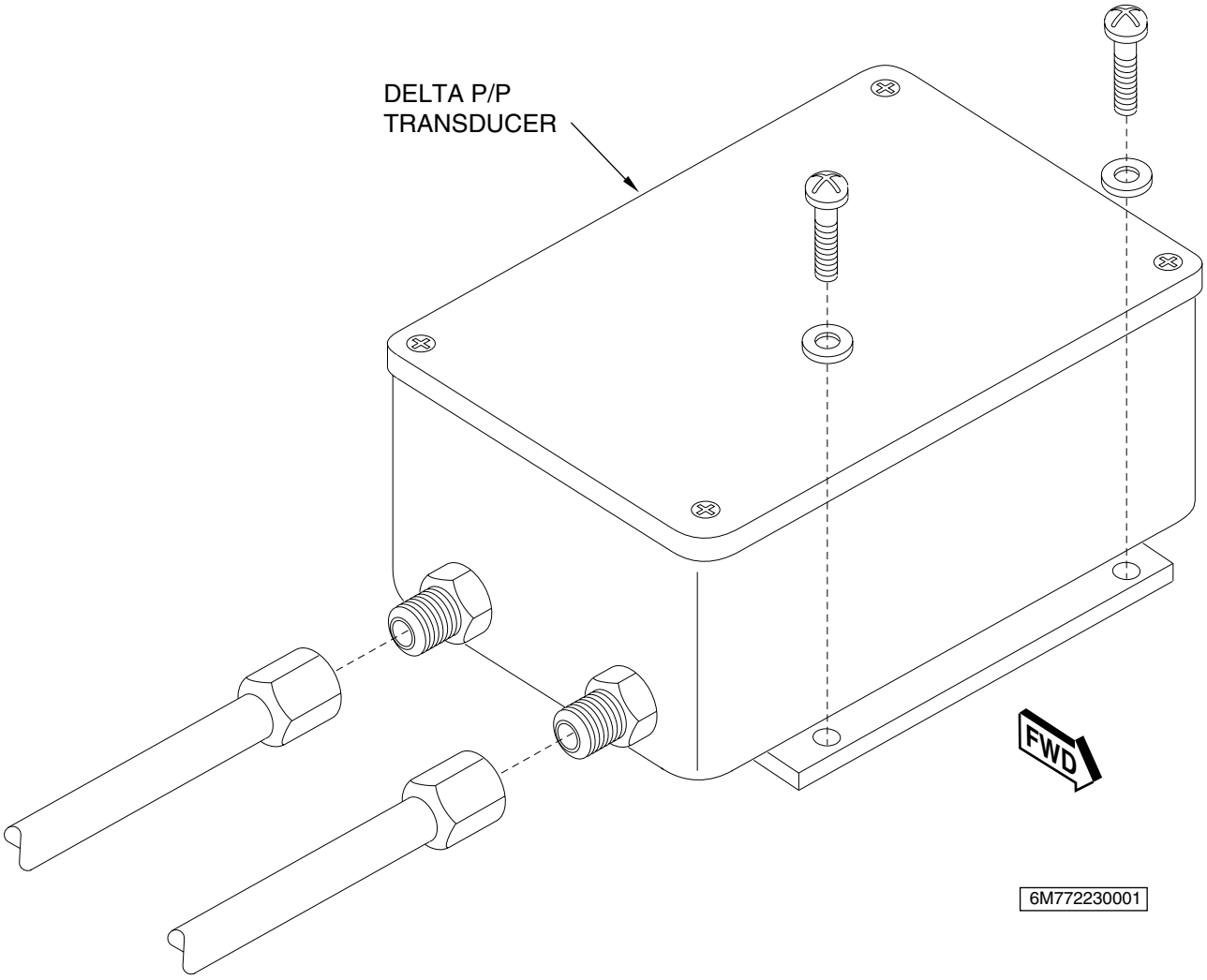
A. Inspect Temperature Limiter as follows:

(1) Inspect temperature limiter for general condition of limiters and wiring, and security of mounting.

DELTA P/P TRANSDUCER – DESCRIPTION AND OPERATION

1. General

A. A delta P/P transducer for each engine is mounted in the top access of each wing inboard of the nacelles. The transducer receives signals from the engine P2 and P5 sensors and transmits the engine air density signal to the single red line controller. The transducer is powered from the left and right essential bus through a one amp circuit breaker.



Delta P/P Transducer
Figure 1

DELTA P/P TRANSDUCER – MAINTENANCE PRACTICES

1. Removal/Installation – Delta P/P Transducer

A. Removal – Delta P/P Transducer

(1) Ensure electrical power is OFF.

(2) Remove access panel (top of wing, inboard of the nacelle).

(3) Disconnect the electrical connector.

(4) Disconnect the two instrument air tube assemblies from the transducer.

(5) Remove mounting screws, and remove the transducer.

B. Installation – Delta P/P Transducer

(1) Ensure electrical power is OFF.

(2) Position the transducer on the mounting bracket, and install the four screws and washers.

(3) Connect the instrument air tube assemblies to the transducer.

(4) Connect the electrical connector, and safety.

(5) Turn electrical power ON.

(6) Operational check system.

(7) install access panel.

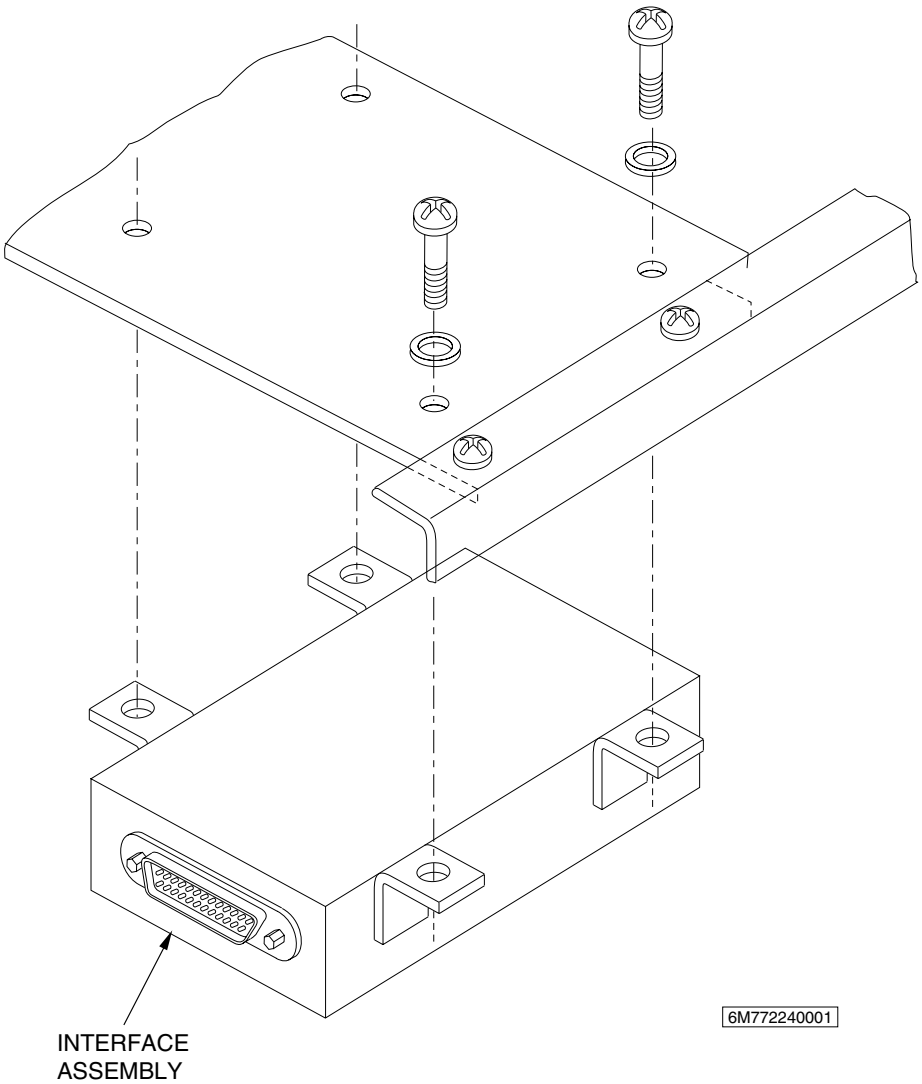
INTERFACE ASSEMBLY, SRL, AND FUEL BYPASS – DESCRIPTION AND OPERATION

1. General

A. The SRL and fuel bypass interface assembly is located under the floorboard forward of the copilot's rudder pedals. The interface assembly is powered from the right nonessential 28VDC bus, and is controlled by the engine temperature warning one amp circuit breaker.

(1) A signal from the SRL controller to the interface assembly operates the annunciator SRL OFF light.

(2) An overtemperature signal from the temperature limiter to the interface assembly activates the fuel bypass valve circuit and illuminates the fuel bypass light on the instrument panel.



Interface Assembly, SRL, and Fuel Bypass
Figure 1

INTERFACE ASSEMBLY, SRL. AND FUEL BYPASS – MAINTENANCE PRACTICES

1. Removal/Installation – Interface Assembly, SRL, and Fuel Bypass

A. Removal – Interface Assembly

(1) Ensure electrical power is OFF.

(2) Remove copilot's seat.

(3) Remove carpet forward of copilot's seat.

(4) Remove floorboard under copilot's rudder pedals.

(5) Disconnect electrical connector from interface assembly.

(6) Support interface assembly, and remove screws and washers securing interface assembly (located forward of copilot's rudder pedals).

(7) Remove interface assembly.

B. Installation – Interface Assembly

(1) Ensure electrical power is OFF.

(2) Position interface assembly on bottom side of support bracket, and secure with screws and washers.

(3) Connect electrical connector to interface assembly.

(4) Turn electrical power ON.

(5) Operational check engine temperature indicating system.

(6) Install floorboard under copilot's rudder pedals.

(7) Install carpet forward of copilot's seat.

(8) Install copilot's seat.

2. Inspection/Check – Interface Assembly, SRL, and Fuel Bypass

A. Inspect Interface Assembly as follows:

(1) Inspect interface assembly for general condition and security of mounting.

ENGINE TEMPERATURE (ET) INDICATOR – DESCRIPTION AND OPERATION

1. General

A. An internally lighted engine temperature indicator that incorporates a digital read out is mounted on the center instrument panel above the torque indicator for each engine. The indicator will read raw, compensated, or conditioned engine temperature.

(1) Raw exhaust gas temperature (EGT) will be indicated when the compensator and SRL system is OFF.

(2) Compensated temperature will be indicated with the compensator circuit breaker engaged and the SRL system OFF.

(3) Conditioned EGT will be indicated when the compensator and SRL systems are on and engine speed is above 80% RPM.

(4) The engine temperature indicator circuit is powered from the left or right 115 VAC bus through a one amp circuit breaker.

(5) The engine temperature indicator light circuit is powered from the left or right 28 VDC bus through a five amp circuit breaker, and a dimming control potentiometer is located above the ET indicators.

ENGINE TEMPERATURE (ET) INDICATOR – MAINTENANCE PRACTICES

1. Removal/Installation – Engine Temperature (ET) Indicator

A. Removal – Engine Temperature (ET) indicator

(1) Ensure electrical power is OFF.

(2) Loosen indicator retaining clamp screw (lower right screw).

(3) Remove indicator from instrument panel.

(4) Disconnect electrical connector from ET indicator.

B. Installation – Engine Temperature (ET) indicator

(1) Ensure electrical power is OFF.

(2) Connect electrical connector to ET indicator.

(3) Position indicator in instrument panel.

(4) Tighten indicator retaining clamp screw (lower right screw).

(5) Turn electrical power ON.

(6) Perform operational check on engine temperature (ET) indicating system.

2. Inspection/Check – Engine Temperature (ET) Indicator

A. Inspect Engine Temperature (ET) indicator as follows:

(1) Inspect ET indicator for general condition and security of mounting.

FUEL BYPASS INDICATOR LIGHT – DESCRIPTION AND OPERATION

1. General

A. A fuel bypass indicator light for each engine is mounted on the center instrument panel above the engine temperature indicators. An over temperature signal from the temperature limiter to the interface assembly activates the fuel bypass circuit and illuminates the fuel bypass light.

(1) The fuel bypass light receives electrical power from the left or right essential bus through a one amp circuit breaker. The circuit also includes a fuel bypass light dim switch located above the fuel bypass lights.

FUEL BYPASS INDICATOR LIGHT – MAINTENANCE PRACTICES

1. Removal/Installation – Fuel Bypass Indicator Light

A. Removal – Fuel Bypass Indicator Light

(1) Ensure electrical power is OFF.

(2) Remove instrument panel glareshield.

(3) Cut the electrical wires.

(4) Loosen the two light cover screws, and remove cover from rear of light.

(5) Withdraw light from the front of the instrument panel.

B. Installation – Fuel Bypass Indicator Light

(1) Ensure electrical power is OFF.

(2) Pull electrical wires through light hole in instrument panel.

(3) Place light cover over wires; then, solder wires to the fuel bypass light assembly (see Wiring Manual, Chapter 76).

(4) Position light into instrument panel.

(5) Install light cover, and tighten the two retaining screws.

(6) Turn electrical power ON.

(7) Perform operational check of fuel bypass light.

(8) Install instrument panel glareshield.

C. Removal – Fuel Bypass Dim Switch

(1) Ensure electrical power is OFF.

(2) Remove glareshield.

(3) Remove dim switch knob.

(4) Remove switch retaining nut and washer.

(5) Withdraw switch from back side of instrument panel.

(6) Cut or unsolder wires from switch.

D. Installation – Fuel Bypass Dim Switch

(1) Ensure electrical power is OFF.

(2) Solder electrical wires to switch (refer to Wiring Manual, Chapter 76).

(3) Position switch in instrument panel.

(4) Install retaining nut and washer.

(5) Install switch knob.

(6) Turn electrical power ON.

(7) Perform operational check fuel bypass lights.

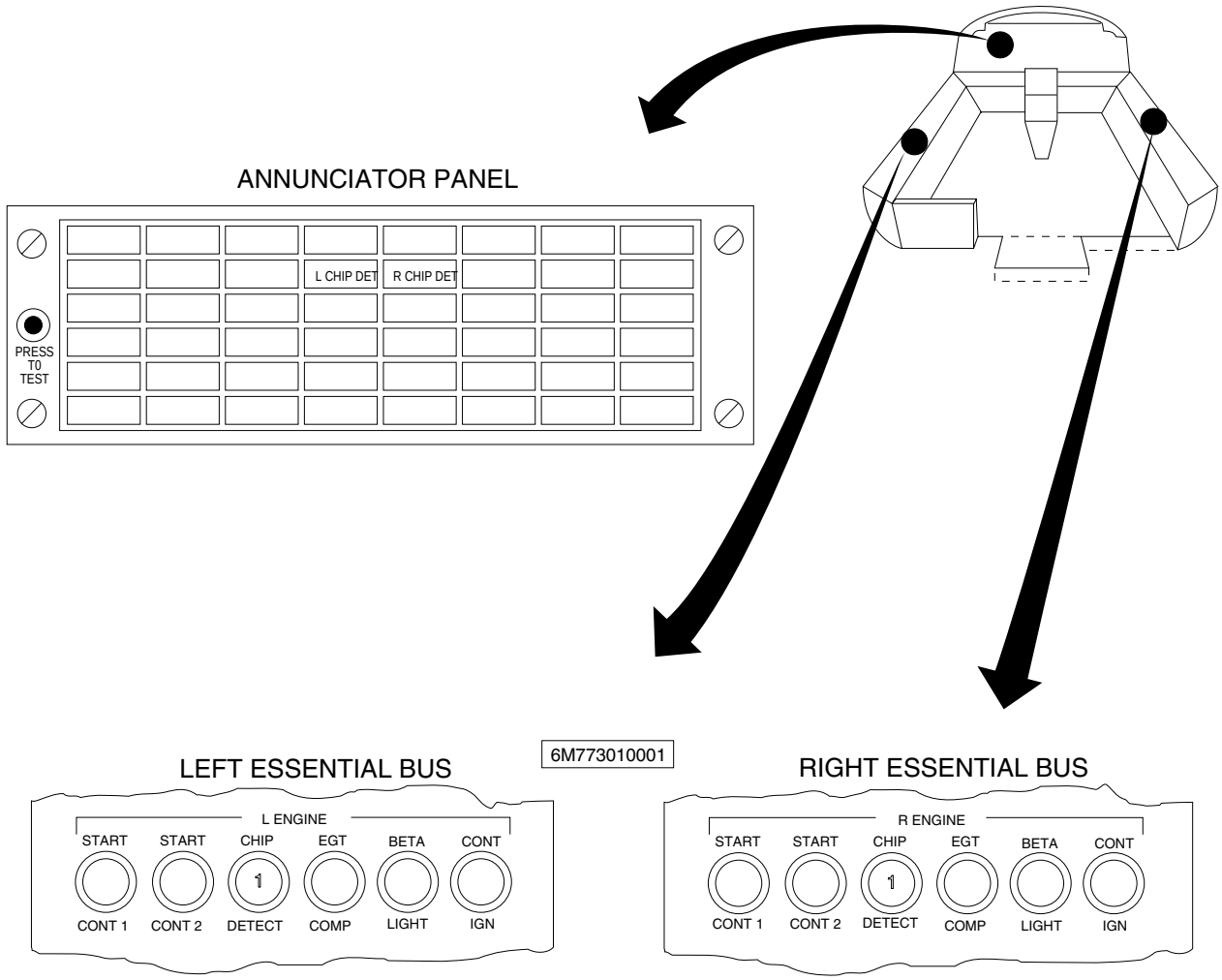
MAGNETIC DRAIN PLUG (CHIP DETECTOR) – DESCRIPTION AND OPERATION

1. General

A. Magnetic Drain Plug (Chip Detector)

(1) A chip detector system is installed on all SA226 aircraft. The system furnishes a visual warning indication of impending internal engine failure.

(2) A magnetic chip detector plug is installed in the lower front center of each engine reduction gear case. The plug attracts ferrous metal particles which may be present in the lower gear sump area.

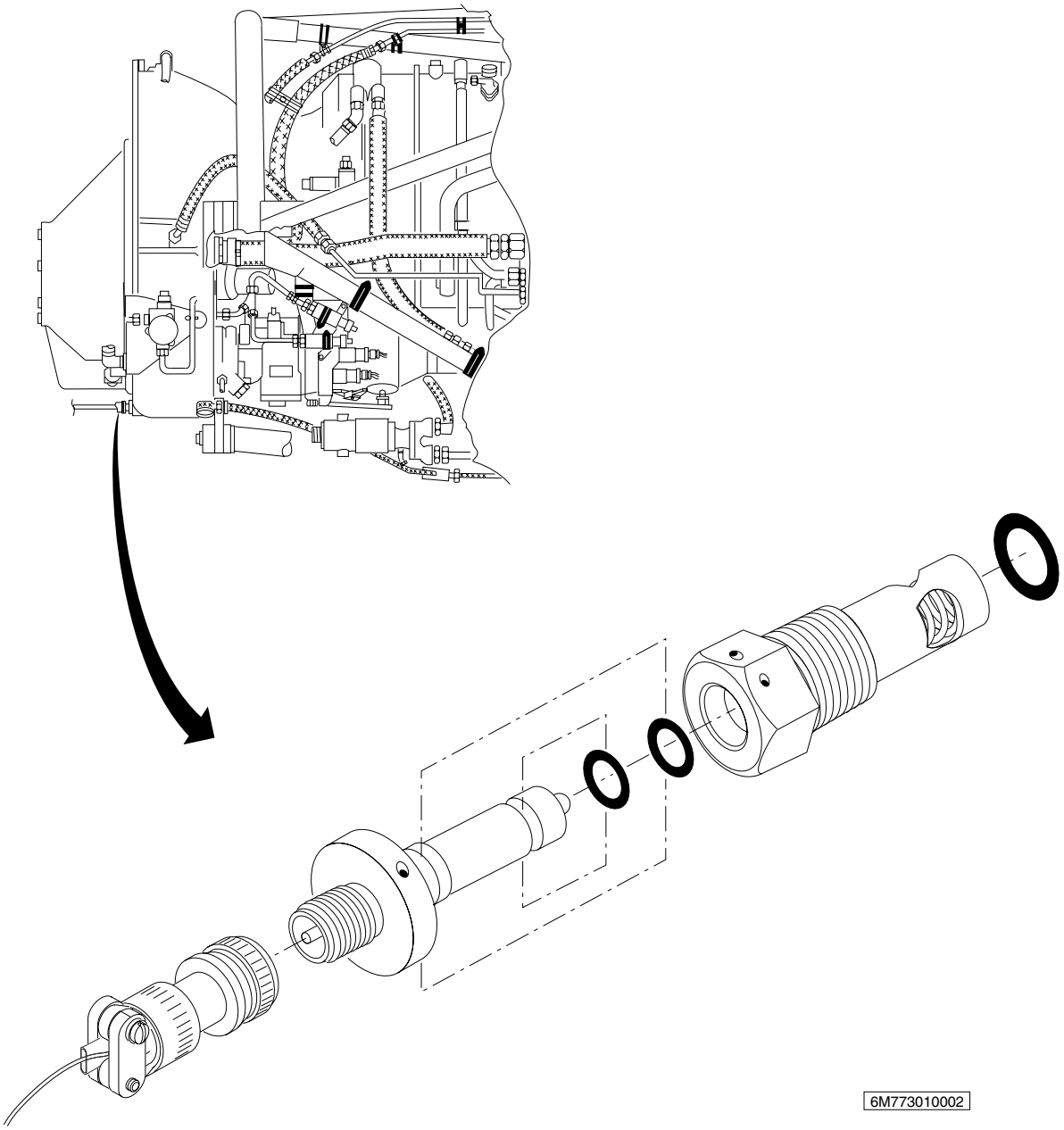


Magnetic Chip Detector (Cockpit)
Figure 1

MAGNETIC DRAIN PLUG (CHIP DETECTOR) – DESCRIPTION AND OPERATION

(3) The magnetic center post of the plug is electrically insulated from the surrounding housing. Metal particles attracted by the plug short out the insulator and provide a ground circuit for the chip detector warning light in the annunciator panel.

(4) The system is powered by 28 VDC from the left and right essential buses controlled by one (1) amp circuit breakers.



Magnetic Chip Detector
(Engine)
Figure 2

MAGNETIC DRAIN PLUG (CHIP DETECTOR) – MAINTENANCE PRACTICES

1. Removal/Installation – Magnetic Drain Plug (Chip Detector)

A. Removal – Chip Detector

(1) Ensure electrical power is OFF.

(2) Remove lower forward cowling.

(3) Disconnect electrical connector from chip detector.

(4) Remove chip detector plug and packings.

(5) Remove plug body and gasket.

A. Installation – Chip Detector

NOTE: Lightly coat packings and gasket with clean engine oil. Apply lubricating compound (Liqui-Moly, Graden V) to threads of plug and body prior to installation.

(1) Ensure electrical power is OFF.

(2) Install plug body with new gasket (S8316AD8),

(3) Torque to 75 inch-pounds, and safety.

(4) Install magnetic plug with new packings (MS29561-012) into body and safety.

(5) Connect electrical connector, and safety.

(6) Install lower forward cowling.

2. Inspection/Check – Magnetic Drain Plug (Chip Detector)

A. Inspection/Check – Chip Detector

(1) Visually inspect magnetic plug for metal, particles.

(2) Inspect body and plug for cracked or damaged threads.

(3) Inspect electrical connector for bent, broken, or burned pin.

(4) Check continuity from electrical connector to ground. Magnetic plug should not indicate any continuity to ground.

(5) Inspect body, plug, and pin for evidence of oil leakage.

(6) Replace part that does not meet inspection requirements.

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CHAPTER 78 – EXHAUST

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EXHAUST – DESCRIPTION AND OPERATION

1. General

A. The exhaust system on SA226 series aircraft directs engine exhaust gasses overboard at the top of the wing nacelle area. It consists of an AiResearch engine exhaust pipe assembly mounted to the rear of the engine turbine section, an exhaust assembly (tailcone) mounted to the engine exhaust pipe, and an aspirator exhaust duct assembly, which mounts between the nacelle firewall, extending, to just aft of the upper nacelle skin. The aspirator exhaust duct assembly is wrapped with an insulated metal and fiberglass blanket to prevent excessive heat from entering nacelle area.

B. When the engine is installed, the tailcone is partially inserted into the aspirator exhaust duct assembly, and a heat seal mounted to the aspirator assembly seals around the outer circumference of the tailcone to prevent exhaust escape. The aspirator exhaust duct assembly is designed with a 1.50° rise so that exhaust from the engine is diverted slightly upward and overboard. Additionally, air entering the scoop in the lower engine cowling, used to cool the oil cooler, is drawn through oil cooler, around the tailcone, within the aspirator assembly, then vented out through the aspirator exhaust duct assembly.

1. Removal/Installation – Exhaust

NOTE: For removal and installation of exhaust pipe assembly, see AiResearch Maintenance Manual, Chapter 72-50-02.

A. Removal – Exhaust Assembly, Tailcone

(1) Remove engine, see Maintenance Manual, Chapter 71.

(2) Remove the 18 retaining bolts, 36 washers, and 18 nuts; remove tailcone from engine.

B. Installation – Exhaust Assembly, Tailcone

(1) Install tailcone with 18 bolts and 18 nuts, and 36 washers. Install with head of bolts facing toward front of engine; install one washer under the head of each bolt and one washer beneath each nut.

NOTE: Place flat side of washers against flange of exhaust duct on both sides. Apply C5A anti-seize compound, MIL-A-907, or equivalent to bolt thread during installation.

(2) Torque bolts to 70 inch-pounds (7.91 N·m).

NOTE: If symmetry of tailcone is not perfectly round, install assembly such that the flat side of the assembly will be at the one o'clock position when engine is installed.

C. Removal – Aspirator Exhaust Duct Assembly

(1) Remove engine, see Maintenance Manual, Chapter 71.

(2) Remove insulation blankets from exhaust duct assembly.

(3) Remove mounting bolts securing exhaust duct assembly to nacelle firewall.

(4) Withdraw exhaust duct assembly forward through nacelle firewall.

D. Installation – Aspirator Exhaust Duct Assembly

(1) Insert exhaust duct assembly through nacelle firewall opening.

(2) Ensure that the distance between nacelle skin opening and the exhaust assembly is symmetrically even around the assembly circumference.

(3) Install and tighten retaining bolts at nacelle firewall, and check installation as described in step **1.**, D., 2.

(4) Ensure exhaust assembly extends aft of the nacelle skin 1 inch, -0, +0.50 inches. Trim as necessary.

(5) Install insulation blankets onto duct assembly.

(6) Install engine, see Maintenance Manual, Chapter 71.

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CHAPTER 79 – OIL

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OIL – DESCRIPTION AND OPERATION

1. General

A. The oil system provides lubricating oil to the various engine components.

(1) An oil tank mounted on the forward left center of each nacelle firewall stores oil for system operation.

(2) Hoses of various sizes route oil to and from each engine.

B. The oil system includes an internal pressure pump and internal scavenge pump for each engine.

(1) The internal pressure pump provides jet and mist lubrication to engine bearings and gears.

(2) There are three scavenge pumps.

(a) Two pumps are located in the reduction gear section. The third pump is located in the turbine section.

(b) Scavenge pumps return oil through the oil-to-fuel heater for fuel filter anti-icing to the oil storage tank.

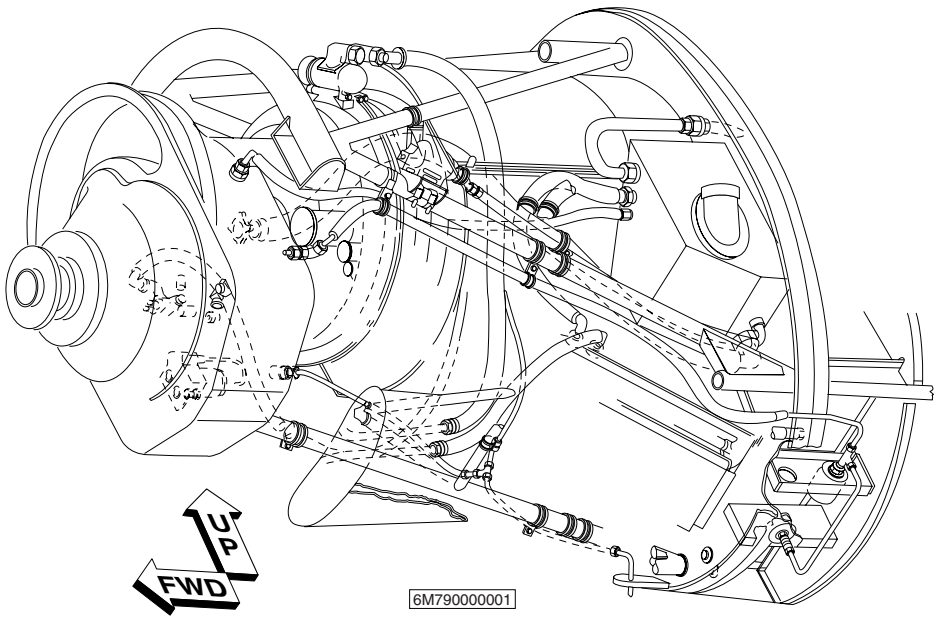
C. The lubrication system also supplies servo oil to each engine propeller control system torque sensing components.

(1) The lubrication includes: an oil temperature bulb, oil pressure transmitter, a low pressure warning switch, and associated plumbing and fittings to route the oil to and from these components.

(2) The lubrication system is a dry sump type.

(a) Excess oil does not accumulate nor is it stored in the sumps.

(b) All excess oil is stored in the externally mounted storage tank.



Oil System
Figure 1

1. Servicing – Oil

WARNING: PROLONGED EXPOSURE TO MIL-L-23699 ENGINE LUBRICATING OIL MAY BE HAZARDOUS TO YOUR HEALTH. FOLLOW MANUFACTURER'S SAFETY PRECAUTIONS.

A. Drain Engine Oil

(1) Position propeller in the feather position to ensure maximum drainage of servo oil into engine oil sump and oil tank.

NOTE: If possible, drain oil while engine is still warm.

(2) Place a catch can or suitable container under drain plug.

(3) Remove magnetic drain plug and drain plug body from lower forward accessory drive housing and allow oil from housing sump to drain into container.

(4) Remove drain plug from bottom of oil tank and drain oil into a suitable container.

(5) Remove oil cooler drain plug and drain oil into a suitable container.

(6) Remove oil filter from left aft accessory drive housing.

NOTE: Inspect magnetic drain plug, filter element and drained oil for contamination.

B. Fill Oil System

(1) Install clean oil filter element with new packing.

(2) Hand tighten filter cap with bushing and safety.

(3) Install oil cooler drain plug and safety.

(4) Install oil tank drain plug and safety.

(5) Install magnetic drain plug body and packing. Torque drain plug 75 inch-pounds (8.47 N·m) and safety.

(6) Install magnetic drain plug and packings. Safety plug and electrical connector.

(7) Remove oil tank cap and dipstick assembly. Fill tank with six quarts of oil and install cap.

(8) Using unfeather pump, position the propeller on start locks.

CAUTION: SHUT DOWN ENGINE IMMEDIATELY IF OIL PRESSURE IS NOT INDICATED WITHIN TEN SECONDS OF ENGINE START.

(9) Start engine IAW AFM and operate at least one minute.

CAUTION: DO NOT MIX DIFFERENT TYPES OR BRANDS OF OIL.

(10) Shut down engine and check oil level; fill oil tank to full mark on the dipstick.

C. Oil Level Check

(1) Check the oil level according to the dipstick attached to the filler cap.

NOTE: With cold oil, the level should be between the FULL and ADD markings on the dipstick. The oil level will tend to be slightly higher with warm oil. Oil will tend to siphon back into the engine after engine shutdown.

CAUTION: DO NOT OPERATE ENGINE(S) WITHOUT A VISIBLE INDICATION OF OIL IN THE OIL FILLER NECK.

(a) Motor the engine 10 to 15% to scavenge oil from the engine into the oil tank.

NOTE: If the oil tank is serviced without scavenging the oil, overservicing will result and the excess oil will be pumped overboard.

(b) Check oil level with the airplane in a level attitude.

2. Inspection/Check – Oil

A. Inspection Procedures

(1) Inspect oil storage tank assembly for condition, security, and evidence of leaks.

(a) Check to ensure oil storage tank clearance from oil cooler is 0.250 inch (6.350 mm).

(b) Check oil level.

(c) Check oil cap and seal for condition.

(2) Check oil cooler for condition, security, and evidence of leaks.

(3) Check oil pressure warning switch for evidence of leaks and security.

(4) Check oil pressure transmitter for evidence of leaks and security.

(5) Inspect plumbing and connections for cuts, cracks, leaks, chafing, security, and condition.

(a) Check clamps for security and condition.

(b) Replace brittle clamp cushions as required.

(6) Inspect seals for leakage as follows:

(a) Check for leaks around tachometer-generator, starter-generator, fuel control, and accessory mounts. Maximum allowable leakage is approximately eight (8) drops per hour (0.5 cc/hr).

(b) Check for compressor and turbine seal leakage. Maximum allowable leakage is approximately fifteen (15) drops per minute (1.0 cc/min).

(c) Check propeller shaft seal for leakage. Maximum allowable leakage is approximately eight (8) drops per hour (0.5 cc/hr).

(d) Maximum oil consumption is one quart per ten hours of operation.

NOTE: Whenever oil filter bypass valve indicator button is extended, perform oil system contamination check IAW Chapter 72, Honeywell Maintenance Manual.

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OIL STORAGE TANK – DESCRIPTION AND OPERATION

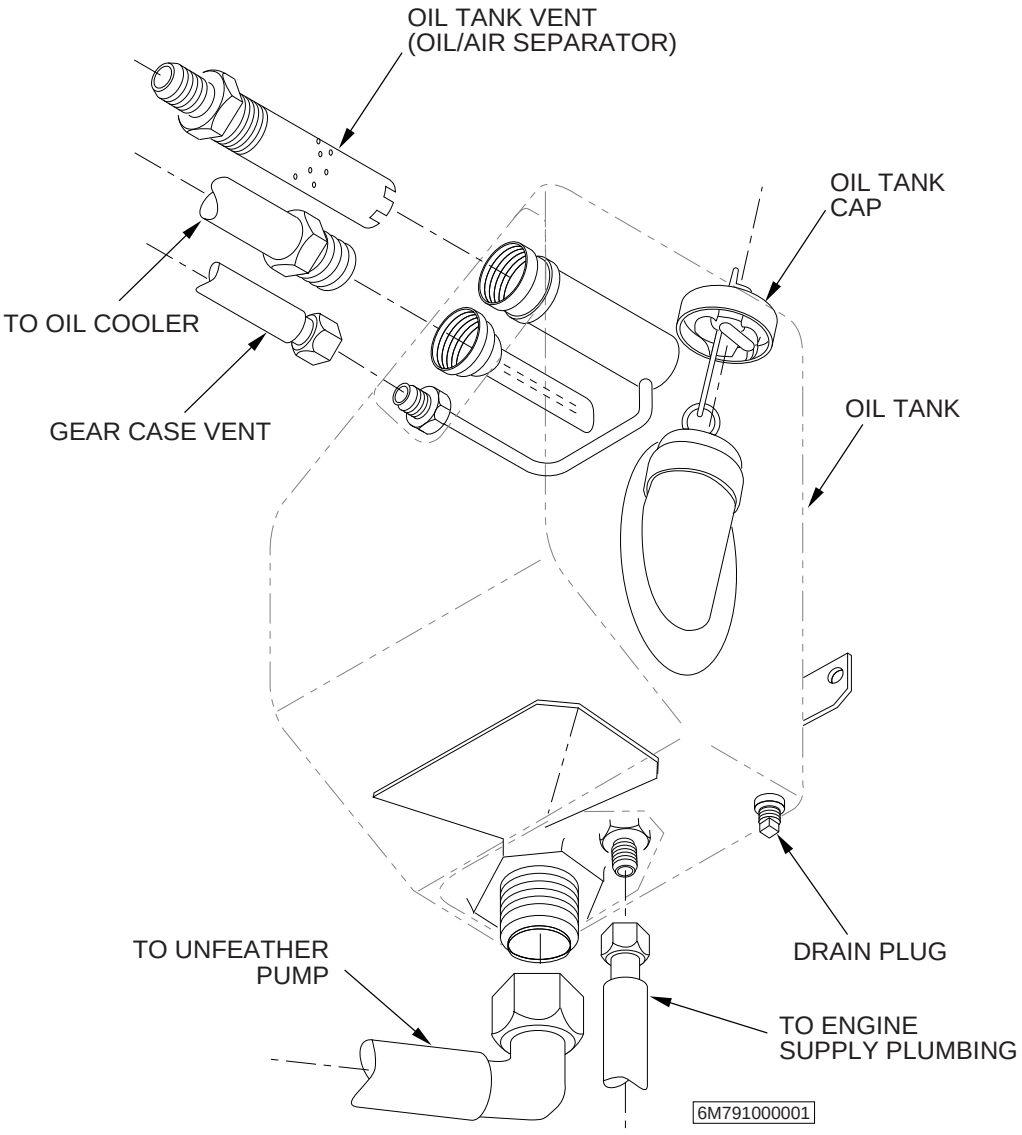
1. General (Figure 1)

A. Each oil storage tank is mounted forward of N.S. 92.69 on the left side of the firewall and has a capacity 3.6 U.S. quarts (3.4 litres). Total oil system capacity for each engine (to include engine, oil lines, and oil storage tank) is approximately 4.5 U.S. quarts (4.3 litres).

B. The oil storage tank filler cap and dipstick assembly is mounted on the left side of the oil storage tank.

(1) The filler cap and dipstick is accessible through an access door located on the left side engine cowl.

(2) A filter screen is incorporated in the filler neck.



Oil Storage Tank
Figure 1

OIL STORAGE TANK – DESCRIPTION AND OPERATION

C. Three threaded fittings welded to the upper front section of the tank provide connections for:

(1) Oil/air separator tank vent tubing.

(2) Oil cooler to tank return plumbing.

(3) Engine gear case vent plumbing.

NOTE: Each hose connection is a different size to ensure proper installation.

D. Three threaded fittings in the bottom of the tank provide connections for:

(1) Tank to unfeather pump supply plumbing.

(2) Tank to engine supply plumbing.

(3) Oil storage tank drain plug.

NOTE: Each hose connection is a different size to ensure proper installation.

E. The oil storage tank vent incorporates an oil/air separator.

(1) The oil/air separator fits into a perforated chamber located inside the tank.

(2) The perforated chamber, containing seven vent holes at the top and one drain hole at the bottom is connected to the engine crankcase vent boss.

(3) Return oil entering the tank through a slotted chamber is directed away from the oil/air separator chamber, thus assisting in oil/air separation.

OIL STORAGE TANK – MAINTENANCE PRACTICES

1. Removal/Installation – Oil Storage Tank

A. Removal – Oil Storage Tank

(1) Ensure electrical power is OFF.

(2) Remove left and right side engine cowlings.

(3) Place drip pan under engine.

(4) Remove lower engine cowling.

(5) Position a suitable container to capture drained oil.

(6) Open oil storage tank filler cap.

(7) Remove drain plug from tank. Drain oil into container.

(8) Disconnect oil storage tank vent tube from fitting on upper front section of tank. Cap open line and fitting.

(9) Disconnect engine gear case vent hose from fitting on upper front section of tank. Cap open line and fitting.

(10) Disconnect oil cooler return hose from fitting on upper front section of tank. Cap open line and fitting.

(11) Disconnect unfeather pump supply hose from fitting on bottom of tank. Cap open line and fitting.

(12) Disconnect main engine oil supply hose from fitting on bottom of tank. Cap open line and fitting.

(13) Remove screws and washers from mounting tabs securing tank to firewall.

(14) Remove oil storage tank.

(15) Clean up any oil spills.

B. Installation – Oil Storage Tank

(1) Ensure electrical power Is OFF.

(2) Position the tank against the firewall and secure with screws and washers.

(3) Remove caps from fittings, hoses, and tubes.

(4) Connect main engine oil supply hose to fitting on bottom of tank.

(5) Connect unfeather pump supply hose to fitting on bottom of tank.

(6) Connect oil cooler return hose to fitting on upper front section of tank.

(7) Connect engine gear case vent hose to fitting on upper front section of tank.

(8) Connect oil storage tank vent tube to fitting on upper front section of tank.

(9) Install drain plug into bottom of oil storage tank. Secure with safety wire.

NOTE: Wrap drain plug threads with teflon tape.

(10) Service oil storage tank IAW Chapter 12, this manual.

(11) Install oil storage tank cap.

(12) Operate the engine and check for leaks. If leaks are detected, shut down engine and correct.

(13) Recheck oil level. Service if required.

(14) Replace the lower engine cowling.

(15) Replace the engine side cowlings.

(16) Remove container and drip pan from under engine.

(17) Clean up any oil spills.

2. Inspection/Check – Oil Storage Tank

A. Leak Check

(1) Turn electrical power ON.

WARNING: ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF OBJECTS AND PERSONNEL DURING ENGINE RUN.

CAUTION: SHUT DOWN ENGINE IMMEDIATELY IF OIL PRESSURE IS NOT INDICATED WITHIN TEN (10) SECONDS OF ENGINE START.

(2) Operate engine IAW AFM for approximately one (1) minute.

(3) Check for leaks during engine run.

(4) Immediately shut down engine if leaks are detected.

(5) Shut down engine IAW AFM.

(6) Open left and right side engine cowlings.

(7) Place drip pan under engine.

(8) Thoroughly check all connections for oil leaks. Correct any leaks found.

(9) Check oil level. Service as required.

(10) Close and secure upper and lower engine cowling.

(11) Clean up oil spills.

(12) Remove drip pan from under engine.

B. Inspection – Oil Storage Tank

(1) Inspect oil storage tank assembly for condition, evidence of leaks, and security.

(2) Check oil storage tank for clearance from oil cooler 0.250 inch (6.350 mm).

(3) Check oil cap assembly for condition and security.

(4) Inspect plumbing and connections for cuts, cracks, leaks, chafing, general condition, and security.

(5) Check clamps for condition and security. Replace brittle clamp cushions.

(6) Check oil level. Service as required.

3. Repair – Oil Storage Tank

A. Repair for chafing Oil Tank assembly (Max allowable depth of chafing is 0.015 inch)

(1) Remove oil tank assembly from engine per step 1.A.

(2) Clean the tank with cleaning solvent per SA226/227 Structural Repair Manual (SRM) chapter 51-30-01. Dry inside and out when complete.

(3) Fill chafes with weld per SA226/227 Structural Repair Manual (SRM) chapter 51-70-02. During weld process, purge tank with argon.

(4) Blend welds flush with the surface +0.002/-0.00 inch.

(5) Visually inspect the interior surface where the tank was welded. Ensure all areas around the weld penetration are smooth and no weld matter is present. If area is not smooth, the weld is not acceptable and the tank is SCRAPPED.

(6) Leak test the tank by applying 5.0 +0.5/-0.0 psig of air pressure for 1 minute. No leakage is allowed.

(7) Reinstall tank assembly step 1.B.

(8) Reorient clamps around the area of the tank to preclude any further chafing.

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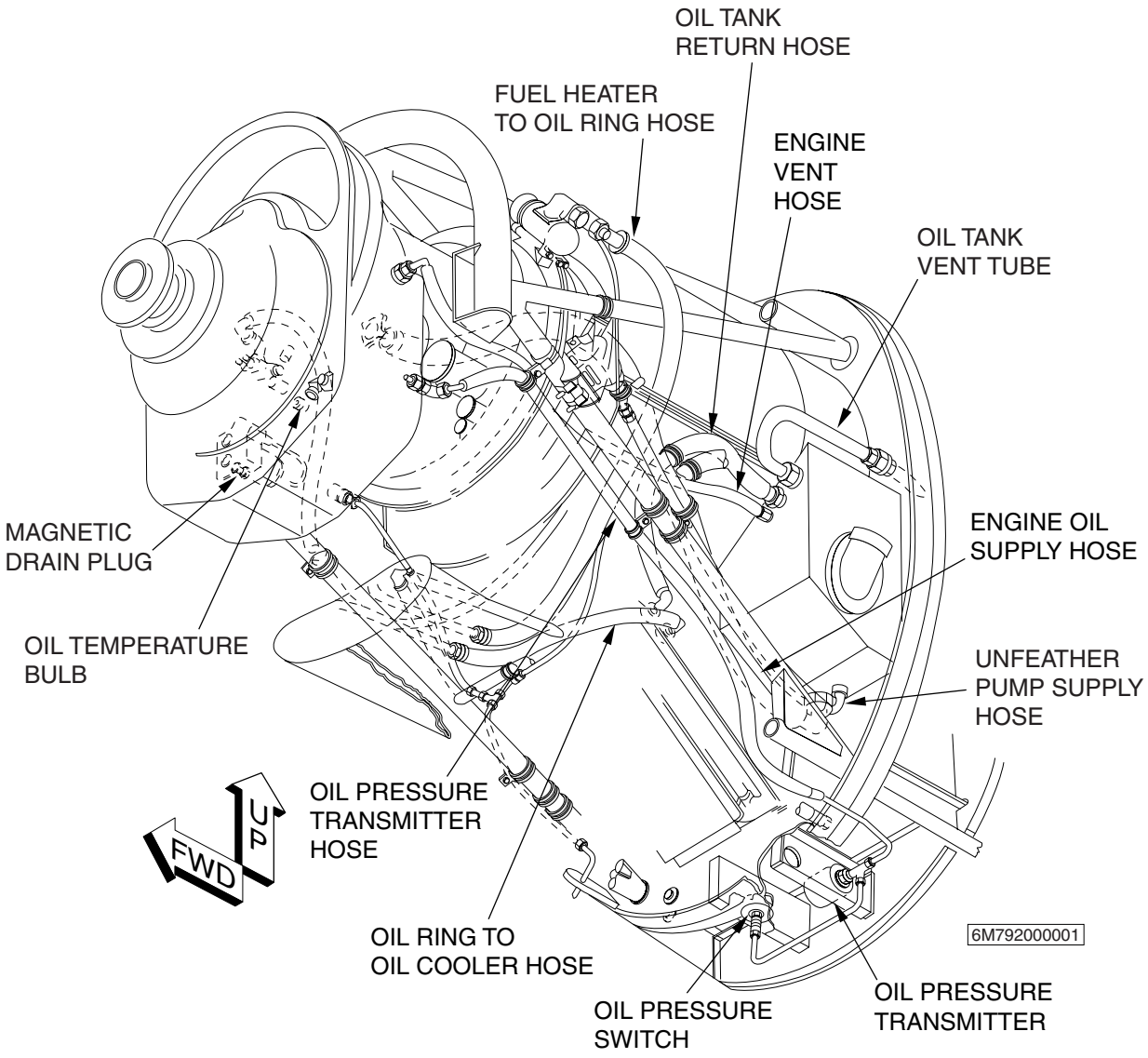
OIL DISTRIBUTION – DESCRIPTION AND OPERATION

1. General

A. Oil distribution consists of the oil cooler, associated plumbing, and Honeywell oil system components.

(1) In addition to the oil cooler and hoses, a ten micron filter element enclosed in a metal housing functions to filter all oil entering the system from the oil pressure pump.

(2) An oil filter bypass valve, located on the left side of the accessory drive housing, is connected in parallel with the filter. Should impurities clog the filter the bypass valve will open, allowing oil to bypass the filter to avoid possible bearing starvation. When the valve is in the bypass position an indicator pin will extend from the oil filter bypass valve flange.



Oil Distribution
Figure 1

(3) An oil pressure regulator valve located above the bypass valve regulates system pressure.

(4) An oil vent valve is electrically actuated to allow air to enter the oil pressure and scavenge pumps during engine start. This unloads the pumps, reducing the load on the engine. The valve is actuated automatically from zero to 50% engine speed during normal ground starts.

(5) The rear turbine bearing oil line is an externally mounted tube connecting the oil supply to the rear turbine bearing.

NOTE: For additional information refer to the Honeywell Maintenance Manual, Chapter 79.

MAINTENANCE MANUAL

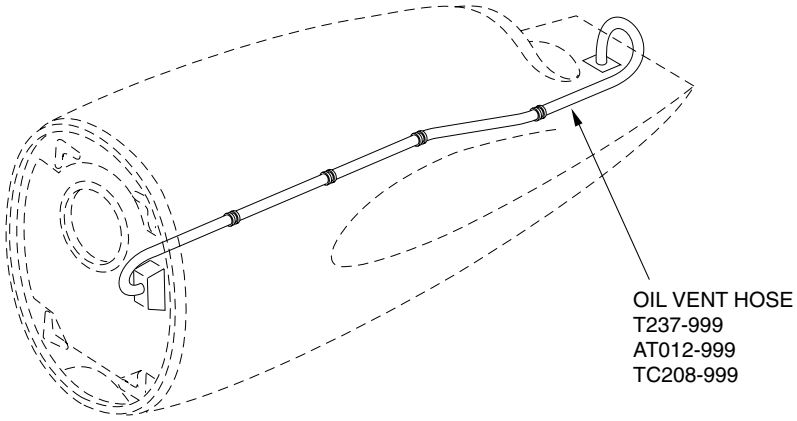
OIL HOSES – DESCRIPTION AND OPERATION

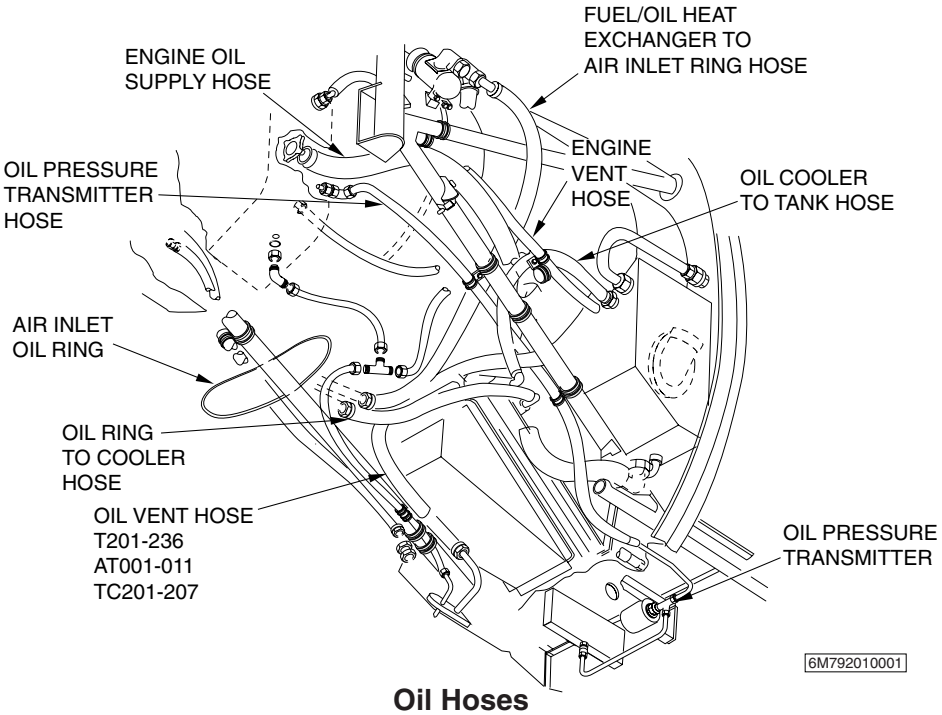
1. General

A. Flexible hoses of various sizes make up the external plumbing of the oil system, carrying oil from the storage tank to the system components and returning to the oil tank.

(1) The tank is vented to an overboard drain. The vent hose is connected to an oil/air separator in the tank and is routed to the engine drain block cluster located on the lower right side of the firewall frame on Pre-Service Bulletin 79-004 models T201-236, AT001--011, TC201-207. All other models have the vent routed aft through the nacelle to drain overboard at the lower aft nacelle, which allows discharge vapors to clear the aircraft.

NOTE: For additional information, refer to Chapter 71, Engine Build-Up.





Oil Hoses
Figure 1

OIL HOSES – MAINTENANCE PRACTICES

1. Removal/Installation – Oil Hoses

A. Removal – Oil Hoses

(1) Ensure electrical power is OFF.

(2) Place drip pan under engine.

(3) Open left and right side engine cowlings as required.

(4) Locate hose to be removed.

NOTE: Engine oil hoses should be removed only as required to facilitate other maintenance, or to replace damaged or contaminated plumbing.

(5) Remove clamps and attaching hardware.

(6) Hold firewall union in place with appropriate wrench, loosen "B" nuts on plumbing.

NOTE: Prior to removal of plumbing, place drip pan under fittings to catch spillage.

(7) Disconnect end fittings from unions/components. Cap open lines.

(8) Carefully remove plumbing.

(9) Clean up oil spills.

B. Installation – Oil Hoses

(1) Ensure electrical power is OFF.

(2) Position drip pan under engine.

(3) Open left and right side cowlings as required.

(4) Carefully position hose in place. Remove caps from open lines.

(5) Connect end fitting to union/component. Tighten to specified torque (See Chapter 91, this manual, for torque values).

(6) Secure plumbing using clamps, bolts, washers, and nuts as required.

(7) Check oil level. Service as required.

(8) Close and secure left and right side cowlings as required.

(9) Perform ground run and check for leaks.

(10) Repair leaks as necessary.

(11) Remove drip pan from under engine.

2. Inspection/Check – Oil Hoses

A. Leak check – Oil Plumbing

WARNING: ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF OBJECTS AND PERSONNEL DURING ENGINE RUN.

CAUTION: SHUT DOWN ENGINE IMMEDIATELY IF OIL PRESSURE IS NOT INDICATED WITHIN TEN (10) SECONDS OF ENGINE START.

(1) Operate engine IAW AFM for approximately one (1) minute.

(2) Shut down engine IAW AFM.

(3) Open engine cowling.

(4) Check thoroughly for leaks. Correct any leaks found.

(5) Check oil level. Service as required.

(6) Operate engine to ensure no leaks.

(7) Close and secure engine cowling.

(8) Clean up oil spills.

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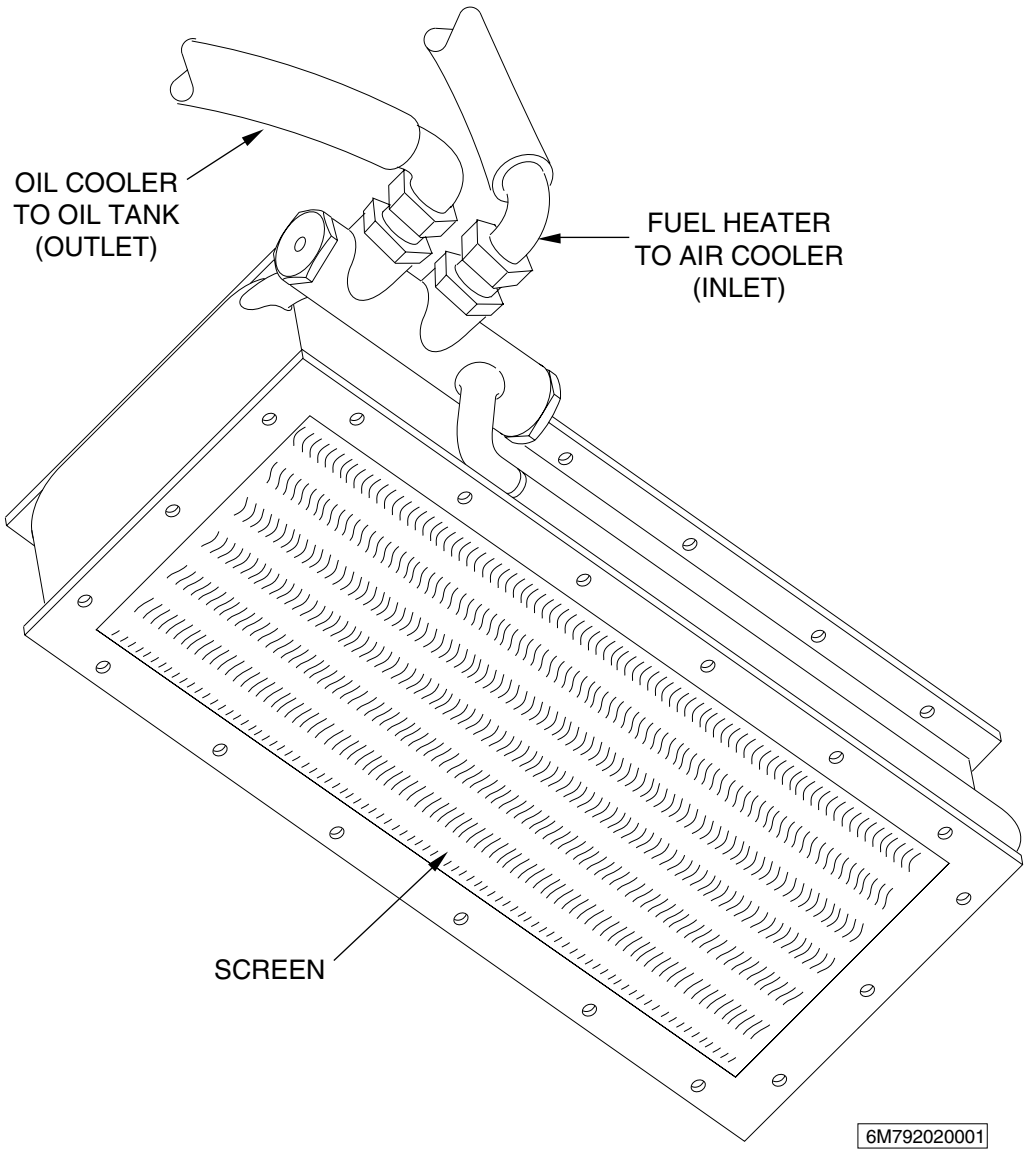
OIL COOLER – DESCRIPTION AND OPERATION

1. General (Figure 1)

A. An aluminum plate type oil cooler is bolted to the lower flange of the aspirator.

(1) An integral temperature bypass valve routes the oil returning from the engine through the cooling coils or directly to the storage tank depending on the oil temperature. The bypass valve is set so that all oil bypasses the oil cooler at outlet oil temperature below 150˚F (66°C) to ensure adequate fuel filter anti-icing.



Oil Cooler
Figure 1

OIL COOLER – MAINTENANCE PRACTICES

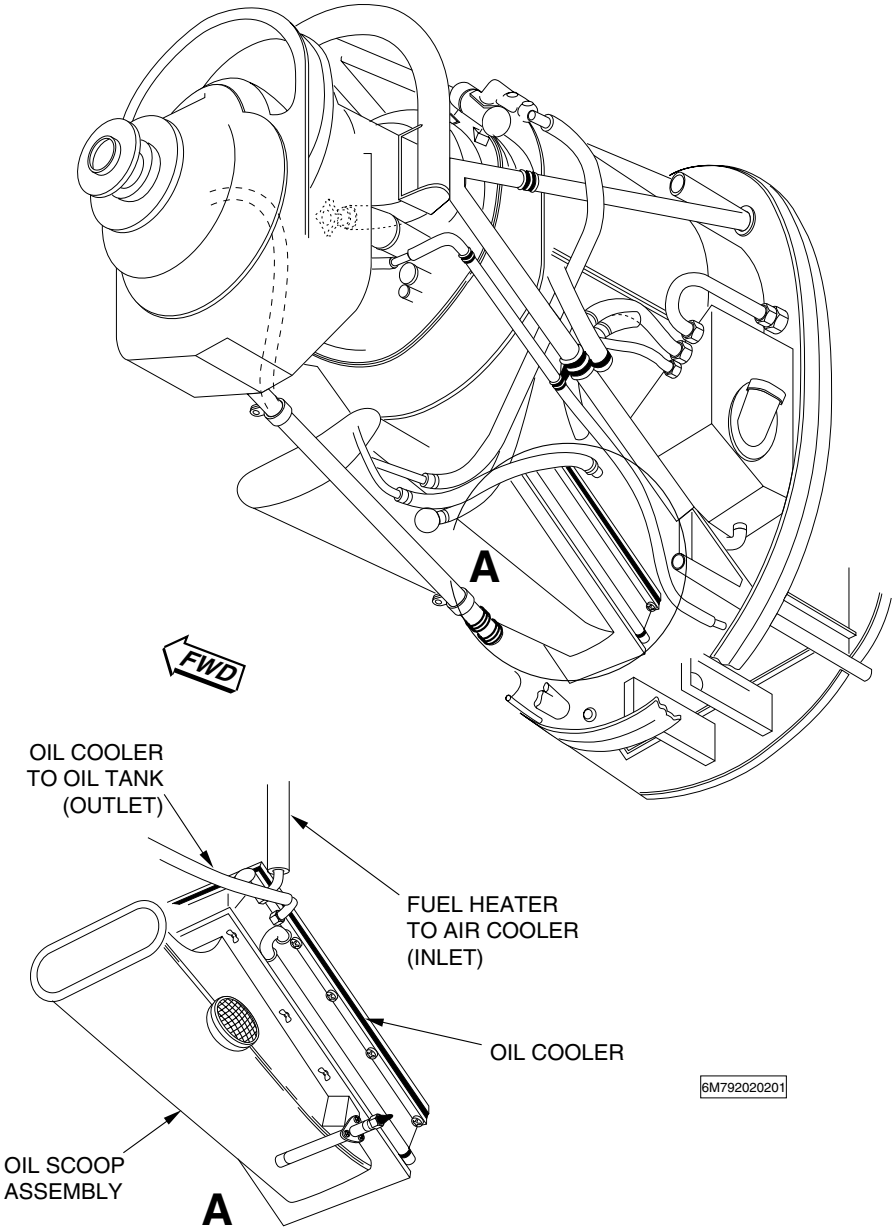
1. Removal/Installation – Oil Cooler

A. Removal – Oil Cooler

(1) Remove engine left and right side engine cowlings.

(2) Place drip pan under engine.

(3) Remove lower engine cowling.



Oil Cooler Assembly
Figure 201

(4) Remove oil cooler scoop assembly.

(5) Disconnect oil cooler to storage tank hose at oil cooler. Cap open line.

(6) Disconnect air scoop lip heater hose at oil cooler. Cap open line.

(7) Remove bolts, washers, and nuts securing oil cooler upper flange to aspirator assembly.

(8) Remove oil cooler.

(9) Clean up any oil spills.

B. Installation – Oil Cooler

(1) Position oil cooler and restrictor plate against aspirator and secure with bolts, washers, and nuts.

NOTE: On models T276, 292-999, the restrictor plate has been eliminated.

(2) Connect air scoop lip heater hose to oil cooler.

(3) Connect oil tank hose to oil cooler.

(4) Connect air scoop lip heater inlet and outlet hoses together with a union to complete oil circulation for ground run-up.

(5) Operate engine and check the oil cooler and connections for leaks. Shut down engine and correct any discrepancies. Recheck if necessary.

(6) Install the oil cooler scoop assembly with the eight airlocks.

(7) Remove the union from the air scoop lip heater hoses and install lower cowling.

(8) Clean up any oil spills.

(9) Recheck oil tank for specified level.

(10) Install engine left and right side engine cowlings.

2. Inspection/Check – Oil Cooler

A. Leak Check – Oil Cooler

WARNING: ENSURE PROPELLERS AND TAIL PIPE AREAS ARE CLEAR OF OBJECTS AND PERSONNEL DURING ENGINE RUN.

CAUTION: SHUT DOWN ENGINE IMMEDIATELY IF OIL PRESSURE IS NOT INDICATED WITHIN TEN (10) SECONDS OF ENGINE START.

(1) Operate engine for approximately one (1) minute IAW AFM.

(2) Shut down engine IAW AFM.

(3) Open left and right side engine cowlings.

(4) Check oil cooler assembly and connections for leaks. Correct any leaks found.

(5) Check oil level. Service as required.

(6) Operate engine to ensure there are no leaks.

(7) Clean up any oil spills.

(8) Close and secure left and right side engine cowlings.

B. Inspection – Oil Cooler

NOTE: Oil cooler need not be replaced at every engine overhaul/change. If the oil cooler passes the following inspection at engine overhaul it may still be used.

(1) Remove oil cooler from aircraft.

(2) Flush cooler assembly solvent thoroughly.

(3) Inspect cooler assembly for leaks and general condition.

(4) X-Ray cooler assembly for metal particles.

(5) If no defect is noted, return unit to service.

OIL INDICATING – DESCRIPTION AND OPERATION

1. General

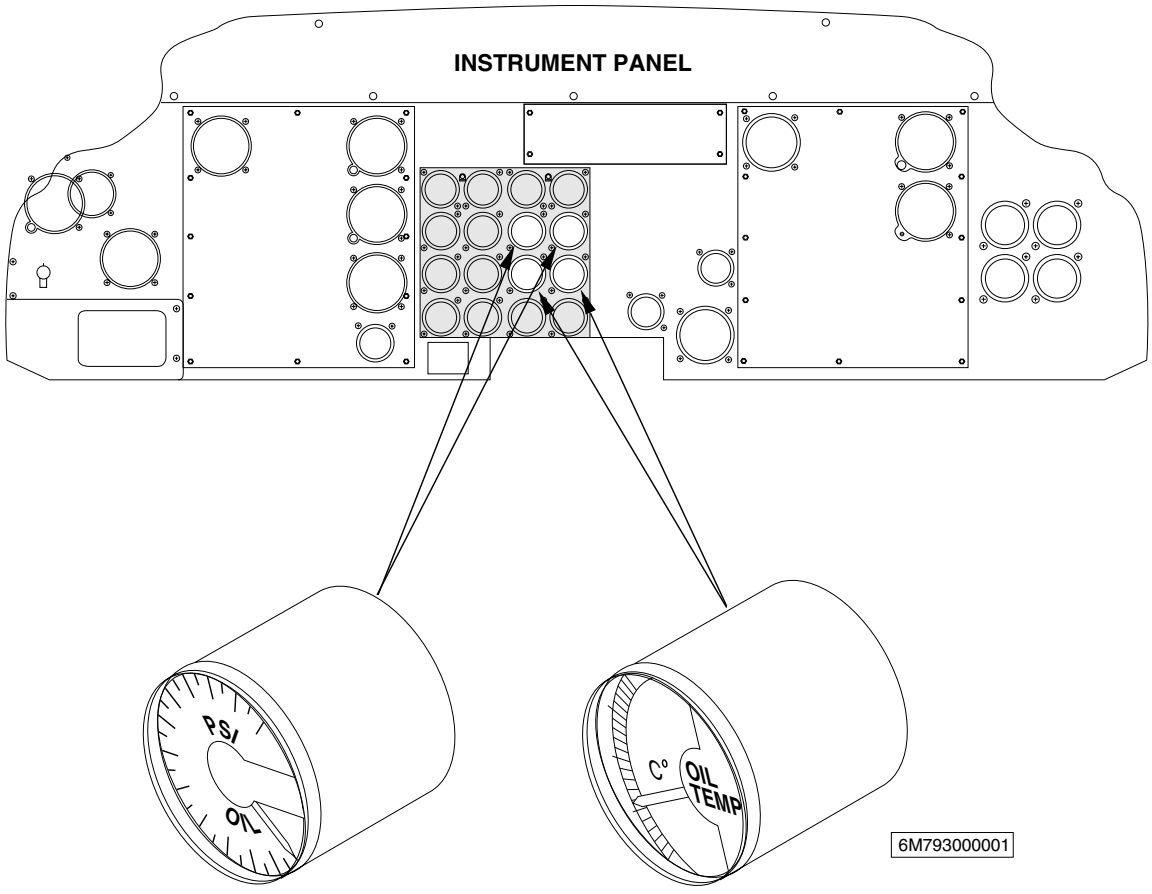
A. Oil indicating consists of: visual indicating, oil pressure indicating, oil temperature indicating, oil pressure transmitters, and oil pressure warning light switches.

(1) The oil pressure and oil temperature indicators, located in the cockpit, provides the crew with visible evidence of system performance for each engine and its oil system. The system provides continuous indication of oil pressure and oil temperature throughout the full range of engine operation (Figure 1).

(a) Indicators are provided for both left and right engines. The indicators are mounted in the center instrument panel.

(b) The pressure indicator reads oil pressure (PSI x 10).

(c) The temperature indicator reads oil temperature in degrees Celsius (°C x 10).

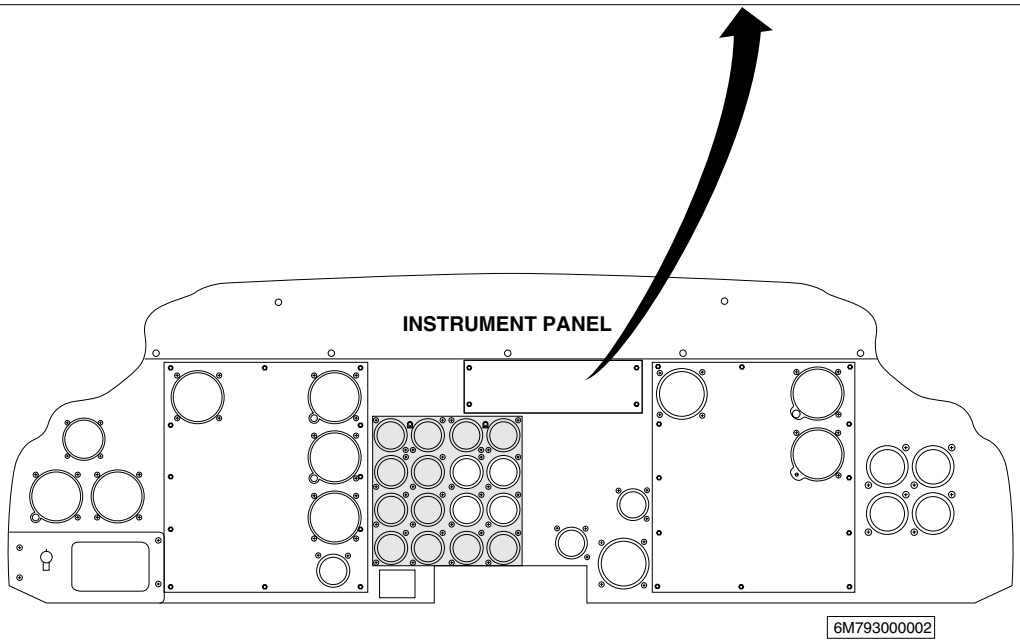


Oil Indicating – Cockpit Indicators
Figure 1

(2) An annunciator panel located in the center instrument panel, contains red warning lights (L OIL PRESS and R OIL PRESS). Warning lights illuminate should oil pressure drop below 40 psi (2.812 kg/cm^2).

ANNUNCIATOR PANEL

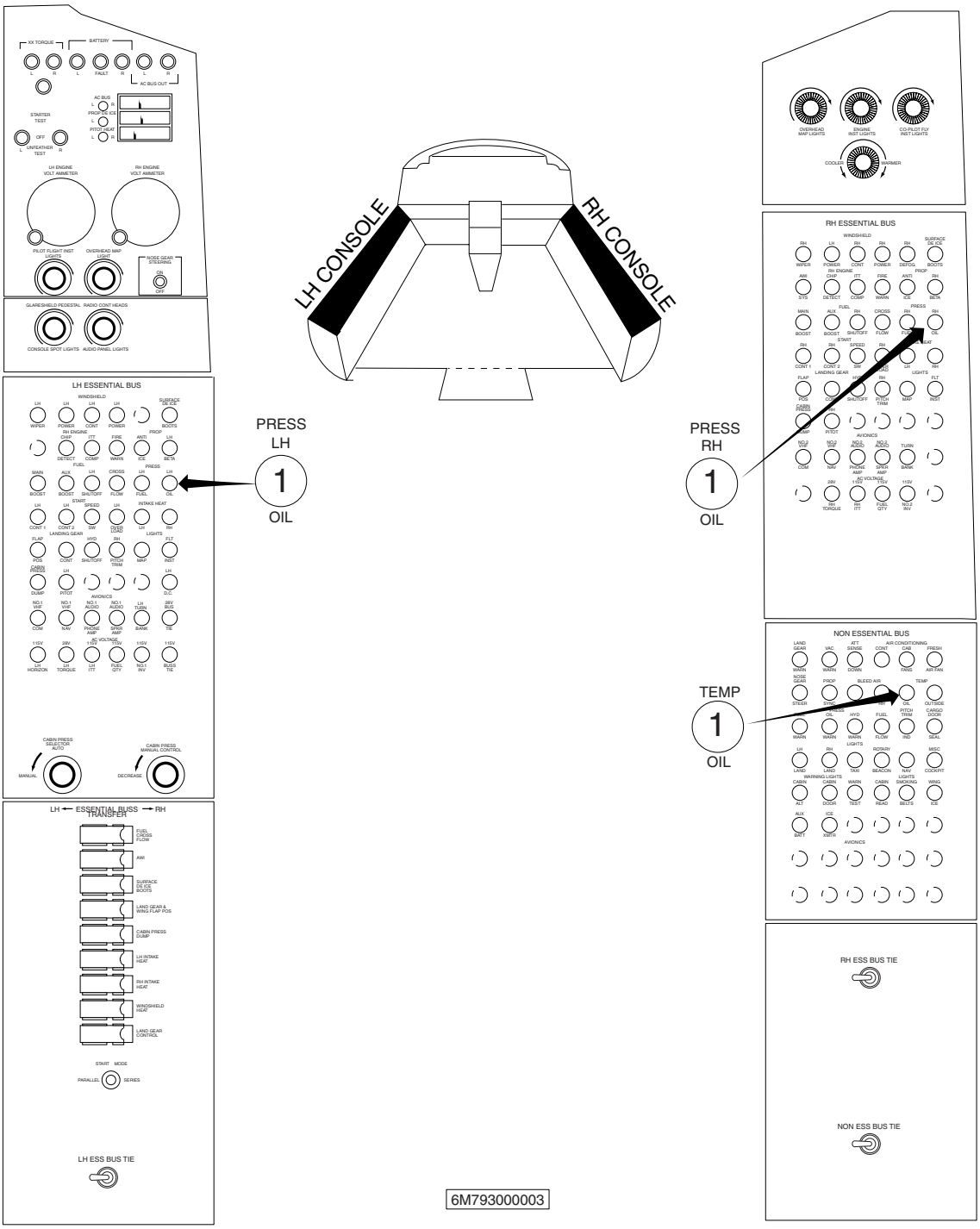
L FIRE WARNING	R FIRE WARNING	CABIN DOOR	L BETA	R BETA	LOW VAC.	L INTAKE HEAT ON	R INTAKE HEAT ON
L WING OVHT	R WING OVHT	BAT FAIL	L CHIP DET	R CHIP DET	CABIN ALT	L ENG HEAT	R ENG HEAT
L OIL PRESS	R OIL PRESS	SAS FAULT	L XFR PUMP	R XFR PUMP	SPARE	L OTS	R OTS
L HYD PRESS	R HYD PRESS	CARGO DOOR	L BAT DISC	R BAT DISC	SPARE	SAS ARM	SAS DEICE
L SAT HOT	R SAT HOT	SPARE	L AC OUT	R AC OUT	SPARE	NOSE STEER	SPARE
SPARE	SPARE	AM LO WTR	L GEN FAIL	R GEN FAIL	SPARE	LOW VAC	SPARE



Oil Indicating – Annunciator Warning Lights
Figure 2

OIL INDICATING – DESCRIPTION AND OPERATION

(3) Electrical power for the indicating system (PRESS and TEMP) is provided by either the LEFT ENGINE INSTRUMENT BUS on the pilot's console or the RIGHT ENGINE INSTRUMENT BUS on the copilot's console, through one ampere circuit breakers.



Oil Indicating – Circuit Breakers
Figure 2

(4) A resistance type oil temperature bulb provides a continuous reading of inlet oil temperature read in celsius on the indicator mounted on the center instrument panel.

(5) Low oil pressure warning is provided by a light in the annunciator panel that receives a signal from the pressure switch mounted in each nacelle forward of the wheel well. Electrical power is supplied from the nonessential bus through a one amp circuit breaker. The pressure switch is set to close at 40 psi (2.812 kg/cm^2) decreasing oil pressure.

NOTE: SA226 models T201-239, AT001-013, TC201-207 were delivered with oil pressure switches (P/N M4011-54) set at 50 psi. A lower oil pressure limit of 40 psi has since been approved for all SA226 aircraft. To employ the lower limits, replace the oil pressure switches P/N M4011-54 with P/N M4011-44 and change the oil pressure gauge marking IAW instrument marking chart for Section 1 of the Aircraft flight manual.

OIL PRESSURE INDICATING – DESCRIPTION AND OPERATION

1. General

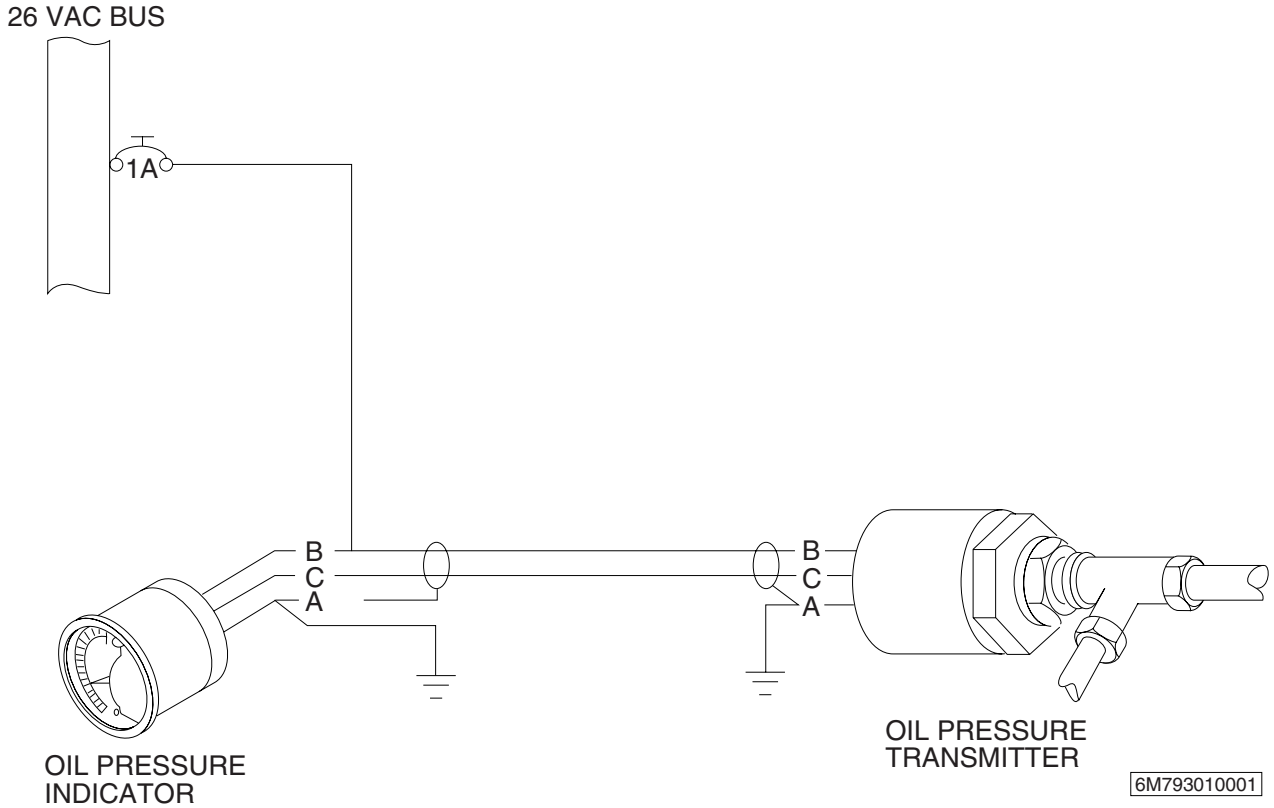
A. An oil pressure transmitter is mounted in a bracket in the lower forward section of each nacelle, between the bulkhead forward of the main landing gear wheel well and firewall.

(1) A hose connects the engine oil pressure sensing port on the left rear of the reduction gear case to the oil pressure transmitter and warning light pressure switch.

(2) A restrictor union is installed at the lower left firewall to reduce pressure fluctuation.

(3) The oil pressure transmitter is connected through the aircraft wiring to the oil pressure indicator.

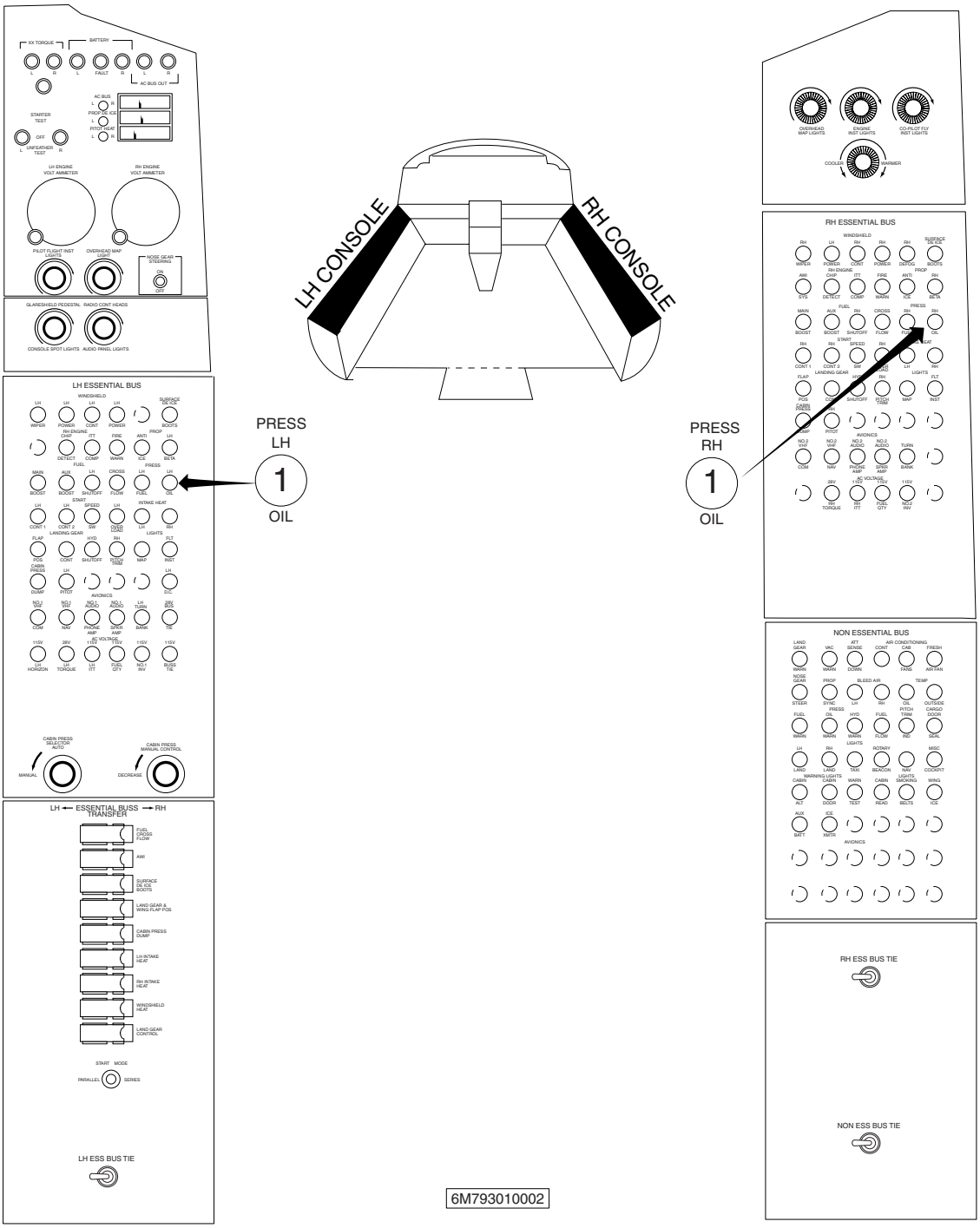
(4) The oil pressure transmitter is a variable reluctance, type sensing unit with a removable plug cap at the top of the transmitter to give access for calibration adjustment. It is; however, a factory sealed non-repairable unit.



Oil Pressure Indicator Warning
Figure 1

OIL PRESSURE INDICATING – DESCRIPTION AND OPERATION

(5) The oil pressure indicating system is powered from either the left or right 26 VAC bus through a one amp circuit breaker. Cannon type connectors are provided at the oil pressure transmitter and indicator.



Oil Pressure Circuit Breaker
Figure 2

OIL PRESSURE INDICATING – FAULT ISOLATION

1. Troubleshooting – Oil Pressure Indicating

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No response on oil pressure indicator.	1. No AC power to indicator.	1. If low pressure warning light is extinguished, the indicator circuit is defective. Check wiring (see 79-30-01 of the Wiring Manual).
	2. Defective or stuck indicator.	2. If wiring is correct, tap indicator face lightly, swap indicators left to right.
	3. Defective transmitter.	3. If wiring is correct and indicator good, the oil pressure transmitter is defective.
	4. No oil pressure.	4. If the low oil pressure warning light is illuminated, the pressure system is defective. Install a direct reading pressure gauge between the engine and transmitter. If the gauge indicates pressure, the transmitter is defective.
		5. If the direct reading gauge does not indicate pressure, remove the gear case outlet port dampener and firewall restrictor union and check for clogged condition.
Intermittent indication.	1. Defective or broken wiring.	1. Disassemble electrical connectors at indicator and transmitter. Check for broken wires. Troubleshoot system wiring as necessary.
Circuit breaker trips.	1. Short in system.	1. Disconnect indicator and reset circuit breaker. If breaker trips remove transmitter electrical connector and reset. If breaker trips wiring between cockpit and nacelle is shorted.

OIL PRESSURE INDICATING – MAINTENANCE PRACTICES

1. Removal/Installation – Oil Pressure Indicating

A. Removal – Oil Pressure Transmitter

(1) Ensure electrical power is OFF.

(2) Remove access panel forward of main landing gear wheel well doors.

(3) Disconnect electrical connector.

(4) Remove tube assemblies from tee fitting on oil pressure transmitter.

(5) Remove tee fitting from transmitter.

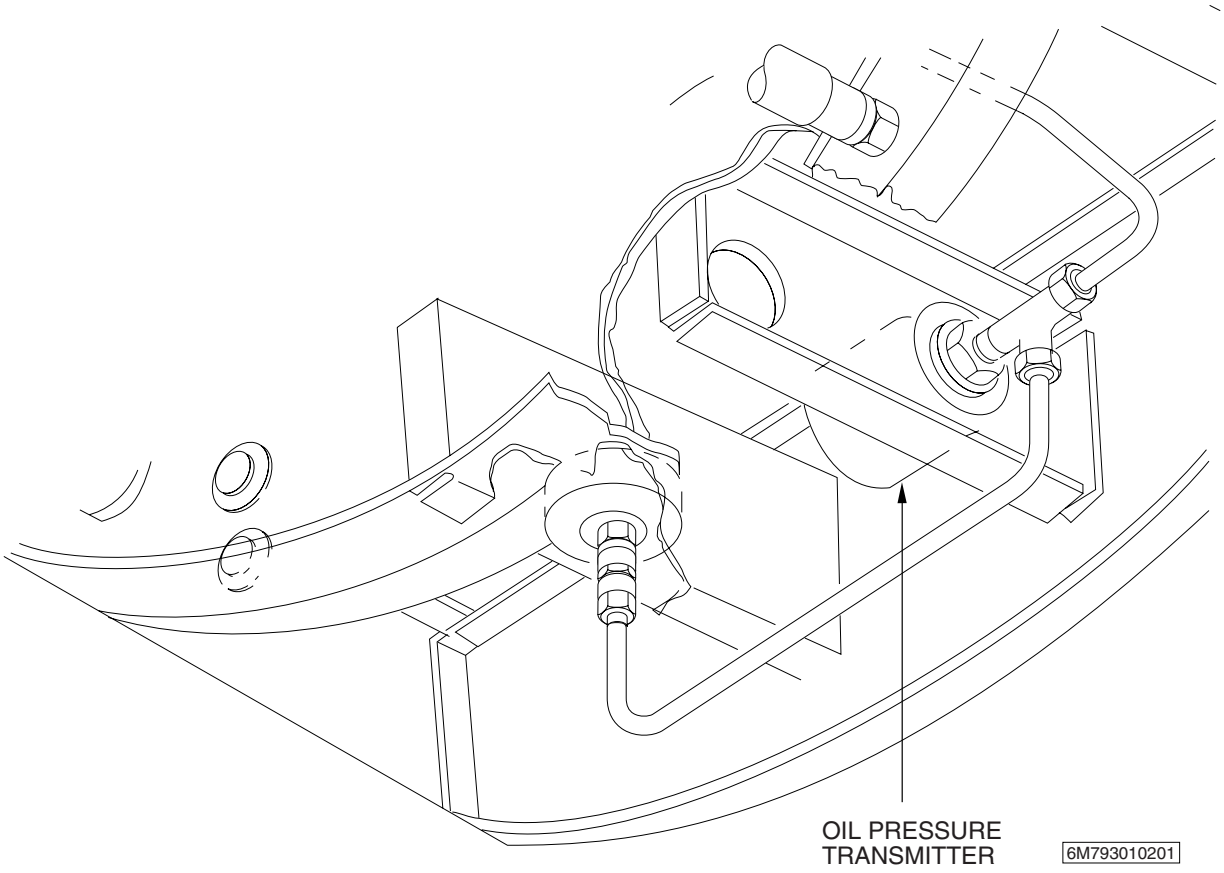
(6) Remove retaining nut, and washer securing transmitter to bracket. Remove transmitter.

B. Installation – Oil Pressure Transmitter

(1) Ensure electrical power is OFF.

(2) Install oil pressure transmitter in bracket and secure with washer and nut.

(3) Install tee fitting and O-ring (P/N MS28778-4) on transmitter.



Oil Pressure Transmitter
Figure 201

(4) Install tube assemblies on transmitter tee fitting.

(5) Connect electrical connector and safety.

(6) Turn electrical power ON.

(7) Operational check by running engine and check for leaks.

(8) Replace access panel.

C. Removal – Oil Pressure Indicator

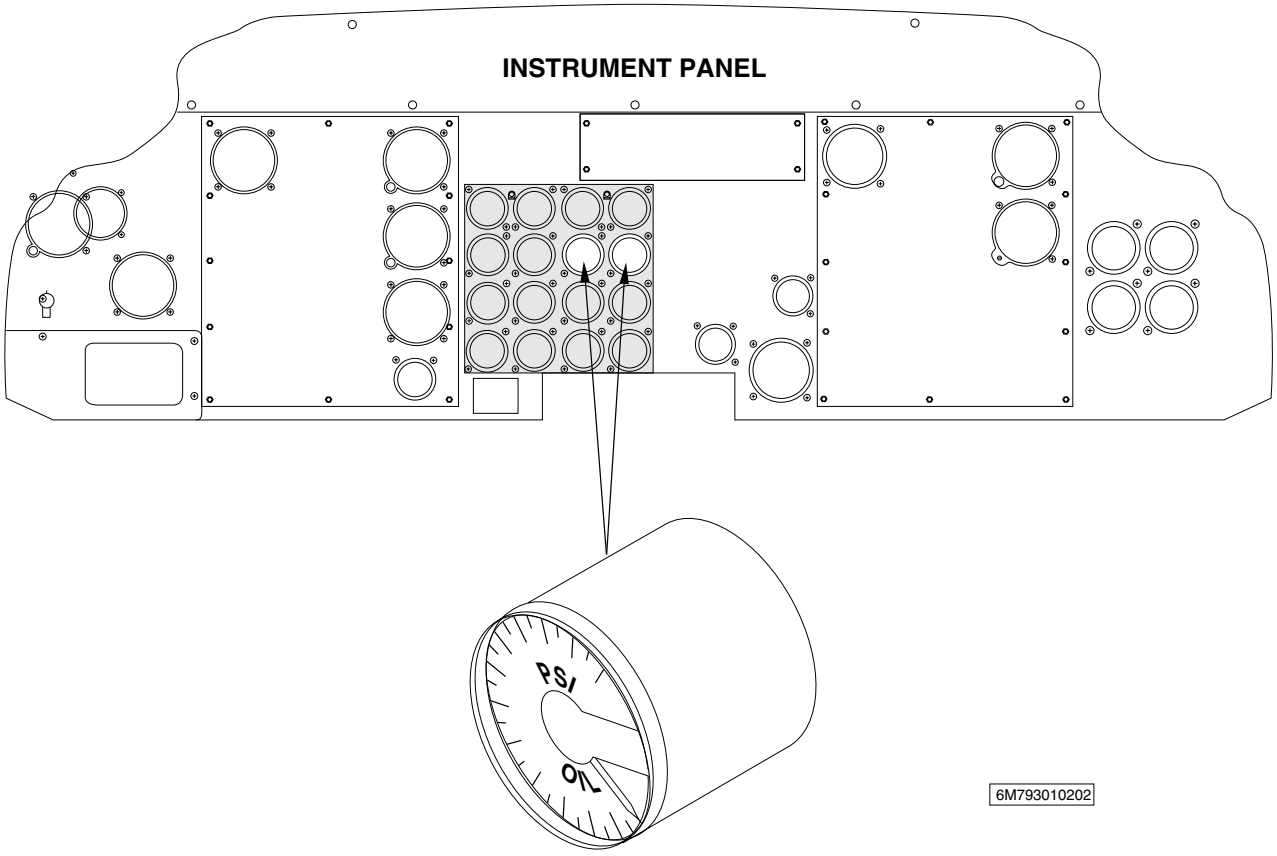
(1) Ensure electrical power is OFF.

(2) Remove the indicator post light cap.

(3) Loosen the retaining clamp screw (lower right screw).

(4) Withdraw the indicator from the instrument panel.

(5) Disconnect the electrical connector.



Oil Pressure Indicator
Figure 202

D. Installation – Oil Pressure Indicator

(1) Ensure electrical power is OFF.

(2) Connect the electrical connector.

(3) Position indicator in instrument panel.

(4) Tighten retaining clamp screw (lower right screw).

(5) Install post light cap.

(6) Turn electrical power ON.

(7) Operational check indicator and light.

2. Adjustment/Test – Oil Pressure Indicating

A. Calibration Test – Oil Pressure Indicating

NOTE: If there is reason to believe the oil pressure indicating system is not functioning properly, the following test may be performed to determine accuracy.

(1) Remove the access panel forward of the main landing gear wheel well doors.

(2) Disconnect oil lines from oil pressure transmitter.

(3) Remove tee fitting from transmitter.

(4) Connect a pressure tester capable of supplying 0-200 psi accurate pressure of dry air.

(5) Energize the circuit.

(6) Apply pressure, testing each applicable value marked on indicator dial. Before reading indicator, lightly tap both transmitter and indicator.

(7) Remove the pressure tester.

(8) Install the tee fitting and oil lines.

(9) Install the access panel.

3. Inspection/Check – Oil Pressure Indicating

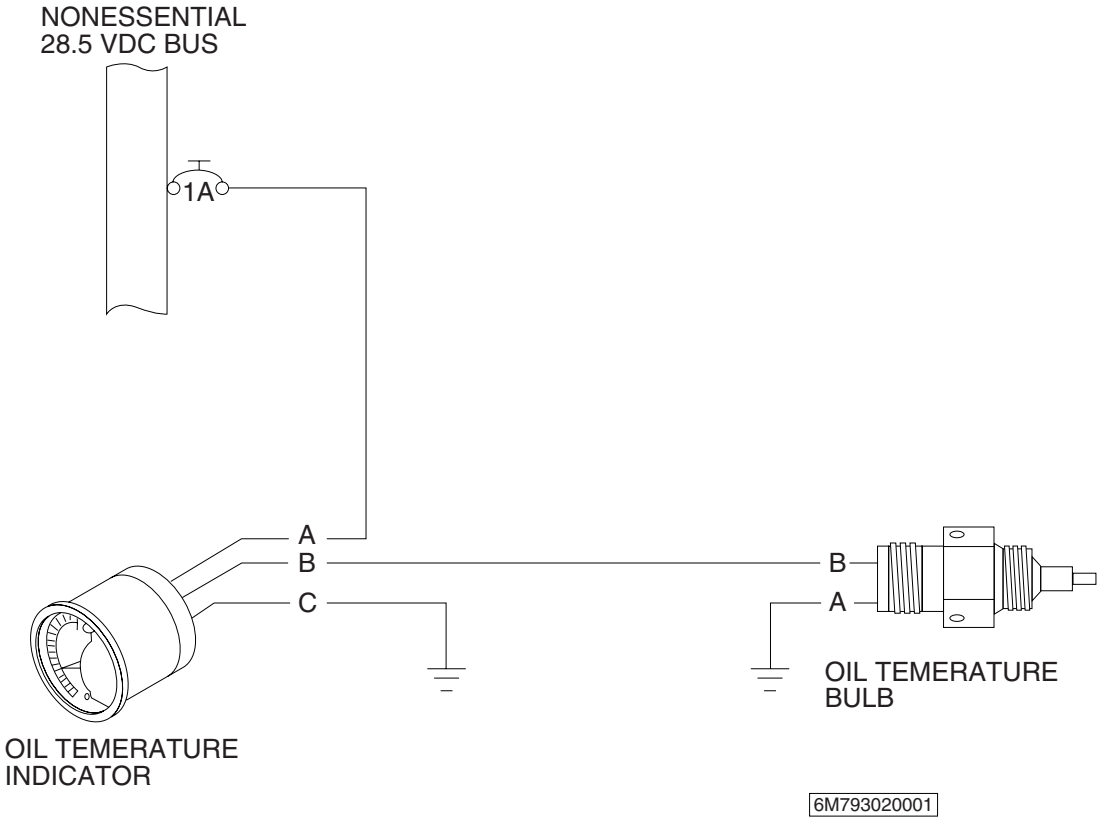
A. Inspect Oil Pressure Transmitter as follows:

(1) Inspect oil pressure transmitter for evidence of leakage, and security.

OIL TEMPERATURE INDICATING – DESCRIPTION AND OPERATION

1. General

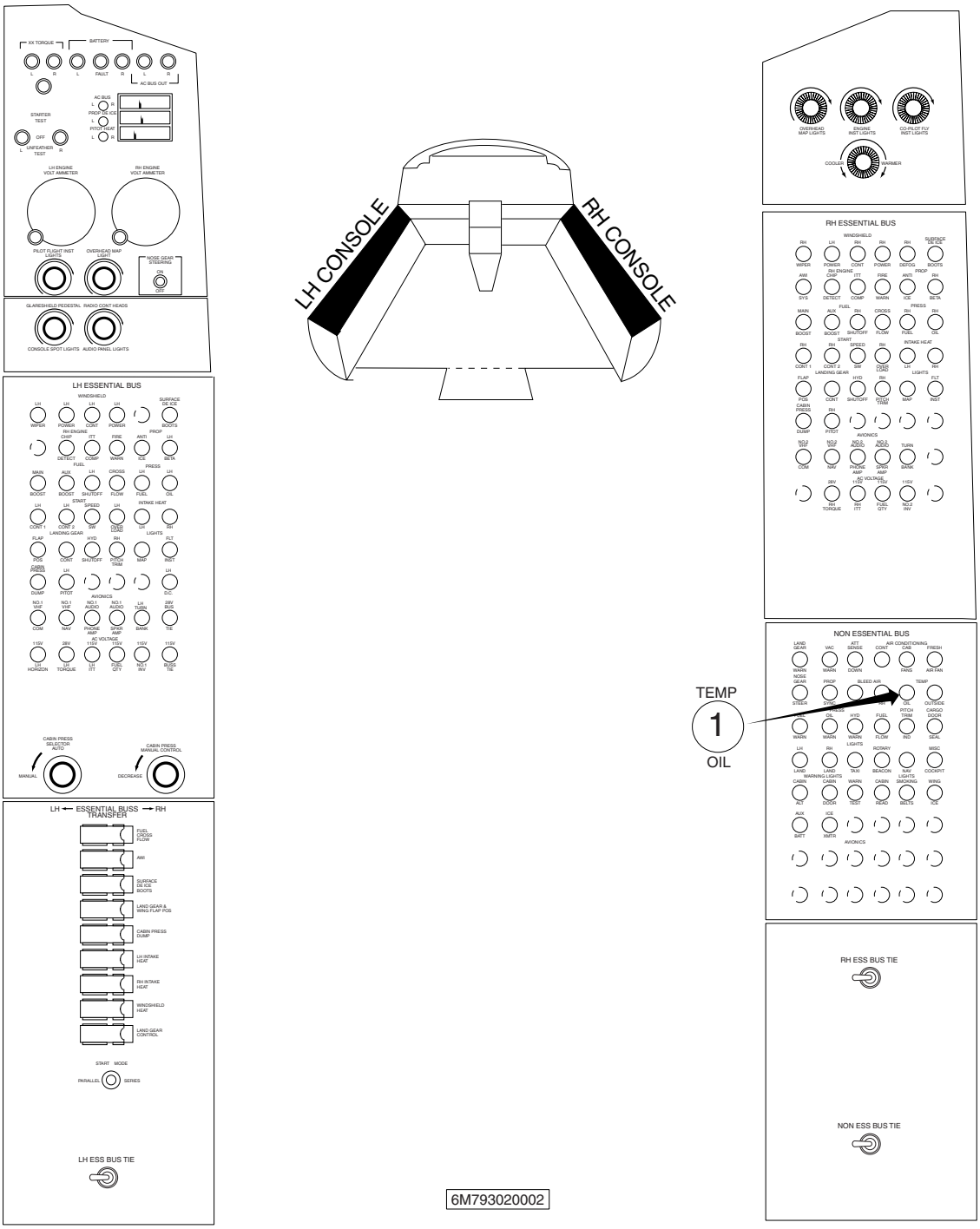
A. An oil temperature resistance type bulb installed in a boss on the rear of each engine gear case housing operates from the aircraft 28 VDC electrical system, and indicates in degrees celsius.



Oil Temperature Indicating
Figure 1

OIL TEMPERATURE INDICATING – DESCRIPTION AND OPERATION

B. The oil temperature indicating system is protected by a one ampere circuit breaker located on the copilots nonessential bus circuit breaker panel.



Oil Temperature Indicating Circuit Breaker
Figure 2

OIL TEMPERATURE INDICATING – FAULT ISOLATION

1. Troubleshooting – Oil Temperature Indicating

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No response on oil temperature indicator.	1. No DC power to indicator.	1 Check for response on other DC powered instruments.
	2. Defective or stuck indicator.	2. Tap indicator face, swap indicators left to right.
	3. Defective oil temperature bulb.	3. Remove electrical (hand shake) connectors for oil temperature bulb, Located in wire bundle left forward engine mount. Check wiring (see 79-30-02 of the Wiring Manual). If wiring is correct, check continuity of oil temperature bulb.
Intermittent indication on oil temperature indicator.	1. Defective or broken wiring.	1. Disassemble electrical connector at indicator and check No. 3 above, checking for broken wires. Repair system as necessary.
Circuit breaker trips.	1. Short in oil temperature system.	1. Disconnect indicator and reset breaker if breaker trips. Remove temperature bulb electrical connectors and reset breaker. If breaker trips, system wiring between cockpit and nacelle is shorted.

OIL TEMPERATURE INDICATING – MAINTENANCE PRACTICES

1. Removal/Installation – Oil Temperature Indicating

A. Removal – Oil Temperature Bulb

(1) Ensure electrical power is OFF.

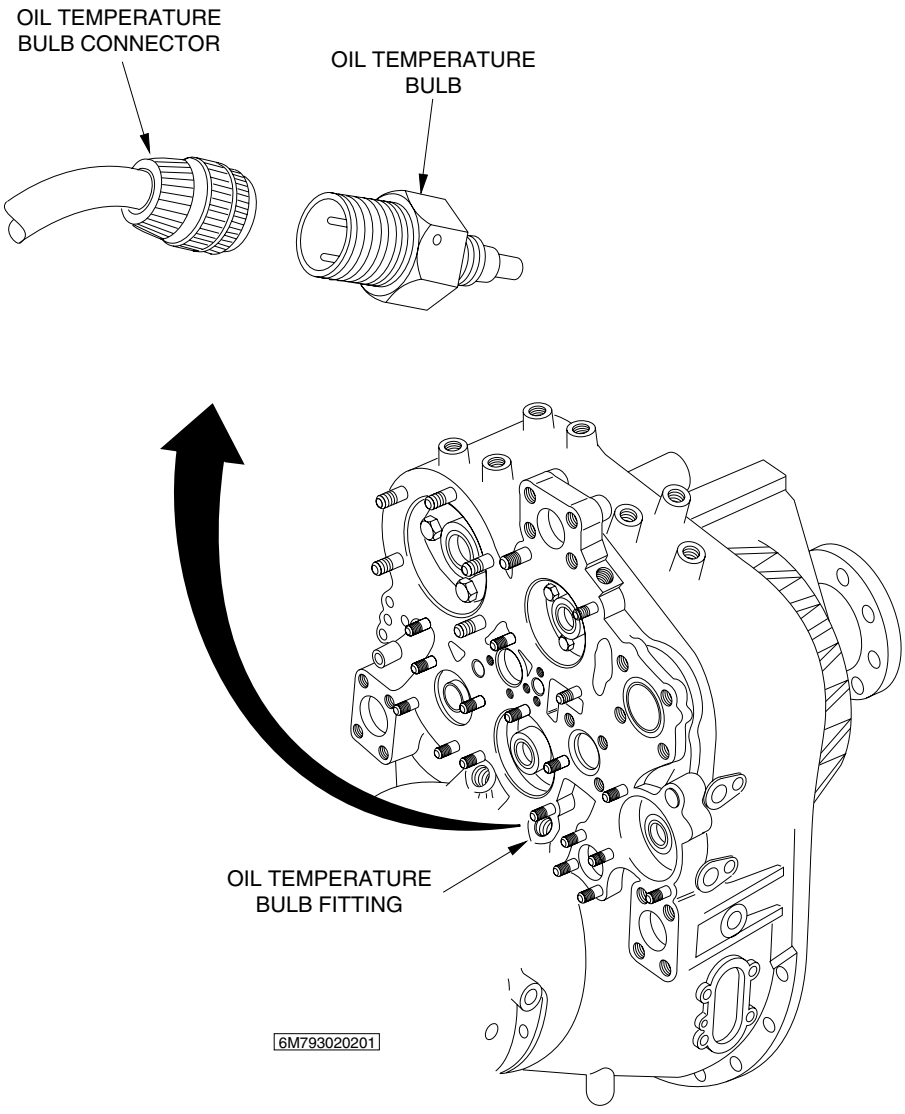
(2) Remove starter-generator (see Chapter 24).

(3) Remove hydraulic pump (see Chapter 29).

(4) Remove tachometer-generator (see Chapter 77).

(5) Disconnect oil temperature bulb electrical connector.

(6) Unscrew the oil temperature bulb from the gear case fitting and remove.



Oil Temperature Bulb
Figure 201

B. Installation – Oil Temperature Bulb

NOTE: Oil temperature bulb is supplied with a metal washer. Discard washer and use packing P/N 59413-554.

(1) Ensure electrical power is OFF.

(2) Install oil temperature bulb with new packing. Torque temperature bulb 65 to 70 inch-pounds (7.34 to 7.91 N·m).

(3) Connect electrical connector to the oil temperature bulb and safety.

(4) Install tachometer-generator (see Chapter 77).

(5) Install hydraulic pump (see Chapter 29).

(6) Install starter-generator (see Chapter 24).

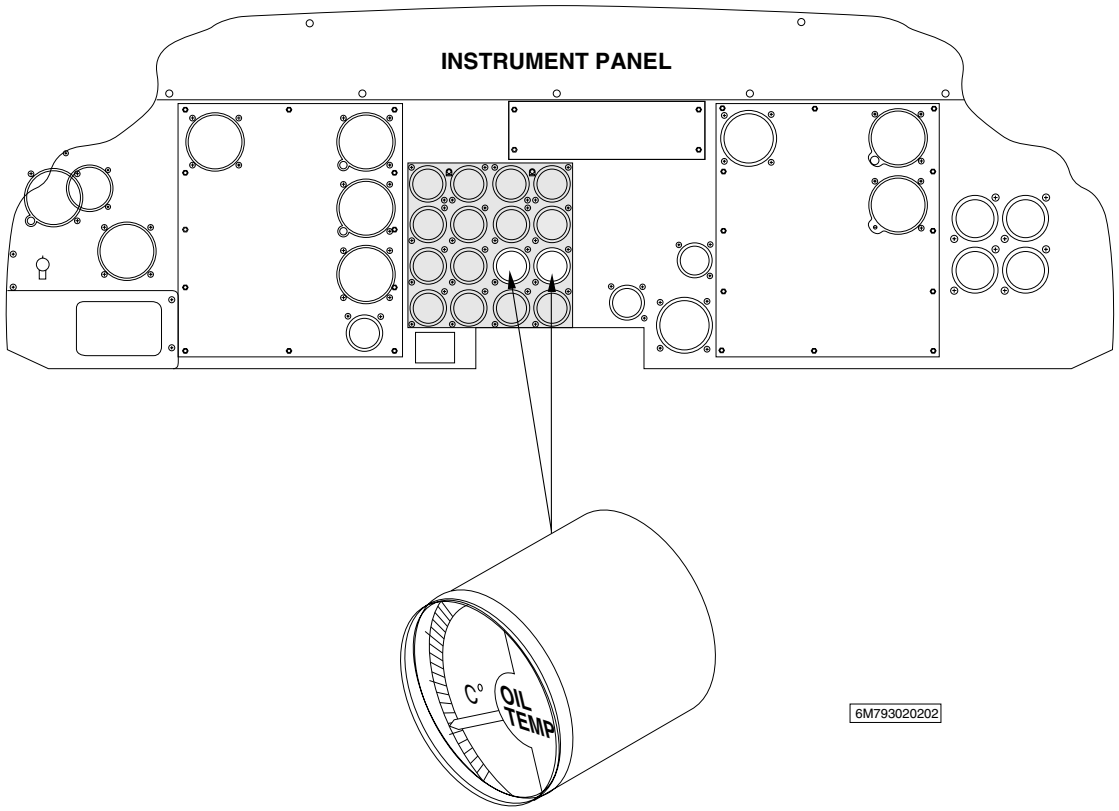
(7) Turn electrical power ON.

(8) Operate engine and check for leaks and proper temperature indication.

C. Removal – Oil Temperature Indicator

(1) Ensure electrical power is OFF.

(2) Remove the indicator post light cap.



Oil Temperature Indicator
Figure 202

(3) Loosen the retaining clamp screw (lower right screw).

(4) Withdraw the indicator from the instrument panel.

(5) Disconnect the electrical connector from the indicator.

D. Installation – Oil Temperature Indicator

(1) Ensure electrical power is OFF.

(2) Connect the electrical connector to the indicator.

(2) Position the indicator in the instrument panel.

(3) Tighten the retaining clamp screw (lower right screw).

(4) Install the post light cap.

(5) Connect the electrical connector to the indicator.

(6) Turn electrical power ON.

(7) Operational check indicator and light.

2. Inspection/Check – Oil Temperature Indicating

A. Inspection/Check – Oil Temperature Bulb

NOTE: If there is reason to believe the oil temperature indicating system is not functioning properly, the following check may be performed to determine accuracy.

(1) Visually inspect oil temperature bulb for cracks, damage, corrosion, or contamination that would affect temperature sensing.

(2) Check resistance across oil temperature bulb electrical pins, see Table 201 for ohm values.

(3) Inspect oil temperature bulb for bent, broken, or burned pins.

(4) Inspect threads for stripped, galled, or peened threads.

OHMS	C°	F°
68.27	-70	
74.24	-50	
80.56	-30	-22
87.04	-10	+14
90.38	0	32
93.80	10	50
100.91	30	86
108.39	50	122
120.36	80	170
128.85	100	212
137.78	120	
151.91	150	

Ohm Values
Table 201

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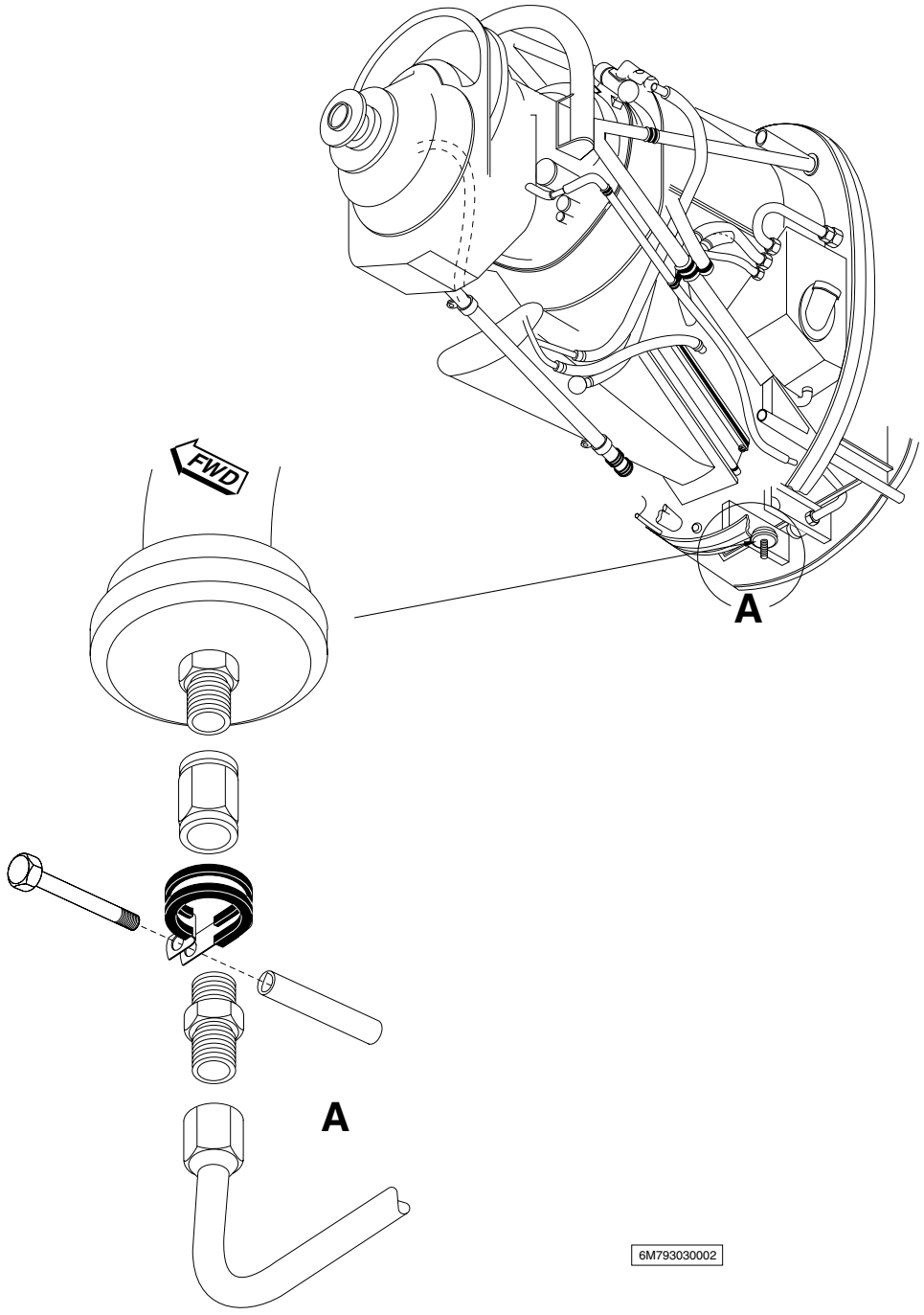
OIL PRESSURE SWITCH – DESCRIPTION AND OPERATION

1. General

A. The pressure switch is mounted to the bulkhead in the lower forward section of the nacelle forward of the main landing gear wheel well. Oil pressure to the switch is through a tube assembly connected to the pressure transmitter tee fitting.

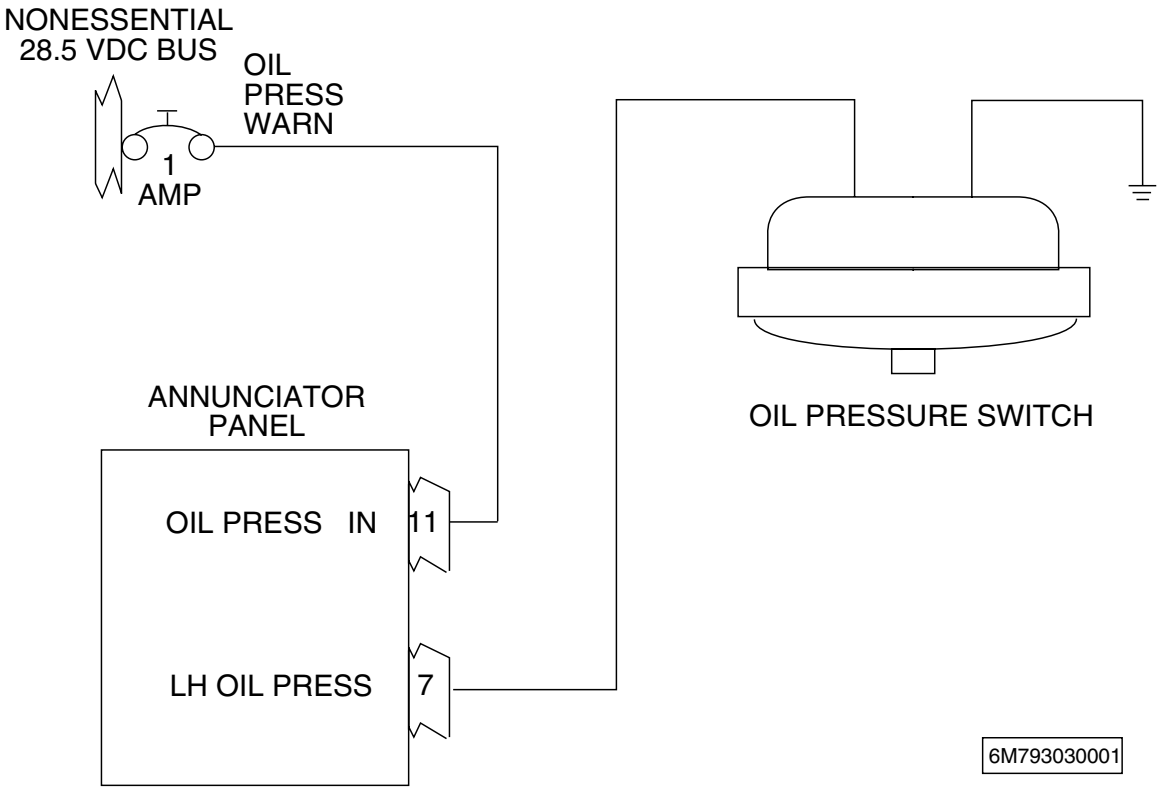


Oil Pressure Switch
Figure 1

OIL PRESSURE SWITCH – DESCRIPTION AND OPERATION

(1) The oil pressure switch, set to close at 40 psi decreasing oil pressure, illuminates the oil pressure warning light on the annunciator panel.

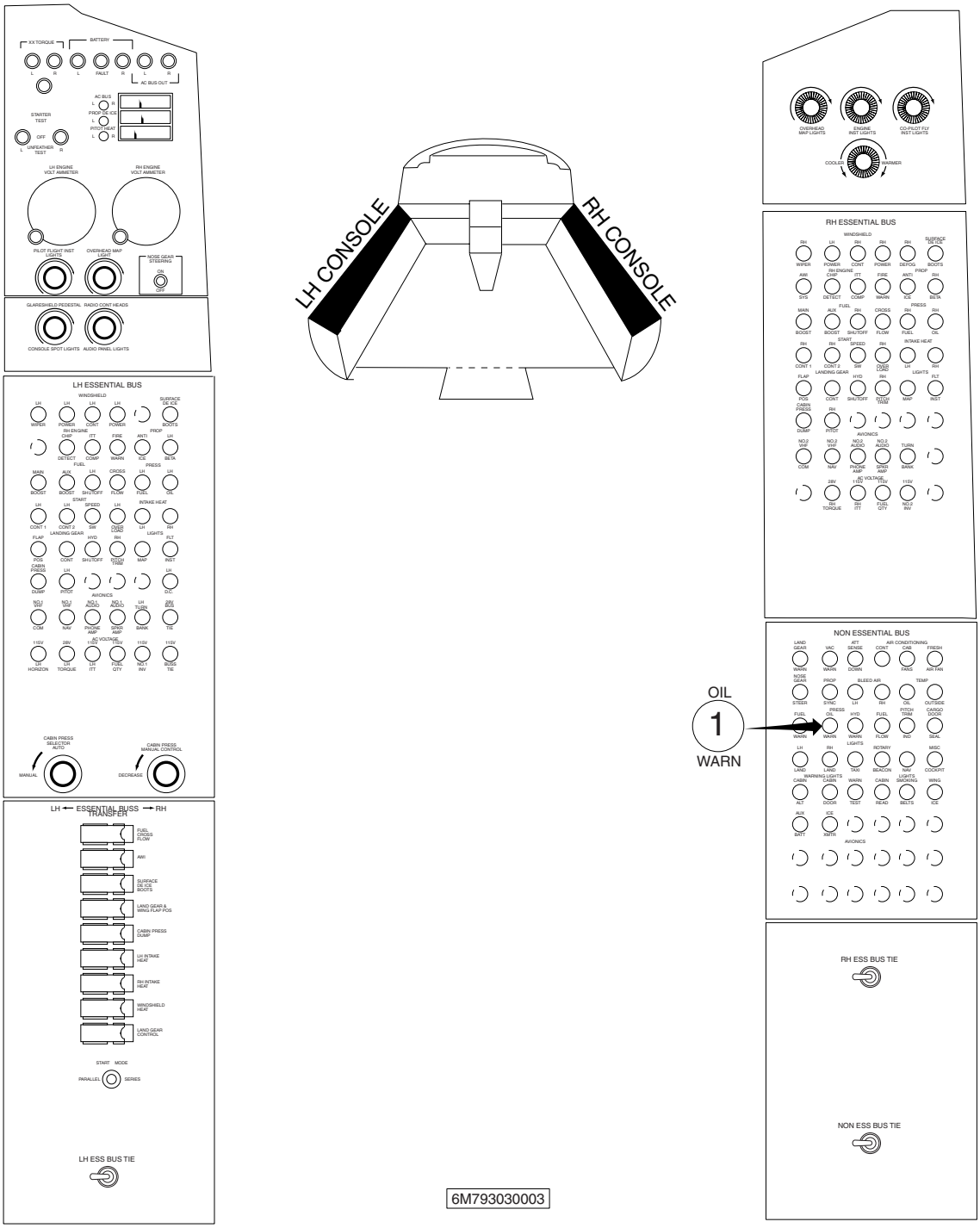
NOTE: SA226 models T201-239, AT001-013, TC201-207 were delivered with oil pressure switches (P/N M4011-54) set at 50 psi. A lower oil pressure limit of 40 psi has since been approved for all SA226 aircraft. To employ the lower limits, replace the oil pressure switches P/N M4011-54 with P/N M4011-44 and change the oil pressure gauge markings.



Oil Pressure Switch Wiring
Figure 2

OIL PRESSURE SWITCH – DESCRIPTION AND OPERATION

(2) Electrical power for the system is supplied by the nonessential DC bus through a one amp circuit breaker.



Oil Pressure Switch Circuit Breaker Figure 3

OIL PRESSURE SWITCH – FAULT ISOLATION

1. Troubleshooting – Oil Pressure Switch

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Low oil pressure light will not illuminate with engine shut down.	1 Defective bulb.	1. Check annunciator panel press-to-test if light fails to illuminate replace bulb.
	2. Defective oil pressure switch.	2. Remove oil pressure switch wiring. See 79-30-03 of the Wiring Manual. If wiring is correct check continuity of oil pressure switch.
Intermittent light indication.	1. Defective or broken wiring.	1. Troubleshoot system as necessary. See 79-30-03 of Wiring Manual.

OIL PRESSURE SWITCH – MAINTENANCE PRACTICES

1. Removal/Installation – Oil Pressure Switch

A. Removal – Oil Pressure Switch

(1) Ensure electrical power is OFF.

(2) Remove access panel forward of main landing gear wheel well doors.

(3) Disconnect electrical leads from pressure switch.

(4) Remove tube assembly from bottom of pressure switch.

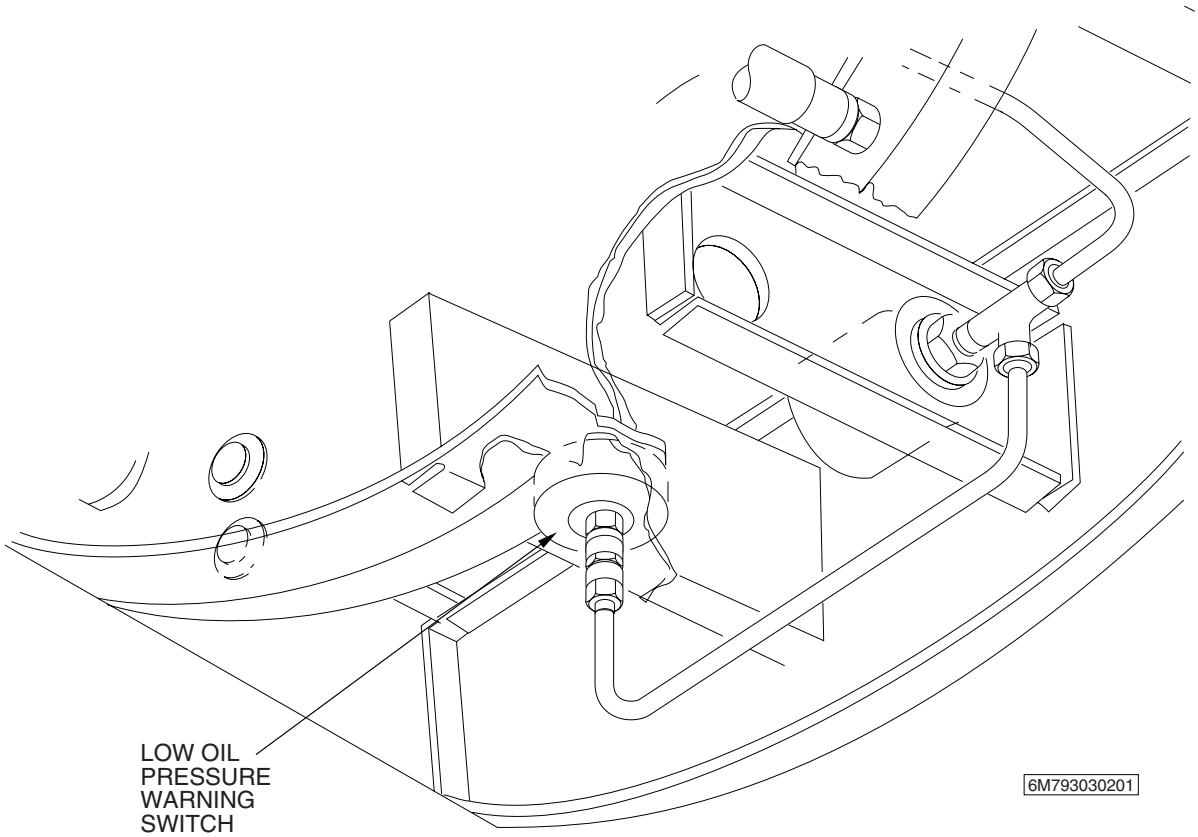
(5) Remove mounting bolt, washer, spacer, and nut securing pressure switch. Remove switch.

B. Installation – Oil Pressure Switch

(1) Ensure electrical power is OFF.

(2) Position oil pressure switch in mounting clamp and secure to bulkhead with bolt, washer, spacer, and nut.

NOTE: On early models, position pressure switch and install mounting bolts, washers, and nuts.



Oil Pressure Switch
Figure 201

(3) Install tube assembly to bottom of pressure switch.

(4) Connect electrical leads to pressure switch.

(5) Install access panel.

2. Adjustment/Test – Oil Pressure Switch

A. Adjustment/Test – Low Oil Pressure Switch

(1) Remove lower forward engine cowling.

(2) Disconnect oil line from oil pressure switch.

(3) Connect an accurate pressure tester capable of supplying 0-60 psi of dry air.

(4) Energize the circuit.

(5) Apply 60 psi pressure; check low oil pressure light on the annunciator panel; light should be extinguished.

(6) Slowly bleed off pressure; light on annunciator panel should illuminate at 40 psi (50 psi for models T201-239, AT001-013, TC201-207).

(7) Remove the pressure tester.

(8) Install the oil line to the pressure switch.

(9) Install the lower forward engine cowling.

3. Inspection/Check – Oil Pressure Switch

A. Inspect Oil Pressure Switch as follows:

(1) Inspect oil pressure switch for evidence of leaks, and security.

MAINTENANCE MANUAL

CHAPTER 80 – STARTING

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MAINTENANCE MANUAL

CHAPTER 80 – STARTING

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CHAPTER 80 – STARTING

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STARTING – DESCRIPTION AND OPERATION

1. General

A. SA226 series aircraft use a separate and automatic start system for each engine. An electric starter is used to crank the engine when the aircraft is on the ground. An oil pump is used to unfeather the prop in the air. DC power for the starter, the unfeather pump as well as the aircraft electrical buses, is provided by two 24 volt batteries. (For more information on the batteries, see Chapter 24). If the batteries are in good condition, they provide enough power to start both engines completely unassisted. However, a receptacle is provided for a ground power unit (GPU), and it is normal procedure to cross generator start the second engine after a battery start. In addition, for cold weather battery starts, circuitry is provided to switch the batteries in series (48 volts) during the start cycle.

B. Definition of Terms

(1) Aborted Start - A start that is ceased before the completion of the start cycle either by the operator or by malfunction.

(2) Air Start - A start accomplished while the aircraft is airborne during which the unfeather pump is used to position the prop blades to allow airflow to spin (crank) the engine.

(3) APU Start – See GPU Start.

(4) Auto Start - An engine start controlled automatically from 10% to 55% by the speed switch or 10% to 60% by the speed switch in the Single Red Line computer.

(5) Battery Start - Starting an engine using only aircraft battery power.

(6) Cold Start 1 - Cold weather engine start or first start of the day usually done using series start mode.

(7) Cold Start 2 - A start in which the engine temperature (ITT, TIT, EGT, ET) remains well below the start limits.

(8) Cross Generator Start - Starting one engine with the other engine running and it's generator on the line.

(9) Emergency Shutdown – Stopping the engine using the stop and feather pull knob.

(10) GPU Start - Starting an engine with the assistance of a ground power unit.

(11) Ground Start - A start accomplished when the aircraft is on the ground in which the starter is used to crank the engine.

(12) High Residual Start - A start in which starter test switch is used to spin and cool the engine before the start cycle is initiated.

(13) Hot Start - A start in which the engine temperature (ITT, TIT, EGT, ET) meets or exceeds the start limits.

(14) Hung Start - A start in which normal acceleration of the engine during the start has ceased due to a malfunction. Usually between 18% to 28% for no fuel or no ignition.

(15) Intentional Emergency Shutdown – Stopping the engines for maintenance, troubleshooting, or training purposes using both the stop button and the stop and feather pull knob.

(16) Light Off - The point at which the combustible mixture of air and fuel is ignited. This point of combustion is noticed during the start cycle by a rapid rise in temperature on the (ITT, TIT, EGT, ET) indicator.

(17) Manual Start - An engine start controlled manually by the speed switch override switch on those aircraft so equipped or by the manual 10% to 60% switch on aircraft T276, 292-999.

(18) NTS Start – A start in which the negative torque system is checked, usually the first start of the day.

(19) Normal Start - A completed start in which nothing out of the ordinary occurred during the start cycle.

(20) Normal Shutdown - Stopping the engine using the stop switch (button).

(21) Parallel Start - Starting an engine when the batteries remain connected in parallel during the entire start cycle.

(22) Series Start - Starting an engine when the batteries are automatically connected in series at 10% during the start cycle.

(23) SRL – Single Red Line.

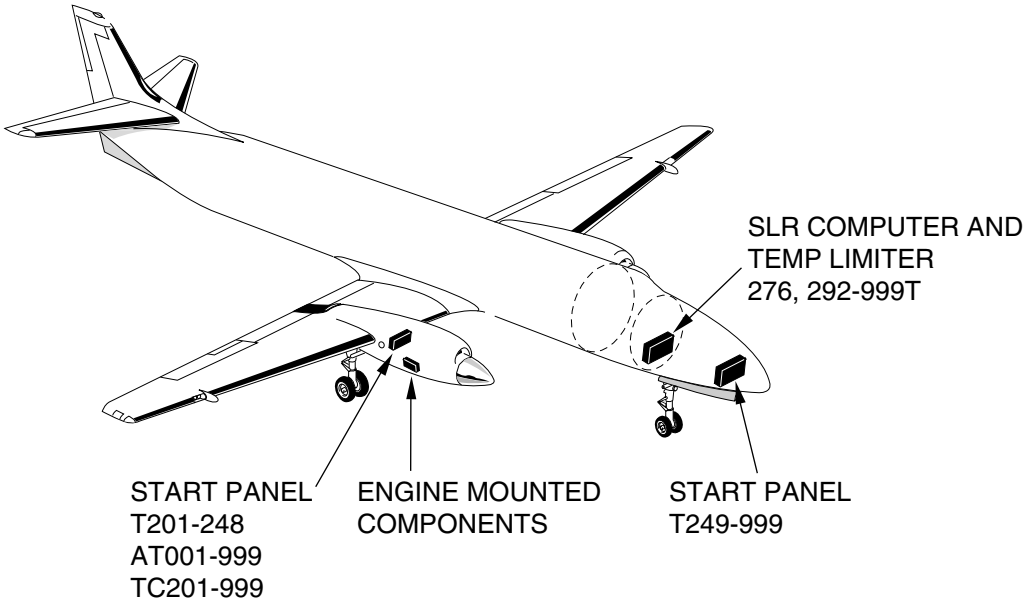
(24) Start Cycle - The time from initiation to the completion of the start. 0% RPM to 55% RPM for aircraft T201-275, 277-291, AT001-999, TC201-999. 0% RPM to 60% RPM for aircraft T276, 292-999. The start cycle is initiated by manually engaging and holding the start switch (button) until the RPM is above 10%. From 10% and up, the start system is normally automatically controlled by operation of engine speed sensing components.

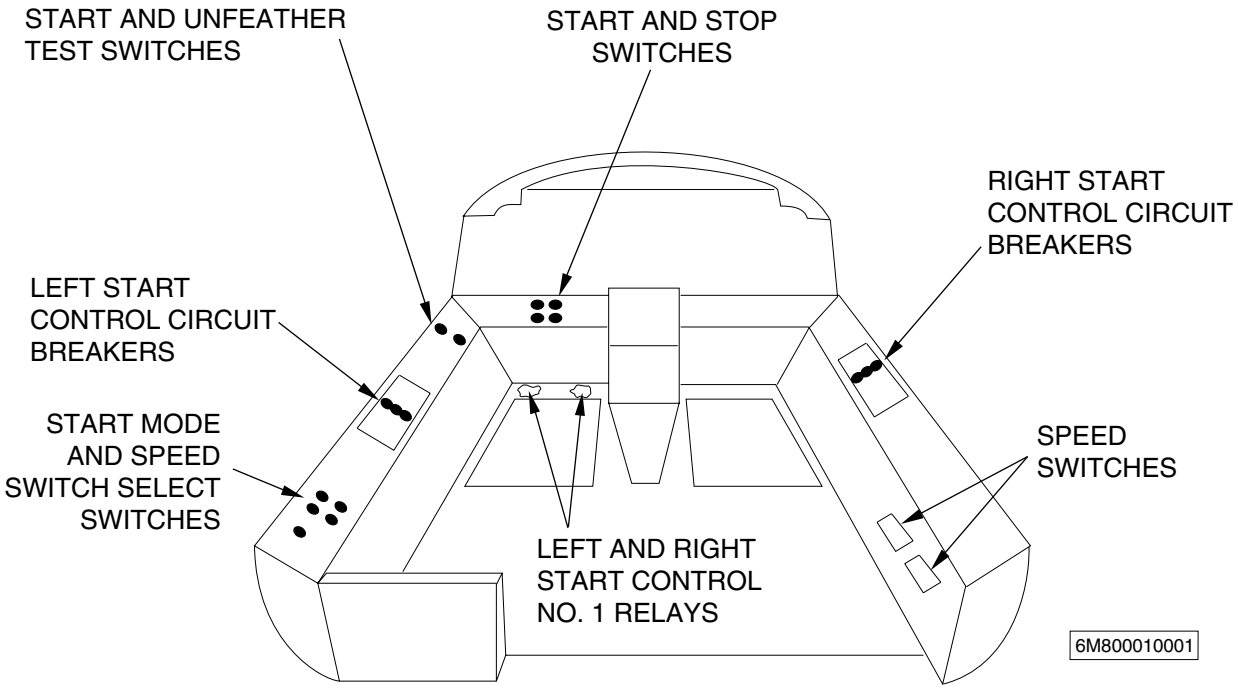
(25) Start Locks - Pins that hold the prop blades at a near zero blade angle to reduce starter load. The pins are operated by spring and centrifugal force.

(26) Start Mode - Series or parallel start when selected before initiation of the start.

(27) Starter Test - Using the starter test switch to spin the engine. Usually done to cool the engine or to build hydraulic pressure to operate the flaps.

(28) Unfeather Test - Using the unfeather test switch to operate the unfeather pump. The switch is used for NTS starts, to set flight idle blade angle, and to place the prop on the start locks.





Starting Component Location
Figure 1

STARTING – DESCRIPTION AND OPERATION

1. General

A. Components of the start system are located in the cockpit, engine, and on an electrical equipment panel (start panel). The start panel is located in the forward area of the right wheel well on aircraft T201-248, AT001-999, and TC201-999. On aircraft T249-999, the start panel is located in the nose baggage compartment on the aft side of the radome bulkhead.

B. Most components of the start system are separate to each engine start system. Three components that are common to both left and right start systems are the series-parallel relay, start mode switch, and the ground power or start mode relay.

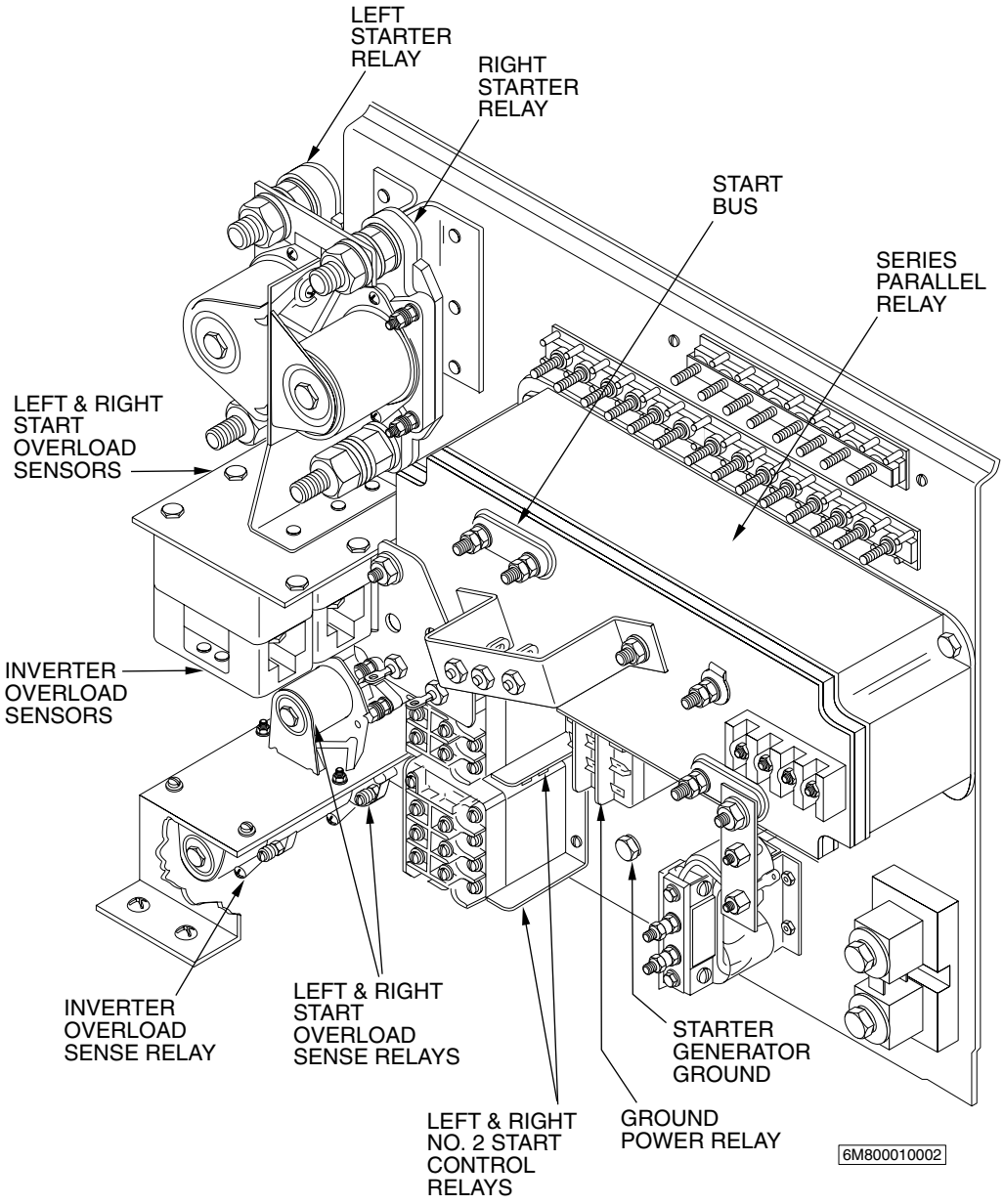
C. Series-Parallel Relay

(1) The series-parallel relay, located on the start panel, controls and buses the high amperage DC power from the batteries and GPU. When the relay is in its normally closed (relaxed) position, 24 volts from the batteries, 28 volts from the GPU is connected to the start bus and battery bus. When the relay is actuated, the batteries are switched to series mode and 48 volts is directed to the start bus, and on aircraft T201-278, AT001-056, TC201-229 to an overload sensor. The series-parallel relay is controlled by the No. 2 start control relay and the ground power relay or the start mode relay. The series-parallel relay will only switch to series mode during the start cycle, when the No. 2 start control relay closes (10%), provided the start mode switch is in series position, and the generators are off or the GPU is disconnected.

D. Ground Power Relay

(1) Aircraft T201-278, AT001-056, TC201-229.

(2) The normally closed contacts of the ground power relay give the coil of the series-parallel relay a ground The ground power relay will open if the start mode switch is positioned to parallel, or if the GPU is connected, or a generator is on. The ground power relay is located on the start panel.



Start Panel
(Right Wheel Well)
Figure 2

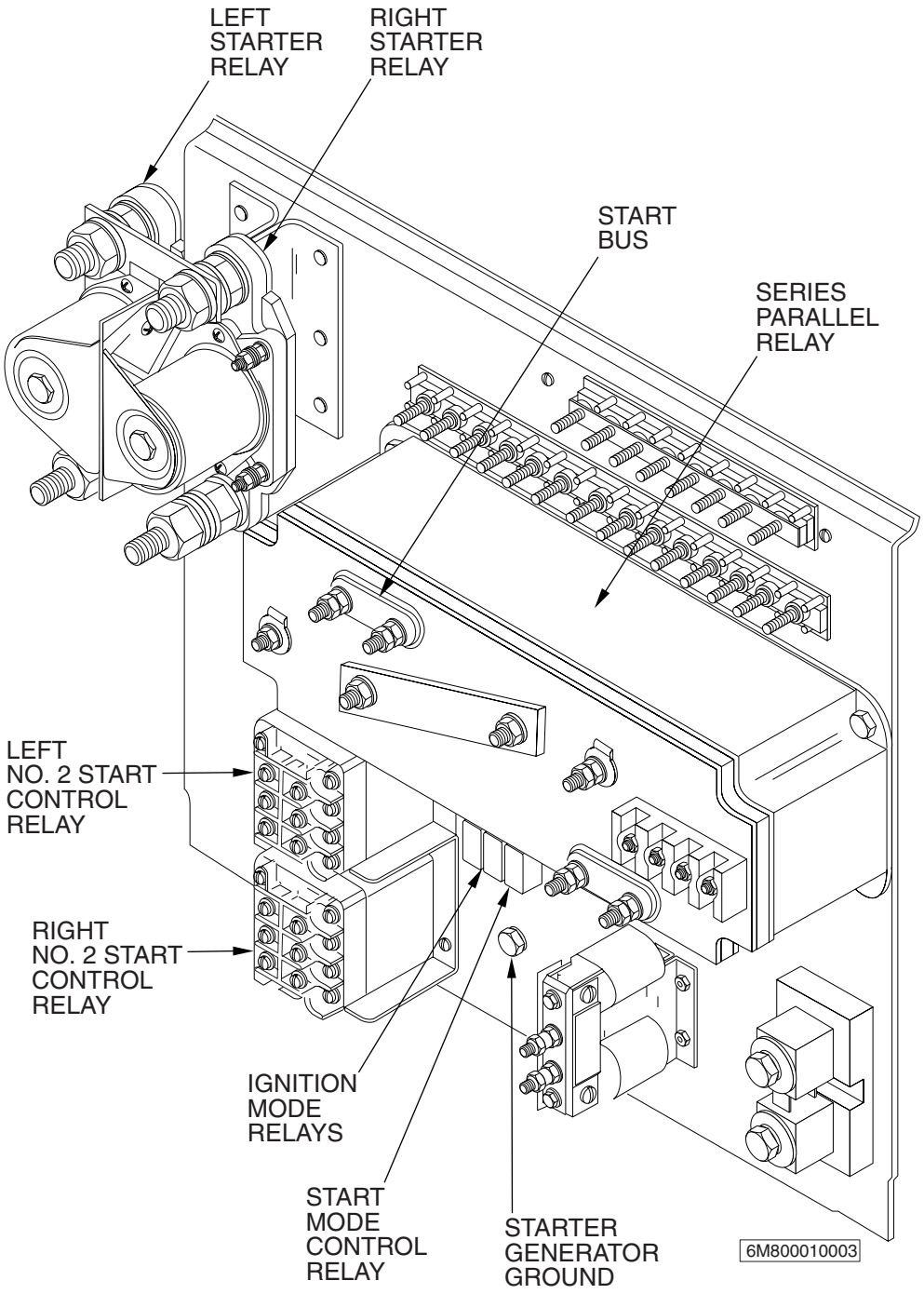
E. Start Mode Control Relay

(1) Aircraft T279-999, AT057-999, TC230-999

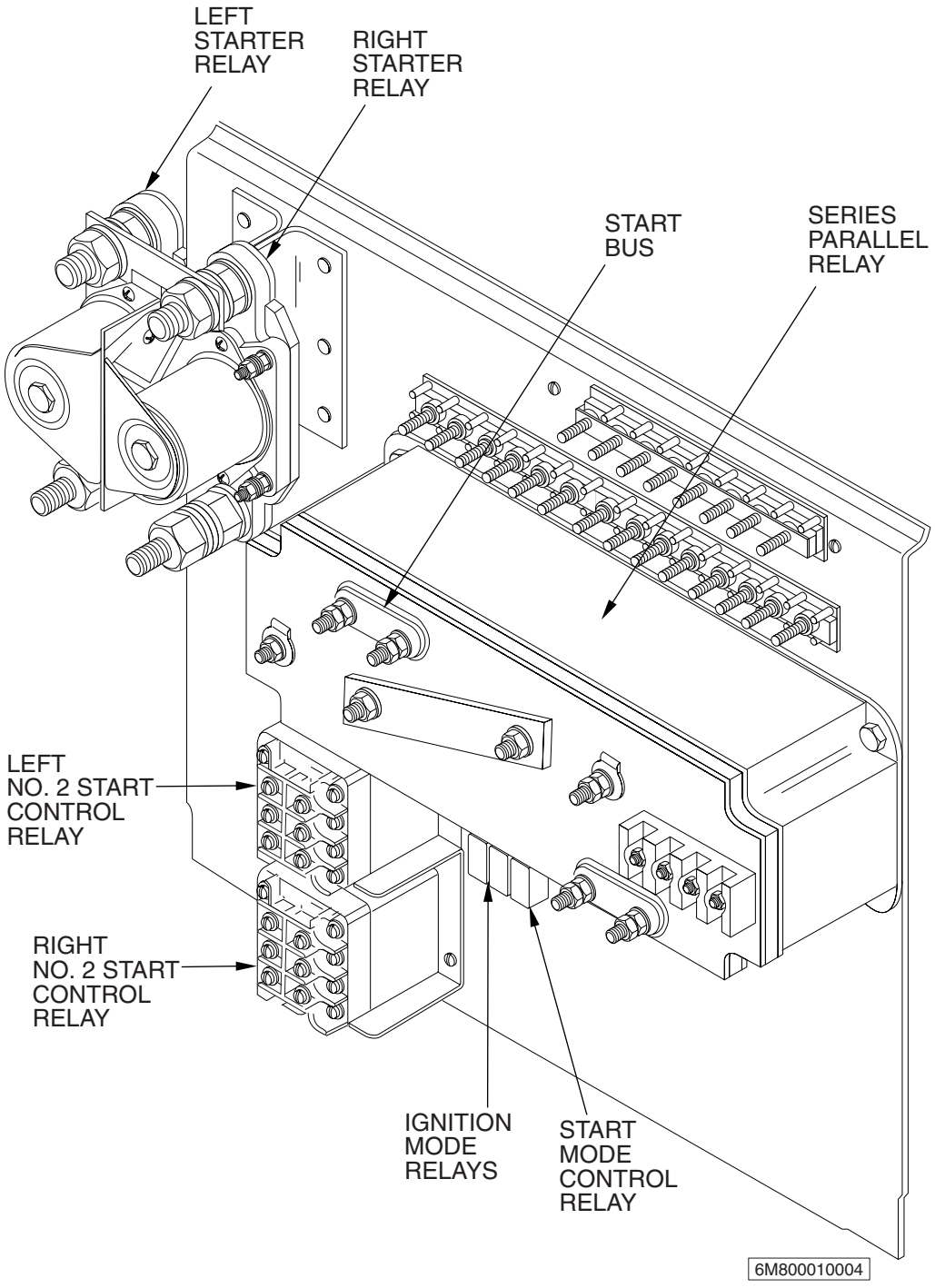
(2) When the start mode control relay closes, it gives the coil of the series-parallel relay a ground. The start mode relay closes during the start cycle at 10%, provided the start mode switch is positioned to series and the generators are off or the GPU is disconnected. The start mode control relay is located on the start panel.

F. Start Mode Switch

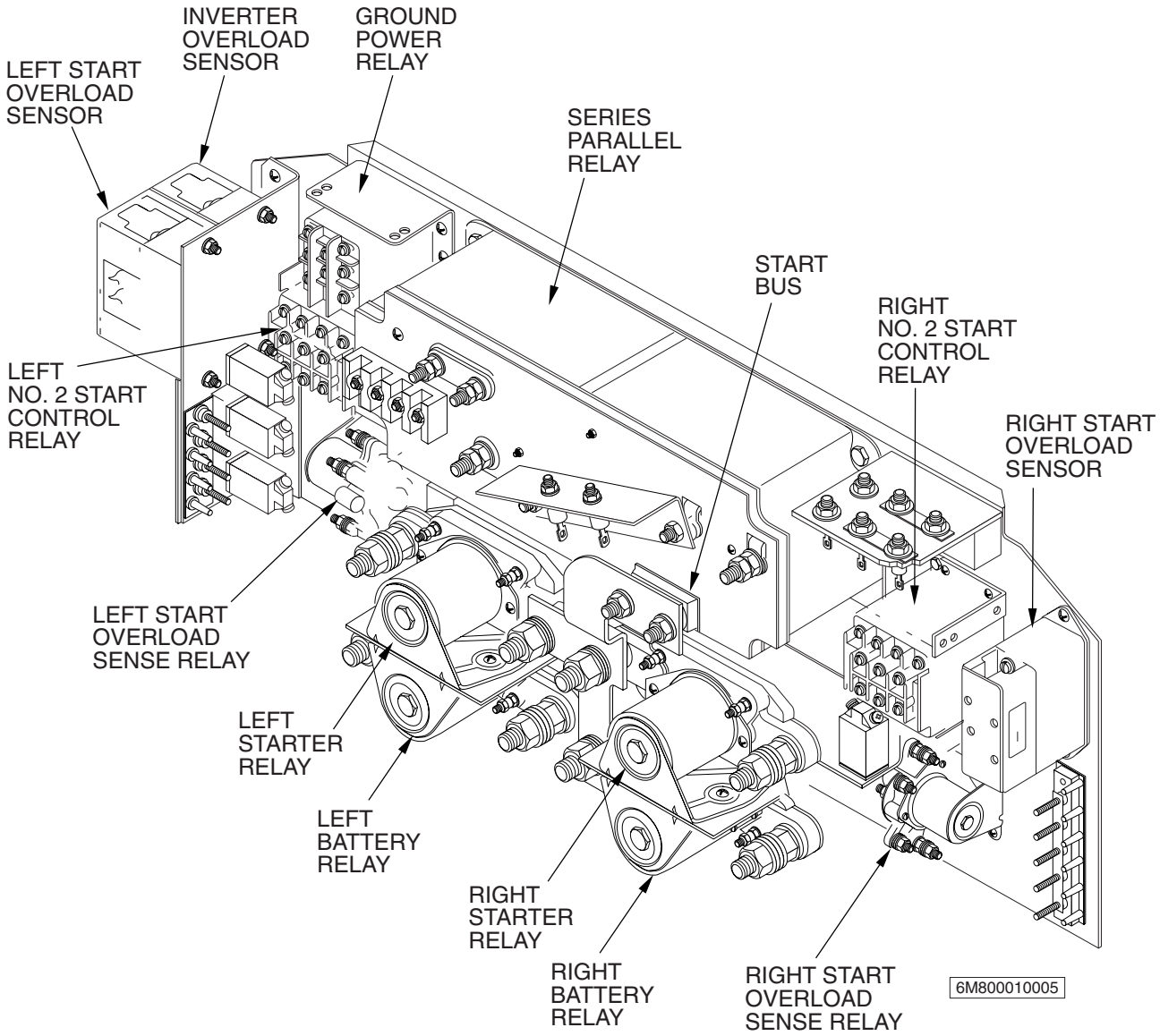
(1) Aircraft T241-999, AT013-999, TC208-999, and aircraft that complied with SB 24-006 have a start mode switch. The switch must be positioned to series to get a series start. The switch controls the ground power relay or the start mode relay. The switch is located on the aft portion of the left console. On aircraft without a start mode switch, a series start is automatic during a battery start.



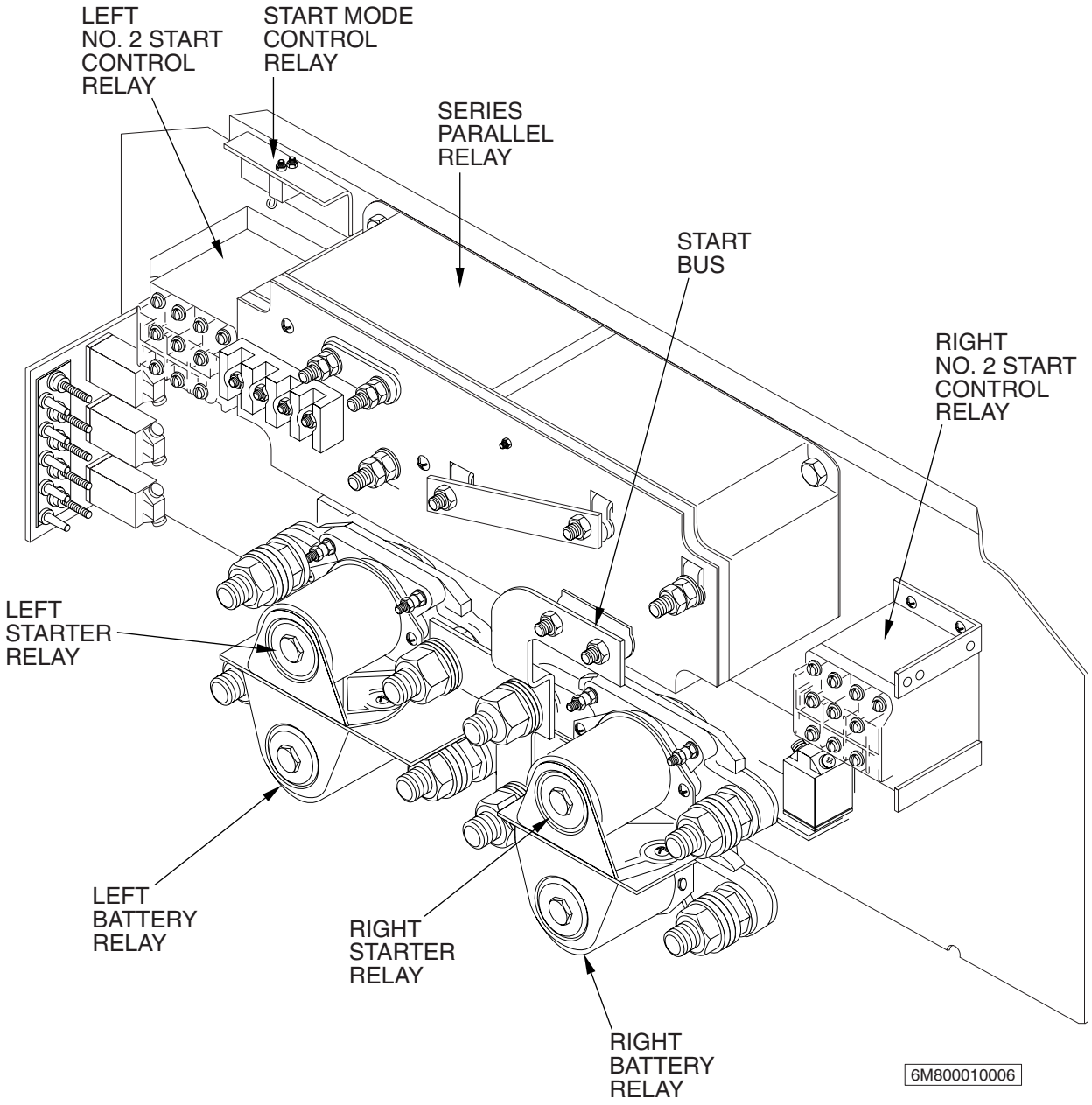
Start Panel
(Right Wheel Well)
Figure 3



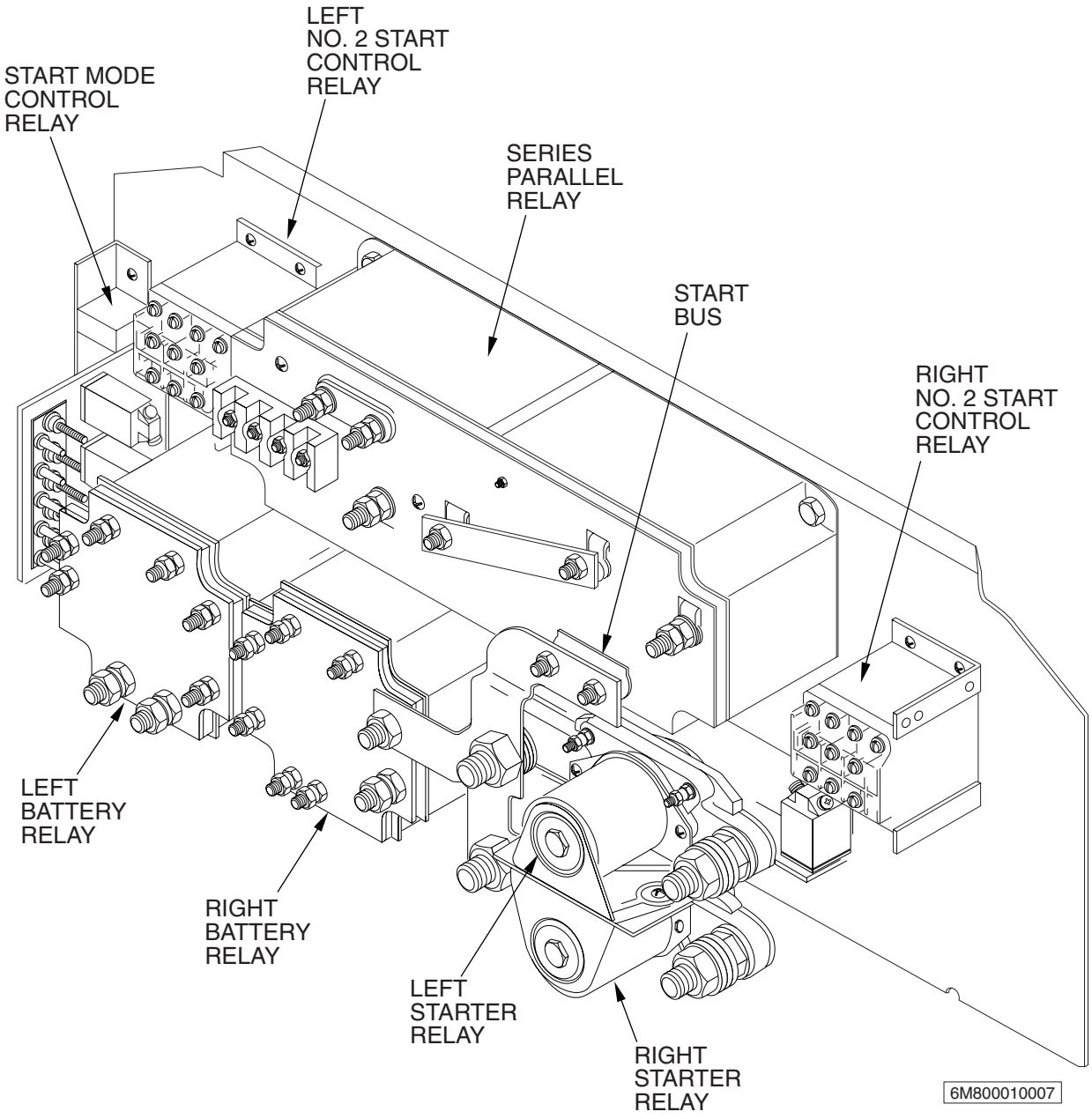
Start Panel
(Right Wheel Well)
Figure 4



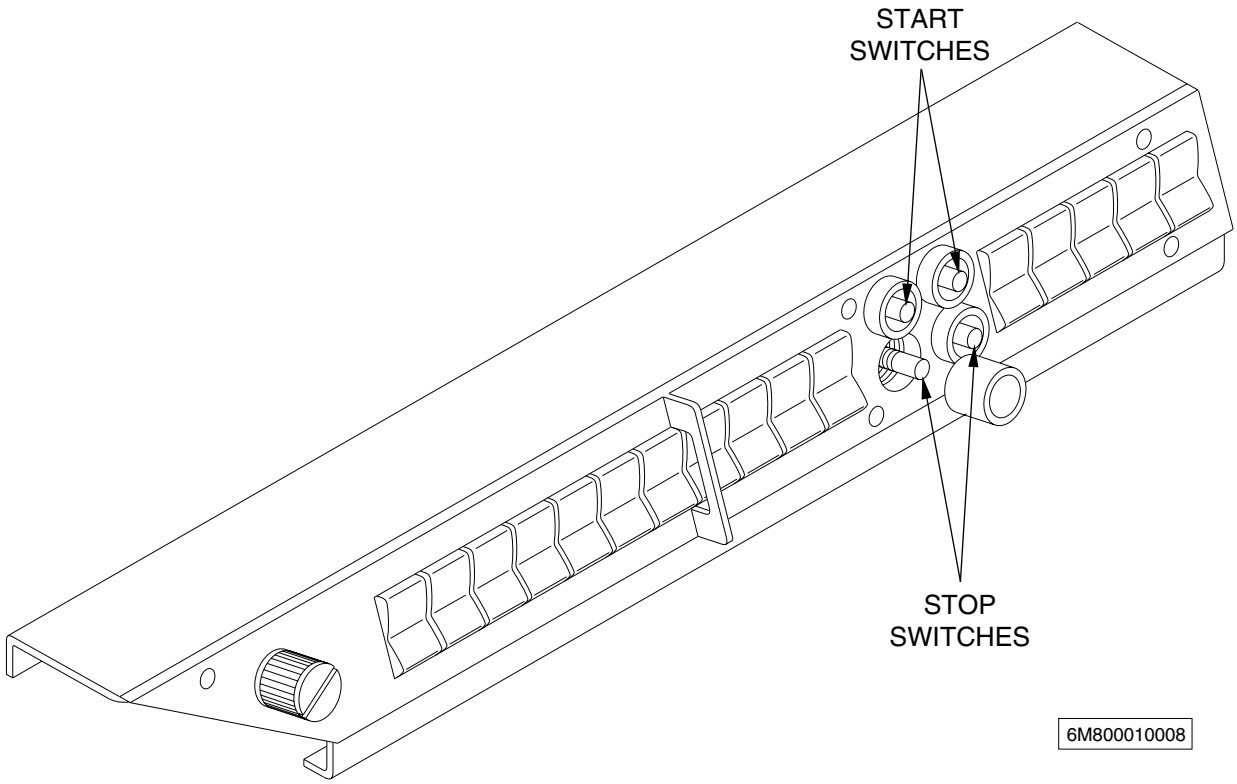
Start Panel
(Nose Baggage Compartment)
Figure 5



Start Panel
(Nose Baggage Compartment)
Figure 6



Start Panel
(Nose Baggage Compartment)
Figure 7



Start/Stop Switches
Figure 8

STARTING – DESCRIPTION AND OPERATION

NOTE: Airframe installed components that are separate to the left and right engine start systems as follows:

G. Starter Relays

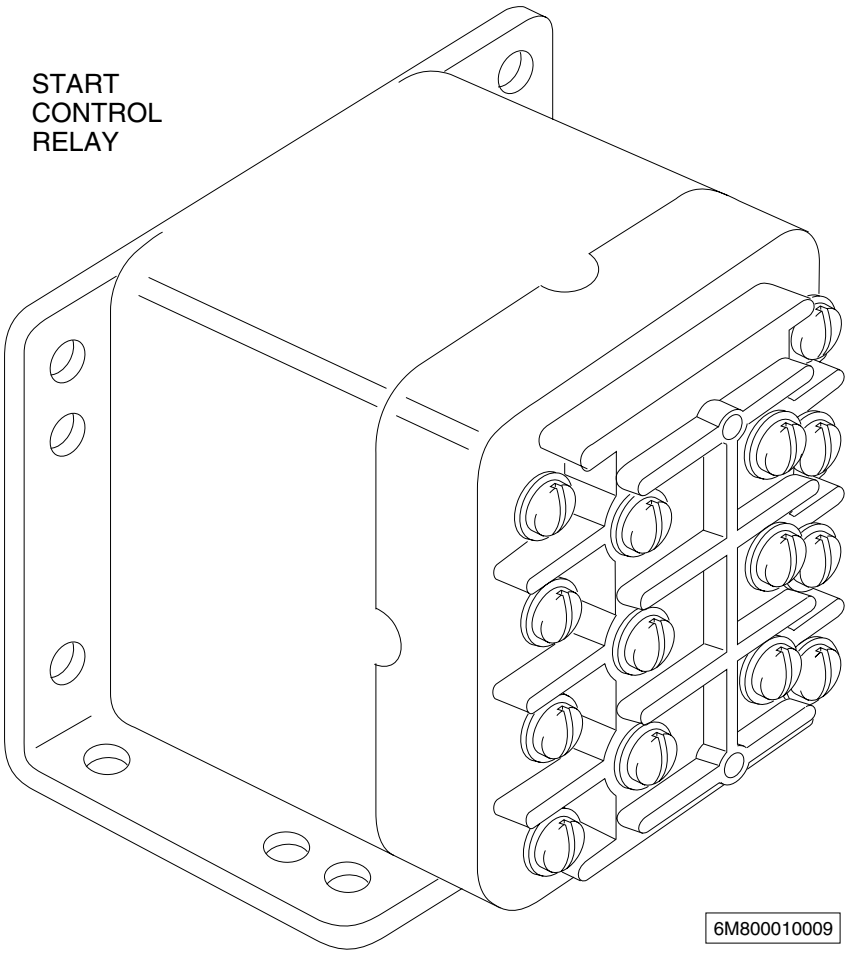
(1) The starter relay connects high amperage power from the start bus to the starter. The starter relay can be actuated from two sources; by the No. 1 start control relay during a normal start, or by the starter test switch. The starter relays are located on the start panel.

H. Start and Stop Switches (Buttons)

(1) The start and stop buttons are located on the left switch panel just inboard of the control column. The start switch initiates the start cycle by closing the No. 1 start control relay and by actuating the anti-ice lockout valve. After 10% RPM, manual actuation of the start button opens the fuel enrichment start pressure regulator (SPR) valve to prime the engine. On those aircraft with a SRL computer, the start button will override the automatic fuel enrichment. The stop button stops the engine by closing the engine fuel shutoff valve. During the start cycle, it also removes the ground from the coils of the No. 1 and No. 2 start control relays.

I. No. 1 Start Control Relays

(1) The No. 1 start control relays are located under the cockpit floor beneath the pilot's rudder pedals. The No. 1 start control relay is closed by the start button at the initiation of the start. It is locked in at 10% by the speed switch, and it remains closed until the completion of the start cycle. During a ground start, the relay closes the starter relay and opens the oil vent valve. For air start, it runs the unfeather pump. Also it connects power to the coil of the No. 2 start control relay and when the No. 2 start control relay closes at 10%, both are locked in.



Start Control Relay
Figure 9

J. No. 2 Start Control Relays

(1) When the No. 2 start control relay closes, it connects power to the engine fuel shutoff valve, to open the valve. It also connects power to the ignition exciter and to the start switch for manual SPR operation. Another contact of the relay connects power to the coil of the series-parallel relay during a ground start. The No. 2 start control relay are located on the start panel.

K. Speed Switches

(1) Aircraft T201-275, 277-291; AT001-999; TC201-999.

(2) The speed switches are located in the right console behind the aft access panel. The speed switch receives a RPM signal from the tachometer generator. This RPM signal is used by electronic circuitry within the speed switch to control three relay contacts. Two of these contacts are used to control the start cycle. The 10% contacts are normally open, and they close at 10% RPM. The 55% contacts are normally closed, and they open at 55% RPM. These contacts are wired to the start control relays so that the contacts close and lock in both relays at 10% and both contacts open at 55%.

L. SRL Computers

(1) Aircraft T276, T292-999.

(2) The SRL computers are located in the nose baggage compartment and are attached to the pressure bulkhead. The SRL computer controls the engine during start as well as when it is running. During start, the computer receives inputs from the EGT compensator and the prop governor monopole. The computer controls the engine temperature during (10 - 60%) and maintains it within limits by automatically controlling the fuel enrichment (SPR) valve.

(3) The computer contains speed sensing circuitry. The circuitry receives an RPM signal from the prop governor monopole. The speed circuitry controls three relay contacts. The 10% - 60% and the 10% to 80% contacts are used during the start cycle. The 10% to 60% contacts are normally open and they close from 10% to 60%. These contacts are wired to the start control relays so that they close and lock in both relays from 10% to 60%. The 10% to 80% contacts only open the primary valve.

M. Speed Switch Select Switch

(1) Aircraft T276, T292-999.

(2) The speed switch select switch is located on the left portion of the left console. The switch is a manual override for the speed switch contacts in the SRL computer.

N. Starter and Unfeather Test Switches

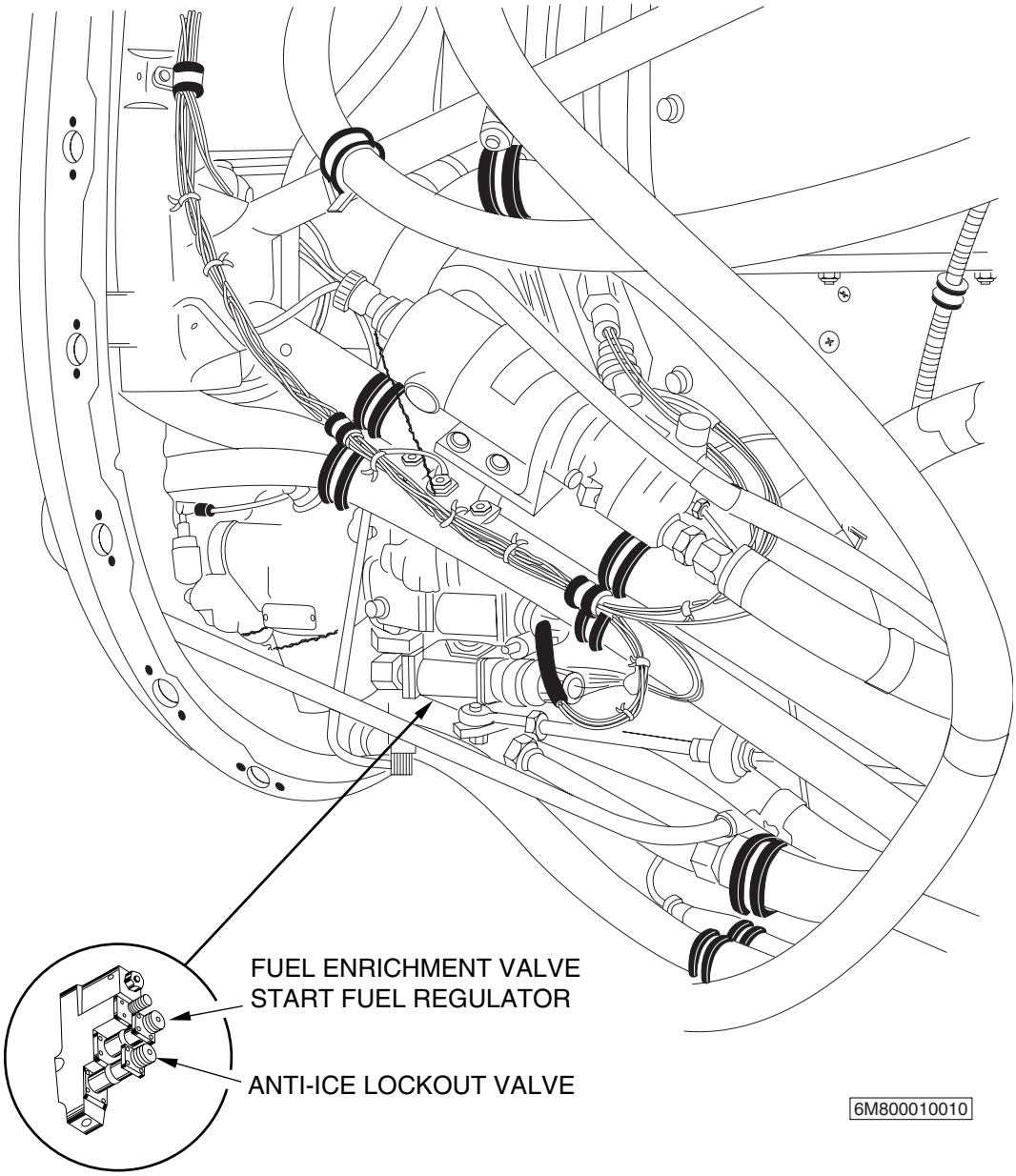
(1) Two guarded toggle or lever lock switches are located on the forward portion of the left console. These switches are used to spin the starter or to run the unfeather pump should it become necessary to cool the engine. They are also used to place the prop on the start locks or for other emergency, maintenance, or test purposes.

O. Scissor Switches (Squat Switch)

(1) A strut compression switch on each landing gear is used to switch power from the starter relay and oil valve during ground starts to the unfeather pump during air starts.

P. Squat Switch Relays (Aircraft T279-999, AT057-999, TC230-999)

(1) A relay is used to switch from ground start to air start rather than switch directly through the squat (scissor) switch. The relays are located behind the center access panel of the left console.



Unfeather Pump
Figure 10

Q. Overload Sensor, Relay, and Diodes (Aircraft T201-278, AT001-056, TC201-229)

(1) The purpose of these components is to power certain components of the start system and the inverter with power directly from the start bus during the start cycle. The diodes are to keep power from feeding back from the essential buses to the start bus. If the overload senses an overload, it disconnects the relay, and, at the same time, shorts the one amp start overload circuit breaker to ground, and it disengages. The circuit breaker is located on the essential buses next to the start control circuit breakers. The overload sensors and the diodes are located on the start panel.

NOTE: Start system components that are installed on the engine are as follows:

The starter and unfeather pump are supplied by Fairchild Aircraft Corporation.

Other components on the engine are supplied by the engine manufacturer.

See Garrett Maintenance Manual for a more complete description.

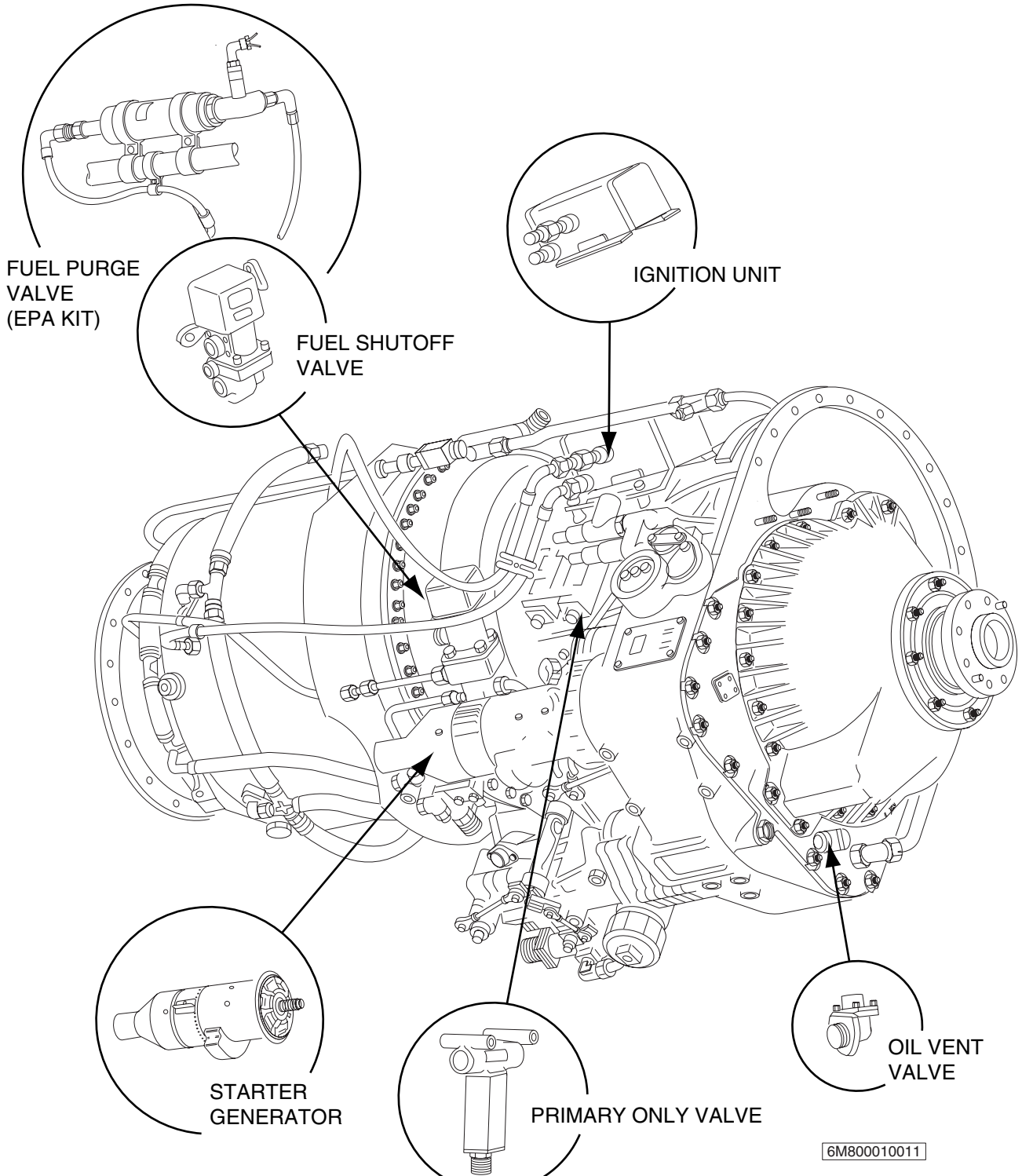
R. Starter

(1) A combination starter-generator is used to crank the engine during the start cycle when the aircraft is on the ground. Once the start cycle is complete, the unit is used as a generator.

(2) The starter-generator is located on the lower right of the engine attached to the aft side of the gear case

S. Unfeather Pump

(1) The unfeather pump is located on the left of the engine, attached to the engine mount. It pumps oil into a cylinder piston arrangement which in turn positions the prop blades. The pump operates during an airstart and can be operated for emergency, maintenance, or test purposes by the unfeather test switch.



Starting Component Locations
Figure 11

T. Anti-Ice Lockout Valve

(1) The anti-ice lockout valve closes during the start cycle. The valve stops fuel from flowing through the oil to fuel heater to make all fuel available for starting and priming purposes rather than fuel filter anti-icing. The valve is located on the left of the engine on the fuel control

U. Start Fuel Pressure Regulator Valve (SPR) Aircraft T201-275, T277-291, AT001-999, TC201-999.

(1) The valve can be opened by actuating the start switch from 10% to 55%. The valve allows fuel to bypass the fuel control and go directly to the primary fuel nozzles to prime the engine. The valve is located adjacent to the anti-ice valve.

V. Fuel Enrichment Valve (SPR) Aircraft T276, T292-999.

(1) When the valve opens, it allows fuel to bypass the fuel control and go directly to the primary fuel nozzles to prime the engine during the start cycle (10% to 60%). The valve is automatically controlled by the SRL computer during an auto start. The valve is also controlled by the start button during a manual start or is used to override the SRL computer. The valve is located adjacent to the anti-ice lockout valve

W. Primary Only Solenoid Valve (Aircraft T276, T292-999)

(1) The valve is opened by the 10% to 80% speed switch contacts in the SRL computer. When the valve is open, it recalibrates the fuel flow divider which increases fuel flow to the primary fuel nozzles during start. The valve is located on the upper right side of the engine near the beta block.

X. Engine Fuel Shutoff Valve

(1) The valve, is opened electrically at 10%. The valve routes fuel from the fuel control to the flow divider. The valve can be closed electrically by depressing the stop button or mechanically by pulling the stop and feather button. The valve cannot be opened mechanically. The valve is located on the right of the engine.

Y. Fuel Purge Valve (EPA Kit)

(1) The fuel purge valve is located on the right of the engine, and it is attached to the mount near the fuel shutoff valve. A small accumulator tank. two hoses, two check valves, and an electrically operated solenoid comprise the EPA kit. The accumulator tank is charged with air pressure from the engine. When the stop button is depressed the purge valve is opened, air pressure from the tank goes into the flow divider drain which pushes the fuel in the manifolds through the fuel nozzles into the engine. Since the engine fuel shutoff is also closed by the stop button, no more fuel is available and the engine shuts down.

Z. Oil Vent Valve

(1) The oil vent valve is located on the front lower left of the engine gear case. The valve is opened when the starter is actuated to allow air to enter the oil pumps and reduce the load on the starter.

AA. Ignition Exciter Unit

(1) The ignition unit is located on the upper right side of the engine just forward of the intake heat valve. The unit provides approximately 18,000 volts to two igniter plugs which are installed in the combustion chamber of the engine. The unit operates from 10% to the completion of the start cycle.

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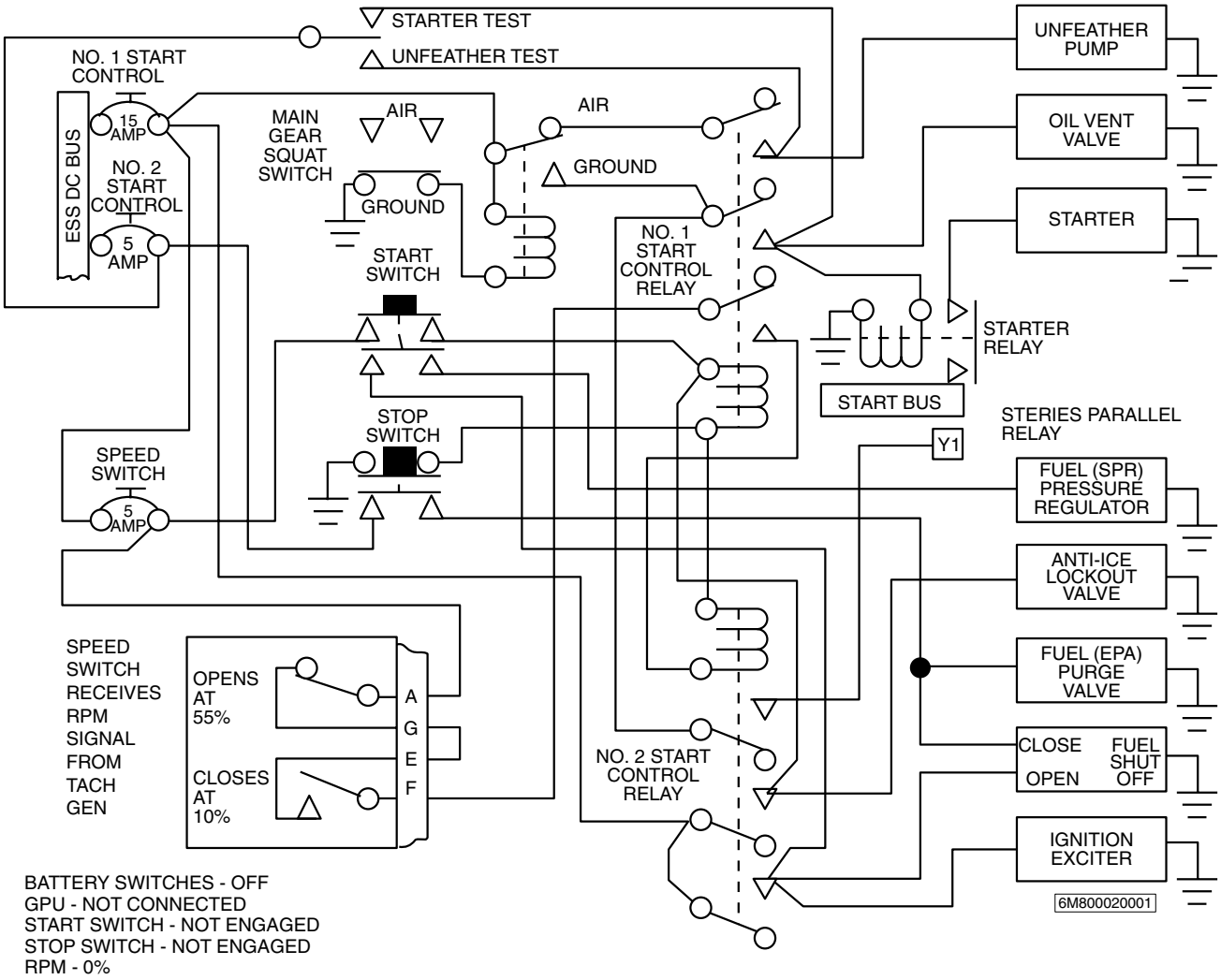


Start System -3
Engine Power Off
Figure 1

STARTING – DESCRIPTION AND OPERATION

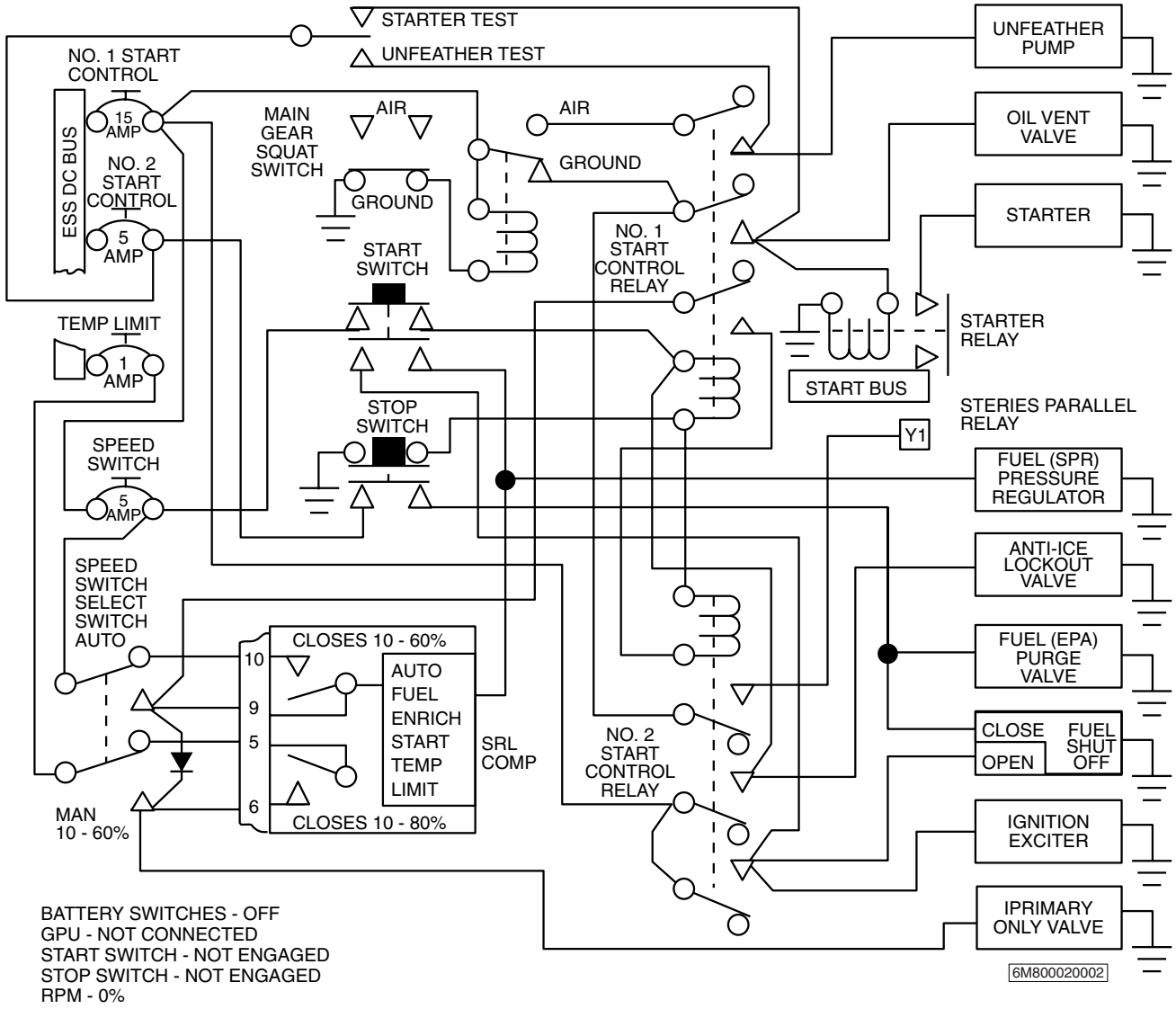
1. General

A. Control power for the start system is provided by two circuit breakers on each essential bus. The No. 1 start control circuit breaker powers the starter relay, oil vent valve, the series-parallel relay for ground starts, and the unfeather pump for air starts. On aircraft T279 and up, AT067 and up, TC230 and up, the No. 1 circuit breaker also powers the ignition exciter, engine fuel shutoff to open, anti-ice lockout valve through the start switch for SPR. The No. 1 circuit breaker also powers the speed switches for the No. 1 and No. 2 start control relay lock in circuitry. On earlier aircraft these components were powered directly from the start bus through an overload sensor and relay.

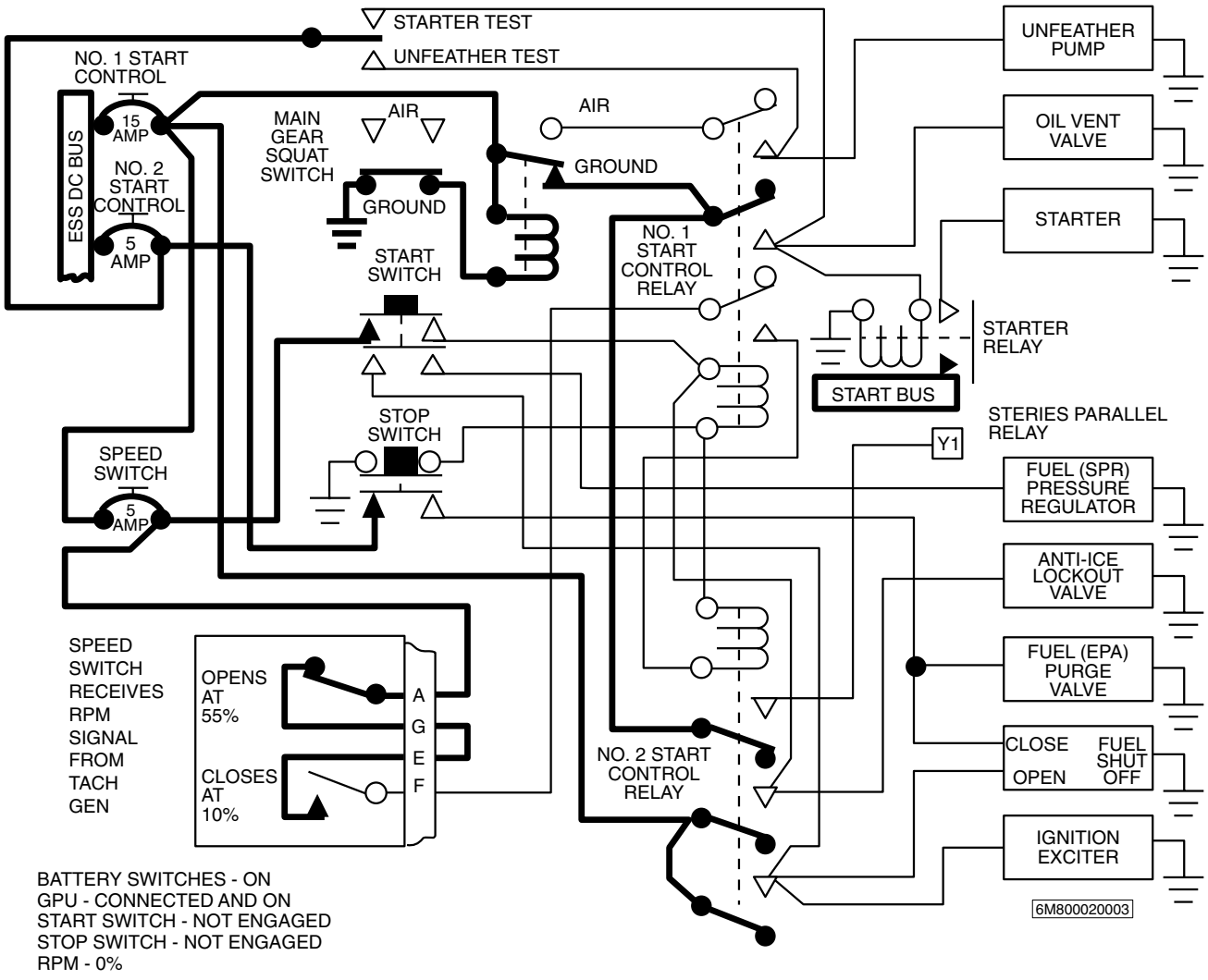
(2) The No. 2 start control circuit breaker powers the start and unfeather test switches, the battery fault detector, and supplies power through the stop switch to close the fuel shutoff.

(3) The temp limit circuit breaker supplies power through the 10% to 80% speed switch only for the primary valve.

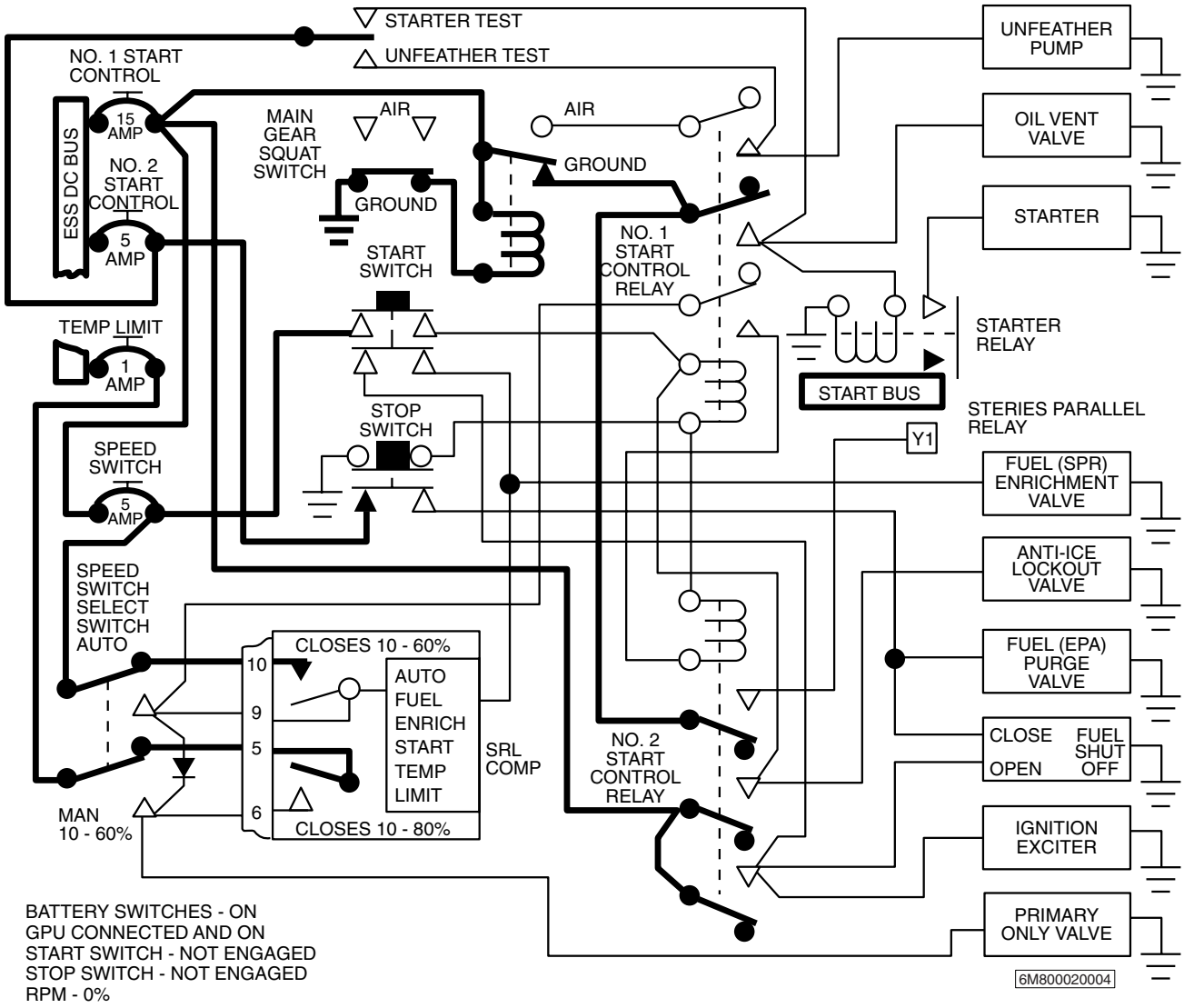
(4) The start bus is powered directly from the GPU. It also gets power through the series–parallel relay from the batteries.



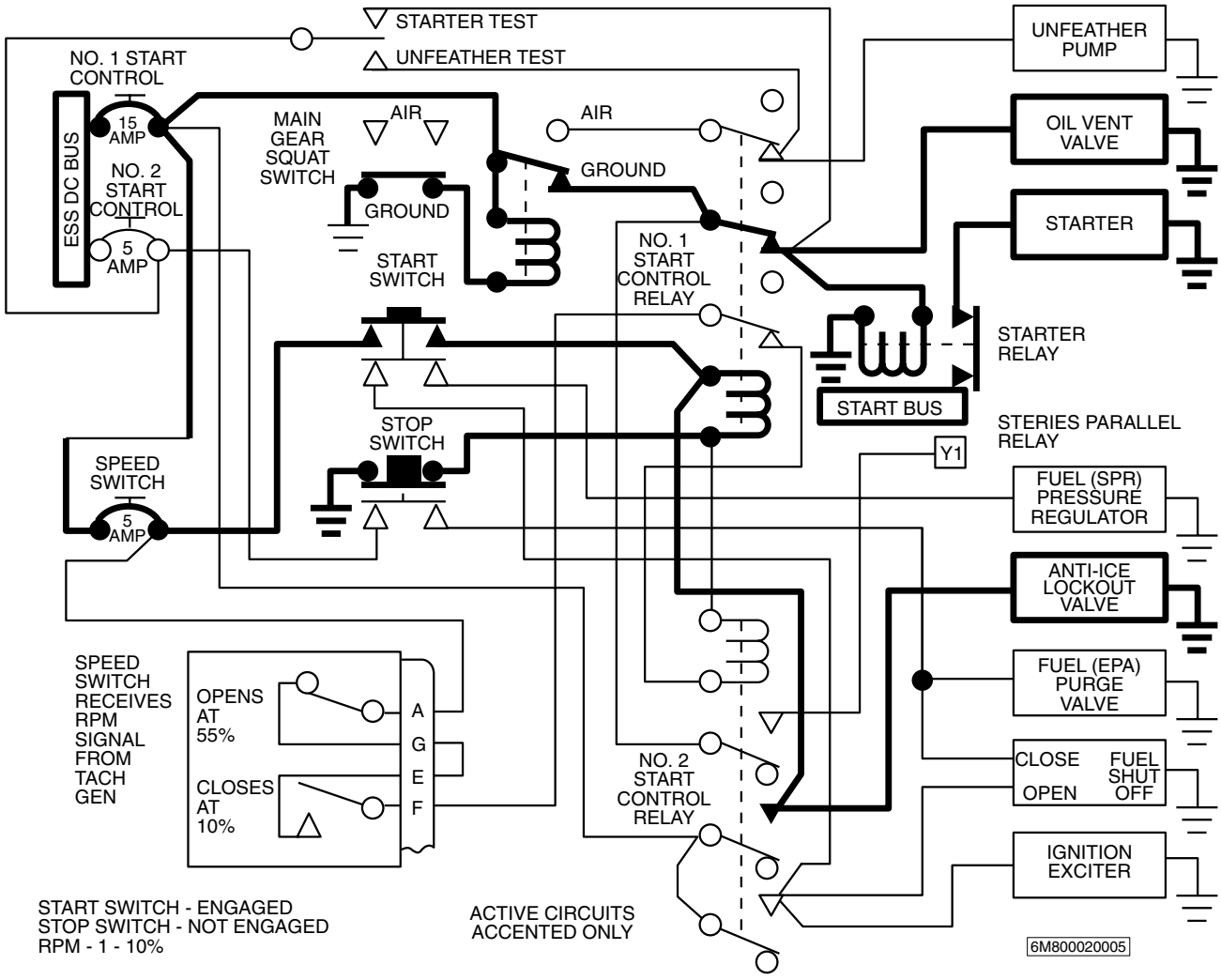
Start System -10
Engine Power Off
Figure 2



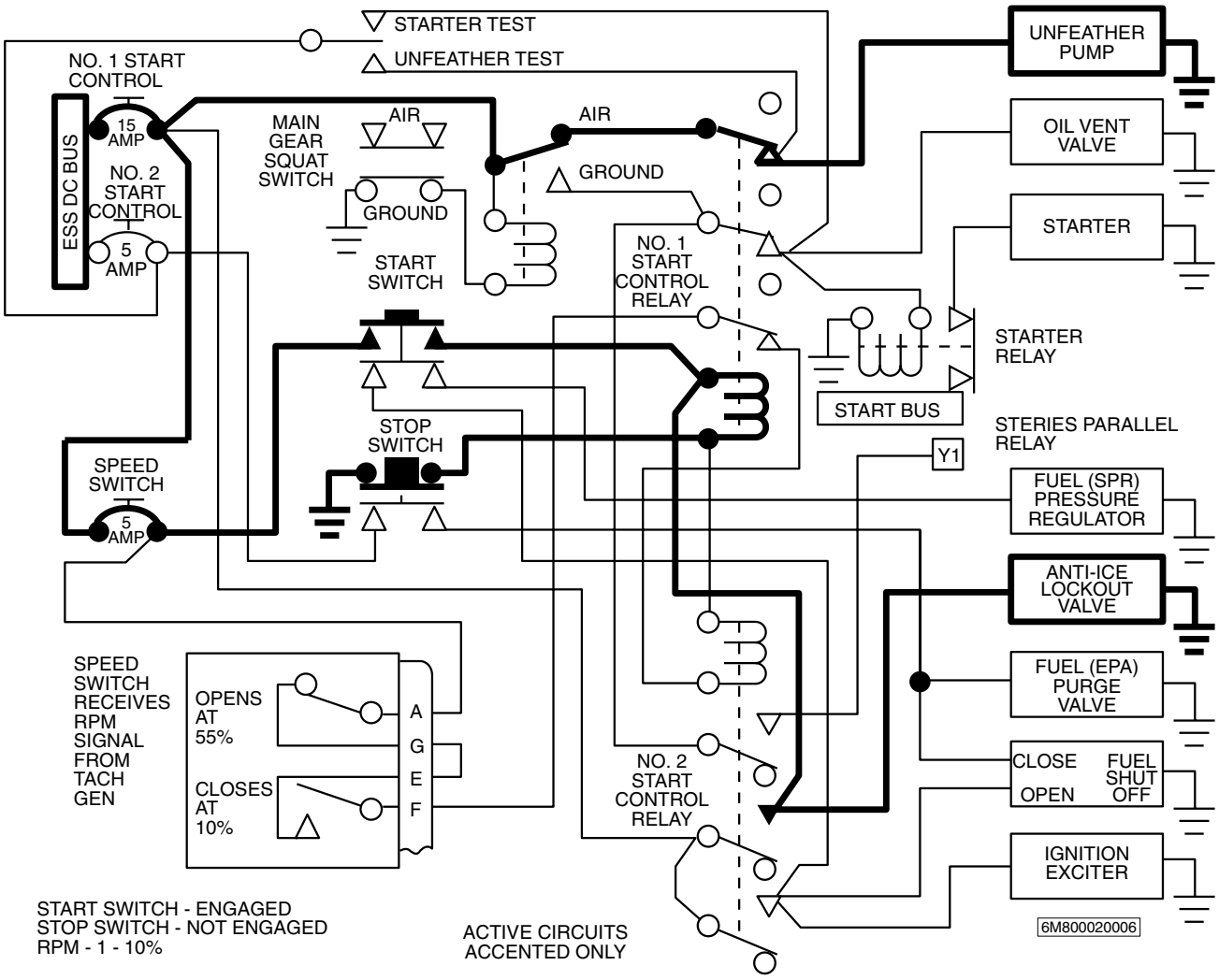
Start System -3
Engine Power On
Figure 3



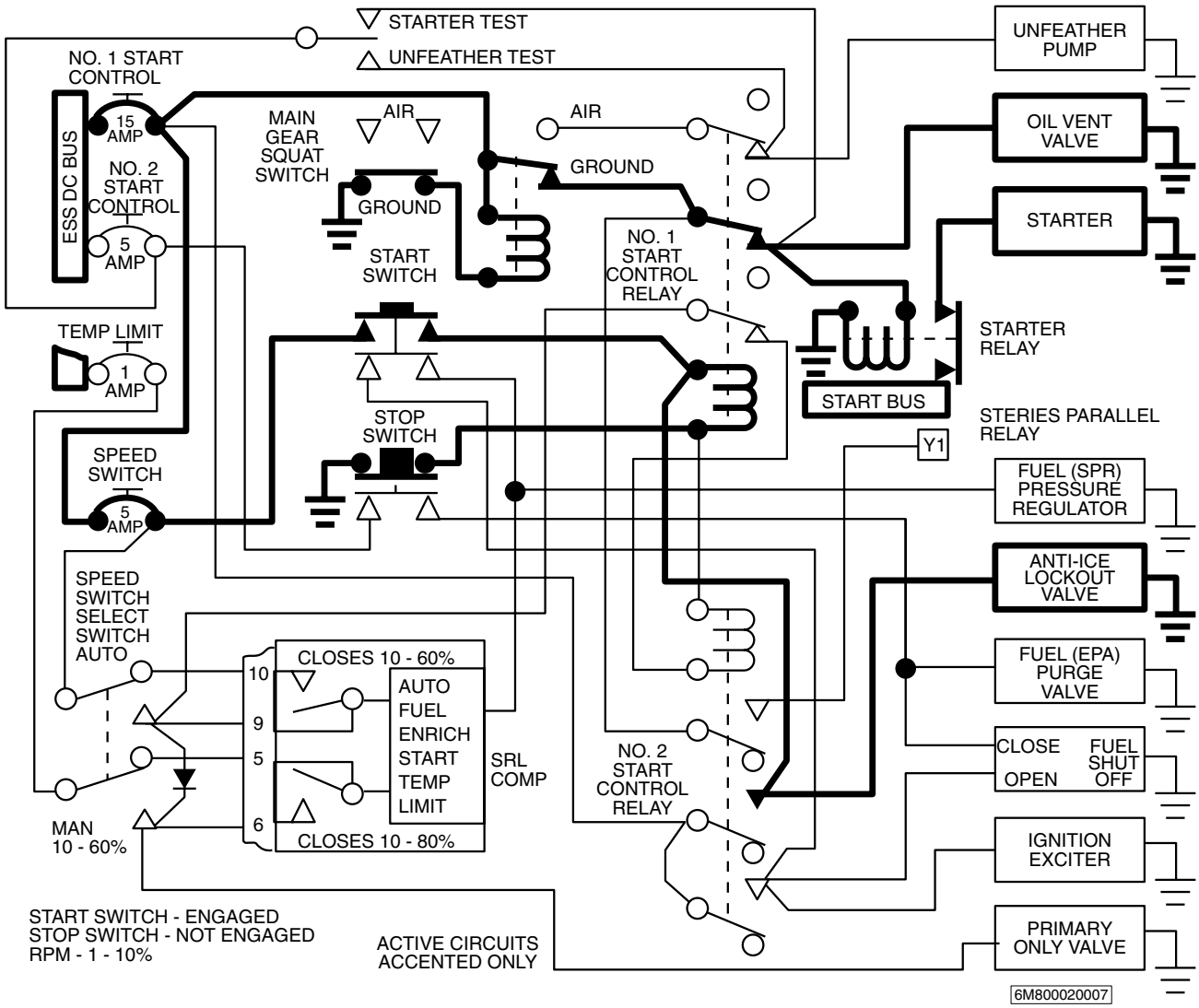
Start System -10
Engine Power On
Figure 4



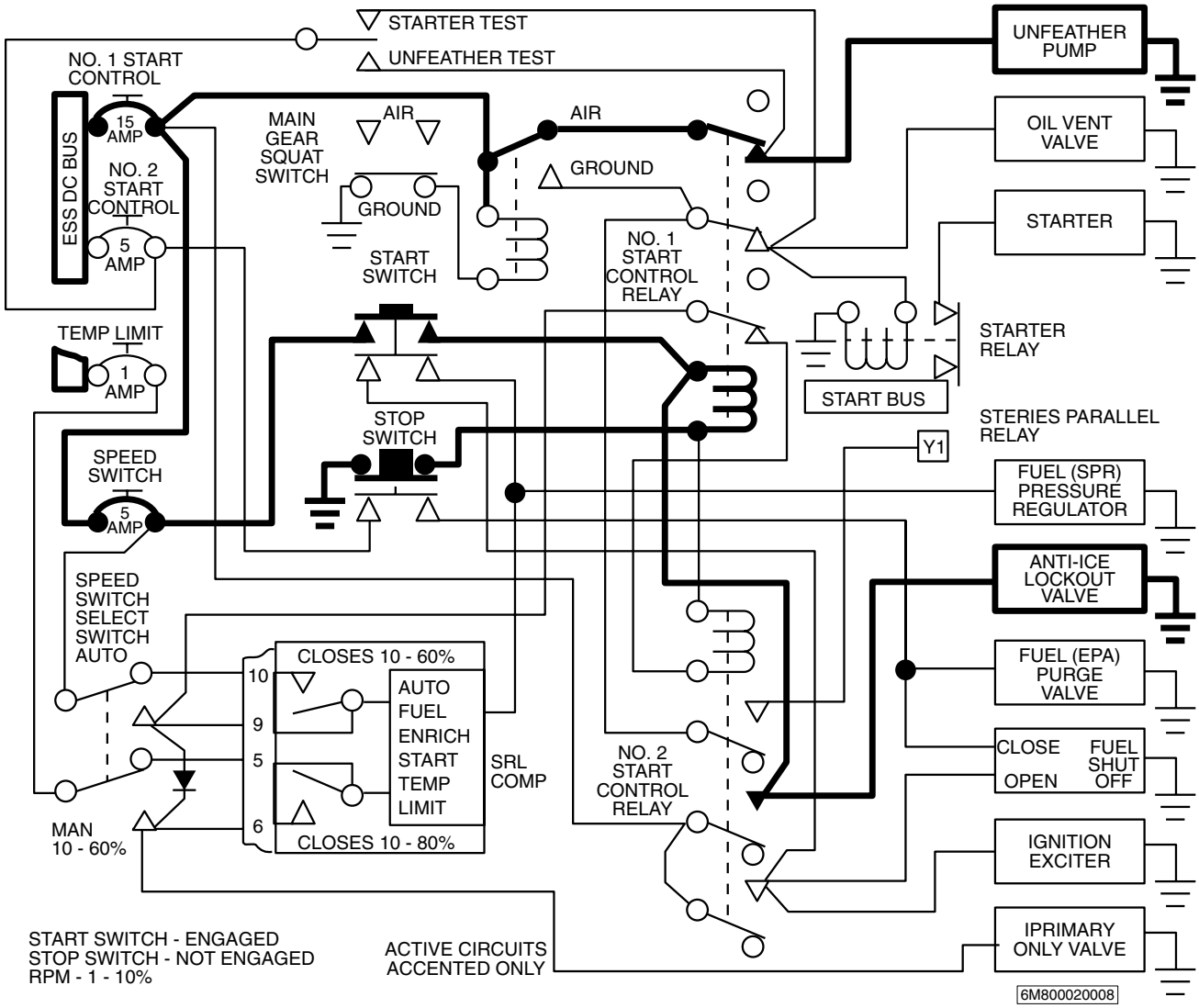
Start System -3 Engine
Ground Start Below 10%
Figure 5



Start System -3 Engine
Air Start Below 10%
Figure 6



Start System -10 Engine
Ground Start Below 10%
Figure 7



Start System -10 Engine
Air Start Below 10%
Figure 8

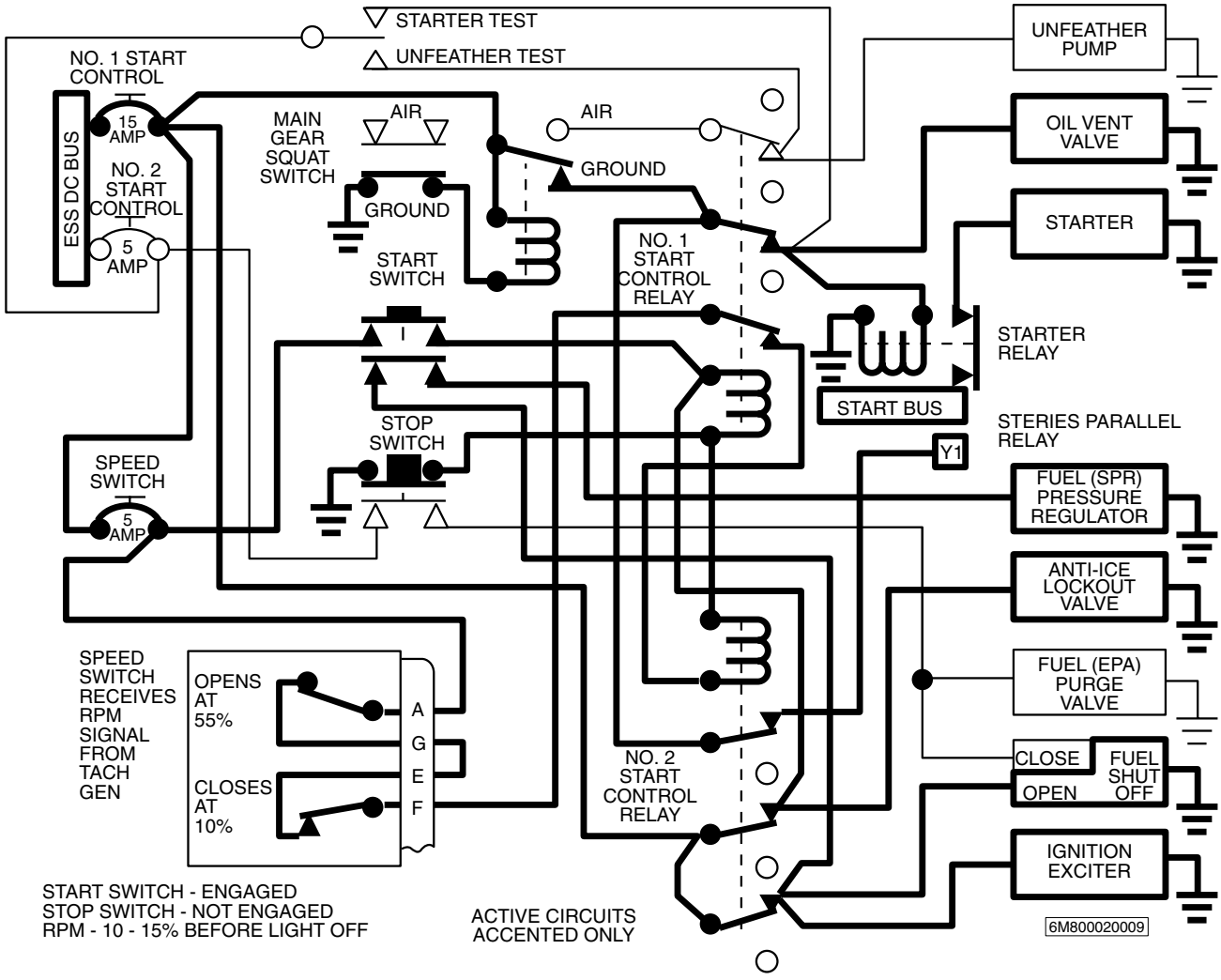
B. Start Below 10%

(1) The start cycle is initiated by manually engaging the start switch. Power is routed through the speed switch circuit breaker to the start switch to actuate the anti-ice lockout valve and the No. 1 start control relay.

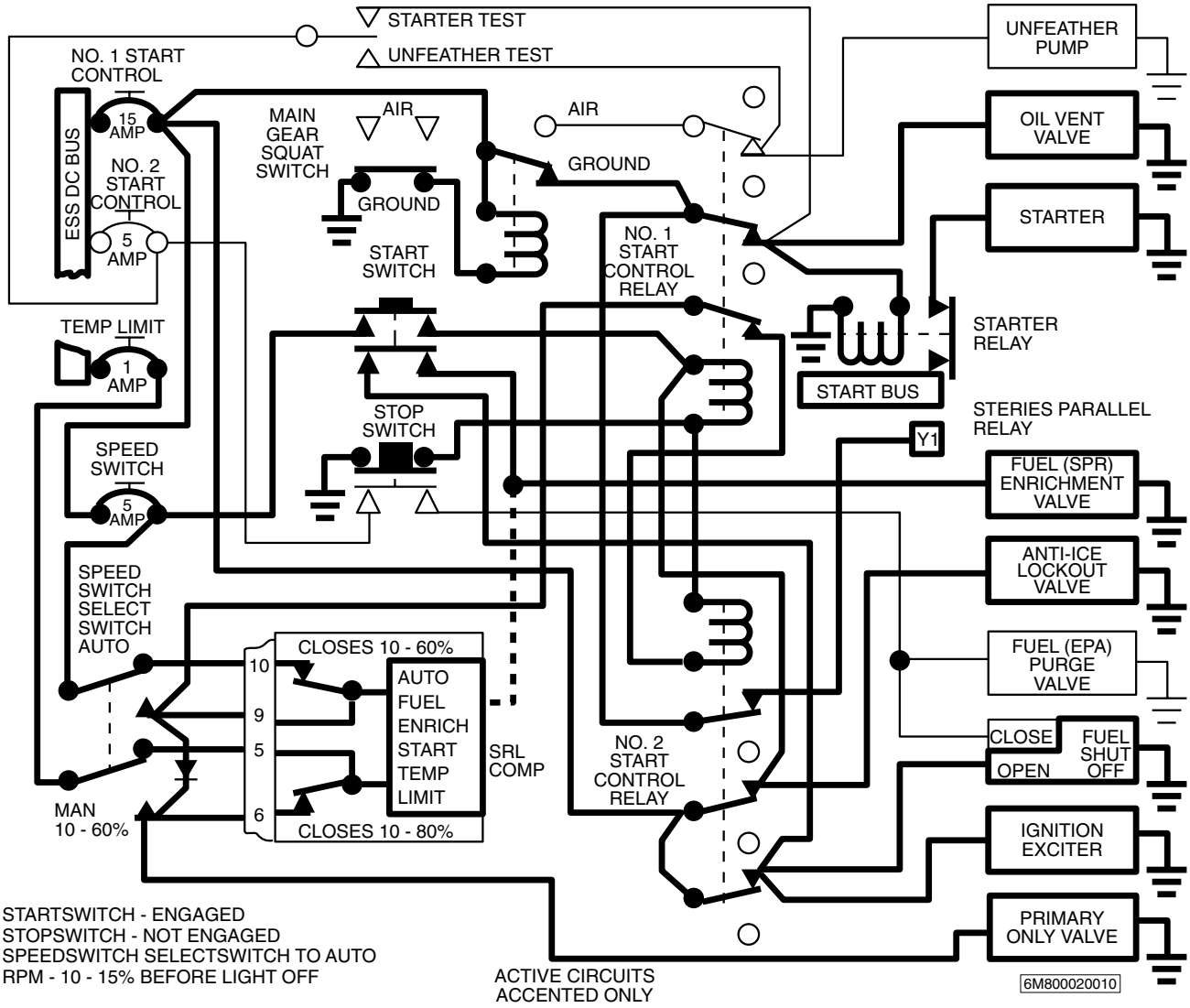
(2) The anti-ice lockout valve is closed during the entire start cycle. When the No. 1 start control relay closes, it routes power (depending on the position of the squat switch) to the unfeather pump or to the oil vent valve and starter relay. The starter relay in turn routes high amperage power directly from the start bus to the starter.

(3) On aircraft T201-278, AT001-056, TC201-229, power is routed directly through the squat switch to the No. 1 start control relay, rather than through the squat switch relay.

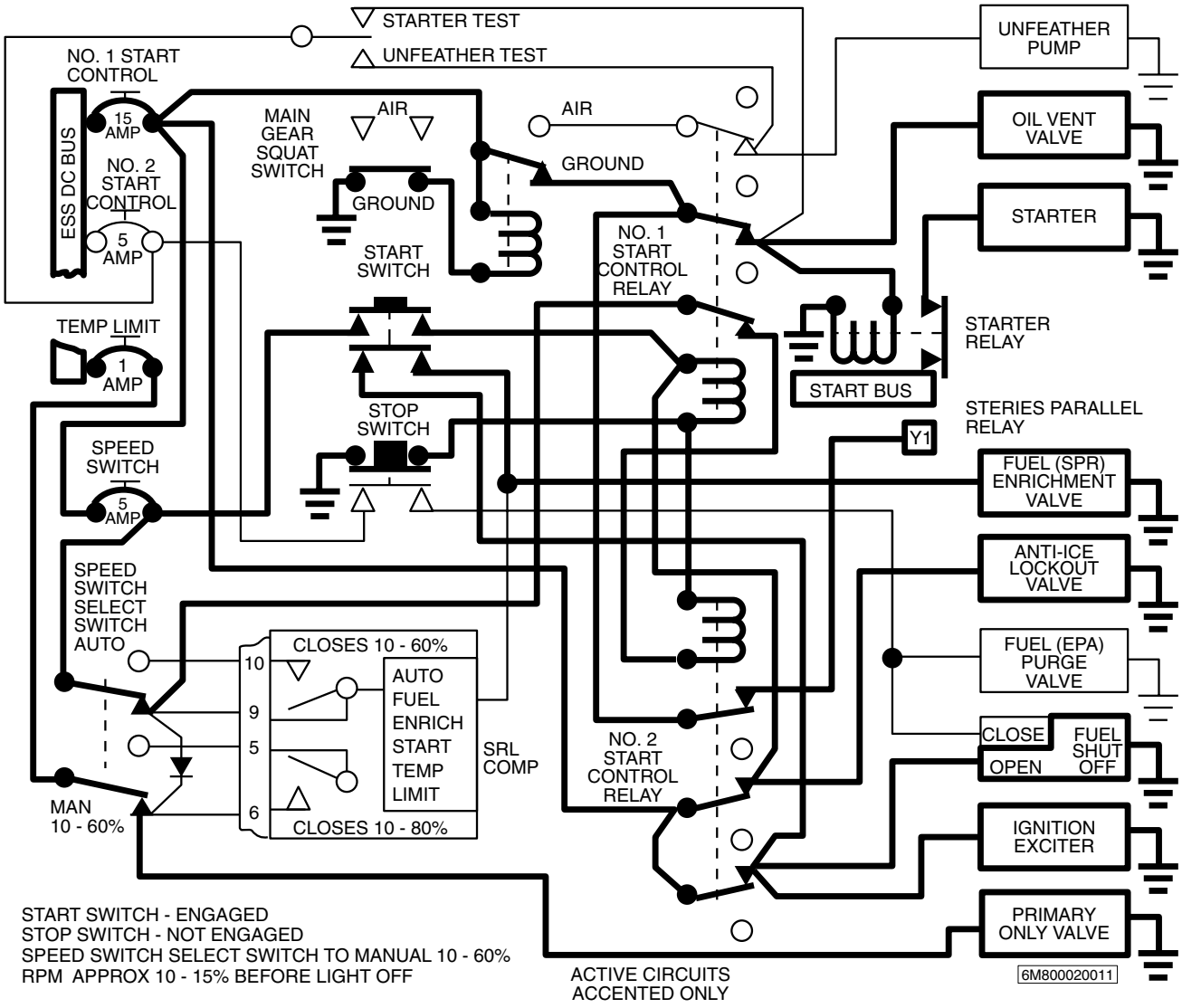
(4) Below 10%, a start may be aborted by merely releasing the start button.



Start System -3 Engine Ground
Start - Above 10% Before Light Off
Figure 9



Start System -10 Engine Auto Ground
Start - Above 10% Before Light Off
Figure 10



Start System -10 Engine Manual Ground Start - Above 10% Before Light Off
Figure 11

C. Start Above 10% Before Light Off

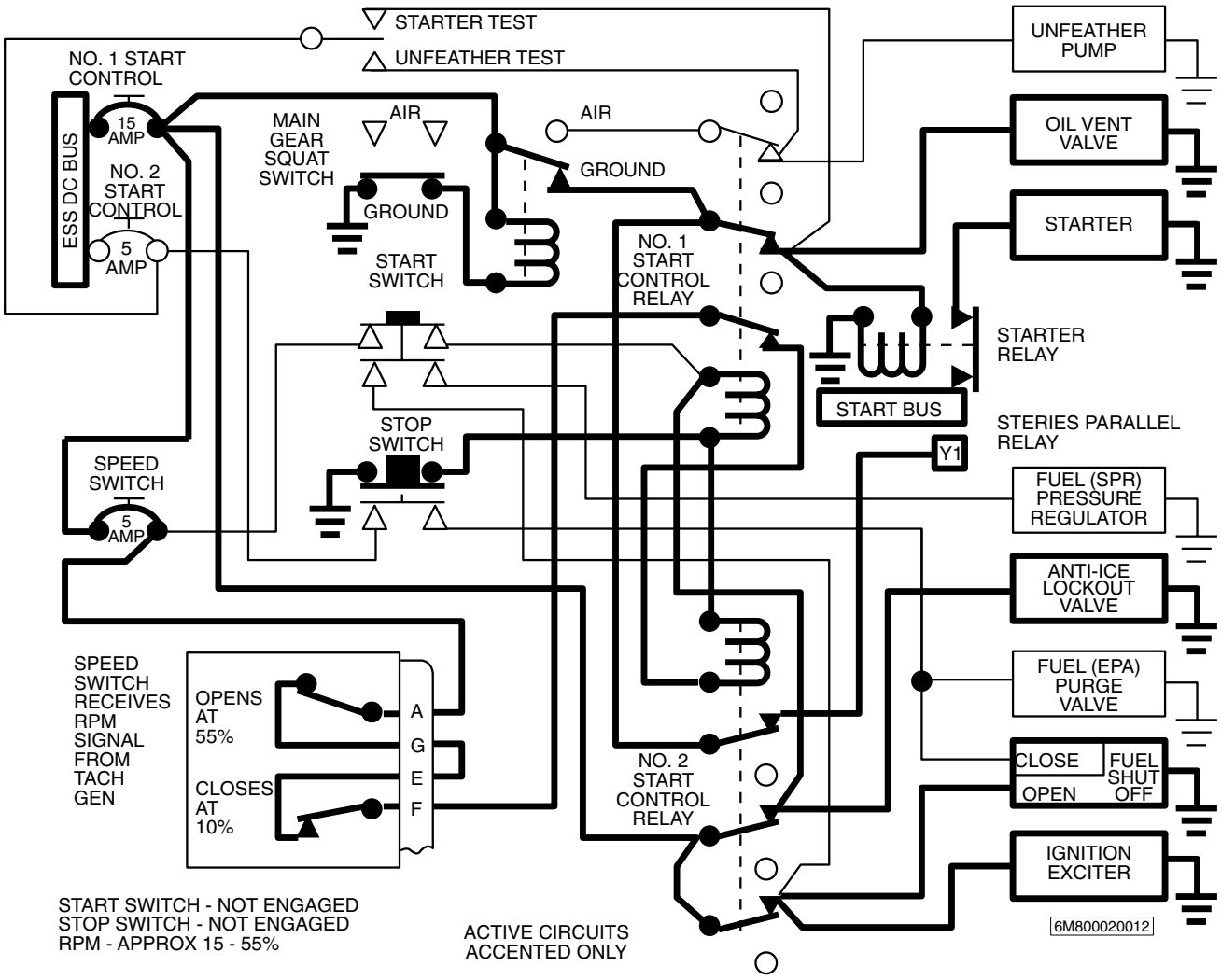
(1) At 10% a contact closes within the speed switch or the SRL computer. This completes a circuit through the No. 1 start control relay to the coil of the No. 2 start control relay, and both relays are locked in.

(2) For the -10 engine start system, another switch contact within the SRL computer is used to open the primary only valve. A switch is provided to manually override these two functions should the SRL computer fail.

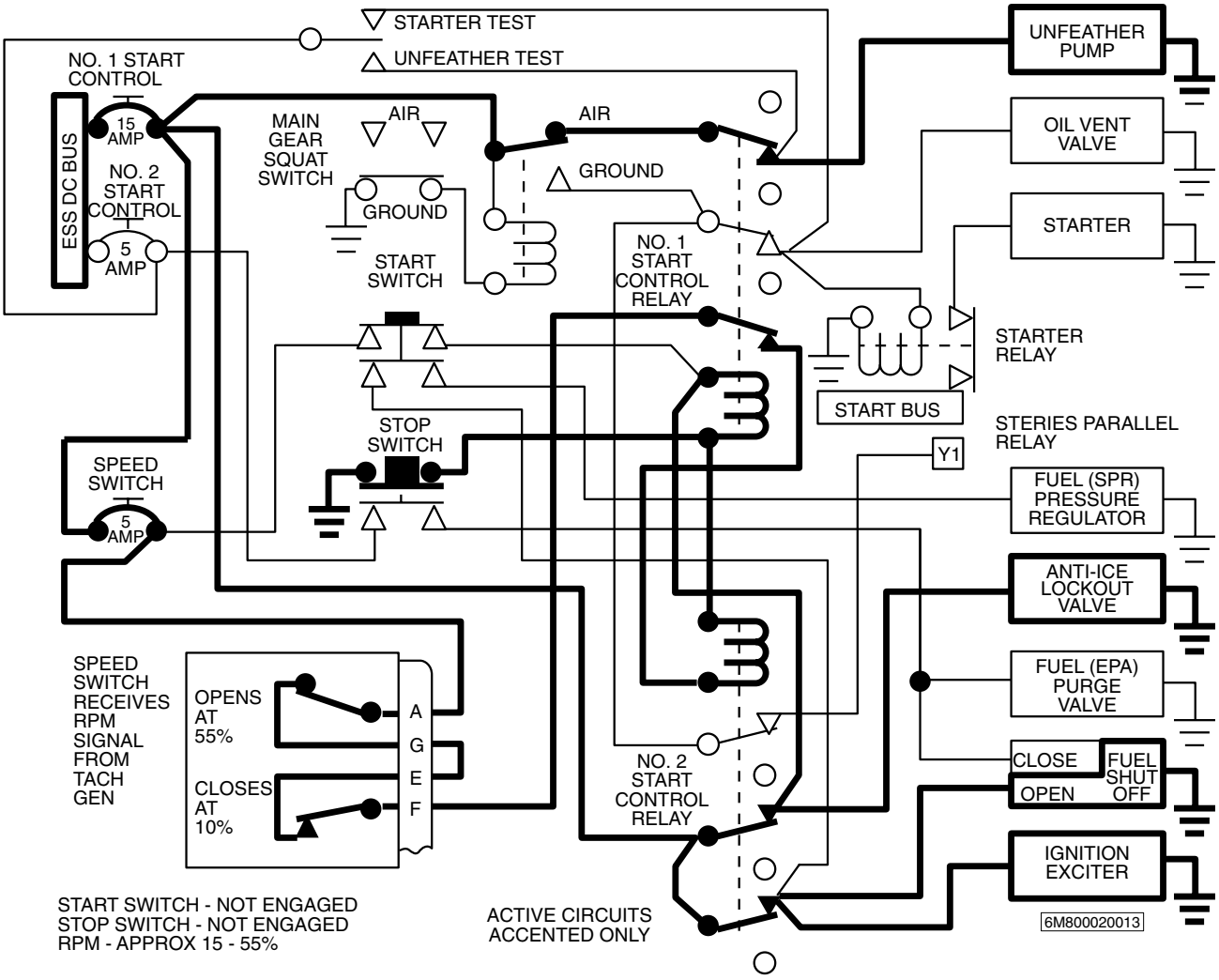
(3) When the No. 2 start control relay closes, power is made available to the coil (Y1) of the series-parallel relay for a series start mode. Power goes to the fuel shutoff to lock it open, and power goes to the ignition exciter.

(4) When light off is confirmed on the ITT, the ET indicator start button is released.

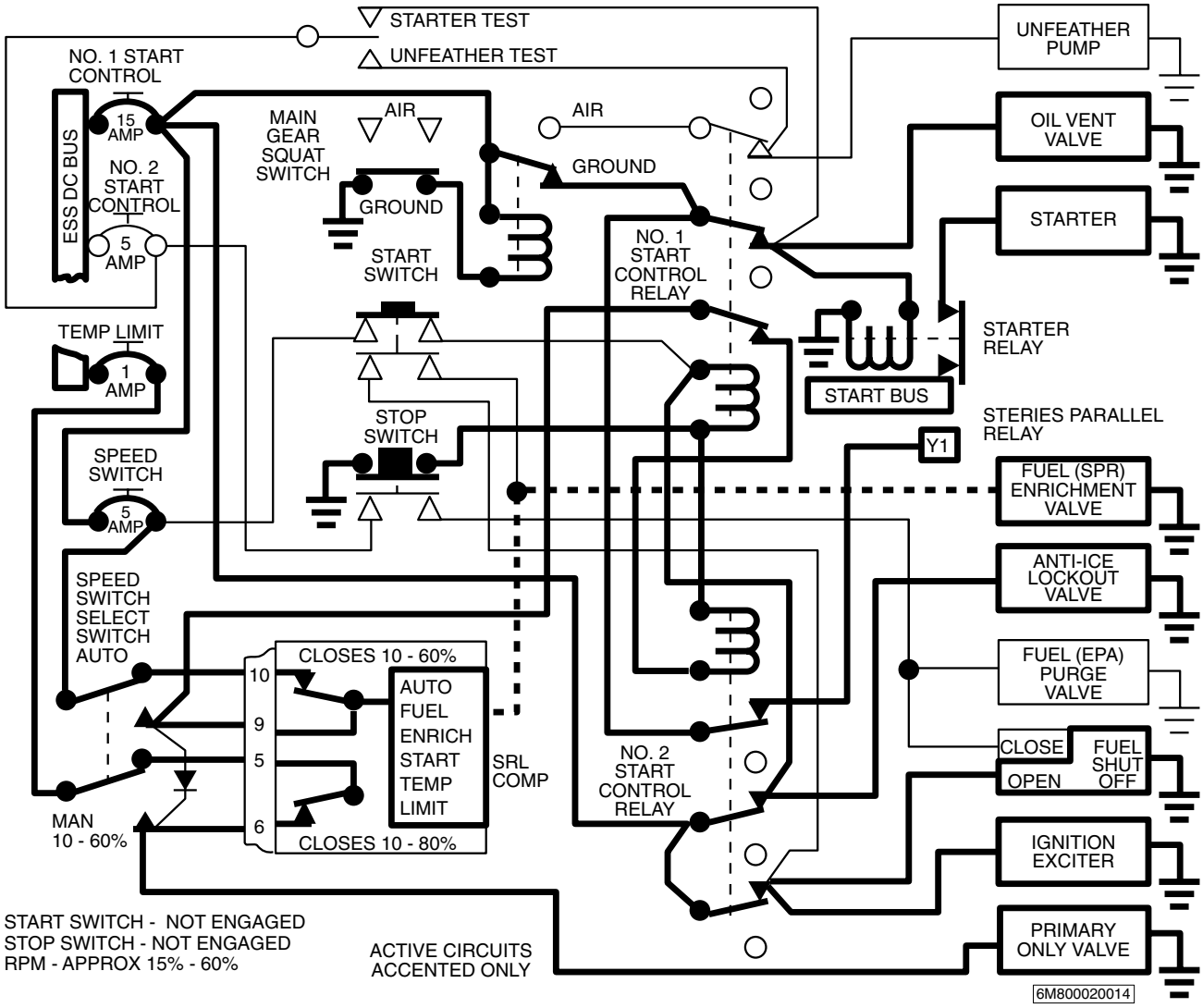
(5) The stop button must be engaged should the start need to be aborted after 10% since both coils of the start control relays are grounded through the stop switch. The stop and feather button will also stop the engine, but it should not be used unless it is necessary. In either case, the engine should be cranked with the starter test switch to clear raw fuel.



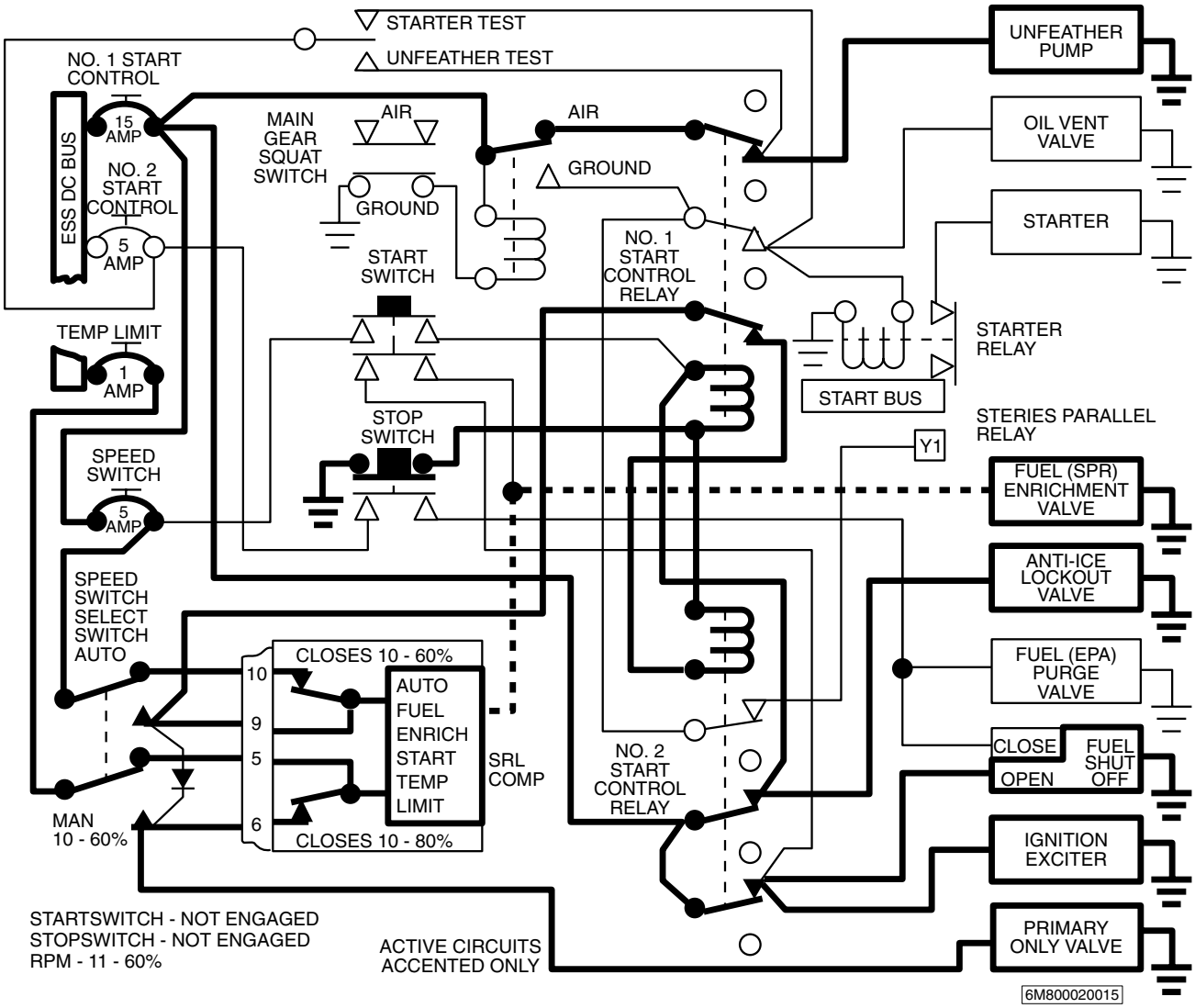
Start System -3 Engine Ground
Start - Above 10% Before Light Off
Figure 12



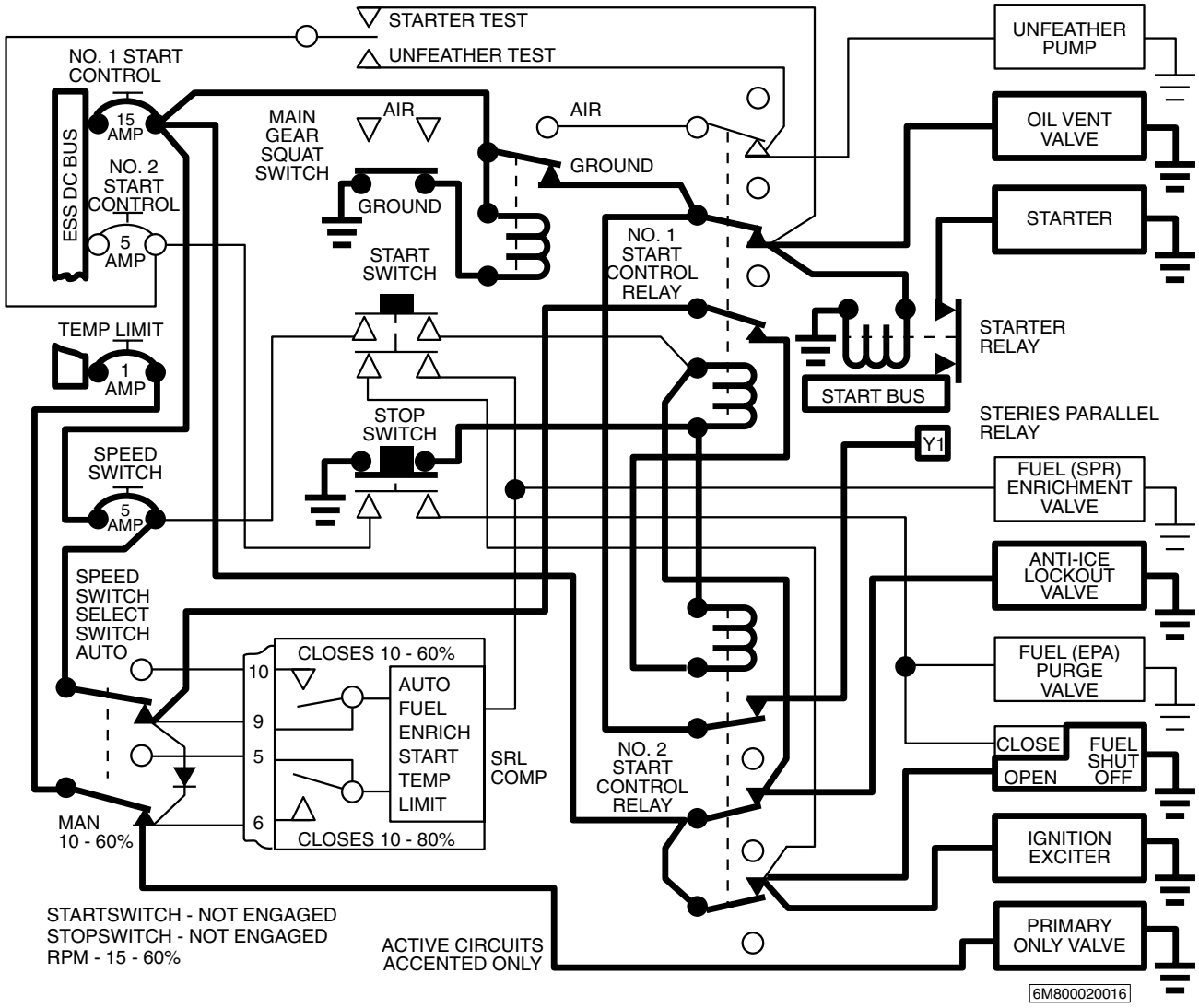
Start System -3 Engine Air
Start - Above 10% After Light Off
Figure 13



Start System -10 Engine Auto Ground
Start - Above 10% After Light Off
Figure 14



Start System -10 Engine Auto
Airstart - Above 10% After Light Off
Figure 15

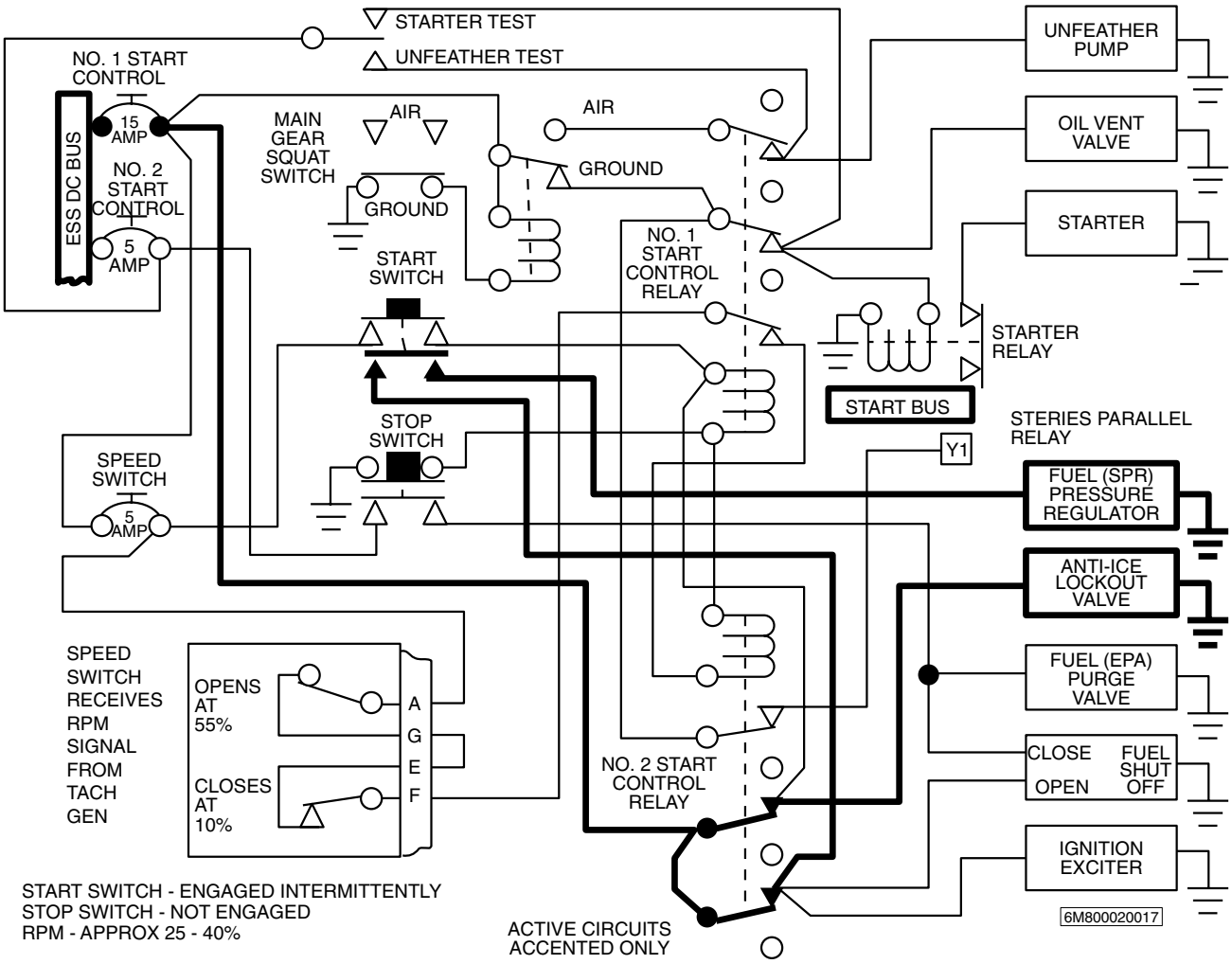


Start System -10 Engine Manual
Ground Start - Above 10% After Light Off
Figure 16

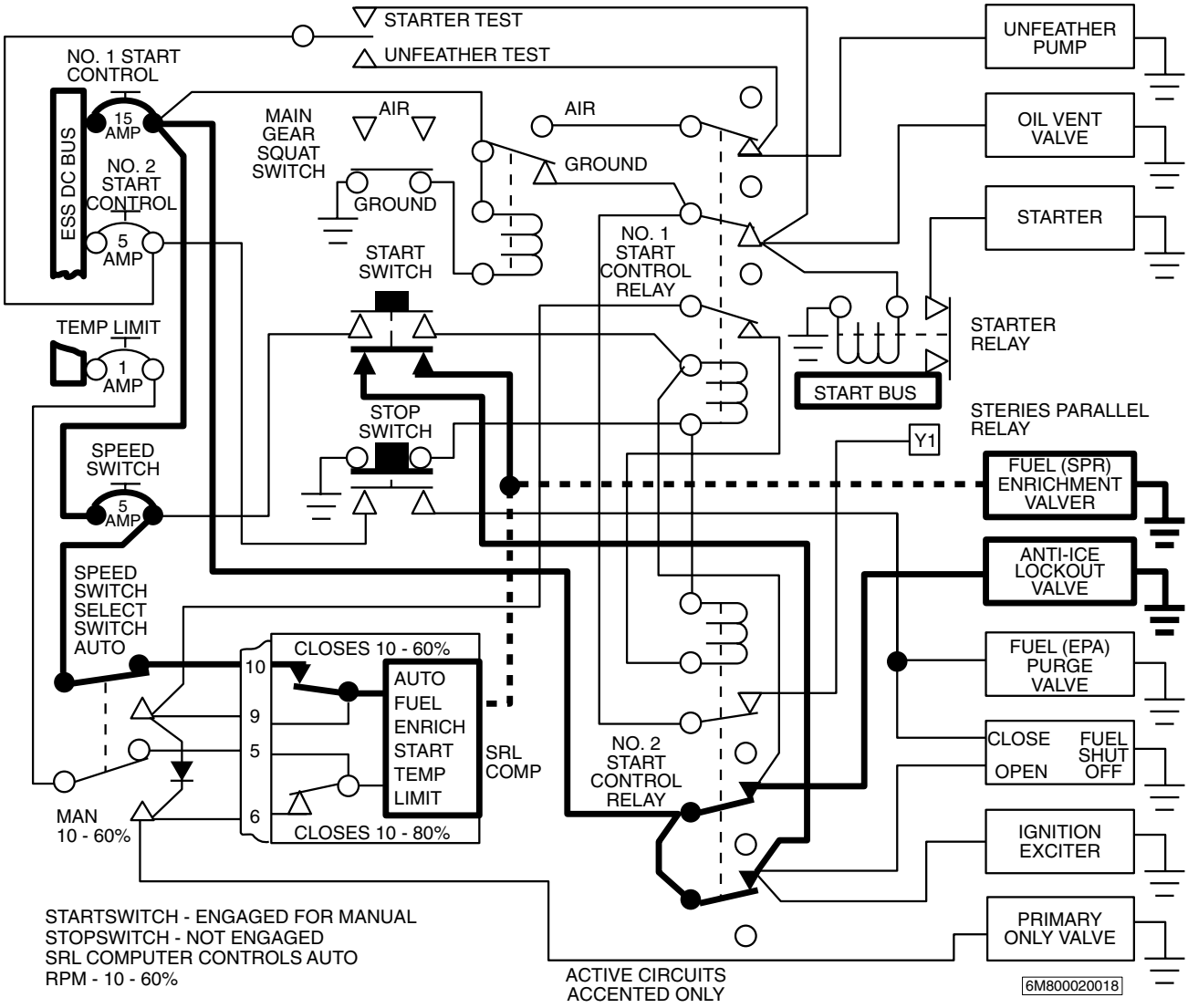
D. Above 10%, After Light Off

(1) When light off is confirmed, release the start button and the engine will accelerate through the start cycle. On the -3 engine the fuel flow divider will allow fuel to flow to the secondary fuel nozzles at approximately 40%, and the engine will accelerate rapidly past 55%. The speed switch contact will open, and the start control relays will return to their normally open position.

(2) On the -10 engine, the fuel flow divider is recalibrated by the primary only valve to allow more fuel to flow to the fuel nozzles during the start cycle. During an auto start, the fuel enrichment valve is controlled automatically and the EGT is held at the start temp light (770°). During a manual start, the fuel enrichment valve must be controlled by the start switch. The engine will accelerate quickly to 60% at which time the speed switch contact in the SRL computer will open and the No. 1 and No. 2 start control relay will return to their normally open position. The primary only valve will remain open until the engine is accelerated above 80%.



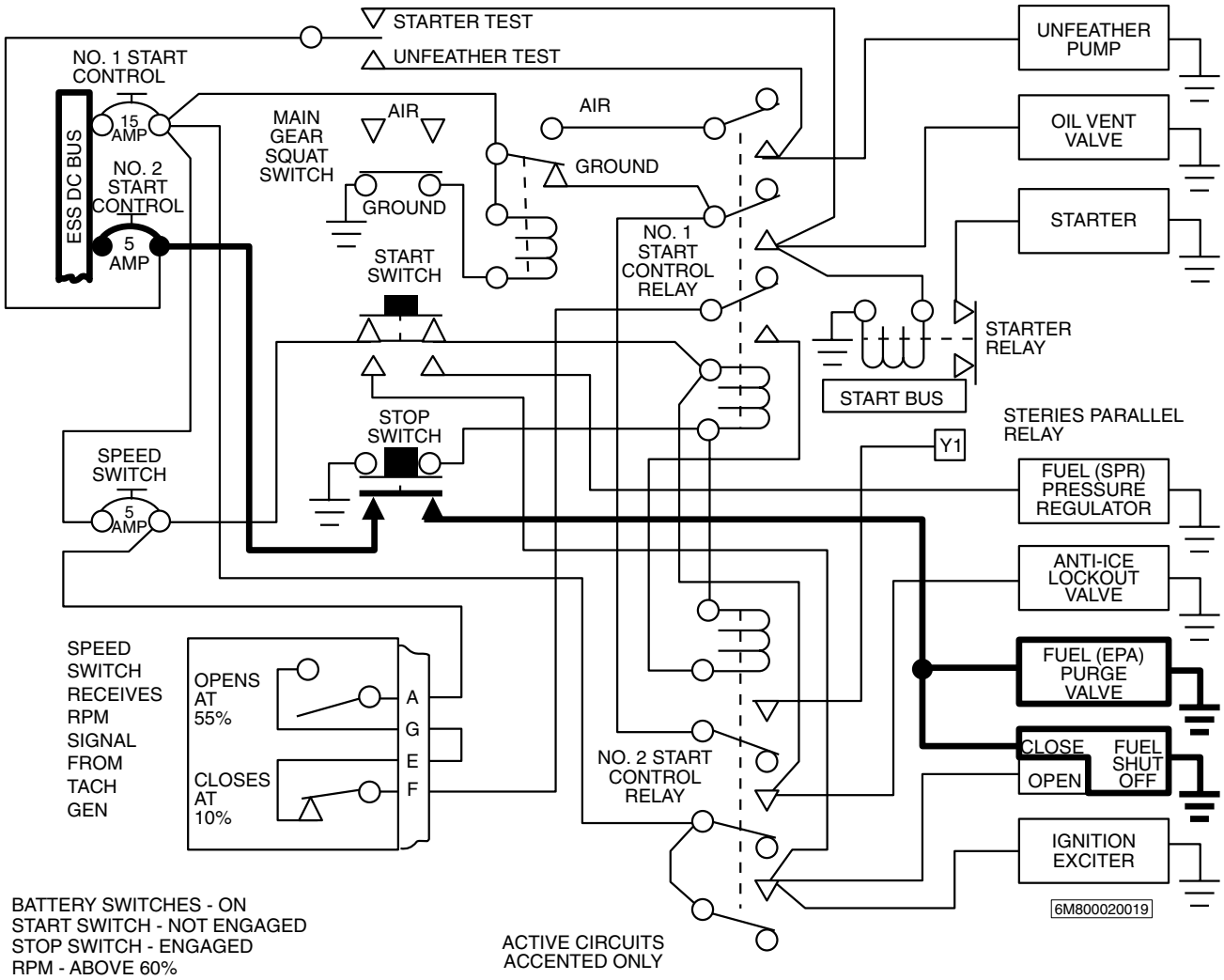
Start System -3 Engine
SPR Operation
Figure 17



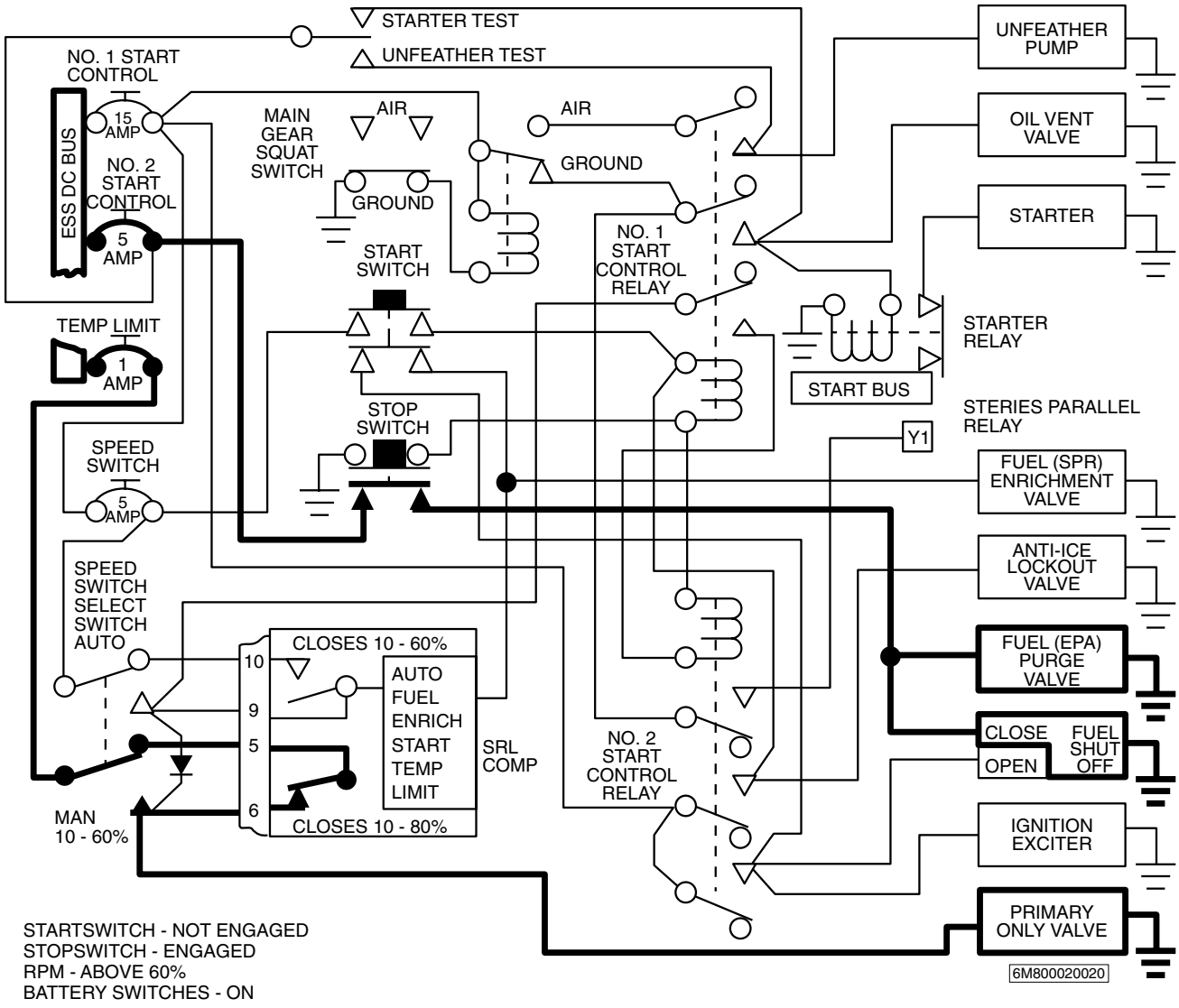
Start System -10 Engine
Fuel Enrichment Auto/Manual
Figure 18

E. Fuel Enrichment (SPR) Operation

(1) On the -3 engine, the start switch is used to manually prime the engine during the start cycle. On the -10 engine, the SRL computer automatically primes the engine and keeps the engine at the temp limit (770°) during the start cycle. The start button should not be used to prime the engine unless the SRL computer has failed, since it will override the temp limit.



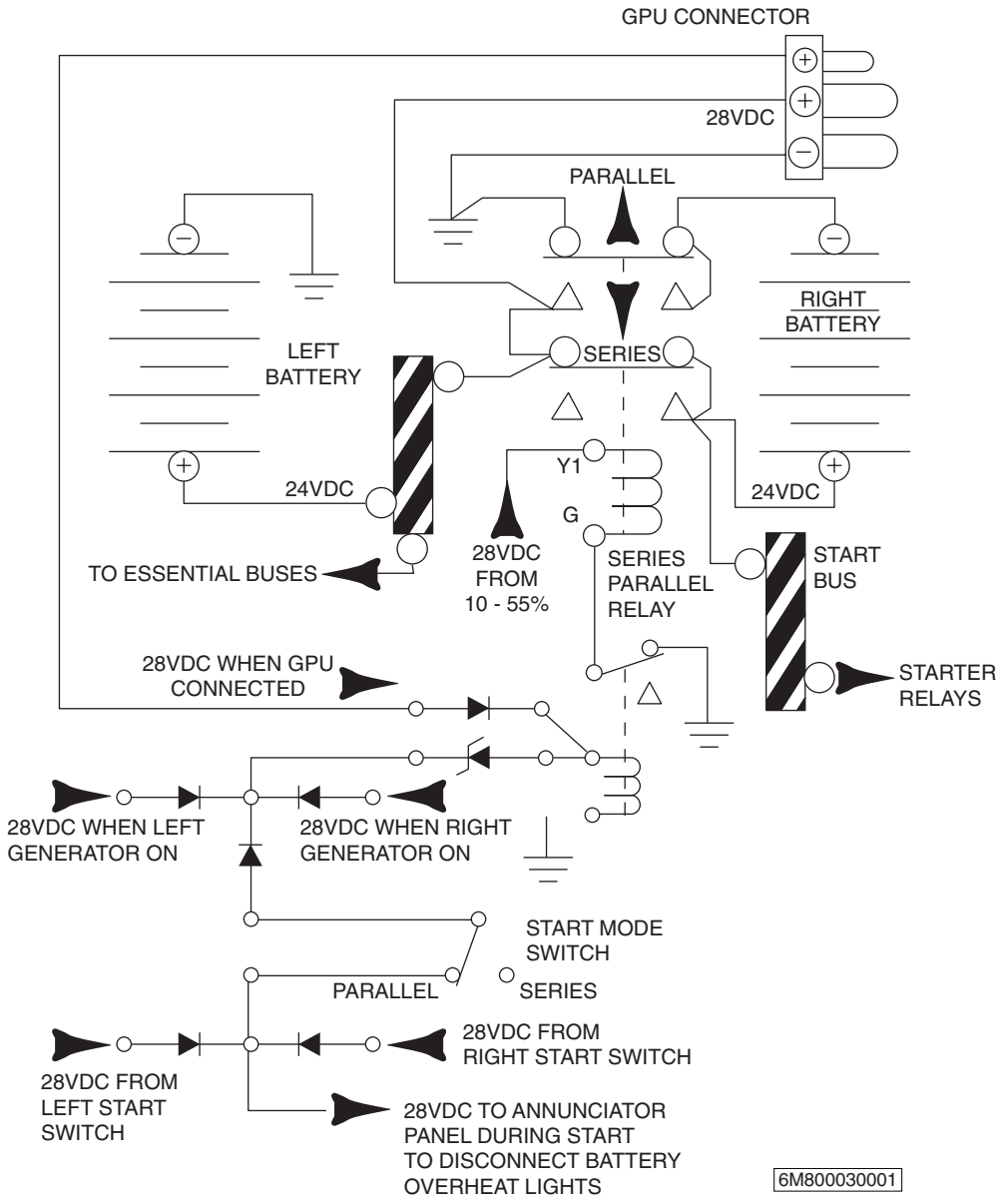
Start System -3 Engine
Normal Shutdown
Figure 19



Start System -10 Engine
Normal Shutdown
Figure 20

F. Normal Shutdown

(1) To stop the engine the speed lever is positioned to low and the power lever to ground idle. The stop button is engaged. Power is routed from the No. 2 start control circuit breakers through the stop switch to the fuel shutoff and fuel purge valves. The fuel shutoff closes and the RPM rises slightly while air pressure from the fuel purge system forces fuel through the fuel flow divider, manifolds, and nozzles and into the engine. Once all the fuel is exhausted, the RPM decays to 0.



Start Mode Control Circuit
Figure 1

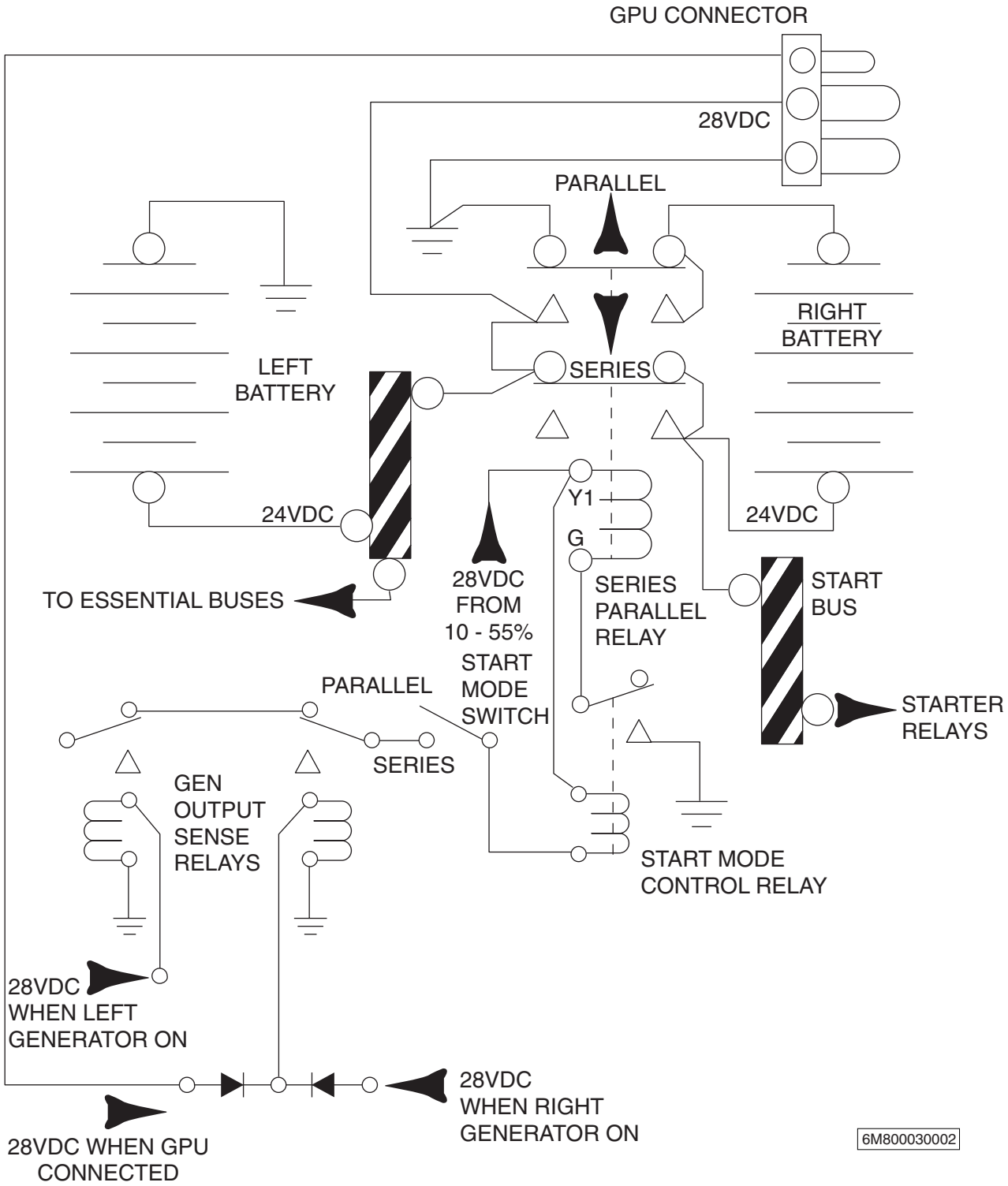
STARTING – DESCRIPTION AND OPERATION

1. General

A. Series/Parallel

(1) Two methods of starting the engine are a ground start and an air start. A ground start has a parallel mode and, a series mode. This refers to the connection of the batteries during the start cycle and not the sequence that the engines are started. A parallel start can be accomplished by batteries alone, or the batteries can be assisted by a GPU or generator.

(2) A series start must be accomplished by the batteries only. A high amperage double contact relay is used to switch the batteries from parallel to series. This doubles the voltage at the starter and allows it to spin faster.



Start Mode Control Circuit Figure 2

1. Troubleshooting – Starting

A. Before you can begin troubleshooting, you must know exactly what the discrepancy is. In most cases this will mean attempting to start the engine yourself. Procedures for starting the engine can be found in the Aircraft Flight Manual. Because the start system is so closely related to the battery control and generator control system, problems with these two systems will usually be related to or surface when the engines are started.

B. Here are some simple rules to follow that will prevent many of these discrepancies and extend the useful life of the batteries and other components of these systems.

(1) Always have the batteries serviced at regular intervals as recommended in Chapter 24 and the Inspection Procedures Manual.

(2) Whenever the nominal battery voltage drops below 20 VDC, remove and deep cycle the batteries.

(3) Never connect a GPU to charge the batteries when their nominal voltage has dropped below 20 VDC.

(4) Always use battery power in conjunction with a GPU or a generator.

(5) Use battery power only when absolutely necessary.

(6) Before connecting a GPU plug to the aircraft, make sure that the polarity is correct and that the small positive pin socket is connected to the large positive pin socket.

(7) Never turn the battery switches on with a GPU connected but not operating.

(8) Make sure the battery switches, the generator switches, and the GPU switches are off before connecting or disconnecting a GPU.

(9) Make sure the contactor switch on the GPU is off before connecting or disconnecting the GPU.

(10) Never turn the generator switches on with a GPU connected.

(11) Never attempt to start or crank an engine in the air or on the ground with both generator switches on.

(12) Never engage the starter test switch with the engine running or spinning.

(13) Never re-engage the start button with the engine spinning below 10% RPM.

(14) Never engage the start button with the engine running.

(15) Never crank the engine above 18% RPM with the starter test switch.

(16) During a normal start, release the start button when light off is confirmed on the TIT, ITT, EGT, ET indicator.

(17) Do not hold the start button in if light off is not confirmed by 18% RPM.

(18) Do not attempt to start an engine the third time. Find out what the problem is before proceeding.

(19) Observe the starter duty cycles as described in the Operating Limitations Section in the Aircraft Flight Manual.

(20) Never attempt to crank the engine with the prop feathered.

(21) Before starting the engines on the ground, move the power levers forward, then move them back over the gate just forward of ground idle.

(22) Before stopping an engine make sure that RPM has gone above 90% (this charges the fuel purge tank).

(23) On the ground, always shut the engines down with the power levers in ground idle.

(24) After pressing the stop button, note that the RPM increases to approximately 70%, then starts to decay (this is the fuel purge). When the RPM starts to decay, pull the power levers back to reverse, and at 20% push the power levers forward (this puts the prop on the start locks).

(25) If the fuel purge does not work or the engine was shut down with the stop and feather pull knob, always crank the engine to clear any raw fuel.

(26) Never shut the engines down with the generator switches on.

(27) Use high residual start procedures when necessary.

(28) Use series start mode only when necessary.

(29) Before starting engines, turn on boost pumps and check pressure on fuel pressure indicator. (This checks the boost pumps, fuel pressure indication, and checks that the aircraft fuel shutoff is open).

(30) Turn the boost pumps off during engine start.

CAUTION: MAKE SURE BOOST PUMPS ARE ON BEFORE TAXIING TO TAKEOFF.

(31) Turn off equipment during engine start that is not actually needed to start the engine, such as lights, avionics, and especially equipment with motors. (This is done because of the low voltages and high currents experienced during a start and will lengthen the service of these components.

C. The following troubleshooting procedures:

(1) Apply to both left and right engine start systems unless otherwise stated.

(2) Apply to both -3 and -10 engine installations unless otherwise stated -3 Engine – T201-275, 277-291; AT001-999, TC201-999, -10 Engine – T276, 292-999.

(3) Apply to all aircraft unless otherwise stated.

NOTE: Early model aircraft are T201-278, AT001-056, TC201-229. Late model aircraft are T279-999, AT057-999, TC230-999.

Start system wiring diagrams must be used in conjunction with these procedures.

Start system wiring diagrams can be found in Chapter 24 or Chapter 80 of Wiring Diagram Manual.

When troubleshooting the start system, make sure the props and engine are cleared and guarded if you crank the engine or it cranks inadvertently.

CAUTION: WHEN CHECKING FOR VOLTAGES AT THE STARTER AND OTHER COMPONENTS ON THE ENGINE, STAY CLEAR OF THE PROP.

D. You may wish to remove the wires from terminal C of the starter and connect a test light to them. This will prevent the prop from spinning while checking components during the start cycle. When this is done, the speed switches will have to be jumped or overridden. This is done on the -3 engine installation by removing the connector from the speed switch and by jumping A to G and E to F. A to G is the 10% speed switch, and is closes at 10%; E to F is the 55% speed switch, and it opens at 55%.

E. On -10 engine installations, the speed switch select switch can be used to override the speed switches, or the connector can be removed from the SRL computer, and pins 9 to 10 and 5 to 6 can be jumped. Pins 9 to 10 are the 10% to 60% speed switch and they lock in the start control relays. Pins 5 to 6 are the 10% to 80% speed switch and they only operate the primary solenoid valve.

F. If a defect exists that complies with the items in the DEFECT column, comply with the PRELIMINARY and follow the steps in the CORRECTIVE ACTION column. If you suspect an item in the PROBABLE CAUSE column, comply with the PRELIMINARY and follow the steps in the CORRECTIVE ACTION column next to the item.

G. When checking for 28 VDC, a voltmeter or a test light may be used. When wiring, wire terminals, or relay contacts are a suspected cause, a test light should be used since it places a slight load on the circuit. The slight load will cause the circuit being checked to open at the trouble point, and the light will extinguish. A voltmeter will place almost no load on the circuit, and the defective item will be harder to find.

CAUTION: SHOULD THE ENGINE START TO SPIN WHEN THE GPU IS CONNECTED OR THE BATTERY SWITCHES ARE TURNED ON, IMMEDIATELY DISCONNECT THE GPU OR TURN OFF THE BATTERIES.

H. Troubleshooting – Starting

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Engine will not crank with start button or starter test switch		PRELIMINARY Make sure that the batteries are in good condition and GPU is connected and operating.
	1. No power at essential bus (current limiter blown)	1. Turn battery switches on and run pilot's and copilot's pitch trim. Should one side not work, open J-box behind pilot's seat and check or replace current limiters.
	2. No power at start bus (A2 of series-parallel relay). Series-parallel relay defective	2. Check for 28 VDC at A2 of series-parallel relay. If not present, check for 28 VDC on bus bar connected to B1 and A3 of series-parallel relay. If present, series-parallel relay is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	3. Wire terminals loose or defective	3. Check all wire terminal connections on start panel, especially four gauge and 0 gauge cable connections; check for 28 VDC at hot side of starter relays. Check wire terminals in X1 and X2 of starter relay, and check X2 for ground.
	4. Starter relay defective	4. With start button engaged, check for 28 VDC at X1 and starter side of relay. If steps 1, 2, and 3 are okay and there is no power at X1, proceed to step 6. If 28 VDC at starter side of relay, proceed to step 5. If 28 VDC at X1 and not at starter side of relay, relay is defective.
	5. Starter Generator defective	5. Remove right engine cowling and check terminal C for 28 VDC when start button is engaged and check terminal E for ground. If power and ground are present, starter is defective.
	6. Wire defective from C1 of No. 1 start control relay to X1 of starter relay	6. Remove floorboard beneath pilot's rudder pedals, and check terminal connections to relays; check continuity of wire.
	7. Engine problem	7. Turn prop thru by hand in the normal direction of rotation; check for binding.

STARTING – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Engine will not crank with start button ,but will crank with starter test switch		PRELIMINARY Make sure that the batteries are in good condition and GPU is connected and operating. If unfeather pump operates on the ground when start button is engages, proceed to step 3.
	1. No. 1 start control (15 amp) circuit breaker defective	1. Pull and reset No. 1 start control circuit breaker and re-engage start button. If engine cranks circuit breaker is defective. (Also see step 3), after checking steps 1 and 3, replace circuit breaker.
	2. Start or stop buttons or wiring defective	2. With battery switches off, push start and stop buttons; check that they click, check that the start clicks and the stop clicks sound the same. Check that the buttons do not feel loose. Replace switches if necessary. Look underneath switch panel at the back of the switches and check for loose wires. Resolder if necessary (also see step 4).

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	3. Squat switch or squat switch relay defective	3. On early model aircraft turn battery switches on, and check in the left wheel well at terminal 13 through the squat switch to 14 for 28 VDC; or check in the right wheel well at terminal 11 through the squat switch to 12 for 28 VDC; if 28 VDC is present at both terminals, proceed to step 4. If 28 VDC not present at terminal 13 or 11, go back to step 1. If 28 VDC not present at terminal 14 or 12, squat switch is defective. On late model aircraft, turn the battery switches on, and pull and reset the No. 1 start control circuit breaker. Listen for the squat switch relay to click (the relays are located in the left console). If relay clicks, proceed to step 4. If the relay does not click, go in the wheel well and ground terminal 13. Go back up in the cockpit and check for the relay to click. If relay now clicks, squat switches or ground wire is defective. If relay still does not click, open left console center door; check squat switch relay for broken wires. Check X2 for ground, and check X! and A2 for 28 VDC. If no ground on X2, wire that goes out to wheel well is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
		If 28 VDC is not at X1 and A2, No. 1 start control ciruit breaker or wiring is defective. If 28 VDC is at X1 and A2, check A1 for 28 VDC. If not at A1 and the relay actuates the relay is defective.
	4. No. 1 start control relay, start or stop button, speed switch circuit breaker, or wiring defective	4. Remove the floorboard below the pilot's rudder pedals. Check C2 of No. 1 start control relay for 28 VDC. If 28 VDC is not present, and steps 1 and 3 check good, check wiring from squat switch or relay to C2. Check X2 for ground, if ground is not present, the stop switch or wiring is defective. Check X1 for 28 VDC when start button is depressed. If 28 VDC is not present when start button depressed, start switch or speed switch circuit breaker is defective. On early model aircraft, if start switch checks good and there is not power on speed switch circuit breaker, proceed to step 5. On late model aircraft, go back and start with step 1.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	5. Overload sense relay, overload sensor overload circuit breaker, diodes or wiring defective, (early model aircraft only)	5. Check A1, A2, and X1 for 28 VDC and check X2 for ground. If 28 VDC is present at S1 and is not present at S2, sensor is defective. If 28 VDC is not present at S1, check overload circuit breaker and wiring. If 28 VDC is not present at A1 and A2 of overload sense relay, check L2 and L1 of overload sensor for 28 VDC. If 28 VDC is present at L1 but not at L2, overload sensor is defective. If 28 VDC is not present at L2, check diodes mounted on series parallel relay, and check wiring.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Start system will not lock in at 10% (auto start)		PRELIMINARY Disconnect starter and install jumper on speed switches for step 3 as per instructions on page 4.
	1. Speed switch/SRL computer defective	1. Swap connectors on speed switches or SRL computer,and try to start again. If start is okay, replace defective component.
	2. No RPM signal	2. On aircraft with -3 engine, the RPM signal for the speed switch comes from the tach generator. Problems with the RPM signal will first show up on the RPM indicator. The first place to look for an RPM problem that effects the RPM indicator as well as the speed switches is on the engine. Two splices in the tach generator wiring are located in the wire bundle on the left of the engine just aft of the nose cowl. These splices are sometimes a cause of problems with the RPM indicating system.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
		On aircraft with -10 engines the RPM signal for the speed switches in the SRL computer comes from the prop governor. The first place to check for a problem is the hand shakes at the prop governor. The black and white wires are for the monopole. The ohm reading from pins 11 and 12 of the SRL computer should be 52-68 ohms. The reading may be slightly higher if the engine is hot.
	3. No. 2 start control relay or wiring defective	3. Check that X2 of No. 2 start control relay is grounded; check D2 for 28 VDC. Check X1 and D1 for 28 VDC above 10%. If no ground at X2, check wiring to X2 of No. 1 start control relay. If 28 VDC is not at D2 on early model aircraft, check wiring back to overload sense relay; on late model aircraft, check wiring back to No. 1 start control of relay. If 28 VDC is not at X1 above 10%, check wiring from D1 and D2 of No. 1 start control relay to the speed switch or speed switch select switch. If ground is at X2 and 28 VDC is at X1 above 10%, the No. 2 start control relay is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Start system locks in at 10%, but no light off		PRELIMINARY Disconnect starter and install jumper on speed switches for steps 2 and 3, as per instructions on page 4.
	1. Operation procedure	1. See procedures in the Flight Manual.
	2. Ignition unit and plugs	2. Listen for audible ignition snap during start, or move continuous ignition switch to CONT and listen for audible snap. Also check for illumination of IGN light. If snap is audible and light illuminates, proceed to step 3. 3. If ignition snap is not audible but IGN light illuminates, check for 28 VDC at B and C, and check for ground at pin A of ignition unit connector. If 28 VDC or ground is not present, check wiring back to ignition mode control relay. If present check and or replace ignitor plugs and unit.
	3. Engine fuel problem engine fuel shutoff)	3. Remove connector from fuel shutoff valve, and check for 28 VDC at pin A and ground at pin B. If present and valve does not open, fuel shutoff valve is defective (see Section 71-00-00 of Maintenance Manual for further troubleshooting and check engine fuel system).

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Propeller will not unfeather (spin) during air start		PRELIMINARY Connect GPU to aircraft and energize. Check oil level for sufficient oil to operate unfeather pump.
	1. No. power at essential bus (current limiter blown)	1. Turn battery switches on, and run pilot's and copilot's pitch trim. Should one side not work, open J-Box behind pilot's seat, and check or replace current limiters, as necessary.
	2. Unfeather pump or ground wire defective	2. With NTS TEST switch to affected side, run unfeather pump with UNFEATHER TEST switch, and check for green NTS TEST light to illuminate. If green light illuminates, check pump pressure in accordance with Maintenance Manual Chapter 61. Check pump ground wire. If greenlight does not illuminate, remove connector from pump and check for 28 VDC. If 28 VDC is not present, proceed to step 3. If 28 VDC is present, unfeather pump is defective.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
	3. Squat switch, squat switch relay, or wiring defective	3. Turn the battery switches on and cheat the outboard squat switch (eight wire switch). Make sure hydraulic pressure is zero and the landing gear handle is in the down position. On early aircraft, check in the left wheel well at terminal 13 through the squat switch to terminal 15 for 28 VDC; or check in the right wheel well at terminal 11 through the squat switch to terminal 13 for 28 VDC. If 28 VDC is present at both terminals. proceed to step 4. If 28 VDC is not present at terminals 13 or 11, check start control No. 1 (15 amp) circuit breaker and wiring. If 28 VDC is not present at terminals 15 or 13, squat switch or wiring is defective. On late model aircraft, check for ground at terminal 13 in the left or right wheel well. If a ground is present, squat switch or wiring is defective. If ground is not present, check X2 of squat switch relay ground. If wheel well is shorted. If ground is not present, check X1, A2, and A3 of squat switch relay for 28 VDC. If 28 VDC is not present at X1 and A2, check start control No. 1 (15 amp) circuit breaker and wiring.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
		If 28 VDC is present at X1 and A2 but not at A3, squat switch relay is defective.
	4. Start control relay No. 1 or wiring defective	4. With squat switch cheated as described in step 3, push start button. With NTS TEST to affected side, check that the green NTS TEST light illuminates. If light does not illuminate and the unfeather pump is not operating, then the start control relay No. 1 or wiring is defective.
Engine crank slow during start and start test		PRELIMINARY Disconnect starter for step 3 in accordance to instructions on page 4.
	1. Batteries defective	1. Check batteries in accordance with instructions in Chapter 24 of Maintenance Manual.
	2. Starter defective	2. Check starter and brushes for condition.
	3. Oil vent valve defective	3. Check for 28 VDC at pin A of oil vent valve when the start button is depressed.

STARTING – MAINTENANCE PRACTICES

1. Removal/Installation – Starting

A. For engine mounted valves, see appropriate chapter in Garrett Maintenance Manual. For starter/generator, unfeather pump, ignition exciter, see appropriate chapter in this Maintenance manual. For relays and switches, remove the components attaching hardware. Mark wires with a piece of tape for the appropriate terminal, and remove wires from component. Or, remove wires from one terminal at a time from the old component, and install them on new component. Then, install component and perform appropriate operational check.

2. Adjustment/Test – Starting

A. There are no adjustments that can be made to the start system. Perform an operational check by starting the engines in accordance with the Aircraft Flight manual.

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AWI HEAT DESCRIPTION AND OPERATION	82-21-06	1	T AT TC	– – – – 005-056, C/O 201-229
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POWER LEVER POSITION SWITCHES DESCRIPTION AND OPERATION	82-21-08	1	T AT TC	– – – – 005-056, C/O 201-229

EFFECTIVITY:
T 201-999, AT 001-999
TC 201-999

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▌AWI DISTRIBUTION DESCRIPTION AND OPERATION	82-22-00	1	T AT TC	– – – – 057-999, C/O 230-999
▌AWI DISTRIBUTION FAULT ISOLATION	82-22-00	101	T AT TC	– – – – 057-999, C/O 230-999
▌AWI DISTRIBUTION MAINTENANCE PRACTICES	82-22-00	201	T AT TC	– – – – 057-999, C/O 230-999
AWI PUMP DESCRIPTION AND OPERATION	82-22-01	1	T AT TC	– – – – 057-999, C/O 230-999
AWI PUMP FAULT ISOLATION	82-22-01	101	T AT TC	– – – – 057-999, C/O 230-999
AWI PUMP MAINTENANCE PRACTICES	82-22-01	201	T AT TC	– – – – 057-999, C/O 230-999
AWI PUMP PRESSURE SWITCH DESCRIPTION AND OPERATION	82-22-02	1	T AT TC	– – – – 057-999, C/O 230-999
AWI PUMP PRESSURE SWITCH FAULT ISOLATION	82-22-02	101	T AT TC	– – – – 057-999, C/O 230-999
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AWI SHUTOFF VALVE MAINTENANCE PRACTICES	82-22-04	201	T AT TC	– – – – 057-999, C/O 230-999
AWI TORQUE SWITCH DESCRIPTION AND OPERATION	82-22-05	1	T AT TC	– – – – 057-999, C/O 230-999
AWI TORQUE SWITCH FAULT ISOLATION	82-22-05	101	T AT TC	– – – – 057-999, C/O 230-999
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DISTRIBUTION SYSTEM MAINTENANCE PRACTICES	82-23-00	200	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
NO. 1 AWI PUMP DESCRIPTION AND OPERATION	82-23-01	1	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
NO. 1 AWI PUMP FAULT ISOLATION	82-23-01	101	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
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AWI PUMP PRESSURE SWITCH DESCRIPTION AND OPERATION	82-23-03	1	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
AWI PUMP PRESSURE SWITCH FAULT ISOLATION	82-23-03	101	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
AWI PUMP PRESSURE SWITCH MAINTENANCE PRACTICES	82-23-03	201	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O

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AWI SHUTOFF VALVE DESCRIPTION AND OPERATION	82-23-05	1	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
AWI SHUTOFF VALVE FAULT ISOLATION	82-23-05	101	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
AWI SHUTOFF VALVE MAINTENANCE PRACTICES	82-23-05	201	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
AWI TORQUE SWITCH DESCRIPTION AND OPERATION	82-23-06	1	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
AWI TORQUE SWITCH FAULT ISOLATION	82-23-06	101	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
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ANNUNCIATOR PANEL MAINTENANCE PRACTICES	82-23-07	201	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
POWER LEVER POSITION SWITCHES DESCRIPTION AND OPERATION	82-23-08	0	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O
POWER LEVER POSITION SWITCHES FAULT ISOLATION	82-23-08	101	T AT TC	276, 292-999, C/O 038E-999, C/O 217-999, C/O

EFFECTIVITY:
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DISTRIBUTION SYSTEM MAINTENANCE PRACTICES	82-24-00	200	T AT TC	279-291,C/O – – – – – – – –
AWI PUMP DESCRIPTION AND OPERATION	82-24-01	1	T AT TC	279-291,C/O – – – – – – – –
AWI PUMP FAULT ISOLATION	82-24-01	101	T AT TC	279-291,C/O – – – – – – – –
AWI PUMP MAINTENANCE PRACTICES	82-24-01	201	T AT TC	279-291,C/O – – – – – – – –
AWI PUMP PRESSURE SWITCH DESCRIPTION AND OPERATION	82-24-02	1	T AT TC	279-291,C/O – – – – – – – –
AWI PUMP PRESSURE SWITCH FAULT ISOLATION	82-24-02	101	T AT TC	279-291,C/O – – – – – – – –
AWI PUMP PRESSURE SWITCH MAINTENANCE PRACTICES	82-24-02	201	T AT TC	279-291,C/O – – – – – – –
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AWI SHUTOFF VALVE DESCRIPTION AND OPERATION	82-24-04	1	T AT TC	279-291,C/O – – – – – – – –
AWI SHUTOFF VALVE FAULT ISOLATION	82-24-04	101	T AT TC	279-291,C/O – – – – – – – –
AWI SHUTOFF VALVE MAINTENANCE PRACTICES	82-24-04	201	T AT TC	279-291,C/O – – – – – – – –
AWI TORQUE SWITCH DESCRIPTION AND OPERATION	82-24-05	1	T AT TC	279-291,C/O – – – – – – – –
AWI TORQUE SWITCH FAULT ISOLATION	82-24-05	101	T AT TC	279-291,C/O – – – – – – – –
AWI TORQUE SWITCH MAINTENANCE PRACTICES	82-24-05	201	T AT TC	279-291,C/O – – – – – – – –
ANNUNCIATOR PANEL MAINTENANCE PRACTICES	82-24-06	201	T AT TC	279-291,C/O – – – – – – – –
POWER LEVER POSITION SWITCHES DESCRIPTION AND OPERATION	82-24-07	0	T AT TC	279-291,C/O – – – – – – – –
POWER LEVER POSITION SWITCHES FAULT ISOLATION	82-24-07	101	T AT TC	279-291,C/O – – – – – – – –
POWER LEVER POSITION SWITCHES MAINTENANCE PRACTICES	82-24-07	201	T AT TC	279-291,C/O – – – – – – – –

EFFECTIVITY:
T 201-999, AT 001-999
TC 201-999

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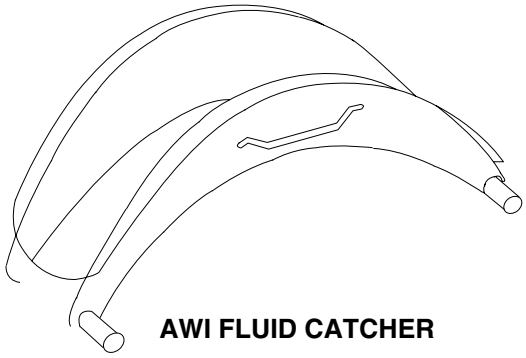
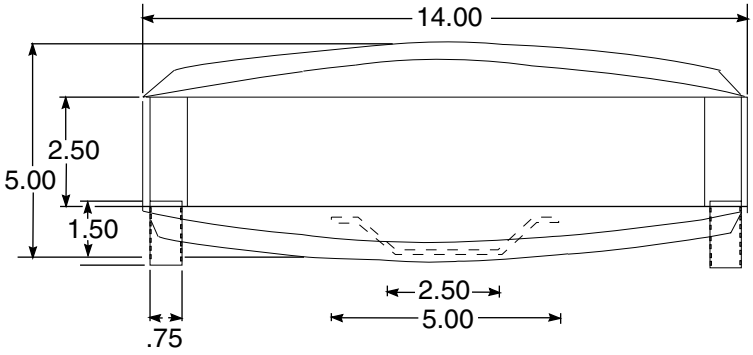
AWI QUANTITY INDICATING SYSTEM FAULT ISOLATION	82-41-00	100	T AT TC	– – – – 005-056,C/O 201-229
AWI QUANTITY INDICATING SYSTEM MAINTENANCE PRACTICES	82-41-00	201	T AT TC	– – – – 005-056,C/O 201-229
AWI QUANTITY INDICATING SYSTEM DESCRIPTION AND OPERATION	82-42-00	0	T AT TC	– – – – 057-999,C/O 230-999
AWI QUANTITY INDICATING SYSTEM FAULT ISOLATION	82-42-00	100	T AT TC	– – – – 057-999,C/O 230-999
AWI QUANTITY INDICATING SYSTEM MAINTENANCE PRACTICES	82-42-00	201	T AT TC	– – – – 057-999,C/O 230-999
AWI QUANTITY INDICATING SYSTEM DESCRIPTION AND OPERATION	82-43-00	0	T AT TC	276, 292-999, C/O 038E-999,C/O 217-999, C/O
AWI QUANTITY INDICATING SYSTEM MAINTENANCE PRACTICES	82-43-00	200	T AT TC	276, 292-999, C/O 038E-999,C/O 217-999, C/O
AWI QUANTITY INDICATING SYSTEM DESCRIPTION AND OPERATION	82-44-00	0	T AT TC	279-291, C/O – – – – – – – –
AWI QUANTITY INDICATING SYSTEM FAULT ISOLATION	82-44-00	100	T AT TC	279-291, C/O – – – – – – – –
AWI QUANTITY INDICATING SYSTEM MAINTENANCE PRACTICES	82-44-00	201	T AT TC	279-291, C/O – – – – – – – –
ALCOHOL WATER INJECTION SERVICE INFORMATION	82-SERVC	0	T AT TC	201-999 001-999 201-999

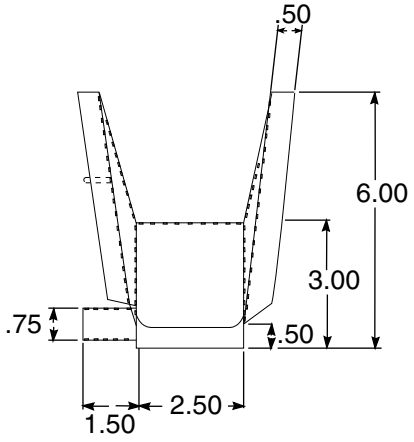
EFFECTIVITY:
T 201-999, AT 001-999
TC 201-999



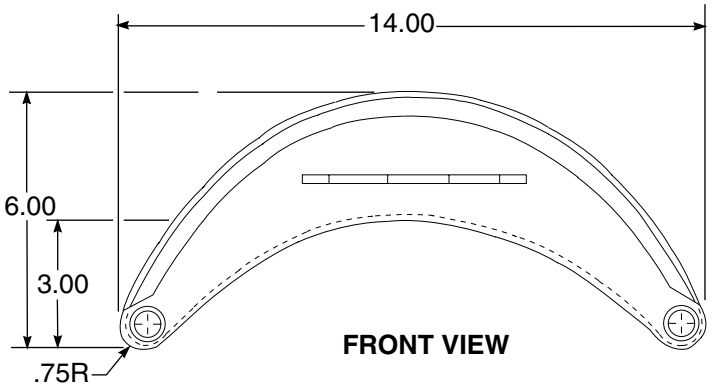
AWI FLUID CATCHER



TOP VIEW



END VIEW



FRONT VIEW

6M82SERVC1

AWI Fluid Catcher
Figure 1

MAINTENANCE MANUAL

WATER INJECTION – SERVICE INFORMATION

1. Introduction

A. The following information is supplied as a supplement to this chapter. Subject matter is included which is not applicable to an ATA 100 format, but is still of interest to Merlin and Metro owner/operators. The information in this section is not part of the basic manual, and may be removed or retained at the discretion of the owner/operator.

B. Revisions and additions to this section will be made as additional information is available. Input from individual owner/operators is encouraged, and may be submitted to:

Fairchild Aircraft Corporation
Post Office Box 790490
San Antonio, Texas 78279-0490
ATTN: Technical Publications Department

C. AWI mixture

(1) The proportion of each ingredient of the AWI mixture, alcohol and distilled water has a great influence upon the torque increase caused by AWI injection. If pure distilled water were injected into the engine, there would be almost no torque increase and a small decrease in ITT. Pure methyl alcohol injection would result in a large increase in both torque and ITT. A mixture which does not fall within the specific gravity limits of .955 to .962 will not produce the desired torque increase when injected into the engine. Alcohol and water may be mixed by volume; however, Fairchild Aircraft Corporation has found an alcohol hydrometer the only accurate method of determining the mixture.

D. Torque Increase During System Calibration

(1) EFFECTIVITY:

AT-005 - 056 (C/0)
TC-201 – 229

(2) The allowable torque increase during AWI firing shown in Table 202, section 82-21-00, will only be valid if the engine torque indicating system is operating properly. With the flow and mixture of the AWI system completely within limits, a defective torque system may cause an indicated torque increase beyond the allowances shown in Table 202. The torque system's accuracy and operation must be satisfactory if the proper amount of torque increase is to be indicated.

E. Engine Installed Metering Valve

(1) Regardless of the type of valve originally installed, all AWI metering valve replacements should be with P/N KLN375SS. The new valve features finer pitch threads for more precise adjustment and a redesigned needle and seat to improve the lineality of the flow variation. The adjusting knob is attached to the needle shaft with a roll pin. The safety wire that locks the valve is passed through this roll pin. The calibration warning tag is also attached to this safety wire.

F. AWI Inlet Fluid Catcher

(1) The pump operated systems (82-02, -03, and 04) require that the inject fluid be caught and measured during calibration. If the nose cowl is not on the engine, the AWI manifold may be positioned over a container. If the cowling is installed, matters become more difficult. One type of catcher is shown in Figure 1. It should be noted that other types of catchers have been designed and manufactured, and have proven to work equally well. Anything that will catch the fluid without restricting the flow or damaging the engine inlet is acceptable.

MAINTENANCE MANUAL

ALCOHOL WATER INJECTION – DESCRIPTION AND OPERATION

1. General

A. Alcohol Water Injection (AWI) enables the engine to produce more power for short periods of time. M7 Aerospace employs two systems:

(1) EMERGENCY SYSTEM – boosts the output of the operative engine in case of engine failure.

(2) CONTINUOUS/EMERGENCY SYSTEM – in addition to the emergency function, allows both engines to be boosted in power output for improved takeoff and climb performance. Both systems operate on the same principles and share many similar components.

B. AWI fluid is stored in a tank and travels through distribution plumbing under pressure. Common plumbing is used to the leading edge of the right wing root, where the plumbing separates and travels to the left and right engines. The flow of fluid is stopped at each nacelle by an AWI shutoff valve.

C. A selector switch installed in the cockpit allows the pilot to control the operation of the system. When the switch is placed in the ARM position, relays and associated circuitry monitor the torque output of the engines. When an engine failure is sensed, the opposite engine shutoff valve is opened, allowing the injection of AWI fluid. The continuous system incorporated circuitry allows the pilot to open both shutoff valves and boost the power output of both engines.

D. A cockpit display consisting of lights in the annunciator panel informs the pilot of the status and operational readiness of the AWI system. Both systems incorporate AWI fluid quantity indicating systems.

2. AWI Systems

A. Four alcohol – water injection (AWI) systems are covered in this chapter. The systems are as follows:

(1) System One:

(a) The Emergency AWI system which was standard on SA226-TC models S/N TC201 through TC229, and optional on SA226-AT models S/N AT005 through AT056.

(2) System Two:

(a) The Emergency AWI system which was standard on SA226-TC models S/N TC230 and up, and optional on SA226-AT models S/N AT057 and UP.

(3) System Three:

(a) The Continuous/Emergency AWI system which was optional on SA226 – T276, 292 and up, TC models S/N TC217 and up, and on SA226 – AT models S/N AT038E and up.

(4) System Four:

(a) The Continuous/Emergency AWI system was optional on SA226-T models S/N 279-291.

3. Awi System Data Arrangement

A. Coverage of each of the four AWI systems is arranged as follows:

(1) General

(2) AWI Storage

(3) AWI Distribution

(4) AWI Quantity Indicating

B. Table 1 is designed to provide a general understanding of the numbering system used throughout this chapter, and to aid the owner/operator in "customizing" this chapter to suit his needs. For example, if your aircraft is equipped with System Three as defined above, you may extract sections 82-00-00, 82-03-XX, 82-13-XX, 82-23-XX, 82-43-XX, and 82-SERVC, and file the remainder of the chapter separately for reference. These sections will provide you with information for maintaining the system installed on your particular aircraft, System three in this example.

C. The numbering system outlined in table 1 is also used in the SA226 Series Illustrated Parts Catalog, and Wiring manual. Thus, if you are the owner/operator of an aircraft equipped with System Two as defined above, and you are troubleshooting an electrical problem, in the distribution system, you will be using wiring diagram 82-22-00; you will find information concerning removal and replacement, maintenance, etc., in the Maintenance Manual under 82-22-XX; and, if replacement parts are needed, you will find these parts in the IPC under 82-22-XX.

DATA	SYSTEM 1	SYSTEM 2	SYSTEM 3	SYSTEM 4
General	82–01–XX	82–02–XX	82–03–XX	82–04–XX
AWI Storage	82–11–XX	82–12–XX	82–13–XX	82–14–XX
AWI Distribution	82–21–XX	82–22–XX	82–23–XX	82–24–XX
AWI Quantity Indicating	82–41–XX	82–42–XX	82–43–XX	82–44–XX

Arrangement of Data
Table 1

4. AWI Fluid

A. Alcohol

(1) The methyl alcohol used in the AWI system must conform to AMS 3004*. Most suppliers can provide alcohol meeting this specification. The quantity of alcohol purchased is dictated by the individual operator's needs; however, it will be less expensive if purchased in larger quantities. If the alcohol is stored in drums or other large containers, it will be necessary to transfer some quantity to smaller containers for ease of mixing. Make sure that the containers used for alcohol are clean and do not leak. Methyl alcohol is flammable and should not be stored near fire.

B. Water

(1) Water may be treated by softening, demineralizing, or distilling to insure conformance to the following:

	Min	Max
Total solids, ppm	–	10
ph	6.0	8.0
Chlorides, ppm	–	1
Sulfates, ppm	–	1

NOTE: The supplier of distilled water or water treatment equipment can provide information of his product's purity. If the above specifications cannot be met, another source of distilled water must be used.

Water Treatment Specifications
Table 2

C. Mixing

(1) Establish an area for storage of AWI fluids and mixing equipment. Do not store other fluids in this area where they may accidentally become mixed with the AWI fluids. Do not use utensils for mixing, pouring, and storing AWI fluid for other liquids. Ensure that all containers are either new, or thoroughly cleaned before being used to mix or store AWI fluid. Ensure that the area is clean and well ventilated; at certain temperatures and pressures AWI fluid is flammable. Check measuring equipment for calibration. The most accurate means of determining the mix is by using an alcohol hydrometer. The hydrometer reading must be between 0.955 and 0.962. When the proper mixture has been obtained, mark the outside of the container with the date and the hydrometer reading. On SA226-T(B) engines, TPE331-10U-501G, the approved mixture consists of methanol (40% + 6 or - 2%) and water (60% + 2% or - 6%) while SA226-T, AT, & TC have a 70/30 mixture. If the mixture is not used for servicing the aircraft, another sample and reading must taken when it is used.

5. Inspection Criteria

A. Corrosion, fungus, and other impurities in the AWI system fall into two categories: deposits and material destruction. In the case of deposits, cleaning of the part is the only requirement. Where material destruction has occurred, a determination of the loss of component reliability caused by the destruction must be made.

(1) Deposits:

(a) White, gray, or black material, having a thick paste-like consistency, are deposits of the AWI mixture. These deposits may be found inside fittings, lines, valves, or the AWI tank. This is not a form of corrosion, since the base metal of the part is not attached. Clean the parts using distilled water and blow dry. Make a thorough inspection to see that no damage to the base metal of the part has occurred. Except for a slight discoloration of the part, deposits leave no indication of having been there. If, after cleaning, the surface is rough or pitted, material destruction has occurred.

(2) Material Destruction:

(a) Pits, checking, roughness, raised ridges, or flaking are indications that corrosion has attacked the base metal of the part. This corrosion may have been caused by several different processes; the result is always a loss in the strength of the part. Hoses, tubing, and other components may continue to function with slight pitting or checking; however, if the pits are near or on the joining surfaces, or affect the sealing capability of the part, this part should be replaced. Valves may have pits in the stationary inlet and outlet ports, but corrosion on the moving parts or sealing surfaces is cause for replacement. Any corrosion which affects sealing, operation, or reliability is cause for component replacement.

EMERGENCY AWI – DESCRIPTION AND OPERATION

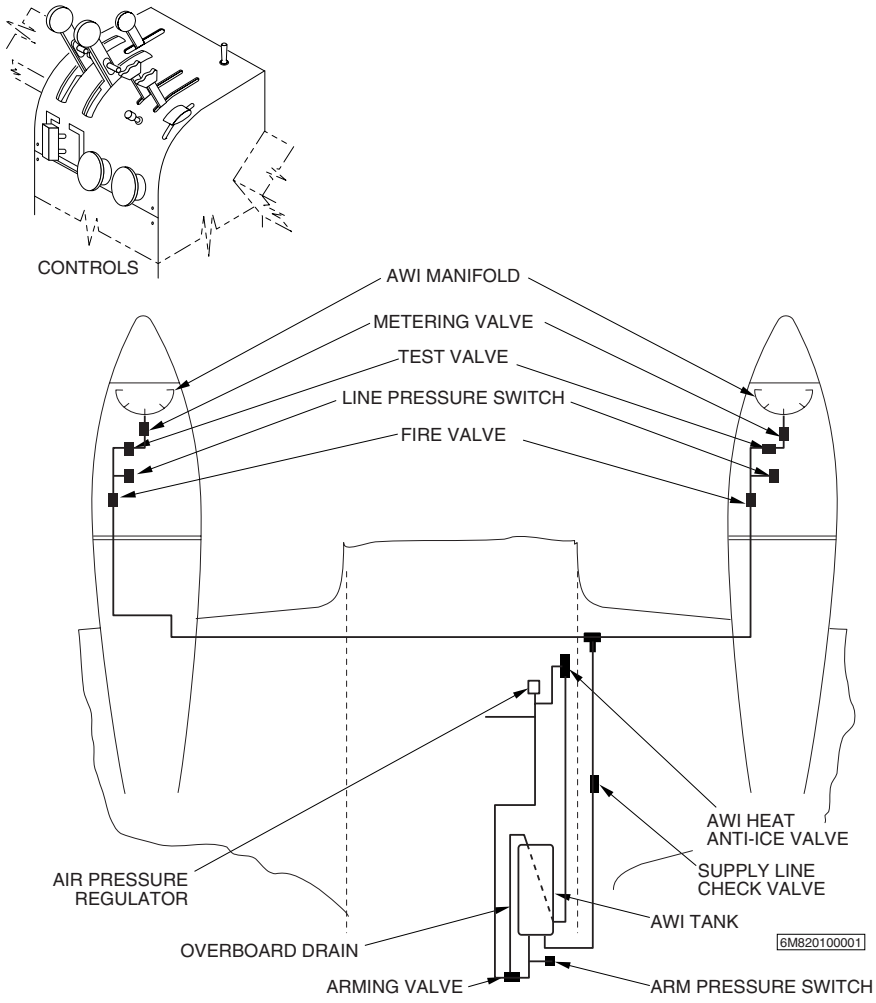
1. General

A. AWI Distribution

(1) AWI fluid is routed from the tank through the AWI supply line over the right wing to the AWI distribution line in leading edge of right hand center section. On aircraft TC-209 and later, and aircraft retrofitted by Service Bulletin 82-003, the AWI supply line contains a check valve to prevent fluid from running back into the tank when engines are stopped. Fluid travels through the leading edge of the center section to left and right engines. Fluid pressure is deadheaded against the fire valve until a signal is received to open the valve.

B. AWI Tank

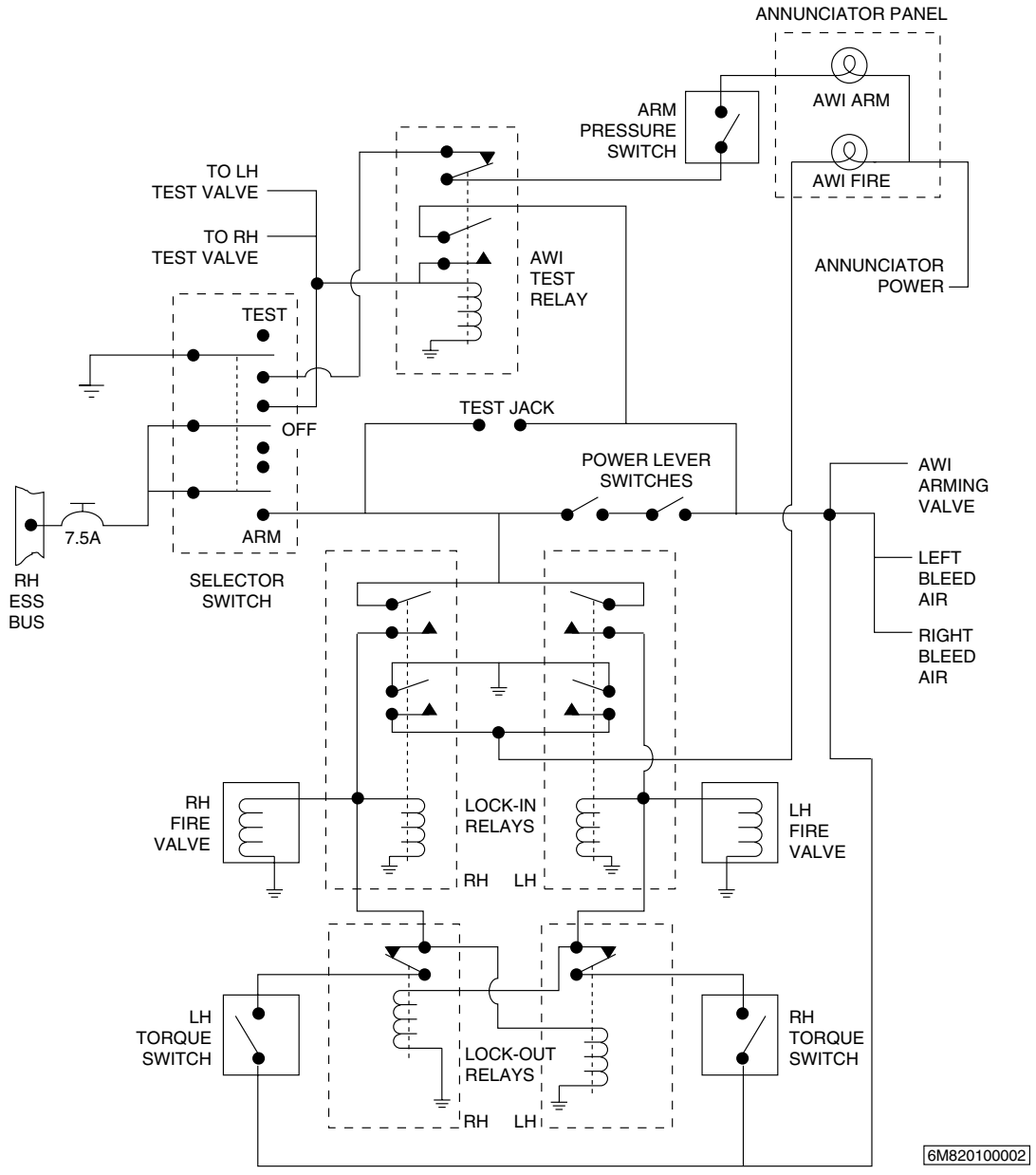
(1) A two gallon tank, located under the RH wing-to-fuselage fairing, incorporates a built-in filler and cap. The tank cannot be overfilled and features a float switch to sense liquid level. When liquid level is too low for a safe takeoff, a float switch closes causing a low water light in the cockpit to illuminate. The tank is secured by two strap type clamps. A line, extending through the tank lengthwise, provides a means of keeping AWI fluid from freezing.



AWI Flow Schematic and Controls
Figure 1

C. AWI Pressurizing Source

(1) Emergency AWI pressurizes the tank as well as the fluid. Engine bleed air is routed to a regulator, located in the RH wing root, which supplies 31 ± 1 psi to the AWI arming valve. The AWI arming valve is normally closed and opens on a signal from the AWI control circuit. When the arming valve opens, the AWI tank is pressurized. A check valve installed in the tank pressurizing line, traps pressure in the tank when the engines are not operating. A pressure sensing switch closes when 30 psi is present in the tank and illuminates the AWI arm light in the cockpit. An over-pressurization relief valve relieves the pressure overboard should it exceed 35 psi.



AWI Electrical Schematic
Figure 2

2. AWI Operation

A. A cockpit installed selector switch and the position of the power levers control the operation of the AWI system. The three positions switch (ARM, OFF, TEST) is used to activate or test the system.

B. Selector Switch in OFF Position

(1) All electrical power to the arming valve and firing circuits is removed. The AWI low water light and the AWI heat circuits are not affected by the selector switch position and continue to function.

C. Selector Switch in ARM Position

(1) With the selector switch in ARM, advancing the power levers to approximately 50% travel applies power to the AWI arming valve allowing the AWI tank to pressurize, shutting off the left and right bleed air (the bleed air flow stops regardless of the position of the bleed air switch). The movement of the power levers results in a corresponding increase in engine torque output; the torque pressure switch opens indicating high torque (engine operating). The open torque switch stops the flow of electrical power to the remainder of the system. The AWI pressure, deadheaded by the closed AWI fire valves, builds until it reaches 30 psi and the AWI ARM light illuminates. The system remains at rest until either an engine failure occurs or the selector switch is moved to off. The three conditions needed for the AWI system to arm are:

(a) Selector switch position – In order for the system to arm, the selector switch must be in the ARM position.

(b) Power lever position – To allow electrical power to be applied to the system, the power levers must be at approximately 50% of their travel. (The power lever position switches may be bypassed by installing the bypass plug. The plug was supplied as part of the aircraft's loose equipment.

(C) Engine power output – By sensing engine torque pressure output, the AWI system can react to engine failure. For the system to arm, it must sense high torque output (engine operating). Power can travel no further because the torque switch opens when high torque is sensed.

D. AWI Firing Sequence

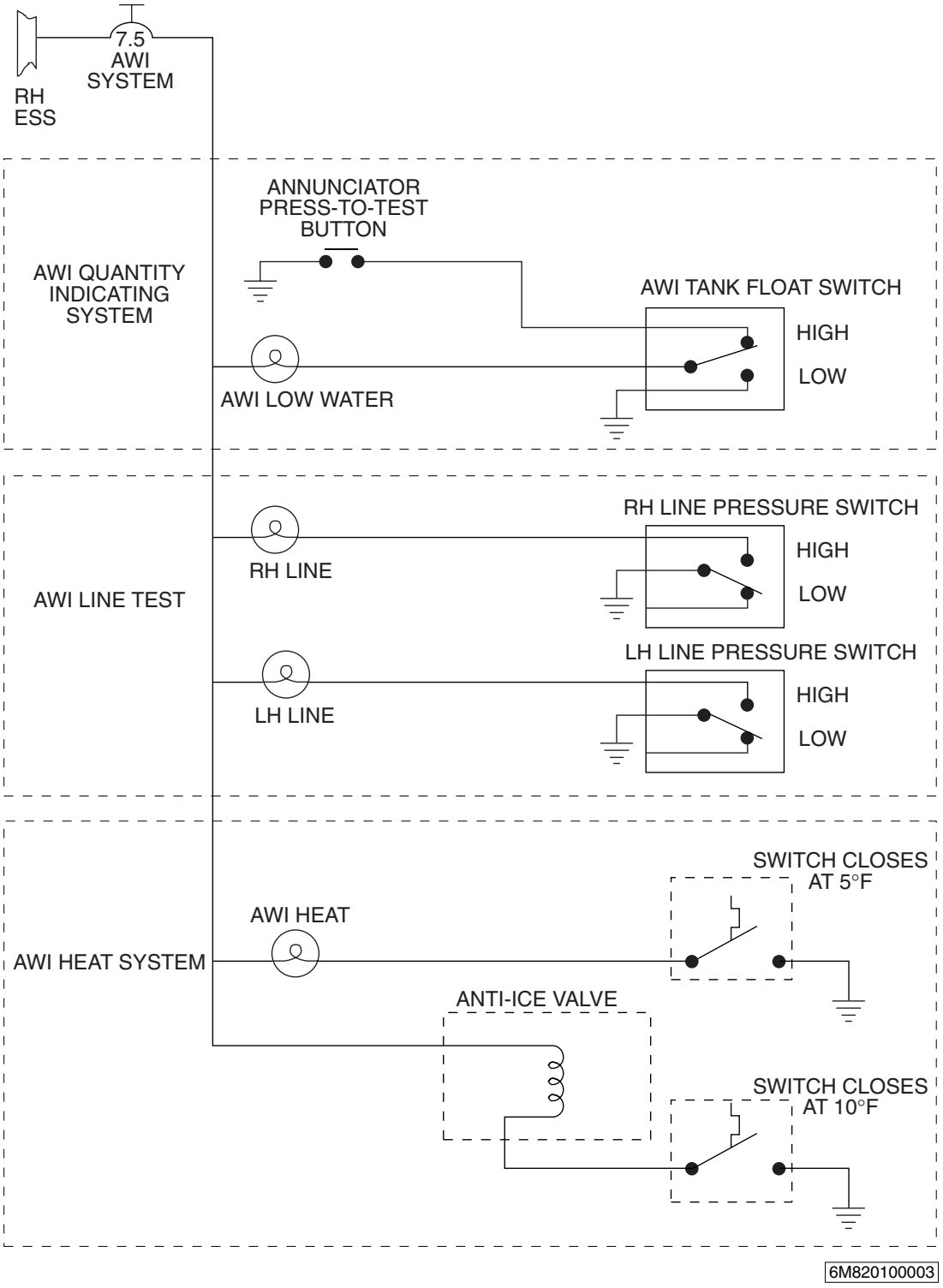
(1) Once armed, the AWI system remains at rest until a low torque output is sensed. A low torque output causes the torque switch to close and supply power to the respective side of the AWI system. The power applied results in three functions occurring:

(a) Fire valve opens – When the torque switch on one engine closes, power is applied to the opposite engine's fire valve. This action allows the power output of the operative engine to be boosted by the AWI.

(b) Lock-out relay closes – The closing of the lock-out relay breaks the electrical circuitry for the operative engine, thus ensuring that the AWI cannot fire into both engines.

(c) Lock-in relay closes – The closing of the lock-in relay provides a separate source of power for the firing valve and ensures the AWI continues to fire even if the failed engine again begins to produce power. The lock-in relay's closing also completes a circuit which illuminates the AWI fire light in the annunciator panel.

NOTE: The firing cycle continues until all AWI fluid is expended or until the AWI selector switch is moved to the OFF position.



AWI Indicating Schematic
Figure 3

E. AWI Functional Test

(1) An AWI test valve is installed on each engine between the fire valve and the AWI manifold (see Figure 1). Installed between the test valve and the fire valve is a line pressure switch. The AWI test valve is normally open; the fire valve is normally closed. When the AWI selector switch is moved to the test position and the engines are at a high power setting, the test circuit is armed. If either power lever is retarded, the respective engine power output goes low and the AWI attempts to fire into the operative engine (the circuitry of the test system does not use the power lever switches for an electrical path).

(2) In the test function, the test valve closes and, if the system is functioning properly, the fire valve opens. This action traps pressure and closes the pressure switch. The closing of the pressure switch is indicated by illumination of the respective line light in the annunciator panel. All components of the system are exercised during this test, and failure of any component will be detected.

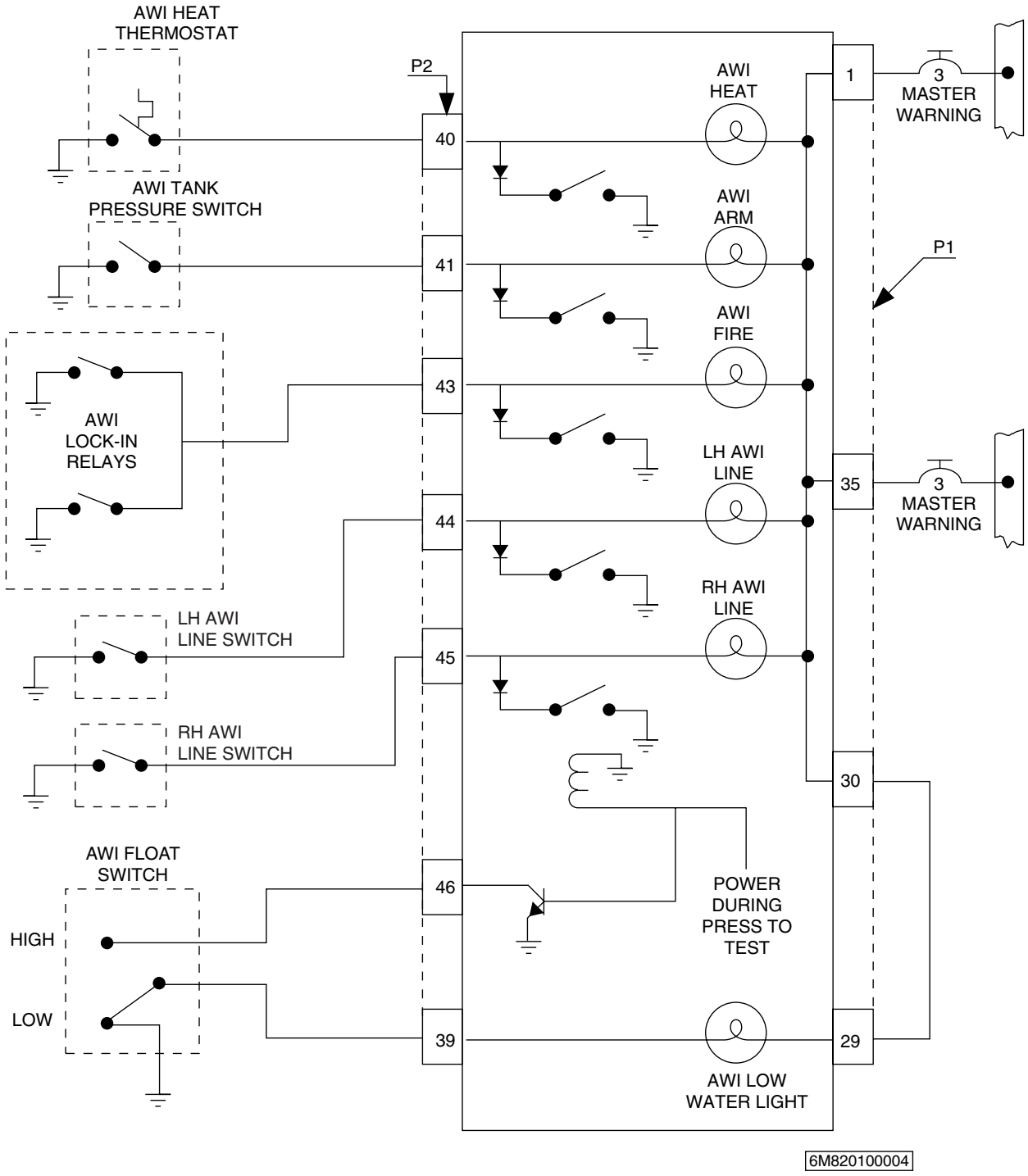
F. AWI Heat

(1) AWI heat is employed to prevent AWI fluid from freezing. Engine bleed air is routed to an anti-ice valve located in the right wing root, near the AWI pressure regulator. Two thermostats are located on the AWI supply line near the AWI tank.

(2) The first thermostat closes at 10°F (-12°C) and completes a circuit which opens the anti-ice valve. Bleed air travels to the AWI tank through the anti-ice valve and is routed over the right wing along with the AWI supply line, deicing the supply line. The AWI tank has a line passing through the center of it through which the bleed air is routed. After flowing through the line in the AWI tank, the bleed air is dumped overboard. The action of the hot bleed air passing through the AWI tank keeps the AWI fluid above its freezing point.

(3) A second thermostat installed on the supply line closes at 5°F (-15°C) and illuminates the AWI heat light in the annunciator panel. In normal operation, the action of the AWI anti-ice system keeps the temperature of the AWI fluid above 5°F and the AWI heat light never illuminates.

(4) If the aircraft was parked overnight and ambient temperatures were lower than 5°F, there is a possibility that the AWI heat light is illuminated. After the engines are operating and bleed air is flowing through the AWI tank, the AWI heat light will go out.



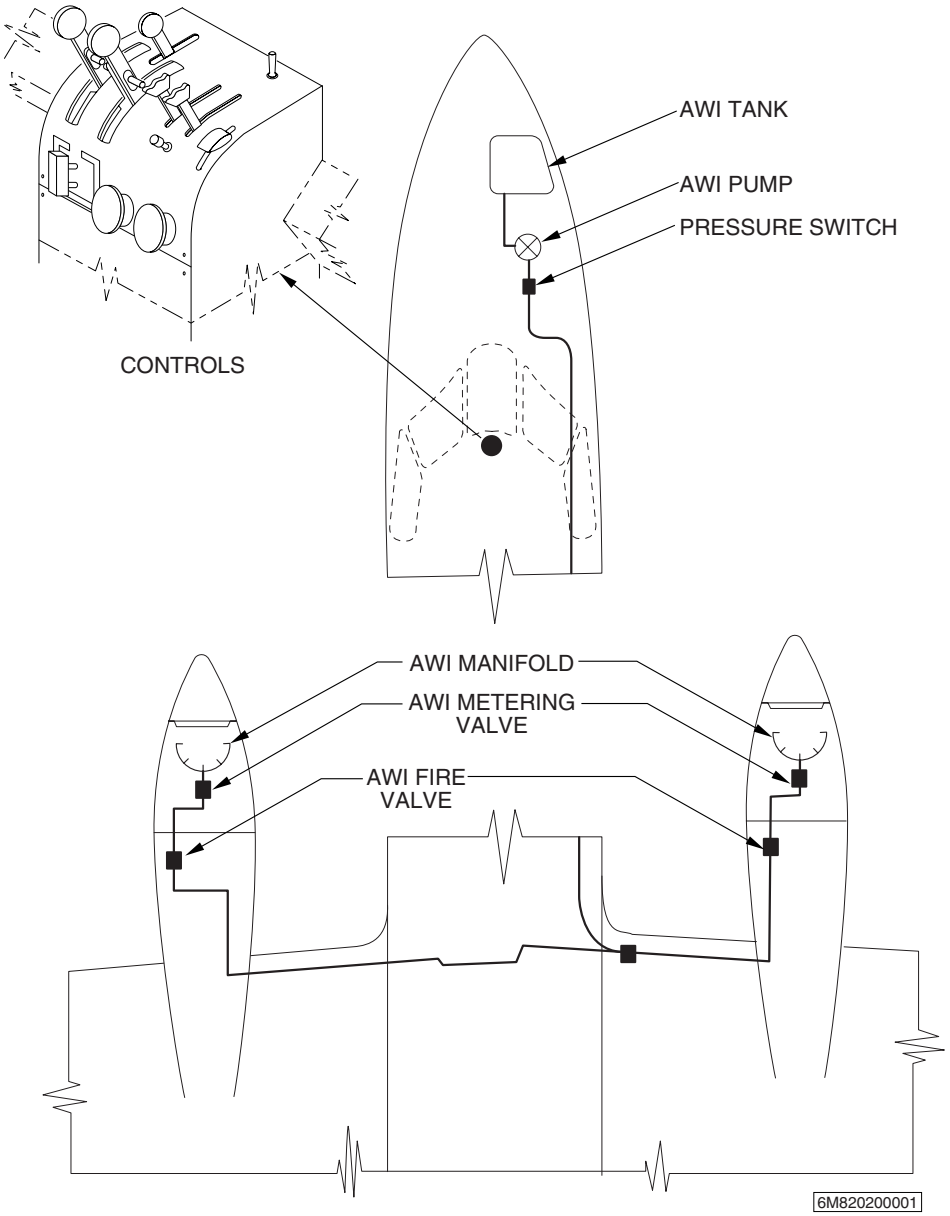
AWI Indicating Schematic
Figure 4

EMERGENCY AWI – DESCRIPTION AND OPERATION

1. General

A. AWI Distribution

(1) The supply line from the AWI pump is routed through the nose section, through the right hand side console in the cockpit and under the right hand passenger compartment floorboards. The supply line passes through the fuselage pressure vessel on the right hand side approximately at the wing root. At the wing root, the supply line is divided and travels through the leading edge of the wing to the left and right engines. The fluid is deadheaded at each engine by an AWI fire valve.



AWI Flow Schematic and Controls
Figure 1

B. AWI Tank

(1) A four gallon fiberglass tank is located in the nose baggage compartment forward area. The tank is under a sealed cover which prevents the escape of AWI fumes. A float switch and light in the annunciator panel are used to inform the pilot of the liquid level in the tank. The tank filler is located on the right hand side of the nose just forward of the baggage door. A sight gauge is used to verify the amount of fluid in the tank. A vent system is incorporated to prevent overfilling of the tank.

C. AWI Pressurizing Source

(1) An electrically driven pump pressurizes the AWI fluid for injection into the engines. The pump output line is routed through the nose section, into and through the right side of the fuselage, and exits at the leading edge of the right wing. A pressure switch is installed in the pump output line. When the pump output pressure reaches 30 psi, the switch closes and causes the AWI armed light in the annunciator panel to illuminate.

2. AWI Operation

A. A cockpit installed selector switch and the position of the power levers control the operation of the AWI system. The two position switch (ARM and OFF) is used to activate the system.

B. Selector Switch in Off Position

(1) All electrical power to the AWI pumping and control circuits is removed. The AWI quantity light is powered separately and functions regardless of the selector switch position.

C. Selector Switch in Arm Position

(1) With the selector switch in the ARM position, advancing the power levers past flight idle supplies power to the LH and RH fire valves and shuts off the LH and RH bleed air systems. Placing the selector switch in the ARM position also supplies power to the AWI pump relay, causing the pump to run and build pressure. When the pump pressure reaches 30 psi, the cockpit AWI armed light illuminates. The LH and RH engine torque switches sense high torque and thus remove the ground necessary to open the respective fire valve. The fluid pressure is trapped by each fire valve and the system remains at rest in this condition until low torque is sensed or the selector switch is moved to OFF.

D. AWI firing Sequence

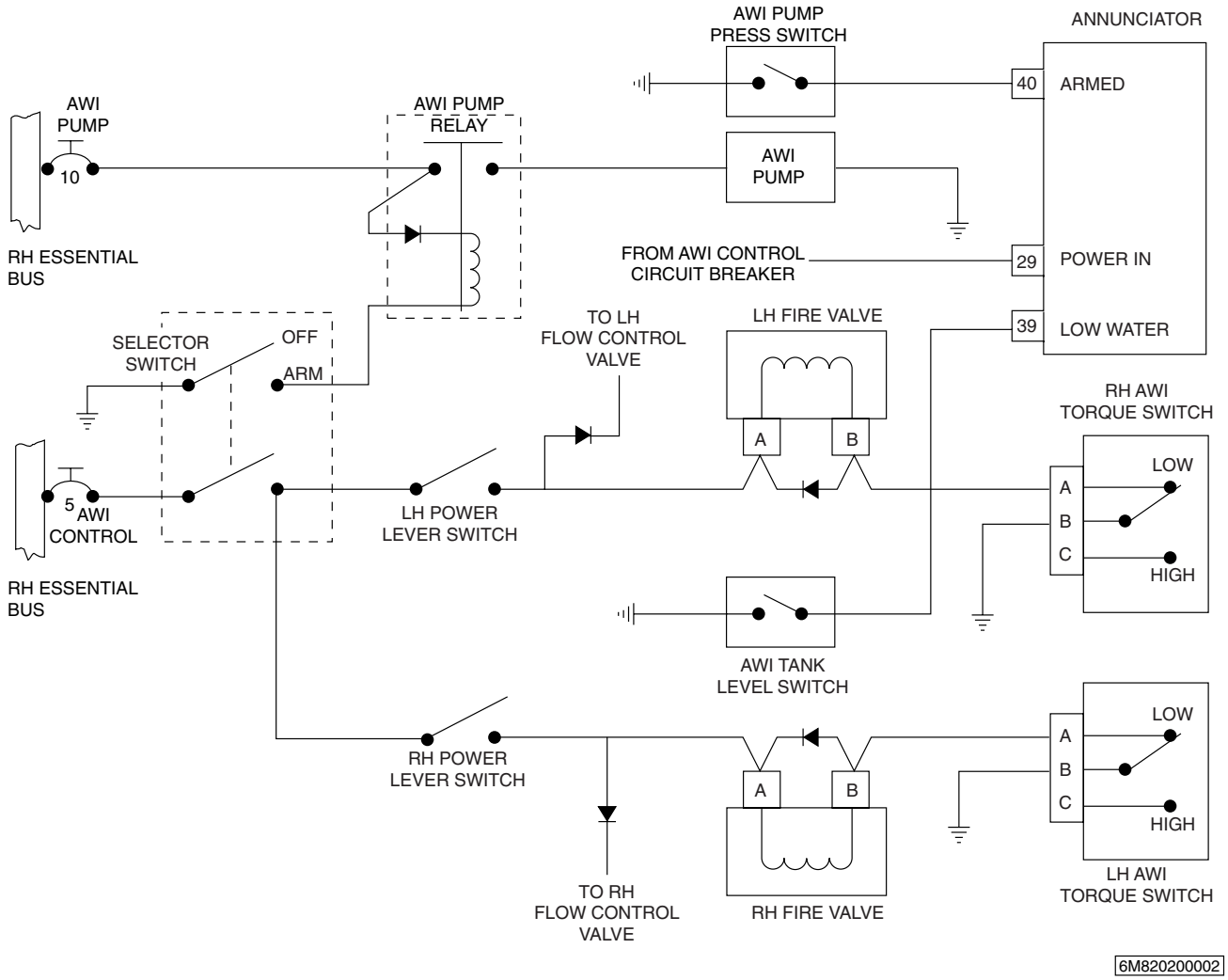
(1) The respective AWI fire valve opens and allows fluid injection only when three conditions are met.

(a) Selector switch position – In order for the system to arm, the selector switch must be in the ARM position.

(b) Power lever position – To apply electrical power to the system, the power levers must be ahead of the flight idle gate.

(c) Engine power output – Each torque switch is connected to the opposite engine's fire valve. When low torque is sensed by the switch (engine failed), the switch closes and completes the electrical path necessary to open the opposite engine's fire valve.

NOTE: Fluid injection continues until the tank is depleted or the selector switch is moved to the of position.



AWI Electrical Schematic
Figure 2

E. AWI Functional Test

(1) The AWI system is dynamically tested before each takeoff during which the system is to be armed. Both power levers are advanced to above flight idle and the selector switch is moved to arm. Each power lever is in turn retarded and an increase in torque is noted on the opposite engine. All components of the system are tested during this sequence and the system may now be armed for takeoff.

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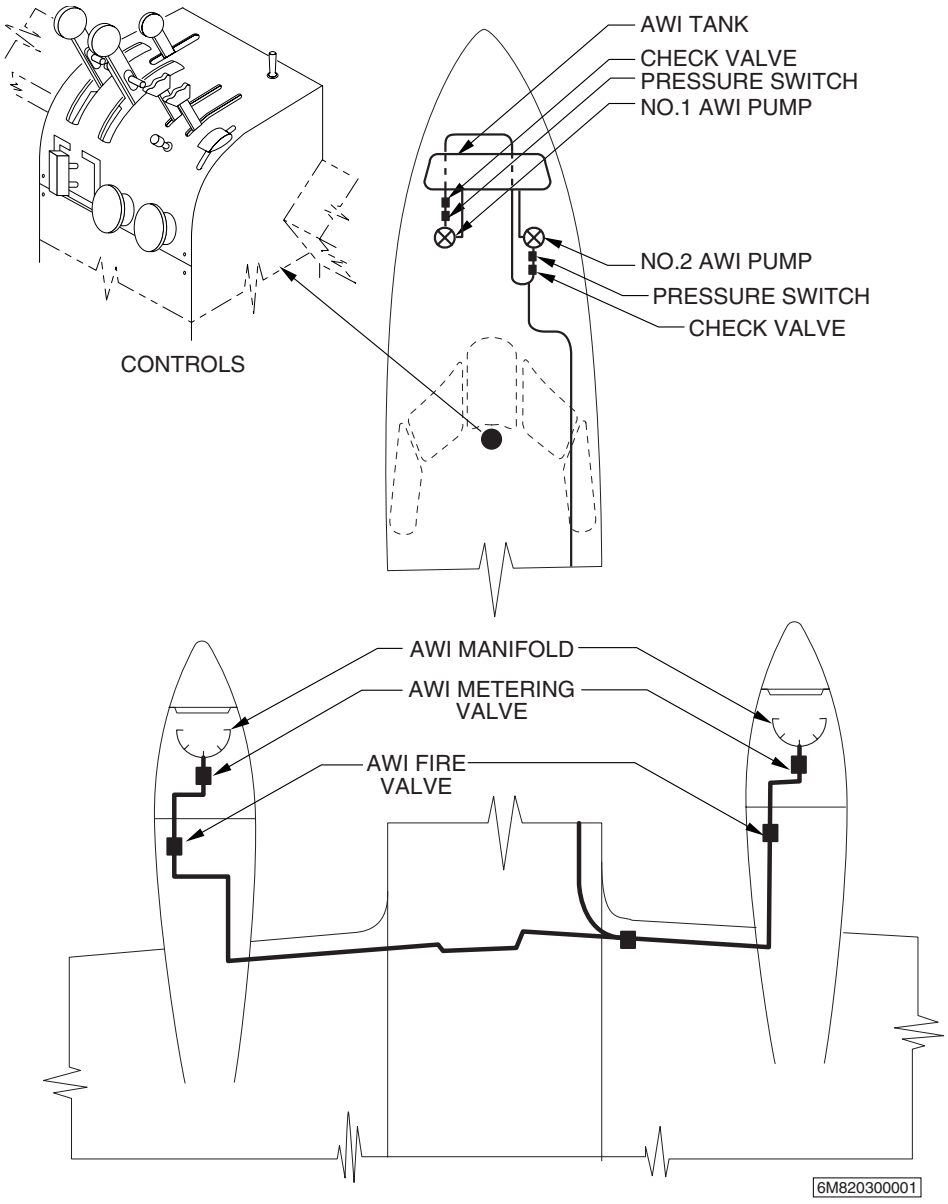
CONTINUOUS/EMERGENCY AWI – DESCRIPTION AND OPERATION

1. General

A. Continuous/Emergency AWI Distribution

(1) The supply line from the AWI pumps travels through the nose section, passing through the RH side console in the cockpit and under the RH passenger compartment floorboards. The supply line passes through the fuselage pressure vessel on the RH side approximately at the wing root. At the wing root, the supply line is divided and travels through the leading edge of each wing to the left and right engines. The fluid is deadheaded at each engine by an AWI fire valve.



AWI Flow Schematic and Controls
Figure 1

B. Continuous/Emergency AWI Tank

(1) A 16 gallon fiberglass tank is located in the nose baggage compartment forward area. The tank is located under a sealed cover which prevents the escape of AWI fluid fumes. A liquid level transmitter and cockpit indicator are used to inform the pilot of the amount of AWI fluid in the tank. The tank is secured by strap type clamps and features internal baffling to prevent the fluid from sloshing. The AWI filler is installed on the RH side of the nose, forward of the baggage compartment door. A vent system with an expansion container is installed on the right hand top side of the tank.

C. Continuous/Emergency AWI Pressurizing Source

(1) Two electrically driven pumps pressurize the AWI fluid for injection into the engines. Each pump is fed from a sump on its side of the tank. Each pump's pressure is routed through a check valve which prevents the pressure from bypassing through the pump should it fail to operate. A connecting "Y" is used to merge the individual pump outlets. From the "Y" the fluid travels through common plumbing until reaching the leading edge of the RH wing. A pressure switch is installed in each pump output to provide a means of testing the system and to indicate to the pilot that the respective pump is in operation.

2. Continuous/Emergency AWI Operation

A. A cockpit installed selector switch and the position of the power levers control the operation of the AWI system. The three position selector switch (ARM, OFF, CONT) is used to select the mode in which the system operates.

B. Selector Switch in OFF Position

(1) All electrical power to the AWI pumping and arming circuits is removed. The AWI quantity indicator is powered separately and functions regardless of the selector switch position

C. Selector Switch in ARM Position

(1) With the selector switch in the ARM position, advancing the power levers past flight idle supplies power to the arming circuit. AWI fluid cannot flow into the engine(s) because the fire valve in each nacelle is closed. The fluid pressure builds to 30 psi and each AWI pump pressure switch closes, illuminating the respective AWI pump light in the cockpit. The power lever position also insures that each engine is producing more than 600 foot-pounds of torque.

(2) The action of the two AWI control relays closing completes an electrical path and illuminates the AWI armed light. The system remains at rest in this condition unless low torque is sensed or the selector switch is moved to off.

D. Continuous/Emergency AWI Firing Sequence

(1) The AWI control box opens the respective fire valve and allows AWI fluid flow only when three conditions are met:

(a) Selector switch position – To function in the emergency mode, the selector switch must be in the arm position.

(b) Power Lever Position – To apply electrical power to the system, the power levers must be forward of the flight idle gate.

(c) Engine Power Output – Each torque switch is connected to the opposite engine's fire valve. When low torque is sensed by the switch (engine failed), the switch closes and completes the electrical path necessary to open the opposite engine's fire valve.

NOTE: Fluid injection continues until the tank is depleted or the selector switch is moved to the OFF position.

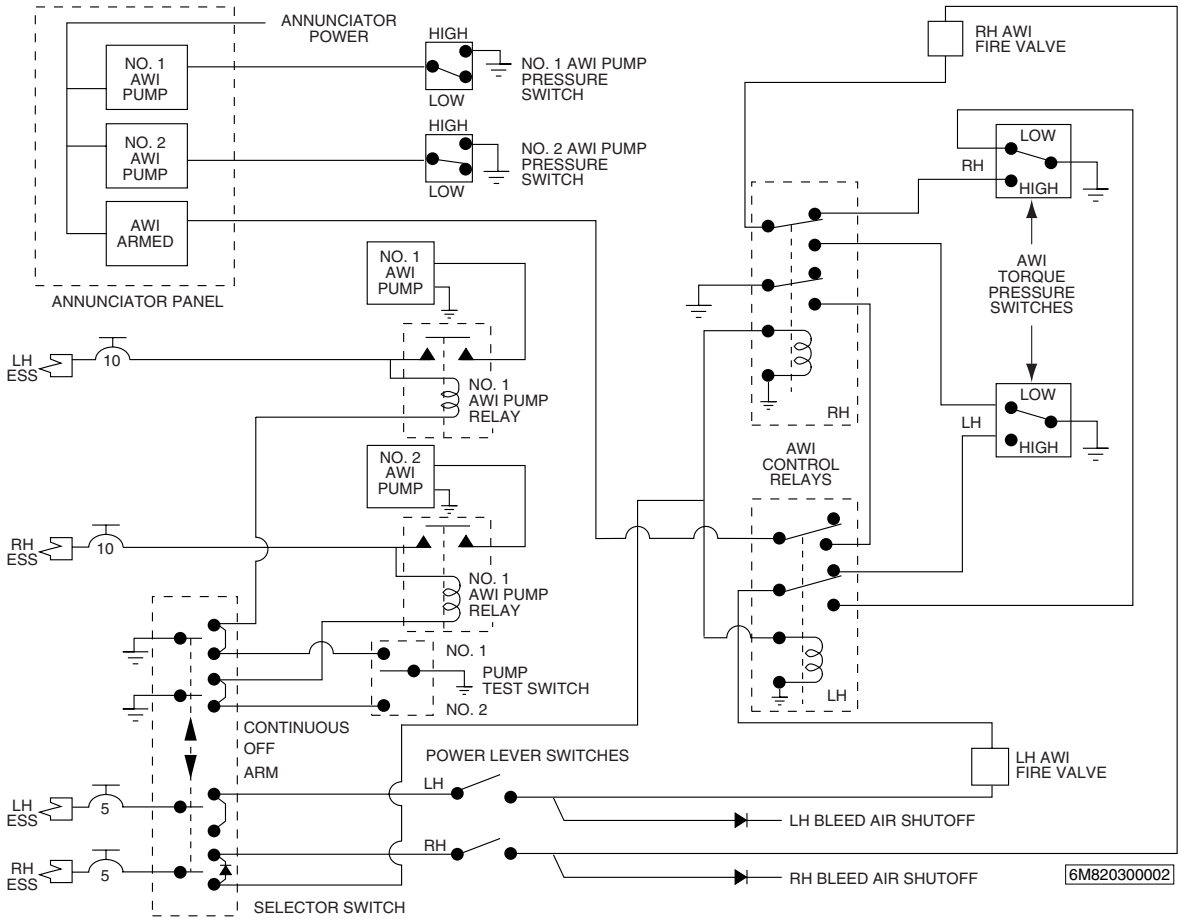
E. Selector Switch in CONT Position

(1) Placing the AWI selector switch in the CONT position fires the AWI into both engines. if three conditions are met.

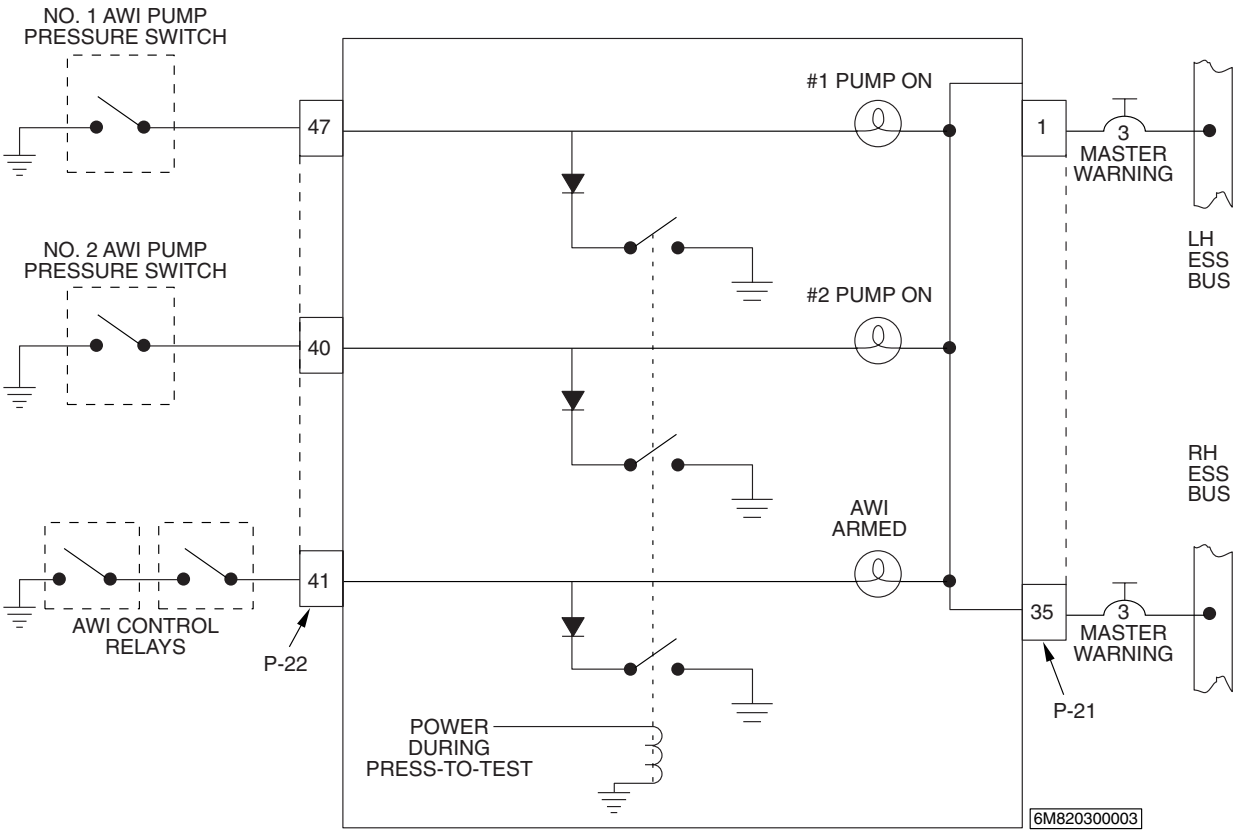
(a) Selector switch position – To function in the continuous mode, the selector switch must be in the CONT position.

(b) Power Lever position – To apply electrical power to the system, the power levers must be forward of the flight idle gate.

(c) Engine Power Output – The engine torque pressure output must be high (engine operating). This is a safety feature during a takeoff with continuous AWI. Should one engine fail, fluid injection to the failed engine stops.



AWI Electrical Schematic
Figure 2



AWI Indicating Schematic
Figure 3

F. AWI Functional Test

(1) The AWI system is provided with a selector switch, located below the main AWI selector switch, to allow the No. 1 and No. 2 AWI pumps to be checked and the two AWI system check valves to be tested. When the switch is placed in the No. 1 position, power is applied to the left hand AWI pump. The pump builds pressure; however, the pressure is deadheaded against the AWI fire valve which is closed. A pressure switch installed at the pump outlet closes and illuminates the No. 1 PUMP ON light in the annunciator panel. The No. 2 pump check valve closes, thus preventing the No. 1 AWI pump's pressure from reaching the No. 2 pump pressure switch. The AWI system is capable of operating in all modes satisfactorily on one pump if the opposite check valve is working. If the opposite check valve is stuck open, the operative AWI pump's pressure bypasses through the inoperative pump. When the AWI pump test switch is selected to a pump, that pump's pressure indication light must come on, but most importantly, the other pump's pressure light must not come on.

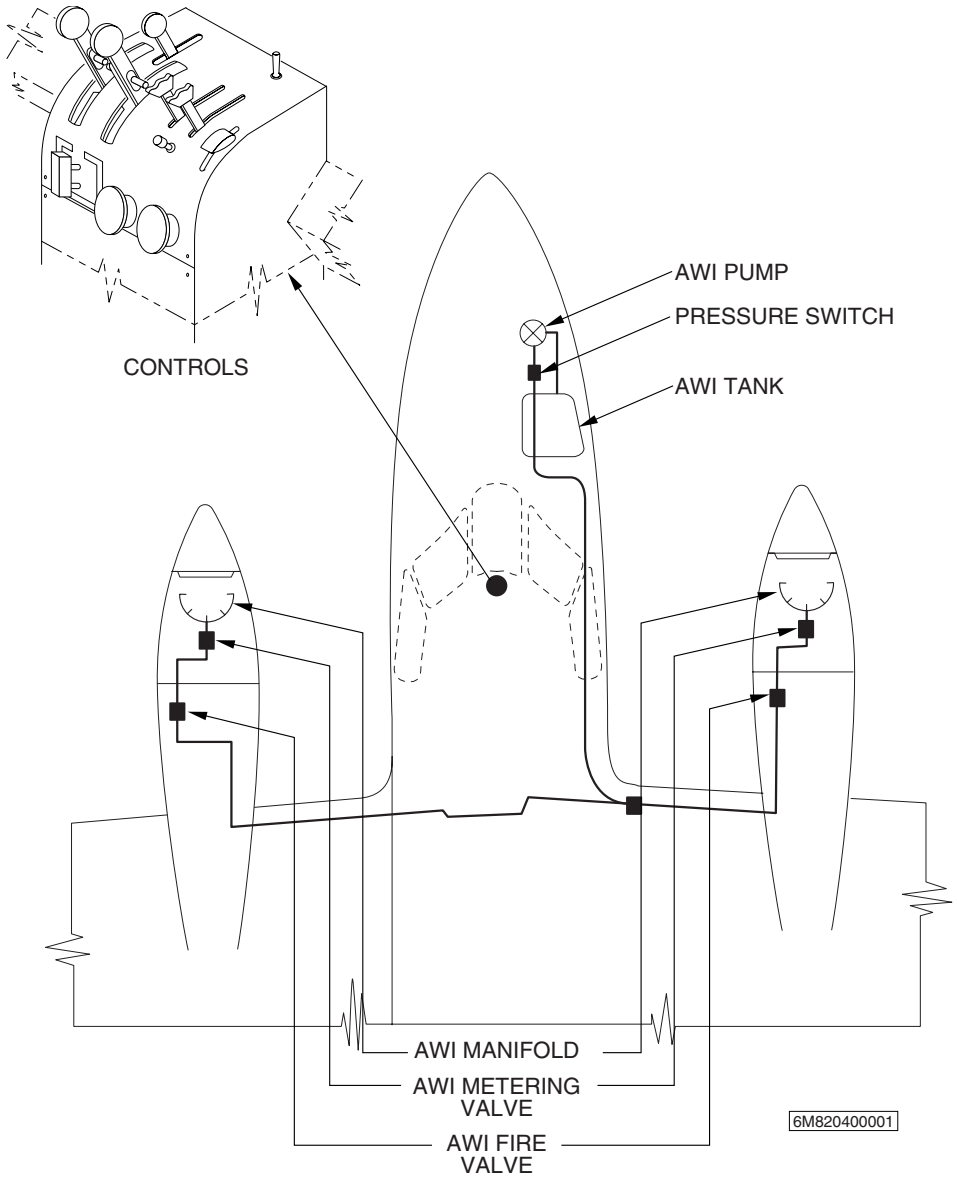
(2) The fire valves and AWI torque pressure switches are tested by arming the system, advancing the power levers until the engines are producing approximately 600 foot-pounds of torque, and then retarding the power levers one at a time, observing the torque rise in the opposite engine. It is not necessary to test the continuous portion of the system separately. During a continuous AWI takeoff, the actual use of the system tests all components.

CONTINUOUS/EMERGENCY AWI – DESCRIPTION AND OPERATION

1. General

A. Continuous/Emergency AWI Distribution

(1) The supply line from the AWI pump is routed through the nose section, passes through the RH side console in the cockpit and under the RH passenger compartment floorboards. The supply line passes through the fuselage pressure vessel on the RH side approximately at the wing root. At the wing root, the supply line is divided and travels through the leading edge of the wing to the left and right engines. The fluid is deadheaded at each engine by an AWI fire valve.



AWI Flow Schematic and Controls
Figure 1

B. Continuous/Emergency AWI Tank

(1) A 14 gallon fiberglass tank is located in the aft RH side of the nose baggage compartment. The tank is located under a sealed cover which prevents the escape of AWI fumes. A liquid level switch and annunciator warning light are used to inform the pilot of a low fluid level. A sight gauge is used to verify the level of fluid in the tank. A filter and scupper drain are located just inside the RH nose baggage door. A vent system is incorporated to prevent overfilling the tank.

C. Continuous/Emergency AWI Pressurizing Source

(1) An electrically driven pump pressurizes the AWI fluid for injection into the engines. The pump output is routed through the nose section, into and through the right side of the fuselage and exists at the leading edge of the wing. A pressure switch is installed at the pump output. When the pump pressure builds up to 30 psi, the pressure switch closes and causes the AWI pump light in the annunciator panel to illuminate.

2. Continuous/Emergency AWI Operation

A. A cockpit installed selector switch and the position of the power levers control the operation of the AWI system. The three position selector switch (ARM, OFF, CONT) is used to select the mode in which the system will operate.

B. Selector switch in OFF position.

(1) All electrical power to the AWI pumping and arming circuits is removed. The AWI quantity warning light is powered separately and functions regardless of the selector switch position.

C. Selector switch in ARM position.

(1) With the selector switch in the ARM position, advancing the power levers past the flight idle gate supplies power to the arming circuit. Placing the selector switch in the ARM position also supplies power to the AWI pump relay, causing the pump to run and build pressure. When the pump pressure reaches 30 psi, the cockpit AWI pump light illuminates. The selector switch position also causes the two AWI control relays to close and complete the path necessary to illuminate the AWI arm light. The fluid pressure is trapped by each fire valve and the system remains at rest in this condition until low torque is sensed or the selector switch is moved to off.

D. Continuous/Emergency AWI Firing Sequence

(1) The respective fire valve opens and allows AWI injection only when three conditions are met:

(a) Selector switch position – To function in the emergency mode, the selector switch must be in the ARM position.

(b) Power Lever Position – To apply power to the system, the power levers must be ahead of the flight idle gate.

(c) Engine power output – Each torque switch is connected to the opposite engine's fire valve. When low torque is sensed by the switch (engine failed), the switch closes and completes the electrical path necessary to open the opposite engine's fire valve.

NOTE: Fluid injection continues until the tank is depleted or the selector switch is moved to the OFF position.

E. Selector Switch in CONT Position

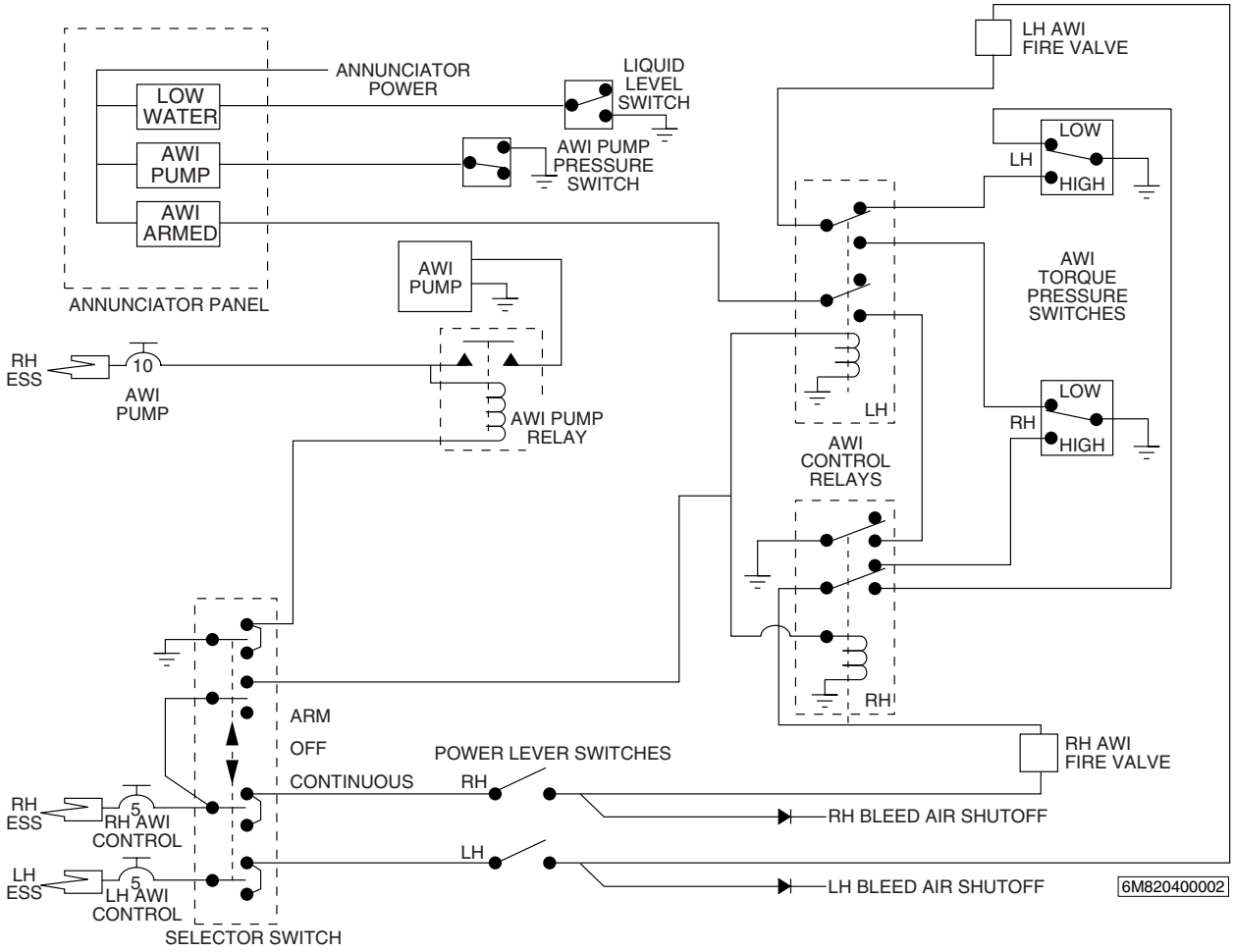
(1) Placing the AWI selector switch in the CONT position will fire the AWI into both engines when three conditions are met:

(a) Selector switch position – To function in the continuous mode, the selector switch must be in the CONT position.

(b) Power Lever position – To apply electrical power to the system, the power levers must be forward of the flight idle gate.

(c) Engine Power Output – Because the AWI control relays are in the relaxed state, each engine's torque pressure switch is connected to the same engine's fire valve. The engine torque pressure must be high (engine operating) to close the torque switch and complete the electrical path necessary to open the fire valve and allow AWI injection. This is a safety feature during a takeoff with continuous AWI. Should one engine fail, the AWI flow to the failed engine will stop.

NOTE: Fluid injection continues until the tank is depleted or the selector switch is moved to OFF position.



AWI Electrical Schematic
Figure 2

F. AWI Functional Test

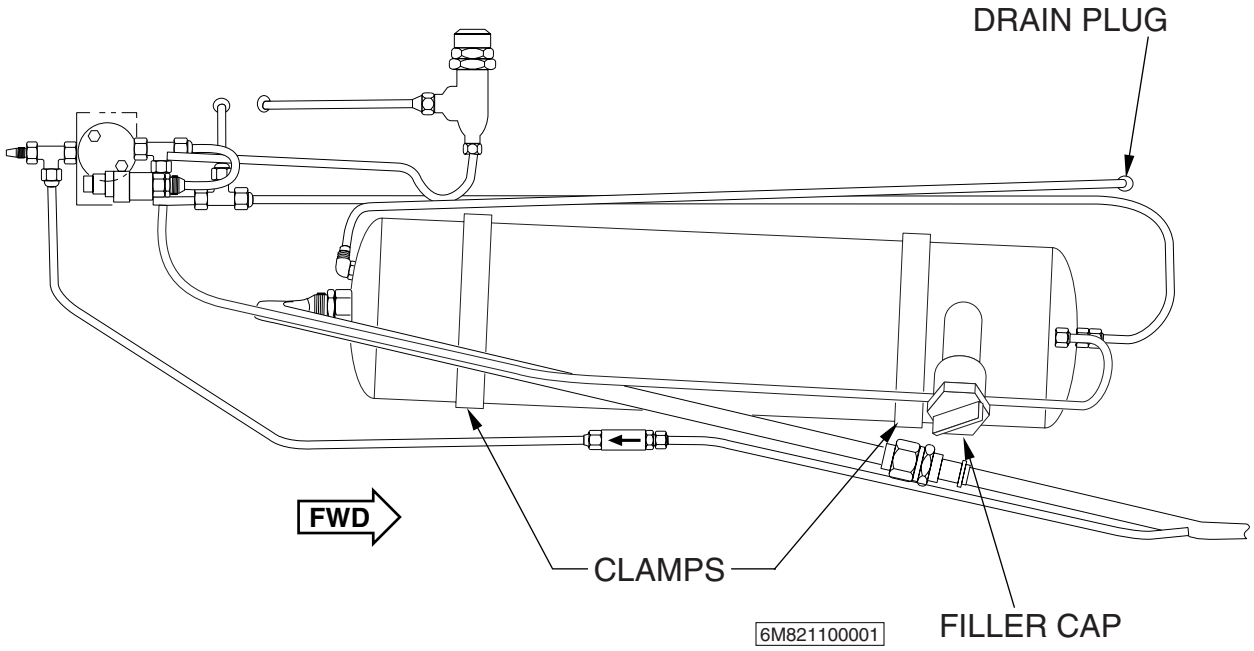
(1) The AWI emergency system is dynamically tested before each takeoff during which the system is to be armed. Both power levers are advanced to above flight idle and the selector switch is moved to ARM. Each power lever is in turn retarded and an increase in torque is noted on the opposite engine. All components of the system have been tested and the system may now be armed for takeoff.

(2) If it is not necessary to test the continuous portion of the system. During a continuous AWI takeoff, the actual use of the system tests all components.

AWI STORAGE – TWO GALLON – DESCRIPTION AND OPERATION

1. General

A. A two gallon stainless steel tank is located under the right hand aft wing-to-fuselage fairing. Two strap type clamps secure the tank to the aircraft structure. A liquid level sensor is installed within the tank and is secured to a boss welded to the tank. A drain line and plug are used to empty the tank.



AWI Storage
(Two Gallon Tank)
Figure 1

AWI STORAGE – TWO GALLON – MAINTENANCE PRACTICES

1. Removal/Installation – Two Gallon AWI Tank

A. Removal – AWI Tank

(1) Remove right hand aft wing-to-fuselage fairing.

(2) Drain AWI tank.

(3) Disconnect all lines attached to the tank.

(4) Loosen clamps securing the tank and rotate tank slightly toward the wing.

(5) Disconnect the electrical wire to the float switch or remove switch from tank.

CAUTION: DO NOT ALLOW THE REMOVED SWITCH TO HANG UNSUPPORTED BY ITS WIRING.

(6) Remove the tank.

B. Installation – AWI Tank

(1) Install the tank loosely into its mounting clamps.

(2) Install the float switch or connect the electrical wire to the switch.

(3) Rotate the tank as necessary to connect the lines.

(4) Torque the tank clamps to 35 inch-pounds (3.95 N·m).

(5) Torque all lines connected to the tank.

(6) Service the tank with AWI fluid and apply air pressure, either by running the engines or connecting to the air pressure regulator.

(7) When the tank installation is free of leaks, install the wing-to-fuselage fairing.

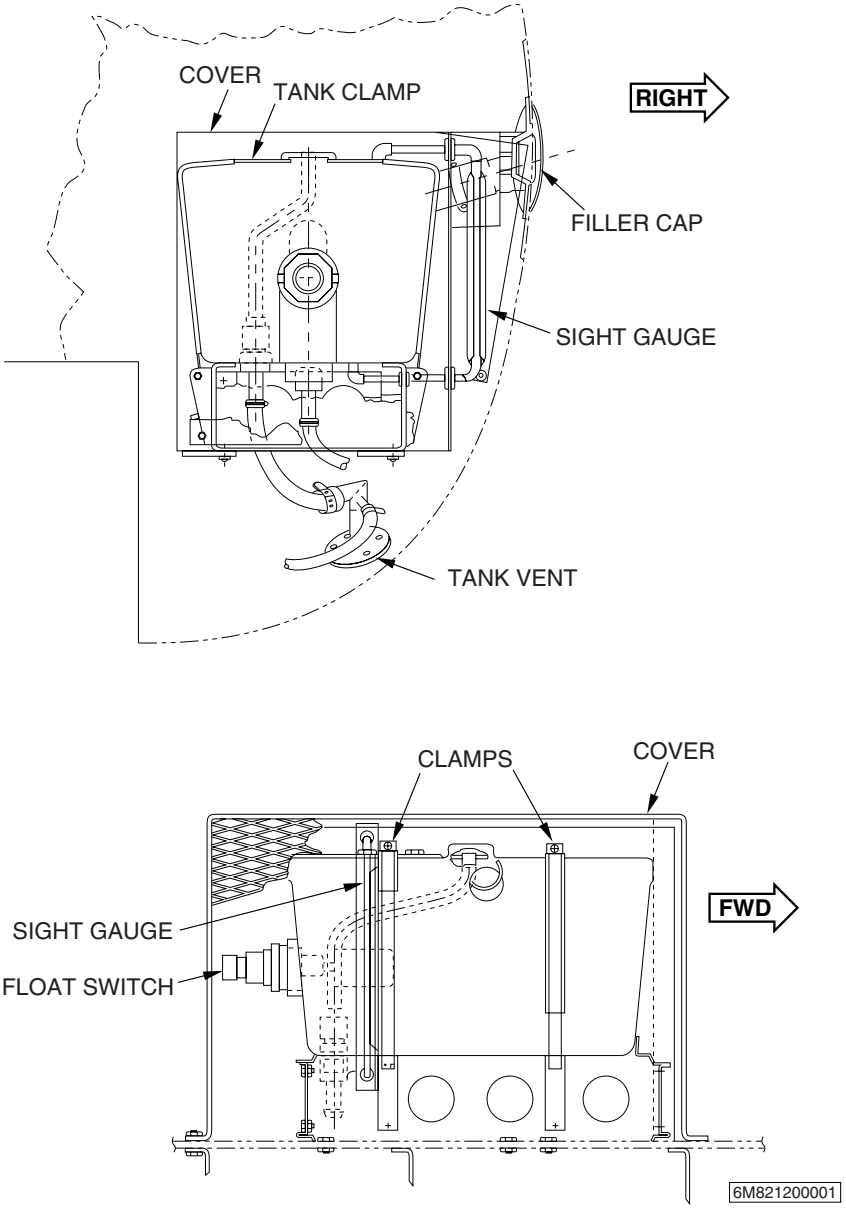
2. Inspection/Check – Two Gallon AWI Tank

A. Inspect tank for security and evidence of leakage.

AWI STORAGE – FOUR GALLON – DESCRIPTION AND OPERATION

1. General

A. A four gallon fiberglass tank is located in the RH side of the nose baggage compartment, just forward of the baggage door. The tank is located under a sealed cover which prevents the escape of AWI fumes into the compartment. The tank is vented through a line running through the center of the tank and connecting to an overboard dump on the lower RH side of the nose. Two strap type clamps secure the tank to the aircraft structure. A liquid level float switch and a tank mounted sight gauge are used to readily determine the quantity of fluid in the tank. The pump supply line fitting is installed in a boss in the bottom of the tank.



AWI Storage
(Four Gallon Tank)
Figure 1

AWI STORAGE – FOUR GALLON – MAINTENANCE PRACTICES

1. Removal/Installation – AWI Storage – Four Gallon

A. Removal – Four Gallon AWI Tank

(1) Remove guard screen around the tank.

(2) Remove tape covering the shroud zipper and unzip the shroud.

(3) Remove clamps securing the filler hose to the tank.

(4) Drain AWI tank. An acceptable method is to open the main system drain on the RH side of the nose, and operate the AWI pump. When the system has been drained, pull the AWI pump circuit breaker.

(5) Disconnect wiring to the AWI quantity switch.

(6) Disconnect AWI sight gauge plumbing.

(7) Loosen strap clamps securing the AWI tank.

(8) Lift and turn the tank as necessary to remove the vent and supply line connections from the bottom of the tank.

(9) Remove tank from the aircraft.

B. Installation – Four Gallon AWI Tank

(1) Place the tank on its mounting bed in the nose compartment.

(2) Lift tank as necessary and connect the supply and vent lines to the bottom of the tank.

(3) Slide the filler connecting hose over the tank filler opening and secure with hose clamps. Torque clamps to 15 inch-pounds (1.69 N·m).

(4) Install tank hold down clamps and torque to 15 inch-pounds (1.69 N·m).

(5) Connect sight gauge plumbing.

(6) Connect wiring to the float switch.

(7) Service tank and check for leaks.

(8) Close shroud cover and cover the shroud zipper with duct tape (Permacel P-12 or equivalent).

(9) Install the guard screen around the tank.

(10) Reset AWI pump circuit breaker and functionally check the system

2. Inspection/Check – AWI Storage – Four Gallon

A. Inspect tank for security and evidence of leakage.

3. Approved Repairs – AWI Storage – Four Gallon

A. AWI Tank Repair

(1) Tools and material Required:

(a) Fiberglass, 1.5 oz Mar P.P.G. Industries, Inc., Rt. 4, Shelby, NC.

(b) Resin, Co Rezyn No 393 (no wax) Polyester Commercial Resins Division (Interplastics Corp.), 205 N.E. Broadway, Minneapolis, MN.

(c) Catalyst, 60% MEKP NORAC, Inc., 405 S. Motor Avenue, Azusa, CA 91702.

(d) Wooden spatula, mixing containers and miscellaneous hand tools.

B. Repair Procedure – AWI Tank

(1) Mix material IAW manufacturer's instructions. Observe manufacturer's listed working life of the materials.

(2) Thoroughly clean area to be repaired using methyl ethyl ketone (MEK). Remove grease and paint completely.

(3) Repair the tank using two layers of the 1.5 oz. cloth that are large enough to cover an area three times the size of the damaged area. Thoroughly saturate the cloth with resin. A heat lamp may be used to speed up the drying process.

(4) Seal tank fittings and fill the tank with water. Pressure test the tank to an internal pressure of 2 psig for 30 minutes. No leakage is allowed.

C. Assembly – AWI Bare Tank

(1) Remove access door from the top of the tank.

(2) Install vent line fitting in the bottom of the tank. Torque fitting just enough to seat the O-ring (35 to 70 inch-pounds [3.95 to 7.91 N·m]).

(3) Install and tighten vent line inside the tank.

(4) Install float switch bushing into the tank and tighten. The bushing may be sealed using teflon tape.

(5) Install float switch into the bushing, making sure the switch is oriented properly (arrow up). The switch may also be sealed with teflon tape.

(6) Working through the access hole, screw the float onto the float switch. Check switch for proper up and down operation.

(7) Install supply line fitting into the bottom of the tank. Tighten finger tight plus 1/4 to 1/2 turn. The O-ring should seat with no gap showing between the fitting and the tank.

(8) Install access cover and gasket. Torque the cover bolts to 15 to 20 inch-pounds (1.69 to 2.26 N·m).

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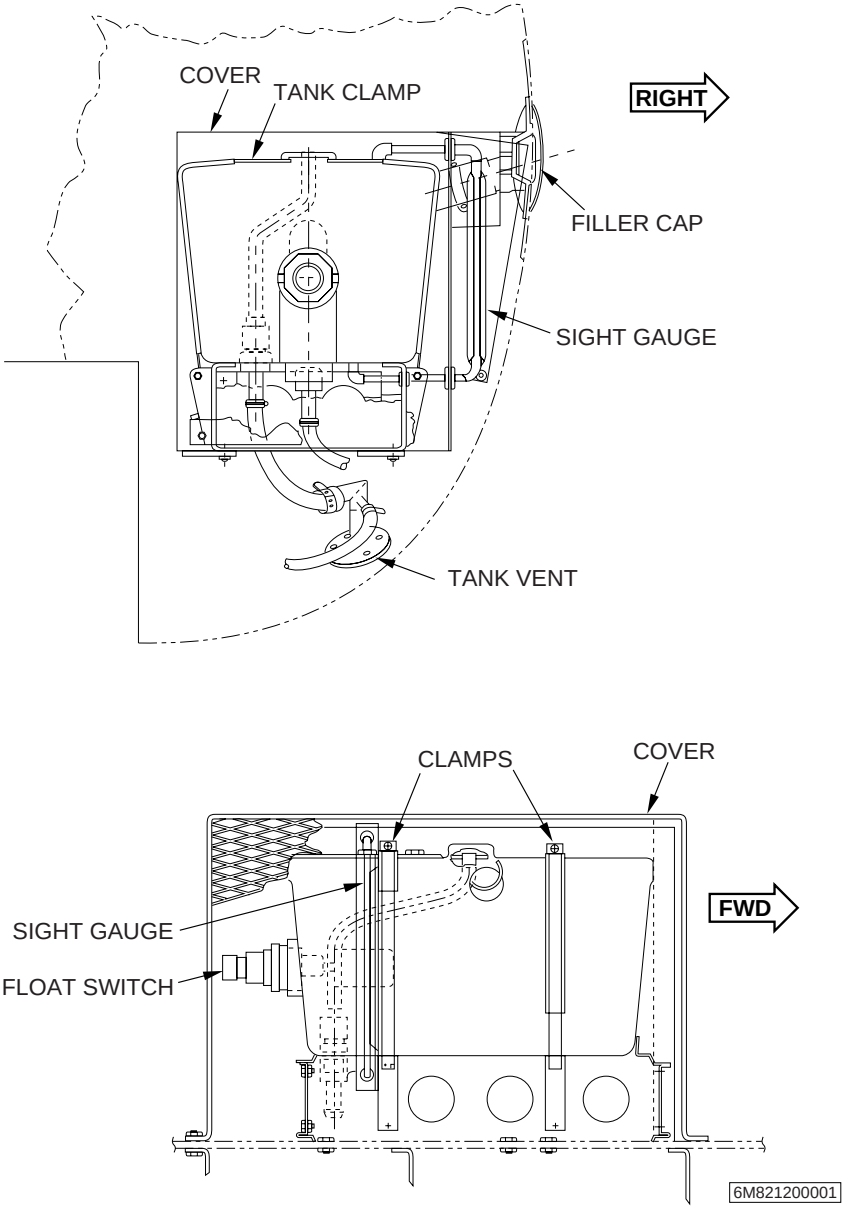
AWI STORAGE – FOUR GALLON – DESCRIPTION AND OPERATION

1. General

A. A four gallon fiberglass tank is located in the RH side of the nose baggage compartment, just forward of the baggage door. The tank is located under a sealed cover which prevents the escape of AWI fumes into the compartment. The tank is vented through a line running through the center of the tank and connecting to an overboard dump on the lower RH side of the nose. Two strap type clamps secure the tank to the aircraft structure. A liquid level float switch and a tank mounted sight gauge are used to readily determine the quantity of fluid in the tank. The pump supply line fitting is installed in a boss in the bottom of the tank.



AWI Storage
(Four Gallon Tank)
Figure 1

AWI STORAGE – FOUR GALLON – MAINTENANCE PRACTICES

1. Removal/Installation – AWI Storage – Four Gallon

A. Removal – Four Gallon AWI Tank

(1) Remove guard screen around the tank.

(2) Remove tape covering the shroud zipper and unzip the shroud.

(3) Remove clamps securing the filler hose to the tank.

(4) Drain AWI tank. An acceptable method is to open the main system drain on the RH side of the nose, and operate the AWI pump. When the system has been drained, pull the AWI pump circuit breaker.

(5) Disconnect wiring to the AWI quantity switch.

(6) Disconnect AWI sight gauge plumbing.

(7) Loosen strap clamps securing the AWI tank.

(8) Lift and turn the tank as necessary to remove the vent and supply line connections from the bottom of the tank.

(9) Remove tank from the aircraft.

B. Installation – Four Gallon AWI Tank

(1) Place the tank on its mounting bed in the nose compartment.

(2) Lift tank as necessary and connect the supply and vent lines to the bottom of the tank.

(3) Slide the filler connecting hose over the tank filler opening and secure with hose clamps. Torque clamps to 15 inch-pounds (1.69 N·m).

(4) Install tank hold down clamps and torque to 15 inch-pounds (1.69 N·m).

(5) Connect sight gauge plumbing.

(6) Connect wiring to the float switch.

(7) Service tank and check for leaks.

(8) Close shroud cover and cover the shroud zipper with duct tape (Permacel P-12 or equivalent).

(9) Install the guard screen around the tank.

(10) Reset AWI pump circuit breaker and functionally check the system

2. Inspection/Check – AWI Storage – Four Gallon

A. Inspect tank for security and evidence of leakage.

3. Approved Repairs – AWI Storage – Four Gallon

A. AWI Tank Repair

(1) Tools and material Required:

(a) Fiberglass, 1.5 oz Mar P.P.G. Industries, Inc., Rt. 4, Shelby, NC.

(b) Resin, Co Rezyn No 393 (no wax) Polyester Commercial Resins Division (Interplastics Corp.), 205 N.E. Broadway, Minneapolis, MN.

(c) Catalyst, 60% MEKP NORAC, Inc., 405 S. Motor Avenue, Azusa, CA 91702.

(d) Wooden spatula, mixing containers and miscellaneous hand tools.

B. Repair Procedure – AWI Tank

(1) Mix material IAW manufacturer's instructions. Observe manufacturer's listed working life of the materials.

(2) Thoroughly clean area to be repaired using methyl propyl ketone (MPK). Remove grease and paint completely.

(3) Repair the tank using two layers of the 1.5 oz. cloth that are large enough to cover an area three times the size of the damaged area. Thoroughly saturate the cloth with resin. A heat lamp may be used to speed up the drying process.

(4) Seal tank fittings and fill the tank with water. Pressure test the tank to an internal pressure of 2 psig for 30 minutes. No leakage is allowed.

C. Assembly – AWI Bare Tank

(1) Remove access door from the top of the tank.

(2) Install vent line fitting in the bottom of the tank. Torque fitting just enough to seat the O-ring (35 to 70 inch-pounds [3.95 to 7.91 N·m]).

(3) Install and tighten vent line inside the tank.

(4) Install float switch bushing into the tank and tighten. The bushing may be sealed using teflon tape.

(5) Install float switch into the bushing, making sure the switch is oriented properly (arrow up). The switch may also be sealed with teflon tape.

(6) Working through the access hole, screw the float onto the float switch. Check switch for proper up and down operation.

(7) Install supply line fitting into the bottom of the tank. Tighten finger tight plus 1/4 to 1/2 turn. The O-ring should seat with no gap showing between the fitting and the tank.

(8) Install access cover and gasket. Torque the cover bolts to 15 to 20 inch-pounds (1.69 to 2.26 N·m).

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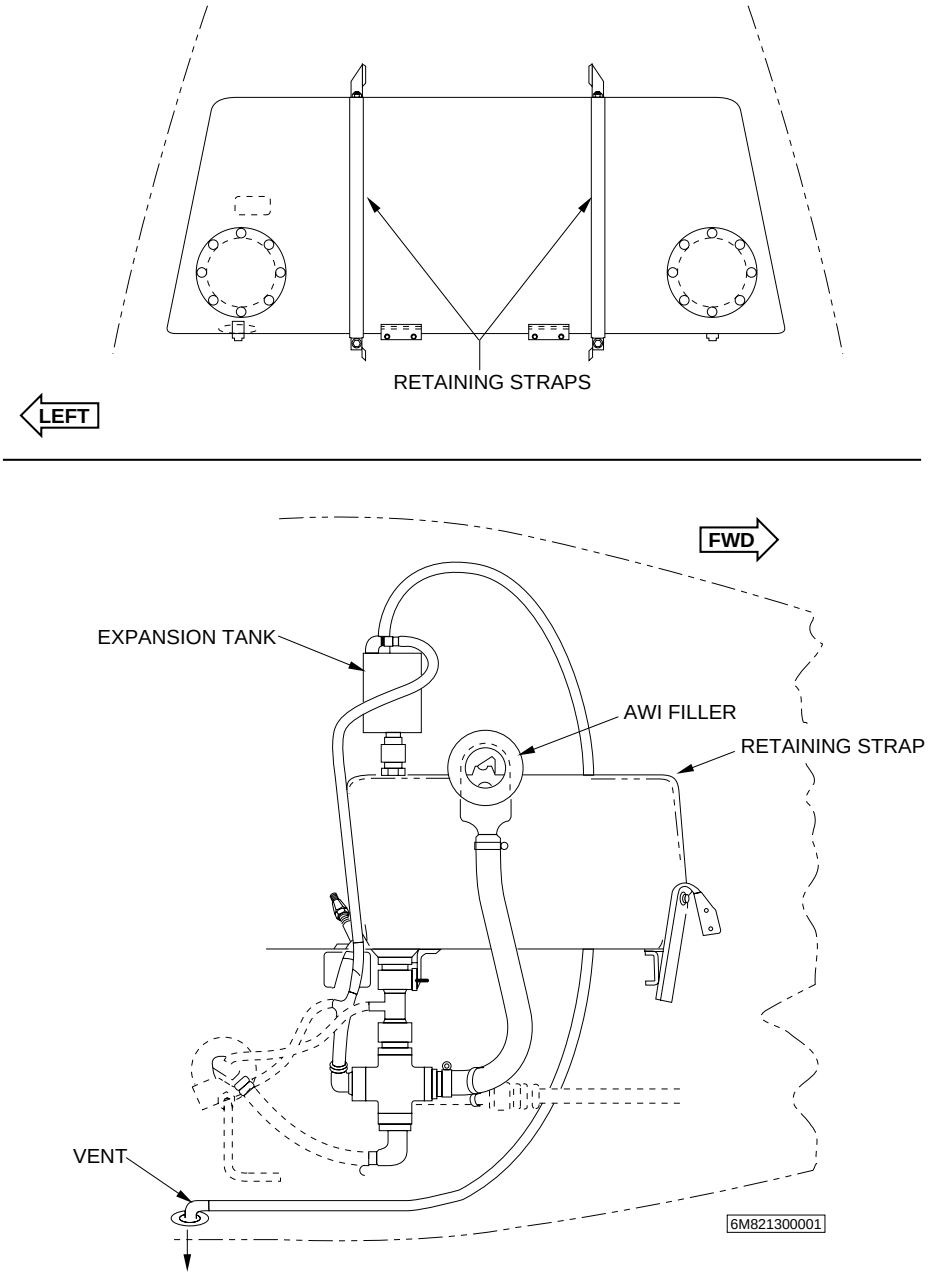
AWI STORAGE – SIXTEEN GALLON – DESCRIPTION AND OPERATION

1. General

A. A sixteen gallon fiberglass tank is installed in the forward nose baggage compartment. The tank is located under a sealed cover which prevents the escape of AWI fumes into the compartment. A liquid level transmitter and a tank mounted sight gauge are used to readily determine the quantity of fluid in the tank. A vent system and expansion tank along with internal baffling in the tank combine to make sure the fluid remains contained during all flight attitudes and conditions.



AWI Storage
(Sixteen Gallon Tank)
Figure 1

AWI STORAGE – SIXTEEN GALLON – MAINTENANCE PRACTICES

1. Removal/Installation – AWI Storage – Sixteen Gallon

A. Removal – Sixteen Gallon AWI Tank

(1) Remove guard screen around the tank.

(2) Remove tape covering the shroud zipper and unzip the shroud.

(3) Drain AWI tank. An acceptable method is to open the main system drain on the RH side, and operate the No. 1 AWI pump. When the tank has been drained, pull the LH and RH AWI pump circuit breakers.

(4) Remove the expansion tank and plumbing from the RH top side of tank.

(5) Loosen strap clamps securing the tank.

(6) Disconnect wiring from the quantity level sensor.

(7) Tilt the tank slightly forward and disconnect the pump supply lines from the LH and RH sides of the tank.

(8) Remove the tank from the aircraft.

B. Installation – Sixteen Gallon AWI Tank

(1) Place the tank loosely in position on its mounting bed in the nose baggage compartment.

(2) Tilt the tank forward and connect the pump supply lines.

(3) Place the tank in position and connect the strap retaining clamps. Torque the clamps 15 inch-pounds (1.69 N·m).

(4) Install the expansion tank and connect the vent tubing.

(5) Connect wiring to the AWI quantity sensor. Refer to section 82-43-00 for instructions on calibrating the sensor.

(6) Service AWI tank and check for leaks.

(7) Close tank shroud and seal the shroud zipper with tape (Permacel P-12 or equivalent).

(8) Reset AWI pump circuit breakers and functionally check the system.

2. Inspection/Check – AWI Storage – Sixteen Gallon

A. Inspect tank for security and evidence of leakage.

3. Approved Repair – AWI Storage – Sixteen Gallon

A. AWI Tank Repair

(1) Tools and Material Required.

(a) Fiberglass, 1.5 oz Mat P.P.G. Industries, Inc., Rt. 4, Shelby, SC.

(b) Resin, Co Rezyn No 393 (no wax) Polyester Commercial Resins Division (Interplastics Corp.), 205 N.E. Broadway, Minneapolis, MN.

(c) Catalyst, 60% MEKP NORAC, Norak, Inc., 405 S. Motor Avenue, Azusa, CA 91702.

(d) Wooden spatula, mixing containers and miscellaneous hand tools.

B. Repair Procedure – AWI Tank

(1) Mix material IAW manufacturer's instructions. Observe manufacturer's listed working life of the materials.

(2) Thoroughly clean area to be repaired using methyl propyl ketone (MPK). Remove grease and paint completely.

(3) Repair the tank using two layers of the 1.5 oz. cloth that are large enough to cover an area three times the size of the damaged area. Thoroughly saturate the cloth with resin. A heat lamp may be used to speed up the drying process.

(4) Seal tank fittings and fill the tank with water. Pressure test tank to an internal pressure of 2 psig for 30 minutes. No leakage is allowed.

C. Assembly – AWI Bare Tank

(1) Remove LH tank access cover.

(2) Install liquid level transmitter (reference section 82-43-00).

(3) Install supply line fittings on the LH and RH side of the bottom of the tank. Install vent line fitting on the RH top side of the tank. All fittings should be tightened finger tight plus 1/4 to 1/2 turn. The O-ring should seat with no gap showing between the fitting and the tank.

(4) Install access cover and gasket. Torque fasteners on both access covers to 15 to 20 inch-pounds (1.69 to 2.26 N·m).

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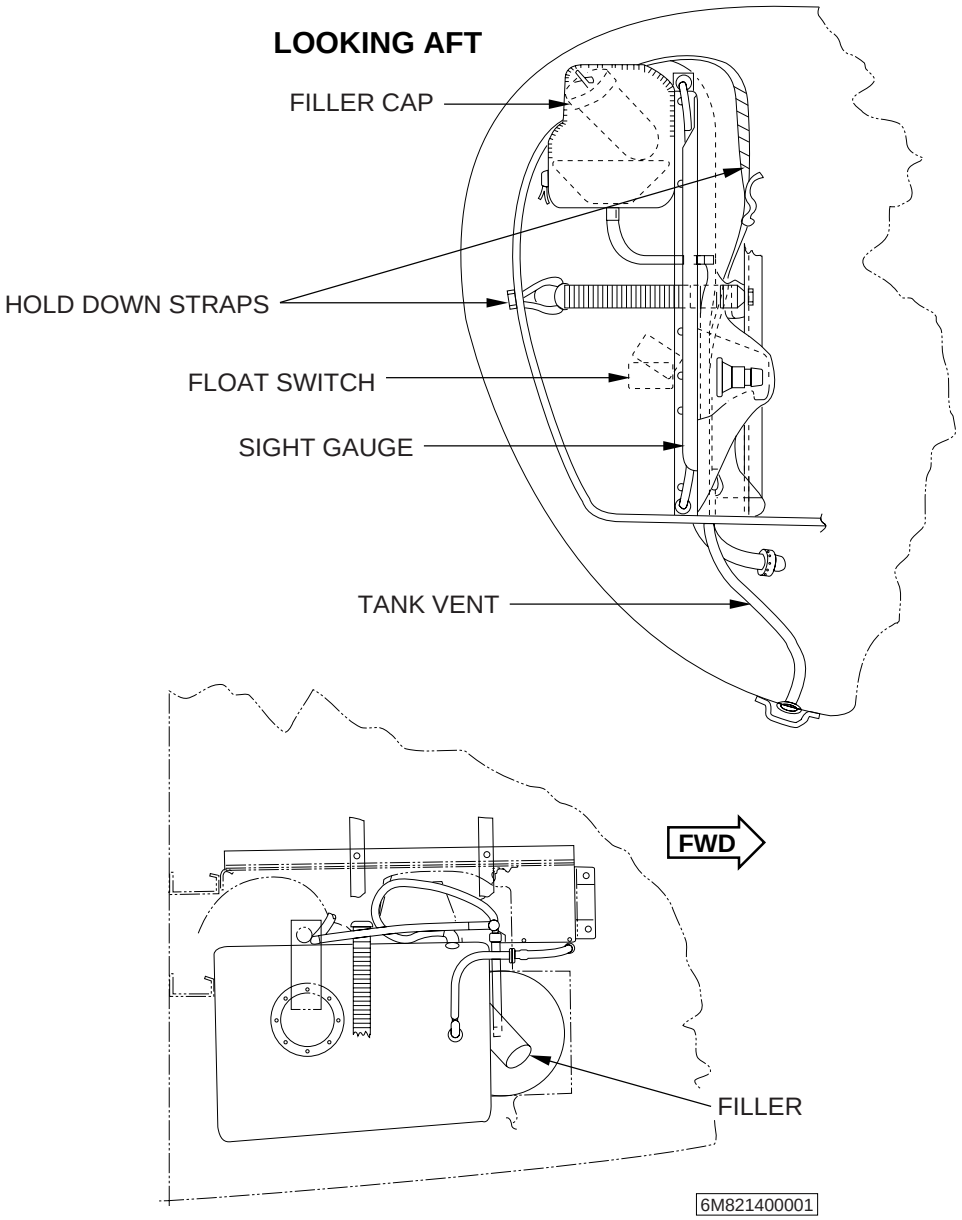
AWI STORAGE – FOURTEEN GALLON – DESCRIPTION AND OPERATION

1. General

A. A fourteen gallon fiberglass tank is installed in the RH aft side of the nose baggage compartment. The tank is located under a sealed cover which prevents the escape of AWI fumes into the compartment. A liquid level float switch and a tank mounted sight gauge are used to allow the quantity of fluid in the tank to be readily determined. The tank is vented from the top by a line running along side the tank to a overboard drain on the RH side of the nose section. A filler with a scupper drain is located on the forward side of the tank is readily accessible through the RH baggage door.



AWI Storage
(Fourteen Gallon Tank)
Figure 1

AWI STORAGE – FOURTEEN GALLON – MAINTENANCE PRACTICES

1. Removal/Installation – AWI Storage – Fourteen Gallon

A. Removal – Fourteen Gallon AWI Tank

(1) Remove guard screen around the tank.

(2) Loosen hold down straps securing the tank.

(3) Disconnect scupper drain and sight gauge hoses.

(4) Disassemble guard screen and sight gauge support structure as necessary to allow removal of the tank.

(5) Remove tape securing shroud zipper and unzip the shroud.

(6) Drain AWI tank. An acceptable method is to open the main system drain on the RH side and operate the AWI pump. When the tank has been drained, pull the AWI pump circuit breaker.

(7) Tilt the tank slightly inboard and remove the supply line connection from the bottom of the tank.

(8) Remove vent line connection from the top of the tank.

(9) Disconnect float switch wiring.

(10) Remove tank from the aircraft.

B. Installation – Fourteen Gallon AWI Tank

(1) Place tank on its mounting bed in the nose compartment.

(2) Tilt tank slightly inboard and connect the supply line to the bottom of the tank.

(3) Connect vent and scupper lines to the tank.

(4) Connect sight gauge lines to the tank.

(5) Reconnect float switch wiring.

(6) Service AWI tank and check for leaks.

(7) Close tank shroud and seal the shroud zipper with tape (Permacel P-12 or equivalent).

(8) Reassemble sight gauge and guard screen support structure.

(9) Connect and tighten tank hold down straps.

(10) Reset AWI pump circuit breaker and functionally check the system.

2. Inspection/Check – AWI Storage – Fourteen Gallon

A. Inspect tank for security and evidence of leakage.

3. Approved Repair – AWI Storage – Fourteen Gallon

A. AWI Tank Repair

(1) Tools and Material Required.

(a) Fiberglass, 1.5 oz Mat P.P.G. Industries, Inc., Rt. 4, Shelby, SC.

(b) Resin, Co Rezyn No 393 (no wax) Polyester Commercial Resins Division (Interplastics Corp.), 205 N.E. Broadway, Minneapolis, MN.

(c) Catalyst, 60% MEKP NORAC, Norak, Inc., 405 S. Motor Avenue, Azusa, CA 91702.

(d) Wooden spatula, mixing containers and miscellaneous hand tools.

B. Repair Procedure – AWI Tank

(1) Mix material IAW manufacturer's instructions. Observe manufacturer's listed working life of the materials.

(2) Thoroughly clean area to be repaired using methyl propyl ketone (MPK). Remove grease and paint completely.

(3) Repair the tank using two layers of the 1.5 oz. cloth that are large enough to cover an area three times the size of the damaged area. Thoroughly saturate the cloth with resin. A heat lamp may be used to speed up the drying process.

(4) Seal the tank fittings and fill the tank with water. Pressure test the tank to an internal pressure of 2 psig for 30 minutes. No leakage is allowed.

C. Assembly – AWI Bare Tank

(1) Remove tank access cover.

(2) Install float switch bushing into the tank and tighten. The bushing may be sealed using teflon tape.

(3) Install float switch into the bushing, making sure the switch is oriented properly (arrow up). The switch threads may be sealed with teflon tape.

(4) Working through the access hole, screw the float onto the switch. Check the switch for proper up and down operation.

(5) Install supply line fitting into the bottom of the tank. Tighten the fitting finger tight plus 1/4 to 1/2 turn. The O-ring should seat with no gap showing between the fitting and the tank.

(6) Install access cover and gasket. Torque cover bolts to 15 to 20 inch-pounds (1.69 to 2.26 N·m).

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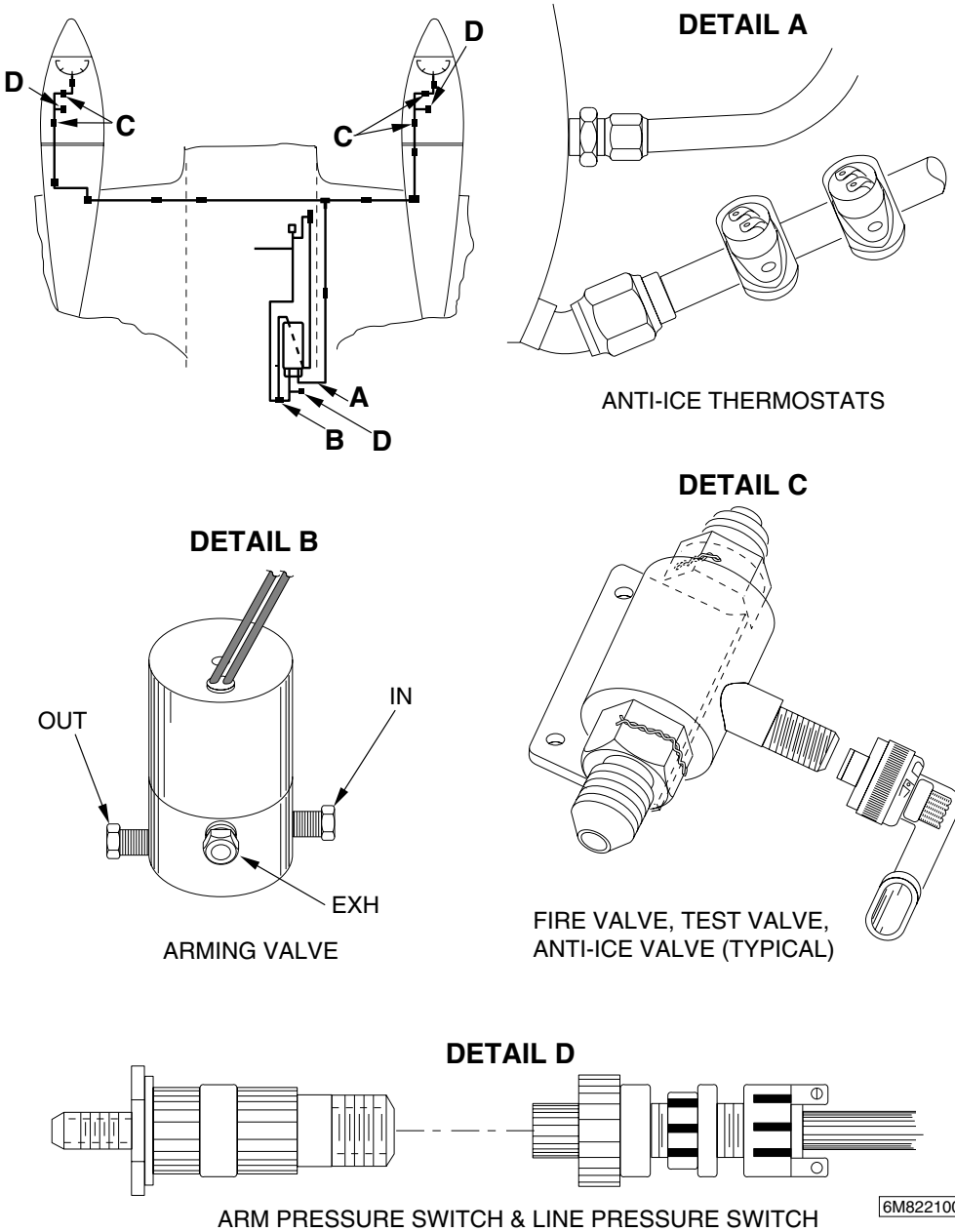
AWI DISTRIBUTION – DESCRIPTION AND OPERATION

1. General

A. The corrosive properties of the alcohol water mixture require that all components of the AWI system be manufactured of materials that do not readily corrode. The part number of some of the valves and fittings used in the system contain prefixes and suffixes indicating that special materials have been used in the part. Exact part number replacements must be used in order to maintain the corrosion resistive properties of the AWI system. Lines, tubing, and connections may be purchased or locally manufactured.



AWI Distribution
Figure 1

B. Hard Tubing and Lines

(1) Replace hard tubing and lines with parts manufactured of the same material (if the line is aluminum, replace it with aluminum) IAW the following specifications.

Aluminum Flared tubing and lines	2024-T3 QQ-A-250/5	Diameter as required
Welded tube assemblies	6061-T6 QQ-A-250/11	Diameter and wall thickness to match original tube assy.
Stainless Steel Flared tubing and lines	304SST MIL T6845	Diameter and wall thickness as required.
Welded tube assemblies	321SST MIL 6721	Diameter and wall thickness as required.

Hard Tubing and Lines Specifications
Table 1

C. Flexible Tubing and Lines

(1) Clear flexible tubing.

'TYGON'
Norton Plastic and Synthetics Division
Specification B-44-3
Diameter and wall thickness to match original part.

(2) Flexible hoses and lines (AEROQUIP TYPE)
Replace only with like part number.

D. Valves, Switches, and Components

(1) Exact part number replacement must be used during maintenance on the AWI system. All component parts that come in contact with the AWI mixture must be treated to prevent corrosion. Anodize aluminum parts and passivate stainless steel parts.

E. Maintenance, Removal, and Replacement

(1) No special precautions are required during maintenance on the AWI system. Use standard aircraft practices in component removal and replacement. Route flexible lines to avoid low spots that may allow AWI mixture to be trapped. Particular care should be given to capping lines when components are removed as the AWI manifold outlets are small orifices and can easily be plugged by foreign material.

AWI DISTRIBUTION – FAULT ISOLATION

1. Troubleshooting – AWI Distribution

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
System will not arm	No air supply to tank	Check for escaping air when tank cap is removed.
	No electrical signal to arming valve	Check to see that LH and RH bleed air shut off when selector switch is moved to arm.
	Arm pressure switch is defective	Apply air pressure to tank and check switch with ohmmeter.
No air supply to AWI tank	No air to or from air regulator or arming valve not open	Check for escaping air when removing hydraulic filler cap.
Air is present at hydraulic filler, not present in AWI tank	Arming valve not opening	Check wiring according to applicable wiring diagram.
Air is not present in the hydraulic reservoir or in the AWI tank	Failed air regulator	Troubleshoot plumbing and repair as necessary.
LH and RH bleed air do not shut off when AWI switch moved to ARM or TEST	Power lever switches not making contact	Insert bypass plug to bypass power lever switches.
	Defective wiring	Troubleshoot wiring according to applicable wiring diagram.
No LINE light in test	Test valve not closing	Monitor engine ITT and torque during test for increases.
	Insufficient pressure in system	Install test gauge to check pressure and check system for leaks.
No FIRE light when AWI fires	Lock-in relay not closing	Fire AWI according to Airplane Flight Manual and advance failed engine to high power. See that AWI continues to fire.
	Defective wiring between lock-in relay and FIRE light	Troubleshoot wiring according to applicable wiring diagram.

1. Servicing – AWI Distribution

WARNING: METHANOL AND ITS VAPOR IS TOXIC AND FLAMMABLE. HANDLING MUST ALWAYS BE IN A WELL VENTILATED AREA. IN CASE OF ACCIDENTAL CONTACT FLUSH WITH WATER AT ONCE. DO NOT SMOKE, CREATE SPARKS, OR HANDLE NEAR AN OPEN FLAME.

A. General

(1) Each time the system is fired, it should be refilled as soon as practical. If the system is allowed to remain with partially filled lines, the chances of corrosion are higher and the system should be inspected more frequently.

(2) Functionally check the system every 100 hours IAW the Airplane Flight Manual instructions.

B. Routine Filling

(1) Mix AWI fluid IAW instructions in section 82-00-00.

(2) Open access panel in right hand aft wing-to-fuselage fairing and remove AWI filler cap.

CAUTION: THE TANK MAY BE UNDER PRESSURE. REMOVE CAP SLOWLY AND ALLOW PRESSURE TO ESCAPE.

(3) Ensure that filler screen inside tank is in place. Fill tank to overflowing.

(4) Install filler cap.

(5) Install access panel.

C. AWI Inactive

(1) When the system is to be left unused for extended periods of time (one month or longer) Perform the following:

(a) Remove right hand aft fuselage-to-wing fairing and open AWI tank drain plug. When tank has drained, install the plug.

(b) Disconnect left and right hand engine AWI hoses at firewall and drain.

(c) If installed, open drain located in left hand forward wing-to-fuselage fairing and drain.

(d) Service the system using distilled water and move aircraft to a run up area.

(e) Fire AWI into one engine for two minutes IAW with Airplane Flight Manual instructions.

(f) Service the tank with distilled water and fire AWI into the other engine for two minutes.

(g) Arm the system and fire AWI into each engine for one minute.

NOTE: Since the supply of distilled water has been exhausted by the previous firing, force air through the lines and valves to dry the system of liquids.

D. Reactivation of AWI

(1) Perform the following before placing the system back in operation:

(a) Service tank with AWI fluid.

(b) Move aircraft to a run up area and fire AWI into each engine.

(c) Service tank and inspect the system for leaks.

(d) When operation is satisfactory and system is free from leaks, the system may be returned to service.

2. Inspection/Check – AWI Distribution

A. Perform a visual inspection of the system for chafing, corrosion, evidence of leakage, and security.

B. Every 9,000 hours, or each nine years, perform the following:

C. The following additional inspection items should be performed on AT-033 AND TC-222 and earlier unless SB 82-003 has been complied with.

(1) Remove AWI float switch and inspect for corrosion, fungus, and any defect which might impair system performance.

(2) If equipped with a check valve in the supply line (TC-209 and later), remove the valve and inspect for corrosion.

(3) Assemble the system and refill tank.

D. Disassemble the system and perform a visual inspection of the wet components of the system IAW the steps in **2.**, B., (1) through (5).

(1) Remove AWI float switch and inspect for corrosion, fungus, and any defect which might impair system performance.

(2) If equipped with a check valve in the supply line (TC-209 and later), remove the valve and inspect for corrosion.

(3) Remove and inspect lines going to AWI fire and test valves on each engine. Inspect lines and valves.

(4) Inspect lines adjacent to the AWI tank. Remove lines attached to the AWI arming valve and inspect lines and valve.

(5) Assemble and service the system. Perform an operational check of the system IAW the Airplane Flight Manual instructions.

E. AWI Calibration

(1) Two types of AWI calibration procedures have been used on Merlin/Metro aircraft. The first procedure used on A/C TC-201 through TC-207, involved using a fixed orifice fitting upstream of the AWI injection manifold to set the flow rate. Effective with TC-208 and up (including TC-202E), the fixed orifice was replaced with a variable orifice. The fixed orifices are no longer available, and when ordered, the variable orifice will be supplied. The variable orifice assembly consists of an adjustable metering valve, a mounting bracket, a warning tag, a tube assembly, and a new AWI injection manifold. These parts may not be ordered separately and must be calibrated as an assembly.

Aircraft Serial Number ———————————

Date ———————————

Pressure Altitude ———————————

Temperature ———————————

	LEFT	RIGHT
WET TORQUE (foot-pounds)		
DRY TORQUE (foot-pounds)		
TORQUE INCREASE (foot-pounds) (see Table 202 for allowable increase)		
ITT WET (°Celsius)		
ITT DRY (°Celsius)		
FLOW TIME (seconds)		
FLOW QUANTITY (fluid ounces)		

AWI Calibration Data Sheet
Table 201

E. Required Equipment

(1) Stop watch.

(2) Measuring equipment.

(3) AWI test jack bypass plug (supplied as part of the aircraft's loose equipment).

(4) Thermometer.

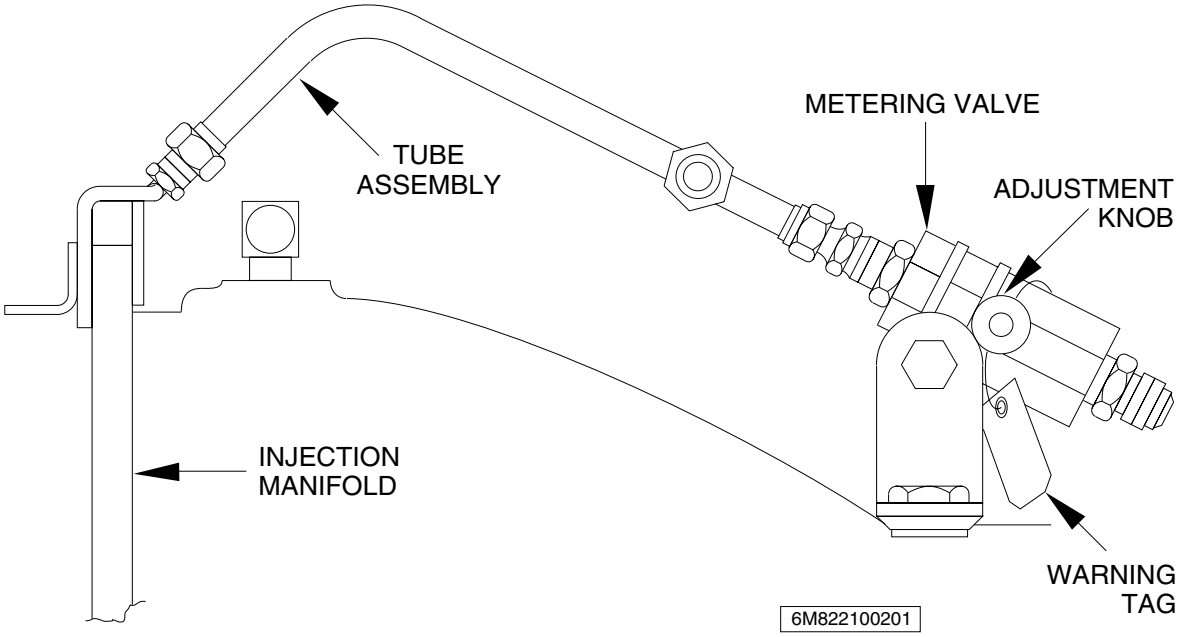
F. Flow Check Procedure

(1) Record the data required for Table 201. The pressure altitude may be obtained by setting the aircraft's altimeter to 29.92 and reading the altitude shown.

(2) Fill the AWI tank with AWI fluid.

(3) Conduct a ground firing of the system IAW the Airplane Flight Manual instructions. Fire each side for 5 to 10 seconds. Do not allow the engine to exceed torque limits.

(4) Refill the AWI tank.



AWI Metering Valve Assembly
Figure 201

CAUTION: ALTHOUGH IT IS NOT NECESSARY TO SHUT DOWN THE ENGINES, EXTREME CARE MUST BE EXERCISED. IF THE RIGHT ENGINE IS PULLED SLIGHTLY INTO REVERSE, THE PROP WASH WILL BE MINIMIZED FOR THE PERSONS FILLING THE TANK.

(5) Fire the AWI into one engine IAW the Airplane Flight Manual instructions with the following exceptions:

CAUTION: AT NO TIME DURING THE FIRING ALLOW THE TORQUE TO EXCEED 2,468 FOOT-POUNDS OR THE ITT TO EXCEED 944°C. IF THESE LIMITS ARE APPROACHED, MOVE THE POWER LEVER TO A LOWER SETTING.

(a) Before firing the AWI, set engine power at 923°C ITT or 2,100 foot-pounds torque, whichever is lower. Record the "dry" torque and ITT in the spaces provided in Table 201. Total firing time for the flow check is 30 seconds per engine. After AWI has fired for approximately 15 seconds, record the "wet" torque and ITT in the spaces provided in Table 201. After the AWI has fired for 30 seconds, return the selector switch OFF, and terminate the firing.

(6) Carefully refill the AWI tank and note the amount of fluid required. Enter this amount in Table 201. The flow should be between 104 and 127 fluid ounces. If the flow is low, open the metering valve on the engine. If the flow is high, close the metering valve. One turn of the valve will cause an approximate change in flow of four fluid ounces.

NOTE: Most aircraft have metering valves installed that have had the valve stem drilled and a roll pin installed to lock the valve adjustment. Valves in this configuration require replacement if adjustment is necessary. See section 82-SERVC.

(7) When the correct flow has been established, repeat steps **2.**, F., (5) and (6) for opposite engine.

(8) After each engine's flow rate has been set, tighten the jam nut on the valve stem. Pass a length of safety wire through the roll pin in the valve knob and the warning tag. Wrap the safety wire around the body of the valve and twist.

ALLOWABLE TORQUE INCREASE (FOOT-POUNDS)

OAT °C	S L To 5999 FT		600 To 1000 FT		1001 To 1500 FT		1501 To 2500 FT		2501 To 3500 FT		3501 To 6000 FT	
	MIN	MAX	MIN	MAX	MIN	MAX	MIN	MAX	MIN	MAX	MIN	MAX
0									253	310	252	302
1									255	312	254	304
2			DO NOT TEST IN THIS AREA						258	314	256	306
3									260	315	258	308
4									263	317	260	310
5							269	321	265	319	263	313
6							270	322	267	321	265	315
7							272	325	269	323	268	318
8							274	326	272	325	271	321
9							275	328	274	327	273	323
10			273	323	273	326	276	330	276	329	276	326
11			274	324	274	328	278	332	279	331	279	329
12			275	325	275	330	280	335	282	334	282	332
13			276	326	276	332	282	337	286	337	286	336
14			278	328	278	334	284	340	289	339	289	339
15	268	326	280	330	280	336	286	342	292	342	292	342
16	272	330	284	334	284	340	290	345	295	345	295	345
17	275	333	286	336	286	343	293	348	298	348	298	348
18	279	335	290	340	290	346	296	352	302	352	302	352
19	283	339	294	344	294	349	299	355	305	355	305	355
20	286	243	297	347	297	352	302	358	208	358	308	358
21	290	346	300	350	300	356	306	361	311	361	311	361
22	294	350	304	354	304	360	310	365	315	365	315	365
23	298	355	308	358	308	364	314	368	318	368	318	368
24	303	358	312	362	312	367	317	372	322	372	322	372
25	307	362	316	366	316	370	320	375	325	375	325	375
26	312	367	320	370	320	375	325	379	328	379	328	378
27	316	371	325	375	325	379	329	383	331	382	331	381
28	321	376	329	379	329	383	333	387	335	386	334	384
29	326	380	333	383	333	387	337	391	338	389	337	387
30	331	385	338	388	338	391	341	395	341	393	340	390
31	337	390	342	392	342	395	345	399	344	397	343	393
32	342	395	347	397	347	400	350	403	348	401	347	397
33	347	400	352	402	352	404	354	407	351	405	350	400
34	353	405	357	407	357	409	358	411	355	409	354	404
35	358	410	362	412	362	414	363	415	358	413	357	407
36	365	416	367	417	367	418	366	420	361	417	359	409
37	372	422	372	422	372	423	370	424	365	421	362	412
38	378	428	377	427	377	428	375	429	368	426		
39	384	434	383	433	383	433	379	433	372	430		
40	390	440	389	439	389	439	384	438	375	434		
41	396	447	395	445	395	445	388	444	379	438		
42	401	454	400	450	400	450	392	449	383	443		
43	408	461	406	456	404	456	396	454				
44	415	468	412	462	410	462	401	460				
45	421	475	419	469	415	468	MAXIMUM OPERATING TEMPERATURE					
46	428	482	425	475	420	475						
47	435	489	430	480								
48	441	496										

Torque Increase Due to Water Injection Table 202

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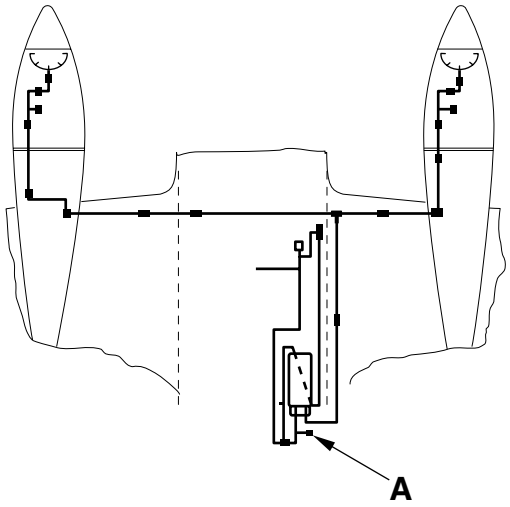
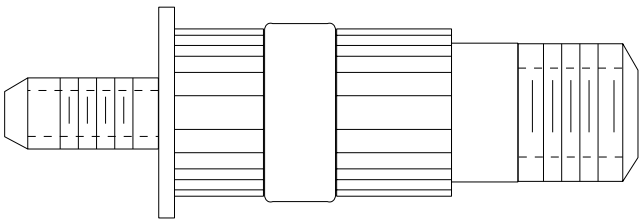
ARM PRESSURE SWITCH – DESCRIPTION AND OPERATION

1. General

A. The arm pressure switch is installed in the air pressure supply line to the AWI tank. When the tank pressure reaches 31.5 ± 0.5 psi the switch closes, completing the path necessary to illuminate the AWI ARM light.



DETAIL A



ARM PRESSURE SWITCH

Arm Pressure Switch
Figure 1

ARM PRESSURE SWITCH – FAULT ISOLATION

1. Troubleshooting – Arm Pressure Switch

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No AWI ARM light	Defective Wiring to switch	Remove the electrical connector and supply a ground to pin B.
	No air pressure or incorrect air pressure in tank	Remove the switch and install a direct reading pressure gauge.
	Defective switch	Replace switch.

ARM PRESSURE SWITCH – MAINTENANCE PRACTICES

1. Removal/Installation – Arm Pressure Switch

A. Removal – Arm Pressure Switch

(1) Gain access to the switch by removing the right hand aft wing-to-fuselage fairing.

(2) Remove safety wire from electrical connector

(3) Remove electrical connector.

(4) Loosen mounting bolt securing the clamp holding the switch in place.

(5) Remove supply line to the pressure switch.

(6) Remove pressure switch.

B. Installation – Arm Pressure Switch

(1) Install the new switch within the clamp and tighten the clamp mounting bolt.

(2) Install the pressure supply line on the switch and tighten.

(3) Install and safety the electrical connector.

(4) Install fairing.

(5) Operational check the AWI system IAW the Airplane Flight manual instructions.

2. Inspection/Check – Arm Pressure Switch

A. Inspect the pressure switch for security of mounting, physical damage, and/or evidence of leakage.

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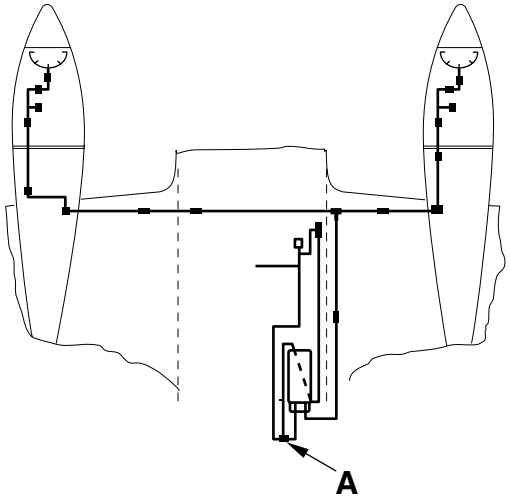
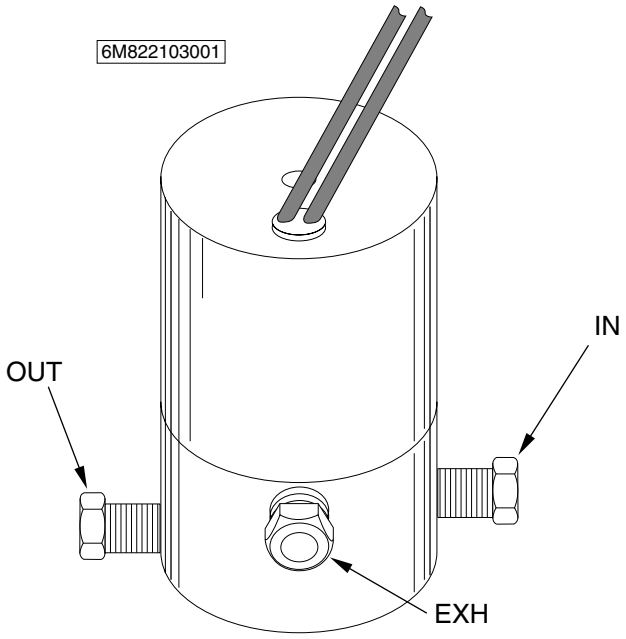
AWI ARMING VALVE – DESCRIPTION AND OPERATION

1. General

A. The AWI arming valve is installed in the air supply line upstream of the AWI tank. The valve is spring loaded normally closed and opens when provided with electrical power from the AWI control circuitry. The valve is located adjacent to the AWI tank under the right hand wing-to-fuselage fairing.



DETAIL A



ARMING VALVE

AWI Arming Valve
Figure 1

AWI ARMING VALVE – FAULT ISOLATION

1. Troubleshooting – Arm Pressure Switch

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No air pressure in AWI tank	No electrical power to valve	Place AWI selector in ARM; both power levers forward until power lever position switches close; 28 volts should be present at yellow wire at valve.
	No supply pressure upstream of the valve	Install a direct reading pressure gauge upstream of the valve.
	Defective valve	Replace valve.

AWI ARMING VALVE – MAINTENANCE PRACTICES

1. **Removal/Installation – AWI Arming Valve**

   A. Removal – AWI Arming Valve

      (1) Gain access to the arming valve by removing the right hand aft wing-to-fuselage fairing.

      (2) Remove lines attached to the valve.

      (3) Disconnect electrical wiring to the valve.

      (4) Remove arming valve from bracket.

   B. Installation – AWI Arming Valve

      (1) Position new valve to bracket and secure.

      (2) Connect lines to the valve.

      (3) Connect electrical wiring to the valve.

      (4) Install fairing.

      (5) Operational check the AWI system IAW the Airplane Flight Manual instructions.

2. **Inspection/Check – AWI Arming Valve**

   A. Inspect the valve for security of mounting, physical damage, and/or evidence of leakage.

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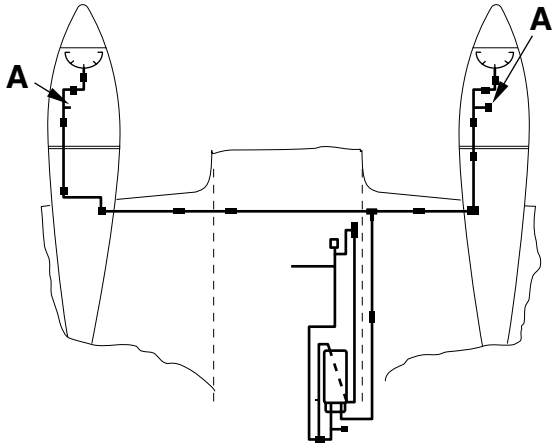
LINE PRESSURE SWITCH – DESCRIPTION AND OPERATION

1. General

A. The line pressure switch is installed between the AWI fire and test valves. During test of the system, the test valve closes to prevent the injection of AWI fluid into the engine when the fire valve opens. Pressure is trapped between the valves and actuates the line pressure switch. Closure of the switch completes the electrical path necessary to illuminate the AWI line light for the respective side and thus inform the pilot that the system is operational. The switch should close by 29 psi increasing and open by 25 psi decreasing.



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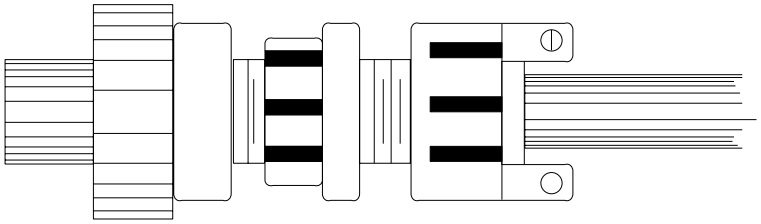
DETAIL A



LINE PRESSURE SWITCH

Line pressure Switch
Figure 1

LINE PRESSURE SWITCH – FAULT ISOLATION

1. Troubleshooting – Line Pressure Switch

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No line light in test	Defective wiring between pressure switch and cockpit	Remove connector at switch and ground pin C of connector. Light should illuminate.
	Insufficient pressure in system	Remove switch and install a direct reading pressure gauge. Test system in accordance with Airplane Flight manual procedures. Gauge should read 29 to 30 psi.
	Defective switch	Replace switch.

LINE PRESSURE SWITCH – MAINTENANCE PRACTICES

1. Removal/Installation – Line Pressure Switch

A. Removal – Line Pressure Switch

(1) Remove the left hand engine side cowl.

(2) Remove safety wire from the electrical connector.

(3) Disconnect the electrical connector.

(4) Remove the supply line connected to the switch.

(5) Loosen bolt on clamp securing the switch.

(6) Remove the switch.

B. Installation – Line Pressure Switch

(1) Position the new switch within the mounting clamp.

(2) Tighten the bolt securing the clamp.

(3) Install the supply line to the switch.

(4) Install and safety wire the electrical connector to the switch.

(5) Install engine cowl.

(6) Operational check the AWI system IAW the Airplane Flight manual instructions.

2. Inspection/Check – Line Pressure Switch

A. Inspect the switch for security of mounting, physical damage, and/or evidence of leakage.

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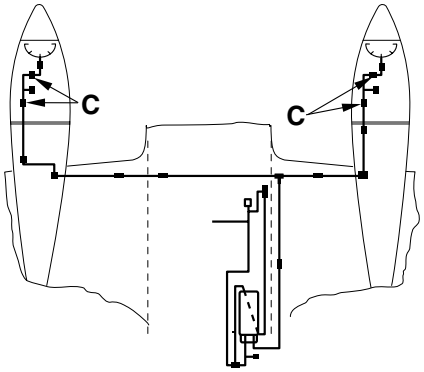
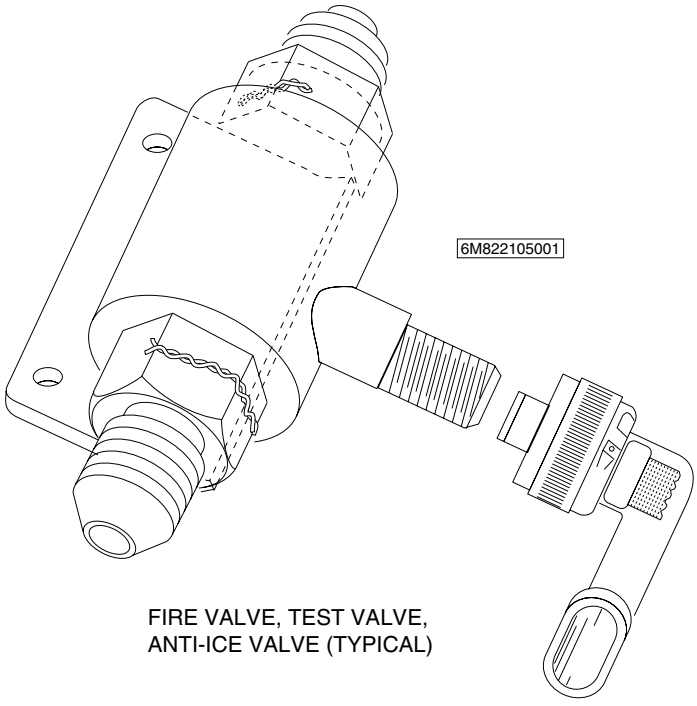
FIRE AND TEST VALVES – DESCRIPTION AND OPERATION

1. General

A. An AWI fire and AWI test valve are installed on each engine to control the flow of AWI fluid. The valves are similar in appearance but opposite in function. The normally closed fire valve is opened by the AWI control circuitry to allow AWI fluid injection. The normally open AWI test valve, installed upstream of the fire valve, closes during AWI system test. Closure of the test valve traps the AWI fluid before the fluid can enter the engine. The trapped fluid pressure actuates the AWI line light pressure switch as part of the system test (reference 82-01-00).



DETAIL C



FIRE VALVE, TEST VALVE,
ANTI-ICE VALVE (TYPICAL)

Fire and Test Valves
Figure 1

FIRE AND TEST VALVES – FAULT ISOLATION

1. Troubleshooting – Fire and Test Valves

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Valve does not function	Defective wiring between cockpit and valve	Remove connector and check for ground at pin B. Check for power at pin A as follows: Fire valve – simulate high torque on opposite engine by removing torque switch connector; selector switch in ARM; power levers high; 28 VDC should be present at pin A. test valve – place selector switch in TEST; power should be present at pin A.
	Defective valve	Replace valve.

FIRE AND TEST VALVES – MAINTENANCE PRACTICES

1. Inspection/Check – Fire and Test valves

A. Inspect the valves for security and evidence of leakage. During inspection outlines in 82-21-00, paragraph B, the inlet and outlet ports of each valve should be checked for corrosion.

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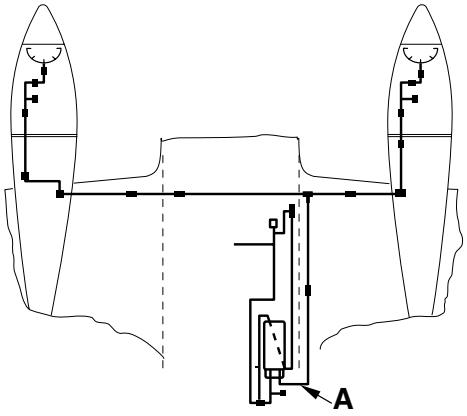
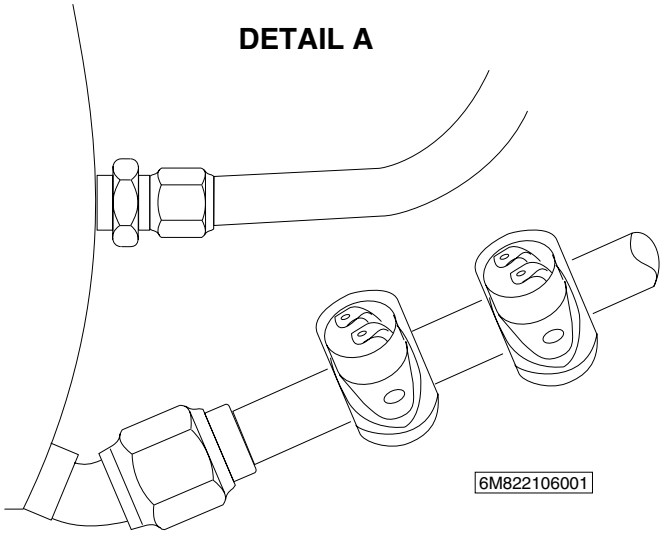
AWI HEAT – DESCRIPTION AND OPERATION

1. General

A. AWI heat prevents the AWI fluid from freezing when the aircraft is operating in ambient temperatures below 5°F (-15°C). Two thermostats are used in the system. One thermostat closes at 10°F (-12°C) and controls the anti-ice valve. The second thermostat closes at 5°F (-15°C) and illuminates the AWI HEAT light at the annunciator. The normally closed anti-ice valve, located in the right hand wing root, is similar in appearance to the engine installed fire and test valves.



DETAIL A



ANTI-ICE THERMOSTATS

Anti-Ice Thermostats
Figure 1

1. Troubleshooting – AWI Heat

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
AWI heat light is illuminated after the engines have been operating for 30 minutes or more	Defective wiring between thermostat and valve	Supply a ground to the -12°C thermostat at wire 7601AC-20 (or wire 7601AD-20). AWI heat valve should open.
	Defective thermostat	Replace thermostat.
	Defective heat valve	Replace valve.

1. Inspection/Check – AWI Heat

A. Inspect the system components for security of mounting and for security of the plumbing and electrical components.

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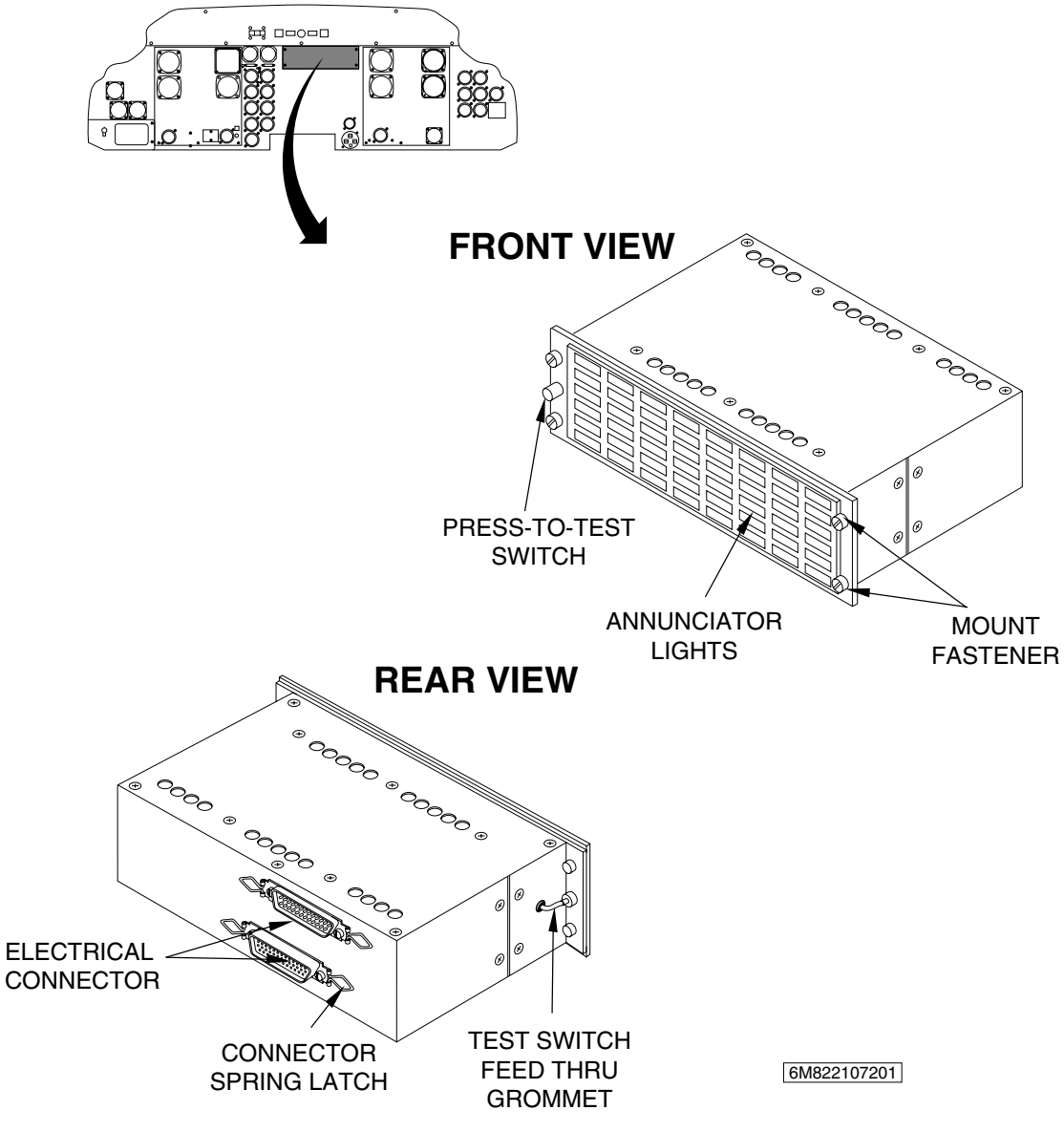
1. Adjustment/Test – Annunciator Panel

A. M7 Aerospace recommends the annunciator be replaced as an assembly. The following test will determine if the problem is with the aircraft wiring or within the annunciator panel:

(1) Remove the two plugs from the back of the annunciator panel. The small plug contains 37 pins; the large plug contains 50 pins.

(2) With the battery switches ON, measure 28 volts at pin 1 and 35 of the small connector.

(3) Turn the battery switches OFF and install the small connector.



Annunciator Panel
Figure 201

(4) Turn the battery switches ON, and push the press-to-test button on the annunciator panel. Make sure the following lights illuminate:

(a) AWI HEAT

(b) AWI ARM

(c) AWI FIRE

(D) L AWI LINE

(E) R AWI LINE

(5) Using a jumper, supply a ground to the following pins of the large receptacle at the back of the annunciator panel:

NOTE: If the problem is within the annunciator, the annunciator assembly should be replaced.

(a) Pin 40 to ground . . . . . AWI HEAT light illuminates.

(b) Pin 41 to ground . . . . . AWI ARM light illuminates.

(c) Pin 43 to ground . . . . . AWI FIRE light illuminates.

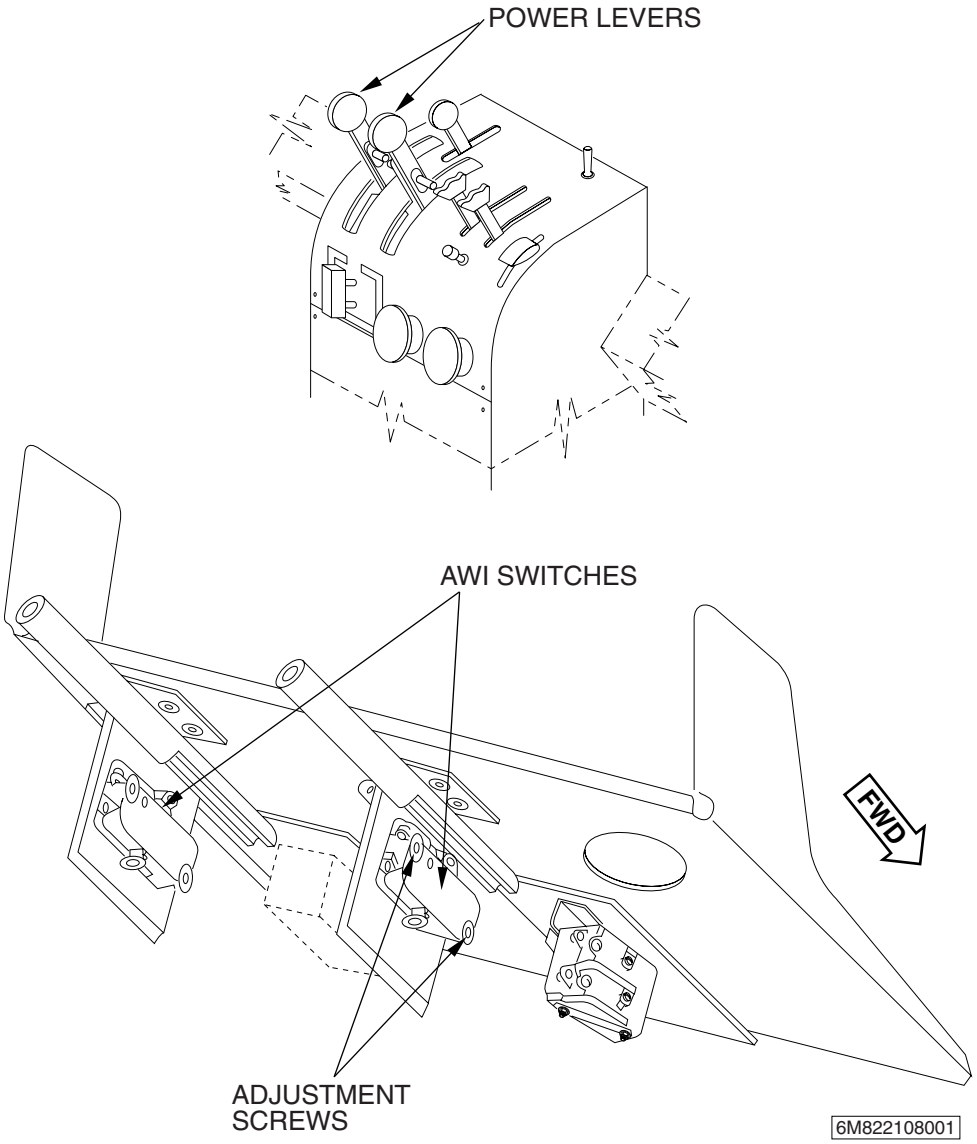
(d) Pin 44 to ground . . . . . .L AWI line illuminates.

(e) Pin 45 to ground . . . . . .R AWI line illuminates.

POWER LEVER POSITION SWITCHES – DESCRIPTION AND OPERATION

1. General

A. Two normally open microswitches are installed in the cockpit pedestal. Each switch is actuated to the closed position by the respective power lever. A slotted mounting plate is installed to allow adjustment of the switches.



ADJUST SWITCHES TO ACTUATE AS POWER LEVERS REACH 50% TRAVEL BETWEEN FLIGHT IDLE AND FULL FORWARD.

Power Lever Position Switch Adjustment Figure 1

POWER LEVER POSITION SWITCHES – FAULT ISOLATION

1. Troubleshooting – Power Lever Position Switches

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
System cannot be armed	One or both switches not closing	Listen for sound of switch closure as respective power lever is moved.
	Defective wiring or defective switch	Insert bypass plug in test jack and attempt to arm system.

POWER LEVER POSITION SWITCHES – MAINTENANCE PRACTICES

1. Adjustment/Test – Power Lever Position Switches

A. Power Lever Position Switch Adjustment

(1) Gain access to the switches either by removing the forward center pedestal access panel or by working from the back of the pedestal.

(2) Adjust the respective switch to actuate when the power lever reaches 50% travel between FLIGHT IDLE and full forward.

2. Inspection/Check – Power Lever Position Switches

A. Inspect the switches for condition and security.

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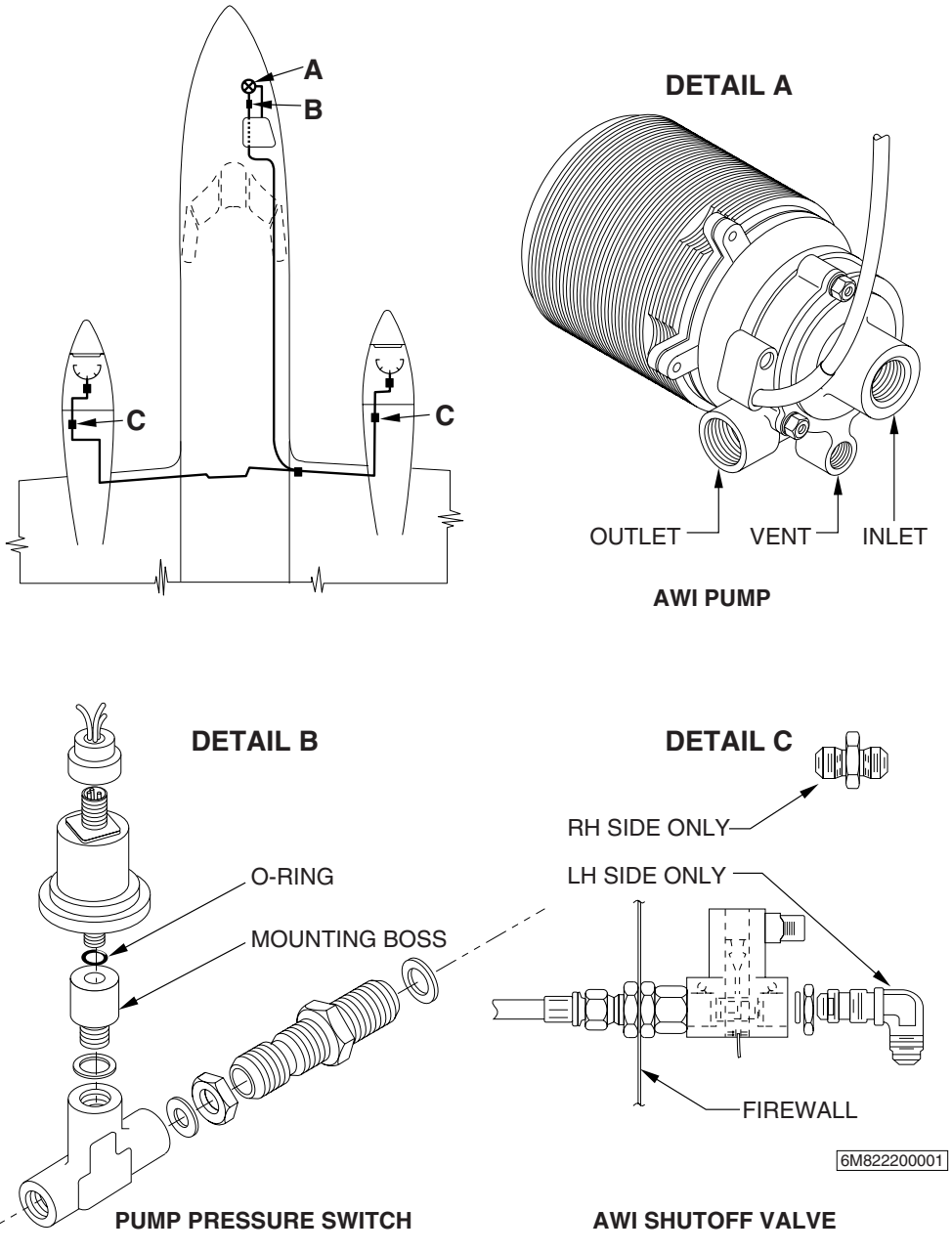
AWI DISTRIBUTION – DESCRIPTION AND OPERATION

1. General

A. The corrosive properties of the alcohol water mixture require that all components of the AWI system be manufactured of materials that do not readily corrode. The part numbers of some of the valves and fittings used in the system contain prefixes and suffixes indicating that special materials have been used in the part. Exact part number replacements must be used in order to maintain the corrosion resistive properties of the AWI system. Lines, tubing, and connections may be purchased or locally manufactured.



AWI Distribution
Figure 1

B. Hard Tubing and Lines

(1) Replace hard tubing and lines with parts manufactured of the same material (if the line is aluminum, replace it with aluminum) IAW the following specifications:

Aluminum Flared tubing and lines	2024-T3 QQ-A-250/5	Diameter as Required
Welded tube assemblies	6061-T6 QQ-A-250/11 original tube assy.	Diameter and wall thickness to match
Stainless Steel Flared tubing and lines	304SST Mil T6845 required.	Diameter and wall thickness as
Welded tube assemblies	321SST Mil 6721 required.	Diameter and wall thickness as

Hard Tubing and Lines Specifications
Table 1

C. Flexible Tubing and Lines

(1) Clear flexible tubing.

'TYGON'
Norton Plastic and Synthetics Division
Specification B-44-3
Diameter and wall thickness to match original part.

(2) Flexible hoses and lines (AEROQUIP TYPE)

Replace only with like part number.

D. Valves, Switches, and Components

(1) Exact part number replacement must be used during maintenance on the AWI system. All component parts that come in contact with the AWI mixture must be treated to prevent corrosion. Aluminum parts will be anodized while stainless steel parts will be passivated.

E. Maintenance, Removal, and Replacement

(1) No special precautions are required during maintenance on the AWI system. Standard aircraft practices should be used in component removal and replacement. Flexible lines should be routed to avoid low spots that may allow AWI mixture to be trapped. Particular care should be given to capping lines when components are removed as the AWI manifold outlets are small orifices and can easily be plugged by foreign material.

1. Troubleshooting – AWI Distribution

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
System will not arm	Low pump output or no pump output	Check for sound of pump operating. Check pump output pressure according to section 82-22-01.
	Defective wiring to pressure switch or defective switch	Remove connector to switch; apply a ground to pin B.
System will not fire either side	Defective selector switch or no power from bus	Selector switch in ARM; power levers forward; check for power at pin A of each engine torque switch connector.
System will not fire one engine	Defective power lever switch	See that moving the suspect power lever forward causes the bleed air on that side to shut off.
	No power to fire valve	Selector switch in ARM; power lever forward; check for power at pin A of torque switch connector.
	Power is present at pin A torque switch connector.	Ground pin A; if valve opens, defective torque or no ground to pin B of torque switch connector. If valve does not open, defective fire valve.

1. Servicing – AWI Distribution

WARNING: METHANOL AND ITS VAPOR IS TOXIC AND FLAMMABLE. HANDLING MUST ALWAYS BE IN A WELL VENTILATED AREA. IN CASE OF ACCIDENTAL CONTACT FLUSH WITH WATER AT ONCE. DO NOT SMOKE, CREATE SPARKS, OR HANDLE NEAR AN OPEN FLAME.

A. Routine Filling

(1) Mix AWI fluid IAW instructions in section 82-00-00.

(2) Remove the AWI filler cap, located on the right hand side of the nose baggage compartment.

(3) Fill tank to desired level using the sight gauge to determine quantity. If desired, the tank may be filled to overflowing.

(4) Install AWI filler cap.

B. AWI Inactive

(1) When the system is to be left unused for extended periods of time (one month or longer):

(a) Open the drain valve located in the left hand wing-to-fuselage fairing.

(b) Open the drain valve on the right hand side of the nose baggage compartment. The pump may be activated for quicker drainage.

(c) Close up and service the system with four gallons of distilled water.

(d) Move aircraft to a run up area and fire the AWI into both engines for three minutes.

(e) After firing, open the drain in the nose baggage compartment and left hand wing-to-fuselage fairing and allow the system to drain.

(f) Remove right hand nose baggage compartment floors.

(g) Disconnect pump inlet lines and allow to drain.

(h) Allow the system to air dry and then reassemble.

C. Reactivation of AWI

(1) Prior to placing the system into operation, perform the following steps:

(a) Service tank with fluid and operate the pump.

(b) Inspect the system and repair any leaks or defects.

(c) Move aircraft into run up area and fire the AWI into each engine using the Airplane Flight Manual instructions.

(d) When operation is satisfactory and system is free from leaks, the system may be returned to service.

2. Inspection/Check – AWI Distribution

A. Visually inspect the system for chafing, corrosion, leakage, and security.

B. Every 9,000 hours, or each nine years, perform the following:

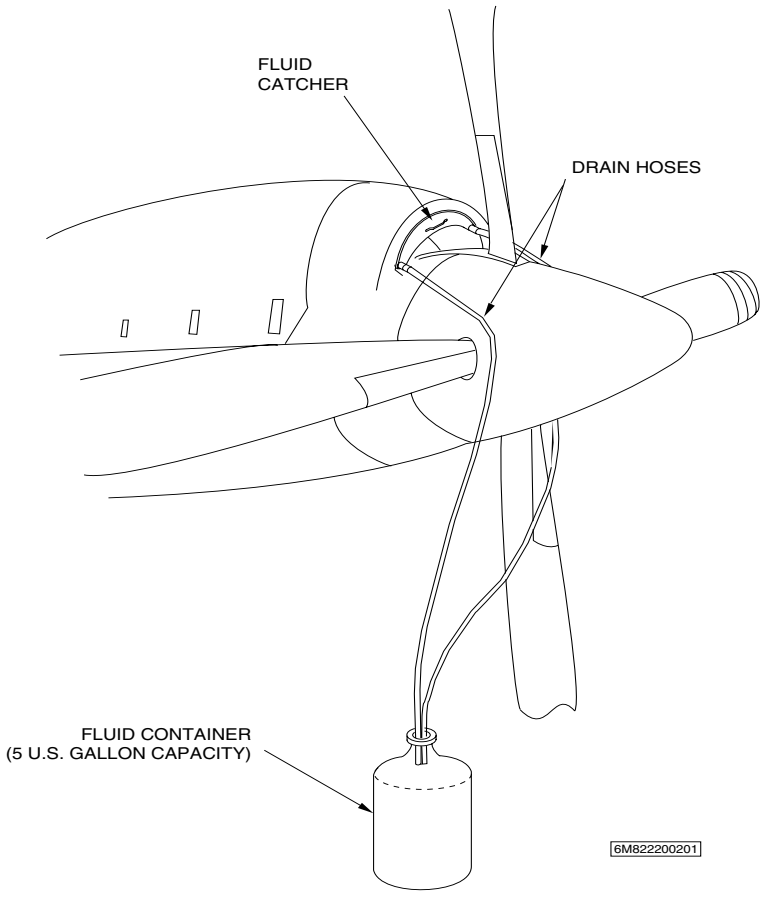
C. Disassemble the system and perform a visual inspection of the "wet" components of the system IAW steps **2.**, B., (1), through (4).

(1) Remove right hand nose baggage compartment floorboard. Remove outlet and inlet lines from AWI pump. Inspect lines and pump for evidence of corrosion.

(2) Remove right hand forward wing-to-fuselage fairing. Remove AWI supply hose from distribution "T" and inspect both fittings.

(3) Remove AWI line from each engine's metering valve and inspect line and valve for evidence of corrosion.

(4) Reassemble and service system. Test system IAW with the Airplane Flight Manual instructions.



Preparation for Flow Check
Figure 201

D. AWI Calibration

(1) The system is calibrated by flowing AWI fluid into each engine for a measured period of time and measuring the flow in fluid ounces. In order to accurately measure the flow of fluid into the engine it will be necessary to remove the engine cowl and install the AWI manifold over a bucket or other container for catching the fluid.

Aircraft Serial Number _____Date: _____

	LEFT (1)	RIGHT (2)
DESIRED FLOW RATE (pounds per hour)	815 ± 35	815 ± 35
FLOW TIME (seconds)		
FLOW QUANTITY (fluid ounces)		
FLOW RATE (pounds per hour) (3)		

NOTES

1. Reference FLOW CHECK PROCEDURE, Paragraph 3.
2. Reference FLOW CHECK PROCEDURE, Paragraph 6.
3. Determine flow rate using the following formula:

$$\frac{225 \times \text{FLOW QUANTITY (fluid ounces)}}{\text{FLOW TIME (seconds)}}$$

Example: $\frac{225 \times 235}{65} = 813.5$

AWI Calibration Data Sheet
Table 201

(2) As an alternate section 82-SERVC provides a catcher that may be locally manufactured and used in the engine inlet to catch the AWI mixture and direct it into a suitable container.

(3) System calibration and flow check will be necessary if either of the AWI metering valves, flow manifolds, or pump is replaced. In the case of a metering valve or manifold, the new components must be adjusted and calibrated as necessary to achieve the required flow limits. If the pump is replaced, the flow through the system must be checked to see if the required limits are met. If the flow is within the required limits, no adjustment to the system will be necessary.

D. Required Equipment

(1) Stop watch.

(2) Measuring equipment (graduated container of 32 fluid ounces minimum capacity.

(3) Source of 28 VDC power.

(4) Calibration data sheet (see Table 201).

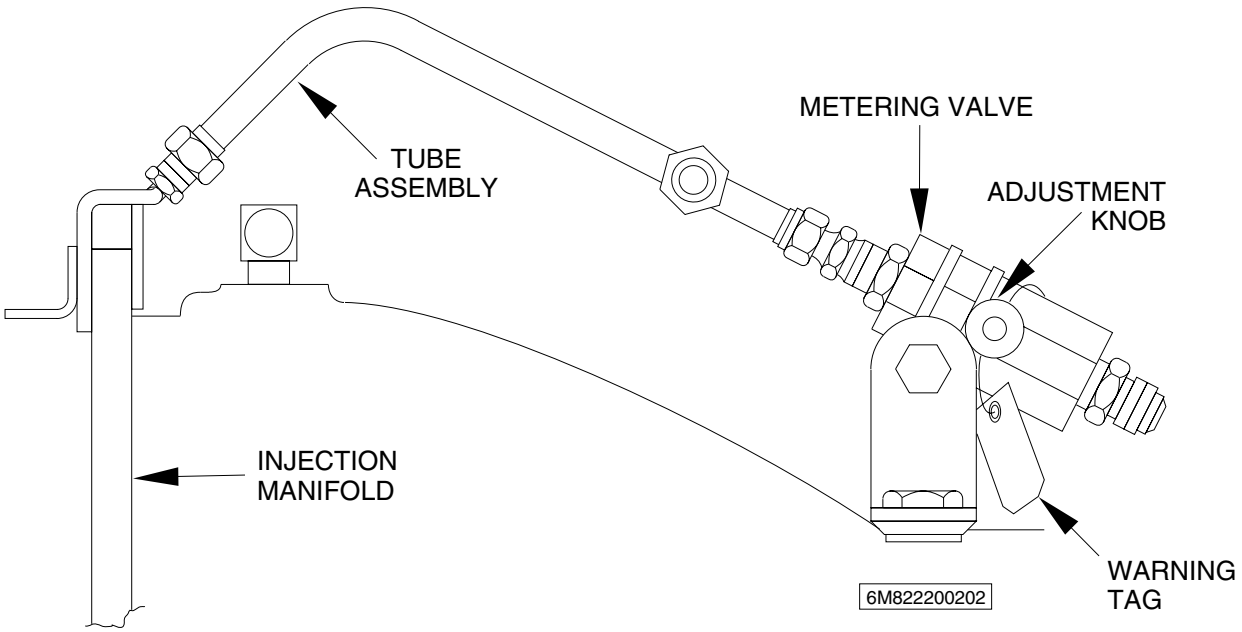
E. Preparation

(1) Install liquid catchers in the inlet of each engine as shown in Figure 201. Two five gallon bottles may be used to catch the mixture. Since all operations deal with only one engine at a time, only one catcher is actually necessary and may be switched from engine to engine.

NOTE: In lieu of the liquid catcher, the nose cowling may be removed and the manifold directed into a bucket or other suitable container.

(2) Fill the tank with AWI fluid IAW instructions in section 82-00-00.

(3) Connect a well regulated source of 28 VDC electrical power to the aircraft. If a power source of sufficient capacity is not available, a smaller source may be connected directly to the pump relay. Voltage fluctuations, as are common with standard APU units, will cause the flow to vary and make precise adjustment difficult.



Metering Valve Assembly
Figure 202

F. Flow Check Procedure

(1) Place AWI selector switch in ARM position. Observe AWI light illuminates.

(2) Advance left hand power lever until firing switch closes (switch can be heard to click). Start timer. After 60 seconds, return AWI switch to OFF position.

(3) Measure amount of liquid collected from left engine and compute the flow from information supplied in Table 201. The flow rate must be 815 ± 35 PPH. Adjust metering valve (see Figure 202) and rerun steps **2.**, F., (1) and (2) if flow is not within limits.

(4) Retard both power levers and place AWI selector switch in ARM position. Observe AWI armed light illuminates.

(5) Advance right hand power lever until firing switch closes (switch can be heard to click). Start timer. After 60 seconds, return AWI switch to OFF position.

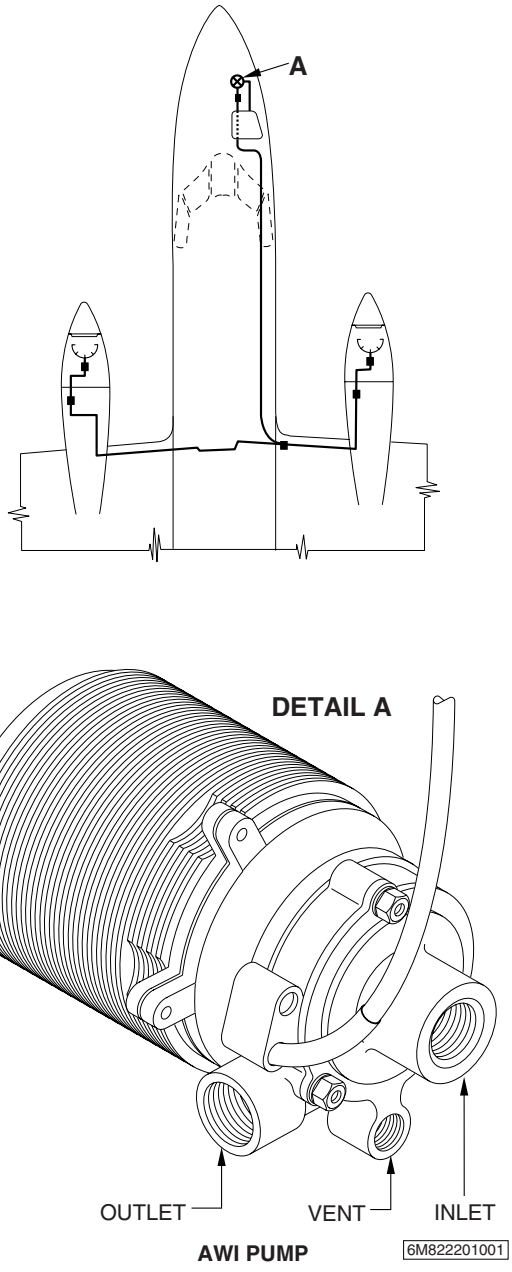
(6) Measure amount of liquid collected from right engine and compute the flow from information supplied in Table 201. The flow rate must be 815 ± 35 PPH. Adjust metering valve (see Figure 202) and rerun steps **2.**, F., (4) and (5) if flow is not within limits.

(7) After each engine's flow rate is within limits, tighten jam nut on the valve stem. Pass a length of safety wire through the roll pin in the valve knob and warning tag. Wrap safety wire around the body of the valve and twist.

AWI PUMP – DESCRIPTION AND OPERATION

1. General

A. The AWI pump is used to provide pressurized water-methanol for delivery to the engine(s). The pump is installed in a mounting bracket located beneath the right hand nose baggage compartment floorboard. The pump is a centrifugal type with no set displacement. Output pressure and flow varies with the output load applied to the pump. Ensure that Pump pressure, running deadheaded at 28.5 volts, is 43 to 48 psi.



AWI Pump
Figure 1

AWI PUMP – FAULT ISOLATION

1. Troubleshooting – AWI Pump

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pump is noisy	Defective pump	Replace pump
Pump output is low as indicated by lack of AWI ARMED light	Defective pump	Install a pressure gauge in the pressure switch mounting boss. If the pressure is low, make sure no leaks exist in the system before replacing the pump.
Pump output is high as indicated by difficulty during system calibration	Mounting bolts to pump bracket overtorqued	Loosen the three mounting bolts securing the pump to the bracket and retorque to no more than 25 inch-pounds (2.82 N·m). If the output remains high, replace the pump.
Pump inoperative	Defective pump relay	Troubleshoot relay according to section 82-22-03.
	Defective wiring	Troubleshoot wiring according to applicable wiring diagram.
	Defective pump	Replace pump.

AWI PUMP – MAINTENANCE PRACTICES

1. Removal/Installation – AWI Pump

A. Removal – AWI Pump

CAUTION: IF THE PUMP IS REPLACED, THE SYSTEM FLOW RATE MUST BE CHECKED AND CALIBRATED IF NECESSARY. REFER TO SECTION 82-22-00 FOR CALIBRATION PROCEDURES.

(1) Pull the AWI pump circuit breaker.

(2) Remove right hand nose baggage compartment floorboards.

(3) Remove supply connection to the pump.

(4) Remove pressure connection to the pump.

(5) Remove vent connection to the pump.

NOTE: If the tank has not been drained, the supply line must be capped. If the pump is to be removed for any purpose other than immediate replacement, cap all lines to prevent the entry of foreign material into the system.

(6) Disconnect electrical connections from the pump.

(7) Remove screws and bolt securing the pump and bracket in place. Remove the pump and bracket.

(8) Remove bolts securing the pump to the bracket and remove the bracket.

(9) Remove the fittings from the pump and discard the O-ring packings.

B. Installation – AWI Pump

(1) Install the pressure fitting and new O-ring (P/N MS28778-10) in the pump.

(2) Install the supply line fitting and new O-ring (P/N MS28778-10) in the pump.

(3) Install the vent line fitting and new O-ring (P/N MS28778-4) in the pump.

(4) Orient all fittings to their original positions.

NOTE: Minor reorientation may be necessary on installation.

(5) Remove the front plate bolts from the pump.

CAUTION: DO NOT OVERTORQUE NOR UNSYMMETRICALLY TORQUE THE BOLTS.

(6) Install the pump mounting bracket using the three front plate bolts. Torque the bolts to 25 inch-pounds (2.82 N·m) and safety.

(7) Install the pump and bracket in aircraft.

(8) Remove the caps from the vent, supply, and pressure lines and connect to the pump. If necessary, service the AWI tank.

(9) Hook the electrical connections to the pump.

(10) Reset the AWI pump circuit breaker.

(11) Operate the pump and check for leaks. Ensure the AWI armed light illuminates (indicates pump pressure). It may be necessary to loosen the pressure and/or supply and vent lines to prime the pump.

(12) Install the nose baggage compartment floorboards.

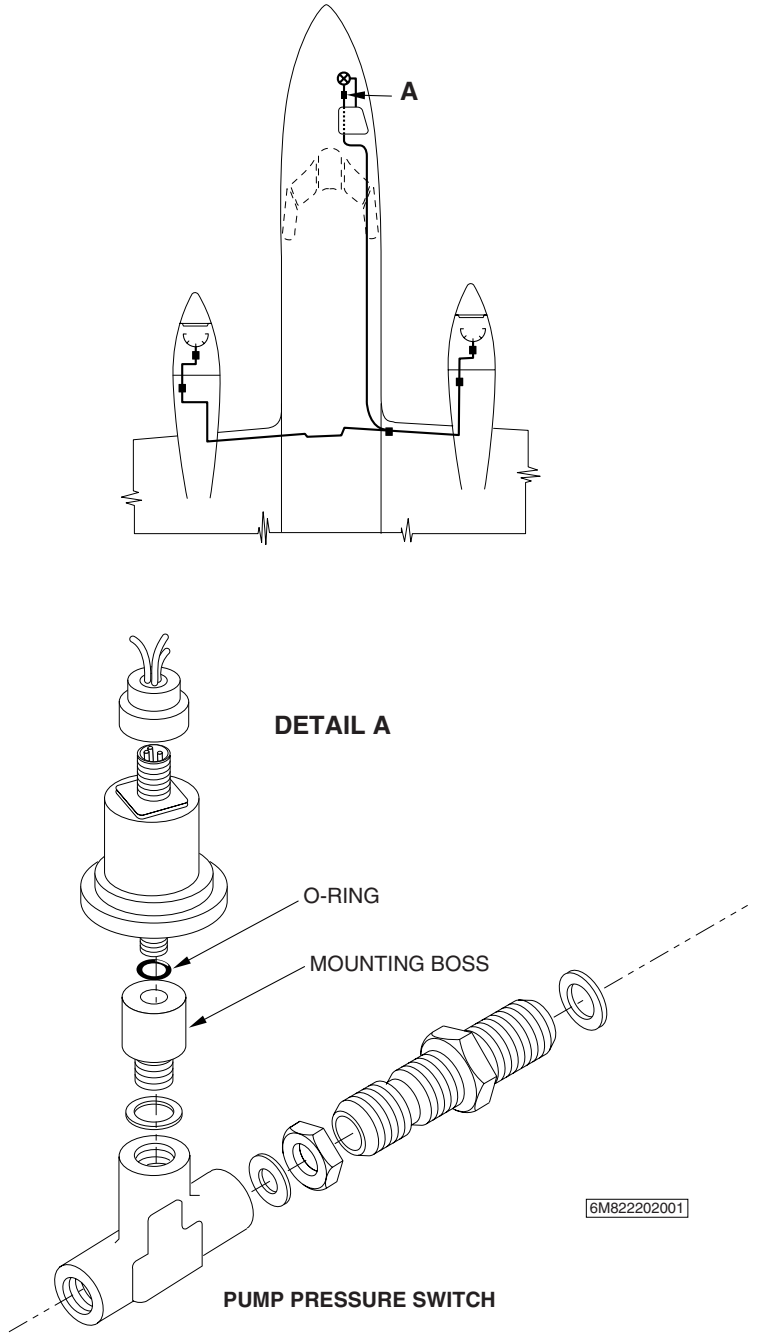
2. Inspection/Check – AWI Pump

A. Check the pump for security of mounting, physical damage and/or evidence of leakage. The AWI pump is life limited in some installations; refer to section 05-10-00.

AWI PUMP PRESSURE SWITCH – DESCRIPTION AND OPERATION

1. General

A. The AWI pump pressure switch is installed in the outlet side of the AWI pump. Closure of the switch completes the patch necessary to illuminate the AWI armed light. The switch is set to close at 30 ± 1 psi increasing pressure and open by 26 psi decreasing pressure.



AWI Pump Pressure Switch
Figure 1

AWI PUMP PRESSURE SWITCH – FAULT ISOLATION

1. Troubleshooting – AWI Pump Pressure Switch

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No AWI armed light, pump is operating	Defective wiring between switch and cockpit	Remove electrical connector from switch, apply ground to pin B. If the annunciator AWI light does not illuminate, problems exist within the wiring or the annunciator panel; refer to the applicable section and troubleshoot as necessary. If the AWI armed light does illuminate, the problem is the switch or the pump output.
	Pump output is low	Install a direct reading pressure gauge in the switch mounting and read pump output pressure (43 to 48 psi with the pump deadheaded).
	Switch is defective	Replace switch.

AWI PUMP PRESSURE SWITCH – MAINTENANCE PRACTICES

1. Removal/Installation – AWI Pump Pressure Switch

A. Removal – AWI Pump Pressure Switch

(1) Gain access to the switch by removing the right hand nose baggage compartment floorboard.

(2) Remove the safety wiring from the electrical connector.

(3) Disconnect the connector.

(4) Unscrew the switch from its mounting boss and remove the switch.

B. Installation – AWI Pump Pressure Switch

(1) Install a new O-ring (P/N MS28778-4) on the new switch and install in the boss. Torque the switch to 30 inch-pounds (3.39 N·m) maximum.

(2) Connect and safety the electrical connector to the switch.

(3) Operate the pump and check for proper operation of the annunciator light.

(4) Install nose baggage compartment floorboard.

2. Inspection/Check – AWI Pump Pressure Switch

A. Check the pump pressure switch for security of mounting, physical damage and/or evidence of leakage.

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AWI PUMP RELAY – DESCRIPTION AND OPERATION

1. General

A. The AWI pump relay controls the application of power to the AWI pump. Circuitry within the AWI control system opens and closes the relay as necessary. Failure of the relay to the open position will not allow the pump to operate. Failure to the closed position causes the pump to operate anytime the battery switches are placed in the ON position.

AWI PUMP RELAY – FAULT ISOLATION

1. Troubleshooting – AWI Pump Relay

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pump will not operate	Defective wiring to pump relay	Battery switches ON; AWI selector in ARM; check for 28 volts at A1 and X1 of the relay. Remove the wire connected to X2 and check for 0 ohms resistance to ground.
	Defective relay or defective pump	Check for 28 volts at A2 of the relay; if voltage is not present, replace relay; if voltage is present, replace pump.

AWI PUMP RELAY – MAINTENANCE PRACTICES

1. Removal/Installation – AWI Pump Relay

A. Removal – AWI Pump Relay

(1) Disconnect the electrical connector to the right hand inverter and remove the inverter.

(2) Remove screws securing the pump relay to the inverter channel.

(3) Tag the electrical wires connected to the relay and disconnect the wires.

B. Installation – AWI Pump Relay

(1) Connect the electrical wires to the new relay.

(2) Install the relay in the inverter channel with screws.

(3) Install the inverter.

(4) Connect and safety the inverter electrical connections.

(5) Functional check the AWI pump and inverter system.

2. Inspection/Check – AWI Pump Relay

A. Check the relay for security of mounting, physical damage and security of the electrical connections to the relay.

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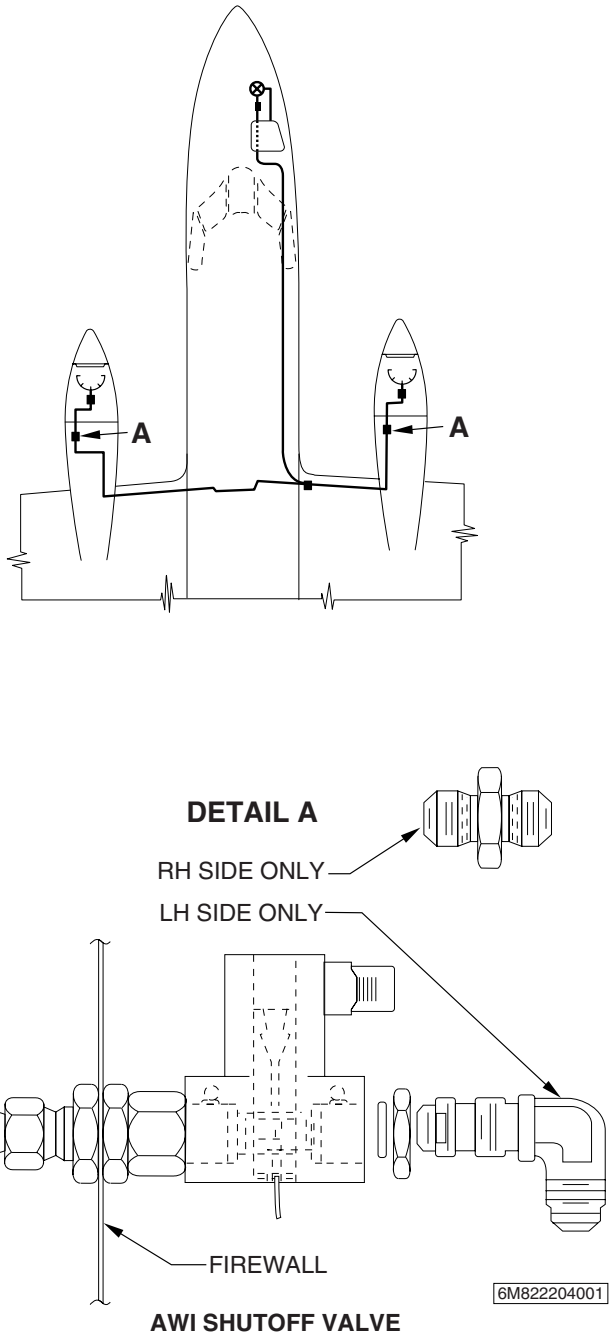
AWI SHUTOFF VALVE – DESCRIPTION AND OPERATION

1. General

A. The AWI shutoff valve installed in each nacelle is used to control the flow of AWI fluid to the engine. The normally closed valve is opened by an electrical signal from the AWI control circuitry.



AWI Shutoff Valve
Figure 1

AWI SHUTOFF VALVE – FAULT ISOLATION

1. Troubleshooting – AWI Shutoff Valve

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Valve will not open	Defective wiring between valve and cockpit	Place AWI selector switch in the arm position; move the respective power lever past flight idle; 28 volts should be present at pin A of the valve connector.
	Defective valve	Replace valve

AWI SHUTOFF VALVE – MAINTENANCE PRACTICES

1. Removal/Installation – AWI Shutoff Valve

A. Removal – Shutoff Valve

(1) Gain access to the shutoff valve through the access door on the left hand side of the nacelle just aft of the firewall.

(2) Remove the AWI hose from the firewall fitting on the forward side of the firewall.

(3) Remove the supply line from the aft side of the valve.

(4) Remove the nut from the firewall fitting on the forward side of the firewall.

(5) Withdraw the valve through the access door opening and remove the electrical connector.

(6) Remove the inlet and outlet fittings from the valve and discard the O-rings.

B. Installation – Shutoff Valve

(1) Install the fittings into the new valve using two new O-rings (P/N MS28778-10). Torque the fittings enough to seat the O-rings (70 inch-pounds [7.91 N·m] maximum).

(2) Install and safety the electrical connector to the valve.

(3) insert the valve through the access door opening.

(4) Connect the inlet hose to the valve and slide the outlet fitting through the firewall hole. Install the nut on the outlet fitting.

(5) Tighten the outlet fitting nut and inlet hose connection. Install and tighten the engine AWI hose to the outlet fitting.

(6) Place the AWI selector switch in ARM and allow the system to build pressure. Move the respective power lever past the flight idle gate. Observe that the shutoff valve opens.

2. Inspection/Check – AWI Shutoff Valve

A. Check the shutoff valves for security of mounting, physical damage, and/or evidence of leakage.

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AWI TORQUE SWITCH – DESCRIPTION AND OPERATION

1. General

A. An AWI torque switch is installed on each engine to sense the power output of the engine and supply this information to the AWI control circuitry. The switch is set to close by 22 psi increasing pressure and open by 15 ± 2 psi decreasing pressure.

AWI TORQUE SWITCH – FAULT ISOLATION

1. Troubleshooting – AWI Torque Switch

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
AWI will not fire	Defective wiring to torque switch	Place AWI selector switch in arm; power levers forward of flight idle; 28 volts should be present at pin A of the torque switch connector; pin B to ground should be 0 ohms resistance.
	Defective torque switch	Replace switch.

AWI TORQUE SWITCH – MAINTENANCE PRACTICES

1. **Removal/Installation – AWI Torque Switch**

   A. Removal – AWI Torque Switch

      (1) Gain access to the switch by removing the right hand side cowl from the engine.

      (2) Cut the safety wire and remove the electrical connector to the switch.

      (3) Unscrew the switch from its mounting boss.

   B. Installation – AWI Torque Switch

      (1) Install a new O-ring (P/N MS29561-012) on the new torque switch.

      (2) Install the new switch in its mounting boss and torque to 30 inch-pounds (3.39 N·m) maximum.

      (3) Install and safety the electrical connector to the switch.

      (4) Operate and test the system IAW the Airplane Flight Manual instructions.

2. **Inspection/Check – AWI Torque Switch**

   A. Check the switches for security, leakage, and condition of the electrical connection to the switch.

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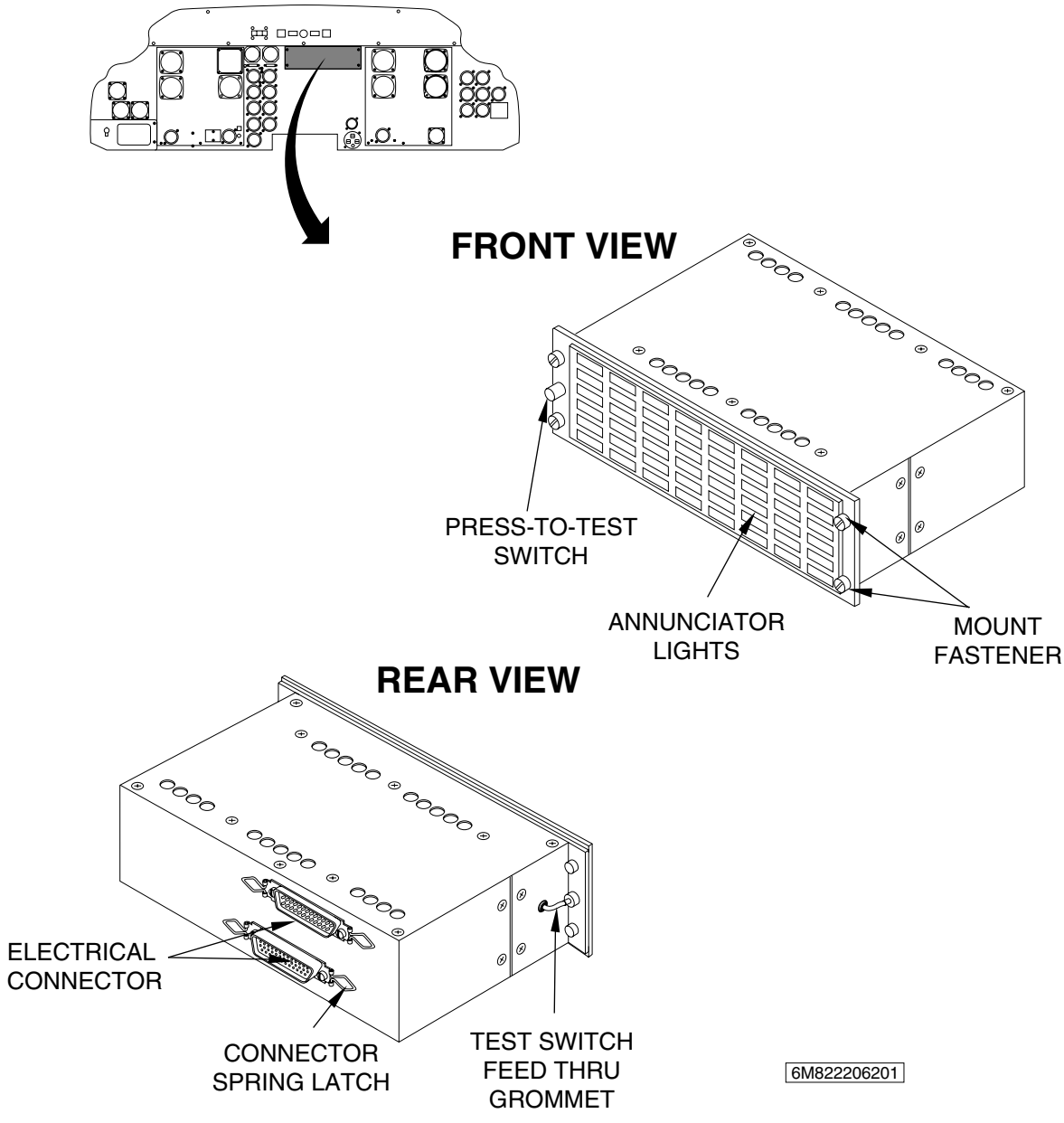
1. Adjustment/Test – Annunciator Panel

A. M7 Aerospace recommends the annunciator be replaced as an assembly. The following test will determine if the problem is with the aircraft wiring or within the annunciator panel:

(1) Remove the plugs from the back of the annunciator panel. The small plug contains 37 pins; the larger plug contains 50 pins.



Annunciator Panel
Figure 201

(2) With the battery switches on, measure 28 volts at pins 1, 35, and 29 of the small connector.

(3) Turn the battery switches off, and install the small connector.

(4) Turn the battery switches on, and push the press-to-test button on the annunciator panel. Make sure the following lights illuminate:

(a) AWI ARMED

(b) AWI LOW WATER

(5) Using a jumper, supply a ground to the following pins of the large receptacle at the back of the annunciator panel:

(a) Pin 40 to ground AWI . . . . . ARMED light illuminates

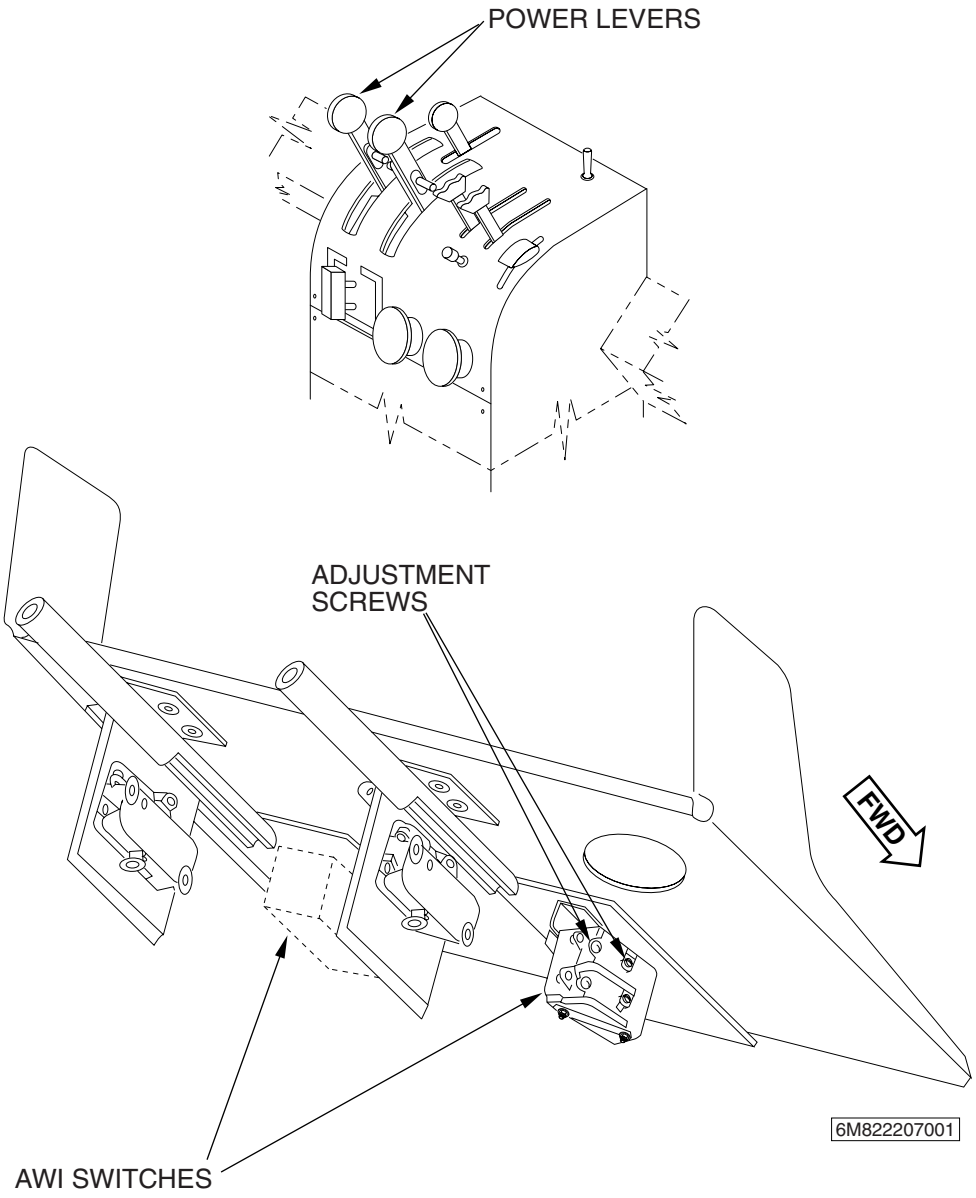
(b) Pin 39 to ground AWI . . . . . .LOW WATER light illuminates

NOTE: If the problem is within the annunciator, the annunciator assembly should be replaced.

POWER LEVER POSITION SWITCHES – DESCRIPTION AND OPERATION

1. General

A. Two normally open microswitches are installed in the cockpit pedestal. Each switch is actuated to the closed position by the respective power lever. A slotted mounting plate is installed to allow adjustment of the switches.



ADJUST SWITCHES TO ACTUATE WHEN THE POWER LEVERS ARE 1/8 INCH BEYOND FLIGHT IDLE.

Power Lever Position Switch Adjustment Figure 1

POWER LEVER POSITION SWITCHES – FAULT ISOLATION

1. Troubleshooting – Power Lever Position Switches

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
AWI will not fire	Power lever switch not closing	Check for bleed air shut off with selector switch in ARM and respective power lever moved beyond flight idle.
AWI fires without power lever beyond flight idle	Shorted bleed air isolation diode	Move bleed air switch to OFF while AWI is firing; If AWI stops; diode is shorted.

POWER LEVER POSITION SWITCHES – MAINTENANCE PRACTICES

1. Adjustment/Test – Power Lever Position Switches

A. Adjustment Procedure

(1) Gain access to the switches either by removing the forward center pedestal access panel or by working from the back of the pedestal.

(2) Adjust the respective switch to actuate when the power lever is 1/8 inch beyond the flight idle gate.

2. Inspection/Check – Power Lever Position Switches

A. Check the power lever switches for condition and security.

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DISTRIBUTION SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The corrosive properties of the alcohol water mixture require that all components of the AWI system be manufactured of materials that do not readily corrode. The part number of some valves and fittings used in the system contain prefixes and suffixes indicating that special materials have been used in the part. Exact part number replacements must be used in order to maintain the corrosion resistive properties of the AWI system. Lines, tubing, and connections may be purchased or locally manufactured.

B. Hard Tubing and Lines

(1) Replace hard tubing and lines with parts manufactured of the same material (if the line is aluminum, replace it with aluminum) in accordance with the following specifications:

(a) Aluminum

Flared tubing and lines	2024-T3 QQ-A-250/5	Diameter as required
Welded tube assemblies	6061-T6 QQ-A-250/11	Diameter and wall thickness to match original tube assy.

(b) Stainless Steel

Flared tubing and lines	304SST MIL-T-6845	Diameter and wall thickness as required.
Welded tube assemblies	321SST MIL-6721	Diameter and wall thickness as required.

C. Flexible Tubing and Lines

(1) Clear flexible tubing

'TYGON'
Norton Plastic and Synthetics Division
Specification B-44-3
Diameter and wall thickness to match original part.

(2) Flexible hoses and lines (AEROQUIP TYPE)

Replace only with like part number.

D. Valves, Switches, and Components

(1) Exact part number replacement must be used during maintenance on the AWI system. All component parts that come in contact with the AWI mixture must be treated to prevent corrosion. Aluminum parts will be anodized while stainless steel parts will be passivated.

E. Maintenance, Removal, and Replacement

(1) No special precautions are required during maintenance on the AWI system. Standard aircraft practices should be used in component removal and replacement. Flexible lines should be routed to avoid low spots that may allow AWI mixture to be trapped. Particular care should be given to capping lines when components are removed as the AWI manifold outlets are small orifices and can easily be plugged by foreign material.

DISTRIBUTION SYSTEM – FAULT ISOLATION

1. Troubleshooting – Distribution System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
System will not arm	No power to AWI control relays	Check for power through selector by noting if the power lever will shut off the respective bleed air.
	Defective annunciator panel light	Troubleshoot according to section 82-23-07.
AWI will not fire to one engine (emergency mode)	No power to fire valve	Remove opposite engine torque switch electrical connector. Selector switch in ARM; power levers forward; power should be present at pin A. Check for ground at pin B.
	Defective fire valve or torque switch	Jumper pin A to B. If valve opens, replace torque switch. If valve does not open, replace valve.
AWI will not fire to one engine (continuous mode)	No power to fire valve	Remove the torque switch electrical connect on the affected engine. Selector switch on CONT; power lever forward; power should be present at pin C. Check for ground at pin B.
	Defective fire valve torque	Jumper pin B to C. If valve opens, replace torque switch. If valve does not open, replace valve.

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No AWI pump light(s)	Defect pressure switch	Troubleshoot according to section 82-23-03.
	Low pump output or no pump output	Troubleshoot according to section 82-23-01 or 82-23-02.

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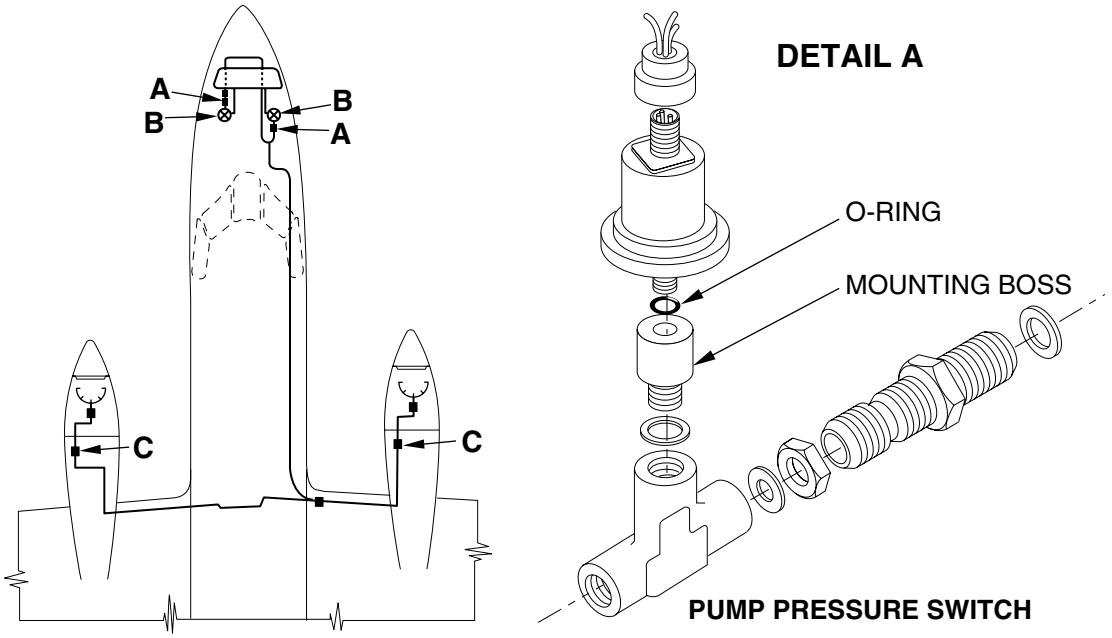


Distribution System
Figure 201

1. Servicing – Distribution System

WARNING: METHANOL AND ITS VAPOR IS TOXIC AND FLAMMABLE. HANDLING MUST ALWAYS BE IN A WELL VENTILATED AREA. IN CASE OF ACCIDENTAL CONTACT FLUSH WITH WATER AT ONCE. DO NOT SMOKE, CREATE SPARKS, OR HANDLE NEAR AN OPEN FLAME.

A. Routing Filling

(1) Mix AWI fluid in accordance with the instructions in section 82-00-00.

(2) Remove the AWI filler cap, located on the right hand side of the nose section, forward of the nose baggage door.

(3) Fill the tank to the desired level using either the cockpit indicator or the sight gauge to determine quantity. Either indicating system will indicate properly only if the aircraft is level. If desired, the tank may be filled to overflowing.

(4) Replace the AWI filler cap.

B. Inactive System

(1) When the system is to be left unused for extended periods of time (one month or longer):

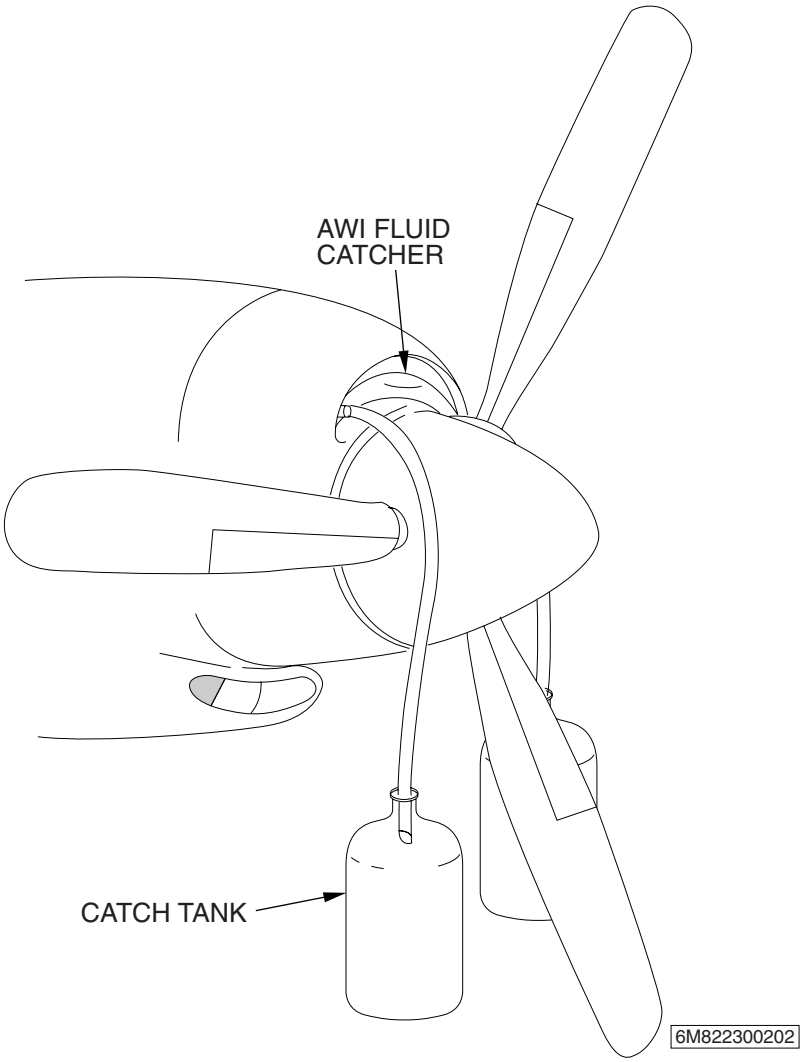
(a) Open the drain valve located in the left hand wing-to-fuselage fairing.

(b) Open the drain valve in the nose wheel well. The No. 1 pump may be activated for quicker drainage.

(c) Close up the system and service with 16 gallons of distilled water.

(d) Move the aircraft to the run up area and fire the AWI into both engines for three minutes.

(e) After firing, open the drains in the nose wheel well and left hand wing-to-fuselage fairing and allow the system to drain.



**Preparation
for Flow Check
Figure 202**

(f) Remove the left and right hand nose baggage compartment floorboards.

(g) Disconnect the pump inlet lines and allow to drain.

(h) Allow the system to air dry and then reassemble.

C. Reactivation of system

(1) Prior to placing the system into operation, perform the following steps:

(a) Service the tank with fluid and operate both pumps.

(b) Inspect the system and repair any leaks or defects.

(c) Move the aircraft to run up area and fire the AWI into each engine IAW the Airplane Flight Manual instructions.

(d) When the operation is satisfactory and the system is free from leaks, the system may be returned to service.

2. Inspection/Check – Distribution System

A. Visually inspect the system for chafing, corrosion, leakage, and security.

B. Each 9,000 hours (or nine years):

(1) Disassemble the system and perform a visual inspection of the "wet" components of the system IAW steps **2.**, B., (1), (a) through (d).

(a) Remove the left and right hand nose baggage compartment floorboards. Remove the outlet and inlet lines from each AWI pump. Inspect the lines and pumps for evidence of corrosion.

(b) Remove the right hand forward wing-to-fuselage fairing. Remove the AWI supply hose from the distribution "t" and inspect both fittings.

(c) Remove the AWI line from each engine's metering valve and inspect the line and valve for evidence of corrosion.

(d) Reassemble and service the system. Test the system IAW the Airplane Flight Manual instructions.

C. System Calibration

(1) The system is calibrated by flowing AWI fluid into each engine for a measured period of time and measuring the flow in fluid ounces. In order to accurately measure the flow of fluid into the engine it will be necessary to remove the engine nose cowl and install the AWI manifold over a bucket or other container for catching the fluid.

(2) As an alternate, section 82-SERVC provides a catcher that may be locally manufactured and used in the engine inlet to catch the AWI mixture and direct it into a bucket or other container.

Aircraft Serial Number _____ Date: _____

			LEFT	RIGHT	
DESIRED FLOW RATE (PPH)	MAXIMUM			950	950
	MINIMUM			780	780
BOTH ENGINES	FLOW	(ounces)			
	TIME	(seconds)			
	RATE	(pph)*			
LH ENGINE ONLY	FLOW	(ounces)			
	TIME	(seconds)			
	RATE	(pph)*			
RH ENGINE ONLY	FLOW	(ounces)			
	TIME	(seconds)			
	RATE	(pph*			

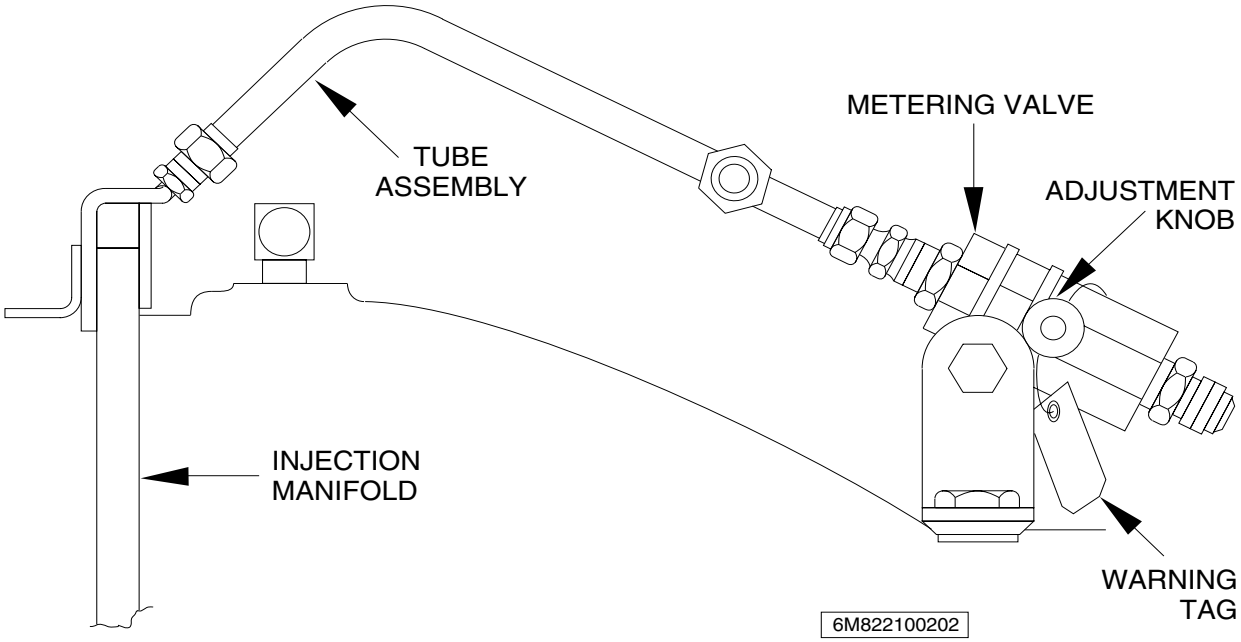
*Determine flow rate using the following formula:

$$\frac{225 \times \text{FLOW (ounces)}}{\text{TIME (seconds)}}$$

EXAMPLE: $\frac{225 \times 256}{65} = 886.1$

Claibration Data Sheet
Table 201

(3) System calibration and flow check will be necessary if either of the AWI metering valves, flow manifolds, or pumps are replaced. In the case of a metering valve or manifold, the new components must be adjusted and calibrated as necessary to achieve the required flow limits. If the pump is replaced, the flow through the system must be checked to see if the required limits are met. If the flow is within the required limits, no adjustment to the system will be necessary.



Metering Valve Assembly
Figure 203

D. Required Equipment

(1) Stop watch.

(2) Measuring equipment (graduated container of 32 fluid ounces minimum capacity).

(3) Source of 28 VDC power.

(4) Calibration data sheet (Table 201).

E. Preparation

(1) Install liquid catchers in the inlet of each engine as shown in Figure 202. Two five gallon bottles may be used to catch the mixture.

NOTE: In lieu of the liquid catcher, the nose cowling may be removed and the manifold directed into a bucket or other suitable container.

(2) Fill the tank with AWI fluid IAW instructions in section 82-00-00.

NOTE: While performing the flow check procedure, monitor the fluid level in the AWI tank. Replenish when the level reaches approximately one-half.

(3) Connect a well regulated source of 28 VDC electrical power to the aircraft. If a power source of sufficient capacity is not available, a smaller source may be connected directly to the pump relay. Voltage fluctuations, as are common with standard APU units, will cause the flow to vary and make precise adjustment difficult.

F. Flow Check Procedure

(1) Place the AWI selector switch in the ARM position.

(2) Observe the AWI pump lights are illuminated.

(3) Advance the left hand power lever until firing switch closes (switch can be heard to click). Start timer.

(4) After 60 seconds, return the AWI selector switch to the OFF position.

(5) Measure the amount of liquid collected from the left engine and compute the flow from the information supplied in Table 201. The flow rate must be 780 to 950 PPH.

(6) If the flow rate is less than 780 PPH or more than 950 PPH, manifold calibration must be accomplished as oulined in **2.**, G. If the flow is correct, proceed to step **2.**, F., (7).

(7) Retard both power levers and place the AWI selector switch in the ARM position. Observe the AWI pump lights illuminated.

(8) Advance the right hand power lever until the firing switch closes (switch can be heard to click). Start timer.

(9) After 60 seconds, return the AWI selector switch to the OFF position.

(10) Measure the amount of liquid collected from the right engine and compute the flow from the information supplied in Table 201. The flow rate must not be less than 780 PPH or more than 950 PPH.

(11) If the flow rate is less than 780 PPH or more than 950 PPH, manifold calibration must be accomplished as outlined in **2.**, G. If the flow rate is correct, proceed to step **2.**, G., (12).

(12) Pull the No. 1 AWI pump circuit breaker. Place the AWI selector switch in the ARM position. Observe the No. 2 AWI pump light illuminated.

(13) Advance both power levers until the firing switches close (switches can be heard to click). Start timer.

(14) After 60 seconds, return the AWI switch to the OFF position.

(15) Measure the amount of liquid collected from each engine and compute the flow rate from Table 201. The flow rate must be at least 780 PPH through each engine.

(16) Reset the No. 1 AWI pump circuit breaker and pull the No. 2 AWI pump circuit breaker. Place the AWI selector switch in the ARM position. Observe the No. 1 AWI pump light illuminated.

(17) Advance both power levers until the firing switches close (switches can be heard to click). Start timer.

(18) After 60 seconds, return the AWI switch to the OFF position.

(19) Measure the amount of liquid collected from each engine and compute the flow rate from Table 201. The flow rate must be at least 780 PPH through each engine.

(20) If the flow requirements called out in steps **2.**, F., (5), (10, (15), and (19) are met, the system is within limits and no further action is required.

G. Manifold Calibration – Maximum Flow

(1) Place the AWI selector switch in the ARM position.

(2) Observe the No. 1 and No. 2 AWI pump lights illuminated.

(3) Advance the left hand power lever until the firing switch closes (switch can be heard to click). Start timer.

(4) After 60 seconds, return the AWI selector switch to the OFF position.

(5) Measure the amount of liquid collected from the left engine and compute the flow rate from Table 201. The flow rate should be very near but no more than 950 PPH. Adjust the engine installed metering valve (Figure 203) and repeat steps **2.**, G., (1) through (5) until the flow rate is correct. Some early aircraft have metering valves installed that have had the valve stem drilled and a roll pin installed to lock the valve adjustment. Valves in this configuration require replacement if adjustment is necessary. See section 82-SERVC for further information on metering valves.

(6) Place the AWI selector switch in the ARM position.

(7) Observe the No. 1 and No. 2 AWI pump light illuminated.

(8) Advance the right hand power lever until the firing switch closes (switch can be heard to click). Start timer.

(9) After 60 seconds, return the AWI selector switch to the OFF position.

(10) Measure the amount of liquid collected from the right engine and compute the flow rate from Table 201. The flow rate should be very near but not more than 950 PPH. Adjust the engine installed metering valve (Figure 203) and repeat steps **2.**, G., (6) through (10) until the flow rate is correct. Some early aircraft have metering valves installed that have had the valve stem drilled and a roll pin installed to lock the valve adjustment. Valves in this configuration require replacement if adjustment is necessary. See section 82-SERVC for further information on metering valves.

H. Manifold Calibration – Minimum Flow

(1) The following steps check the minimum flow condition of the system. One of the AWI pumps will have a lower output than the other. Once it has been determined which pump has the lower output, only that pump need be checked for the minimum flow check portion of the calibration.

(a) Pull the No. 1 AWI pump circuit breaker. Place the AWI selector switch in the ARM position. Observe the No. 2 pump light illuminated.

(b) Advance both power levers until the firing switches close (switches can be heard to click). Start timer.

(c) After 60 seconds, return the AWI switch to the OFF position

(d) Measure the amount of liquid collected from each engine and record on Table 201.

(e) Reset the No. 1 AWI pump circuit breaker and pull the No. 2 AWI pump circuit breaker. Place the AWI selector switch in the ARM position. Observe the No. 1 AWI pump light illuminated.

(f) Advance both power levers until the firing switches close (switches can be heard to click). Start timer.

(g) After 60 seconds, return the AWI switch to the OFF position.

(h) Measure the amount of liquid collected from each engine and compare with the amounts obtained in step **2.**, F., (15). The lowest flow recorded indicates which pump must be used for minimum flow checks and whether minimum system flow is acceptable. If the lowest flow recorded in either step **2.**, F., (15) or (19) was at least 780 PPH per engine, then the system is acceptable. If the flow was less than 780 PPH for either engine, then the engine installed metering valve must be adjusted. When an adjustment is made to a metering valve, the applicable maximum flow checks (steps **2.**, G., [1] through [5] for left engine, steps **2.**, G. [6] through [10] for right engine) must be repeated. The entire minimum flow check must then be repeated.

(i) To be within acceptable limits, the system flow rate must not be more than 950 PPH with both pumps supplying either engine or less than 780 PPH per engine with the lowest output pump (as defined in step **2.**, H., (1) supplying both engines.

(j) After each engine's flow rate is within limits, tighten the jam nut on the valve stem. Pass a length of safety wire through the roll pin in the valve knob and the warning tag. Wrap the safety wire around the body of the valve and twist.

No. 1 AWI PUMP – DESCRIPTION AND OPERATION

1. General

A. The No. 1 AWI pump is used to provide pressurized water-methanol for delivery to the engine(s). The pump is installed in a mounting bracket located beneath the left hand nose baggage compartment floorboard. The pump is a centrifugal type with no set displacement. Output pressure and flow will vary with the output load applied to the pump. Pump pressure, running deadheaded at 28.5 volts, should be 43 to 48 psi.

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No. 1 AWI PUMP – FAULT ISOLATION

1. Troubleshooting – No. 1 AWI Pump

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pump is noisy	Defective pump	Replace pump.
Pump output is low as indicated by lack of a No. 1 AWI pump annunciator light	Defective pump	Install a pressure gauge in the pressure switch mounting boss. If the pressure output is low, make sure no leaks exist in the system before replacing the pump.
Pump output is high, as indicated by difficulty during system calibration	Mounting bolts to pump bracket overtorqued	Loosen the three mounting bolts securing the pump to the bracket and retorque to no more than 25 inch-pounds. If the output remains high, replace the pump.
Pump inoperative	Defective pump relay	Troubleshoot relay according to section 82-23-04.
	Defective wiring	Troubleshoot wiring according to applicable wiring diagram.
	Defective pump	Replace pump.

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No. 1 AWI PUMP – MAINTENANCE PRACTICES

1. Servicing – No. 1 AWI Pump

A. No regularly scheduled maintenance is required of the AWI pump. During inspections, the pump should be checked for security of mounting, physical damage and/or evidence of leakage. The AWI pump is life limited in some installation; refer to section 5-50-82 for specified limits.

2. Removal/Installation – No. 1 AWI Pump

A. Pump removal and replacement

CAUTION: IF THE AWI PUMP IS REPLACED, THE SYSTEM FLOW RATE MUST BE CHECKED AND CALIBRATED IF NECESSARY. REFER TO SECTION 82-23-00 FOR CALIBRATION PROCEDURES.

(1) Pull the No. 1 AWI pump circuit breaker.

(2) Remove the left hand nose baggage compartment floorboard.

(3) Remove the supply, pressure, and vent connections to the pump. If the tank has not been drained, the supply line must be capped. If the pump is to be removed for any purpose other than immediate replacement, cap all lines to prevent the entry of foreign material into the system.

(4) Disconnect the electrical connections from the pump.

(5) Remove the three bolts securing the pump to the bracket and remove the pump.

(6) Remove the fittings from the pump and discard the O-ring packings.

(7) Install the pressure fitting and new O-ring (P/N MS28778-10) in the pump. Install the supply line fitting and new O-ring (P/N MS28778-10 in the pump. Install the vent line fitting and new O-ring (P/N MS28778-4 in the pump. Orient all fittings to their original positions; minor reorientation may be necessary on installation.

(8) Remove the three front plate bolts from the pump. Install the pump in its mounting bracket using the three front plate bolts. Torque the bolts to 25 inch-pounds and safety.

CAUTION: DO NOT OVERTORQUE NOR UNSYMMETRICALLY TORQUE THE BOLTS.

(9) Remove the caps from the vent, supply, and pressure lines and connect to the pump. If necessary, service the AWI tank.

(10) Hook the electrical connections to the pump. Reset the pump circuit breaker.

(11) Operate the pump and check for leaks. Make sure the No. 1 AWI pump light illuminates (indicates pump pressure). It may be necessary to loosen the pressure and/or supply and vent lines to prime the pump.

(12) Reinstall the nose baggage compartment floorboard.

No. 2 AWI PUMP – DESCRIPTION AND OPERATION

1. General

A. The No. 2 AWI pump is used to provide pressurized water-methanol for delivery to the engine(s). The pump is installed in a mounting bracket located beneath the right hand nose baggage compartment floorboard. The pump is a centrifugal type with no set displacement. Output pressure and flow will vary with the output load applied to the pump. Pump pressure, running deadheaded at 28.5 volts, should be 43 to 48 psi.

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No. 2 AWI PUMP – FAULT ISOLATION

1. Troubleshooting – No. 2 AWI Pump

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pump is noisy	Defective pump	Replace pump
Pump output is low as indicated by lack of a No. 2 AWI pump annunciator light	Defective pump	Install a pressure gauge in the pressure switch mounting boss. If the pressure output is low, make sure no leaks exist in the system before replacing the pump.
Pump output is high, as indicated by difficulty during system calibration	Mounting bolts to pump bracket overtorqued	Loosen the three mounting bolts securing the pump to the bracket and retorque to no more than 25 inch-pounds. If the output remains high, replace the pump.
Pump inoperative	Defective pump relay	Troubleshoot relay according to section 82-23-04.
	Defective wiring	Troubleshoot wiring according to applicable wiring diagram.
	Defective pump	Replace pump.

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No. 2 AWI PUMP – MAINTENANCE PRACTICES

1. Servicing – No. 2 AWI Pump

A. No regularly scheduled maintenance is required of the AWI pump. During inspections, the pump should be checked for security of mounting, physical damage and/or evidence of leakage. The AWI pump is life limited in some installation; refer to section 5-50-82 for specified limits.

2. Removal/Installation – No. 2 AWI Pump

A. Pump Removal and Replacement

CAUTION: IF THE AWI PUMP IS REPLACED, THE SYSTEM FLOW RATE MUST BE FLOW CHECKED AND CALIBRATED IF NECESSARY. REFER TO SECTION 82-23-00 FOR CALIBRATION PROCEDURES.

(1) Pull the No. 2 AWI pump circuit breaker.

(2) Remove the right hand nose baggage compartment floorboard.

(3) Remove the supply, pressure, and vent connections to the pump. If the tank has not been drained, the supply line must be capped. If the pump is to be removed for any purpose other than immediate replacement, cap all lines to prevent the entry of foreign material into the system.

(4) Disconnect the electrical connections from the pump.

(5) Remove the two screws and one bolt securing the pump and bracket in place. Remove the pump and bracket.

(6) Remove the three bolts securing the pump to the bracket and remove the pump.

(7) Remove the fittings from the pump and discard the O-ring packings.

(8) Install the pressure fitting and new O-ring (P/N MS28778-10) in the pump. Install the supply line fitting and new O-ring (P/N MS28778-10) in the pump. Install the vent line fitting and new O-ring (P/N MS28778-4) in the pump. Orient all fittings to their original positions; minor reorientation may be necessary on installation.

(9) Remove the three front plate bolts from the pump. Install the pump in its mounting bracket using the three front plate bolts. Torque the bolts to 25 inch-pounds and safety.

CAUTION: DO NOT OVERTORQUE NOR UNSYMMETRICALLY TORQUE THE BOLTS.

(10) Install the pump and bracket in the aircraft.

(11) Remove the caps from the vent, supply, and pressure lines and connect to the pump. If necessary, service the AWI tank.

(12) Hook the electrical connections to the pump. Reset the pump circuit breaker.

(13) Operate the pump and check for leaks. Make sure the No. 2 AWI pump light illuminates (indicates pump pressure). It may be necessary to loosen the pressure and/or supply and vent lines to prime the pump.

(14) Reinstall the nose baggage compartment floorboard.

AWI PUMP PRESSURE SWITCH – DESCRIPTION AND OPERATION

1. General

A. An AWI pump pressure switch is installed in the outlet side of each AWI pump. Closure of the switch completes the path necessary to illuminate the respective AWI pump light. Each switch is set to close at 30 ± 1 psi increasing pressure and open by 26 psi decreasing pressure.

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AWI PUMP PRESSURE SWITCH – FAULT ISOLATION

1. Troubleshooting – AWI Pump Pressure Switch

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No AWI pump light, pump is operating	Defective wiring between switch and cockpit	Remove electrical connector from switch, apply ground to pin B. If the annunciator AWI pump light does not illuminate, problems exist within the wiring or the annunciator panel; refer to the applicable section and troubleshoot as necessary. If the AWI pump light does illuminate, the problem is the switch or the pump output.
	Pump output is low	Install a direct reading pressure gauge in the switch mounting and read pump output pressure (43 to 48 psi with the pump deadheaded).
	Switch is defective	Replace switch.

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AWI PUMP PRESSURE SWITCH – MAINTENANCE PRACTICES

1. Servicing – AWI Pump Pressure Switch

A. No regularly scheduled maintenance is required of the pump pressure switches. During inspections, the pressure switches should be checked for security of mounting, physical damage and/or evidence of leakage.

2. Removal/Installation – AWI Pump Pressure Switch

A. Pressure Switch Removal and Replacement

(1) Gain access to the No. 1 switch by removing the left hand nose baggage compartment floorboard. Gain access to the No. 2 switch by removing the right hand nose baggage compartment floorboard.

(2) Remove the safety wiring from the electrical connector and remove the connector. Unscrew the switch from its mounting boss and remove the switch.

(3) Install a new O-ring (P/N MS28778-4) on the new switch and install in the boss. Torque the switch to 30 inch-pounds maximum.

(4) Install and safety the electrical connector to the switch.

(5) Operate the pump and check for proper operation of the corresponding annunciator light.

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AWI PUMP RELAY – DESCRIPTION AND OPERATION

1. General

A. The AWI pump relay is used to control the application of power to the AWI pump. Each AWI pump is controlled by its own relay. Circuitry within the AWI control system opens and closes the relay as necessary. Failure of the relay to the open position will not allow the pump to operate. Failure to the closed position will cause the pump to operate anytime the battery switches are placed in the ON position.

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AWI PUMP RELAY – FAULT ISOLATION

1. Troubleshooting – AWI Pump Relay

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pump will not operate	Defective wiring to pump relay	Battery switches ON; AWI selector in ARM; check for 28 volts at A1 and X1 of the relay. Remove the wire connected to X2 and check for 0 ohms resistance to ground.
	Defective relay or pump	Check for 28 volts at A2 of the relay; if voltage is not present, replace relay; if voltage is present, replace pump.

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AWI PUMP RELAY – MAINTENANCE PRACTICES

1. Servicing – AWI Pump Relay

A. No regularly scheduled maintenance is required on the pump relays. During inspections, the relays should be checked for security of mounting, physical damage and security of the electrical connections.

2. Removal/Installation – AWI Pump Relay

A. Pump Relay Removal and Replacement

(1) Remove the electrical connector to the right hand inverter and remove the inverter. The No. 1 pump relay is behind the left hand inverter and the No. 2 pump relay is behind the right hand inverter.

(2) Remove the two screws securing the pump relay to the inverter channel.

(3) Tag the electrical wires connected to the relay and disconnect the wires.

(4) Connect the electrical wires to the new relay.

(5) Install the relay in the inverter channel using the hardware removed in step 2.

(6) Reinstall the inverter. Install and safety the inverter electrical connector.

(7) Check the operation of the AWI pump and inverter system.

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AWI SHUTOFF VALVE – DESCRIPTION AND OPERATION

1. General

A. An AWI shutoff valve installed in each nacelle is used to control the flow of AWI fluid to the engine. The normally closed valve is opened by an electrical signal from the AWI control circuitry.

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AWI SHUTOFF VALVE – FAULT ISOLATION

1. Troubleshooting – AWI Shutoff Valve

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Valve will not open	Defective wiring between valve and cockpit	Place AWI selector switch in the ARM position; move the respective power lever past flight idle; 28 volts should be present at pin A of the valve connector.
	Defective valve	Replace valve

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AWI SHUTOFF VALVE – MAINTENANCE PRACTICES

1. Servicing – AWI Shutoff Valve

A. No regularly scheduled maintenance is required of the AWI shutoff valves. During inspections, the shutoff valves should be checked for security of mounting, physical damage and/or evidence of leakage.

2. Removal/Installation – AWI Shutoff Valve

A. Shutoff Valve Removal and Replacement

(1) Gain access to the shutoff valve through the access door on the left hand side of the nacelle just aft of the firewall.

(2) Remove the AWI hose from the firewall fitting on the forward side of the firewall. Remove the supply line from the aft side of the valve.

(3) Remove the nut from the firewall fitting on the forward side of the firewall. Withdraw the valve through the access door opening and remove the electrical connector.

(4) Remove the inlet and outlet fittings from the valve and discard the O-rings.

(5) Install the fittings into the new valve using two new O-rings (P/N MS28778-10). Torque the fittings enough to seat the O-rings (70 inch-pounds maximum).

(6) Install and safety the electrical connector to the valve; insert the valve through the access door opening.

(7) Connect the inlet hose to the valve and slide the outlet fitting through the firewall hole. Install the nut on the outlet fitting.

(8) Tighten the outlet fitting nut and inlet hose connection. Install and tighten the engine AWI hose to the outlet fitting.

(9) Place the AWI selector switch in ARM and allow the system to build pressure. Move the respective power lever past the flight idle gate. The shutoff valve should open.

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AWI TORQUE SWITCH – DESCRIPTION AND OPERATION

1. General

A. An AWI torque switch is installed on each engine to sense the power output of the engine and supply this information to the AWI control circuitry. The switch is set to close by 22 psi increasing pressure and open by 15 ± 2 psi decreasing pressure.

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AWI TORQUE SWITCH – FAULT ISOLATION

1. Troubleshooting – AWI Shutoff Valve

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
AWI will not fire	Defective wiring to torque switch	Place AWI selector switch in ARM; power levers forward of flight idle; 28 volts should be present at pin A of the torque switch connector; pin B to ground should be 0 ohms resistance. Move selector switch to cont pos-tion; 28 volts should be present at pin C of the torque switch connector; pin B to ground should be 0 ohms resistance.
	Defective torque switch	Replace switch.

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AWI TORQUE SWITCH – MAINTENANCE PRACTICES

1. Servicing – AWI Torque Switch

A. No regularly scheduled maintenance is required for the AWI torque switches. During inspections, the switches should be checked for security, leakage, and condition of the electrical connection to the switch.

2. Removal/Installation – AWI Torque Switch

A. Torque Switch Removal and Replacement

(1) Gain access to the switch by removing the right hand side cowl from the engine.

(2) Cut the safety wire and remove the electrical connector to the switch.

(3) Unscrew the switch from its mounting boss.

(4) Install a new O-ring (P/N MS29561-012) on the new torque switch. Install the new switch in its mounting boss and torque to 30 inch-pounds maximum.

(5) Install and safety the electrical connector to the switch.

(6) Operate and test the system in accordance with the Airplane Flight Manual instructions.

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ANNUNCIATOR PANEL – MAINTENANCE PRACTICES

1. Inspection/Check – Annunciator Panel

A. Fairchild Aircraft Corporation recommends the annunciator be replaced as an assembly. The following test will determine if the problem is with the aircraft wiring or within the annunciator panel.

(1) Remove the two plugs from the back of the annunciator panel. The small plug contains 37 pins; the large plug contains 50 pins.

(2) With the battery switches ON, measure 28 volts at pins 1, 35, and 29 of the small connector.

(3) Turn the battery switches OFF, and install the small connector.

(4) Turn the battery switches ON, and push the press-to-test button on the annunciator panel. Make sure the following lights illuminate.

(a) AWI ARMED

(b) NO. 1 AWI PUMP

(c) NO. 2 AWI PUMP

(5) Using a jumper, supply a ground to the following pins of the large receptacle at the back of the annunciator panel:

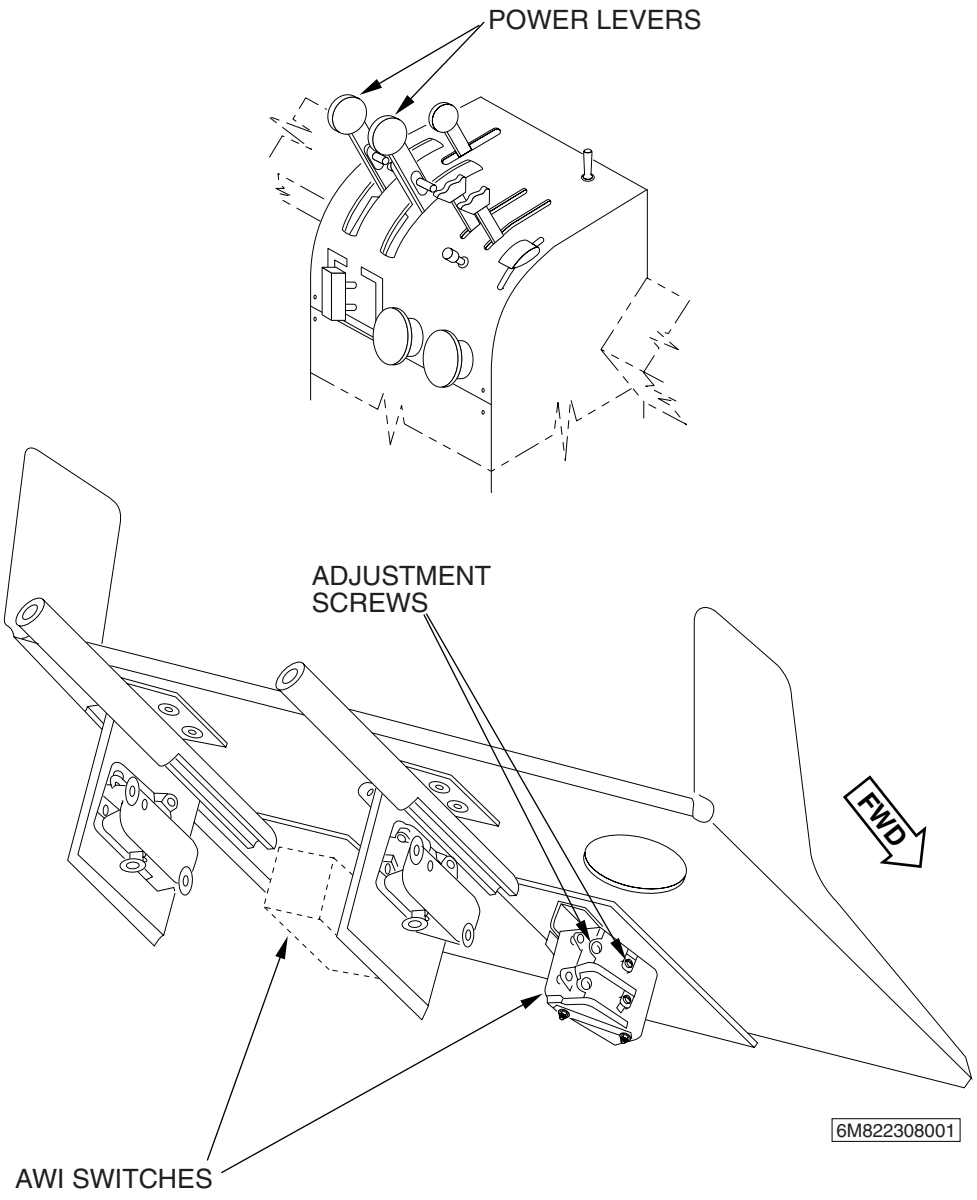
(a) Pin 40 to ground. .NO 2 AWI PUMP light illuminates.

(b) Pin 41 to ground. .AWI ARMED light illuminates.

(c) Pin 47 to ground. .NO 1 AWI PUMP light illuminates.

NOTE: If the problem is within the annunciator, the annunciator assembly should be replaced.

POWER LEVER POSITION SWITCHES – DESCRIPTION AND OPERATION



ADJUST SWITCHES TO ACTUATE WHEN THE POWER LEVERS ARE 1/8 INCH BEYOND FLIGHT IDLE.

Power Lever Position Switch Adjustment Figure 1

POWER LEVER POSITION SWITCHES – DESCRIPTION AND OPERATION

1. General

A. Two normally open microswitches are installed in the cockpit pedestal. Each switch is actuated to the closed position by the respective power lever. A slotted mounting plate is installed to allow adjustment of the switches.

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POWER LEVER POSITION SWITCHES – FAULT ISOLATION

1. Troubleshooting – Power Lever Position Switches

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
AWI will not fire	Power lever switch not closing	Check for bleed air shut off with selector switch in ARM and respective power lever moved beyond flight idle.
AWI fires without power lever beyond flight idle	Shorted bleed air isolation diode	Move bleed air switch to OFF while AWI is firing; if AWI stops, diode is shorted.

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POWER LEVER POSITION SWITCHES – MAINTENANCE PRACTICES

1. Servicing – Power Lever Position Switches

A. No regularly scheduled maintenance is required of the power lever switches. During routine inspections, the switches should be inspected for condition and security.

2. Adjustment/Test – Power Lever Position Switches

A. Power Lever Position Switch Adjustment

(1) Gain access to the switches either by removing the forward center pedestal access panel or by working from the back of the pedestal.

(2) Adjust the respective switch to actuate when the power lever is 1/8 inch beyond the FLIGHT IDLE gate.

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DISTRIBUTION SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The corrosive properties of the alcohol water mixture require that all components of the AWI system be manufactured of materials that do not readily corrode. The part number of some valves and fittings used in the system contain prefixes and suffixes indicating that special materials have been used in the part. Exact part number replacements must be in order to maintain the corrosion resistive properties of the AWI system. Lines, tubing, and connections may be purchased or locally manufactured.

B. Hard Tubing and Lines

(1) Replace hard tubing and lines with parts manufactured of the same material (if the line is aluminum, replace it with aluminum) in accordance with the following specifications:

(a) Aluminum

Flared tubing and lines	2024-T3 QQ-A-250/5	Diameter as required
Welded tube assemblies	6061-T6 QQ-A-250/11	Diameter and wall thickness to match original tube assy.

(b) Stainless Steel

Flared tubing and lines	304SST MIL-T-6845	Diameter and wall thickness as required.
Welded tube assemblies	321SST MIL-6721	Diameter and wall thickness as required.

C. Flexible Tubing and Lines

(1) Clear flexible tubing

'TYGON'
Norton Plastic and Synthetics Division
Specification B-44-3
Diameter and wall thickness to match original part.

(2) Flexible hoses and lines (AEROQUIP TYPE)

Replace only with like part number.

D. Valves, Switches, and Components

(1) Exact part number replacement must be used during maintenance on the AWI system. All component parts that come in contact with the AWI mixture must be treated to prevent corrosion. Aluminum parts will be anodized while stainless steel parts will be passivated.

E. Maintenance, Removal, and Replacement

(1) No special precautions are required during maintenance on the AWI system. Standard aircraft practices should be used in component removal and replacement. Flexible lines should be routed to avoid low spots that may allow AWI mixture to be trapped. Particular care should be given to capping lines when components are removed as the AWI manifold outlets are small orifices and can easily be plugged by foreign material.

DISTRIBUTION SYSTEM – FAULT ISOLATION

1. Troubleshooting – Distribution System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
System will not arm	No power to AWI control relays	Check for power through selector by noting if the power lever will shut off the respective bleed air.
	Defective annunciator panel light	Troubleshoot according to section 82-23-07.
AWI will not fire to one engine (emergency mode)	No power to fire valve	Remove opposite engine torque switch electrical connector. Selector switch in ARM; power levers forward; power should be present at pin A. Check for ground at pin B.
	Defective fire valve or torque switch	Jumper pin A to B. If valve opens, replace torque switch. If valve does not open, replace valve.
AWI will not fire to one engine (continuous mode)	No power to fire valve	Remove the torque switch electrical connect on the affected engine. Selector switch on CONT; power lever forward; power should be present at pin C. Check for ground at pin B.
	Defective fire valve torque	Jumper pin B to C. If valve opens, replace torque switch. If valve does not open, replace valve.

DISTRIBUTION SYSTEM – FAULT ISOLATION

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No AWI pump light(s)	Defect pressure switch	Troubleshoot according to section 82-23-03.
	Low pump output or no pump output	Troubleshoot according to section 82-23-01 or 82-23-02.

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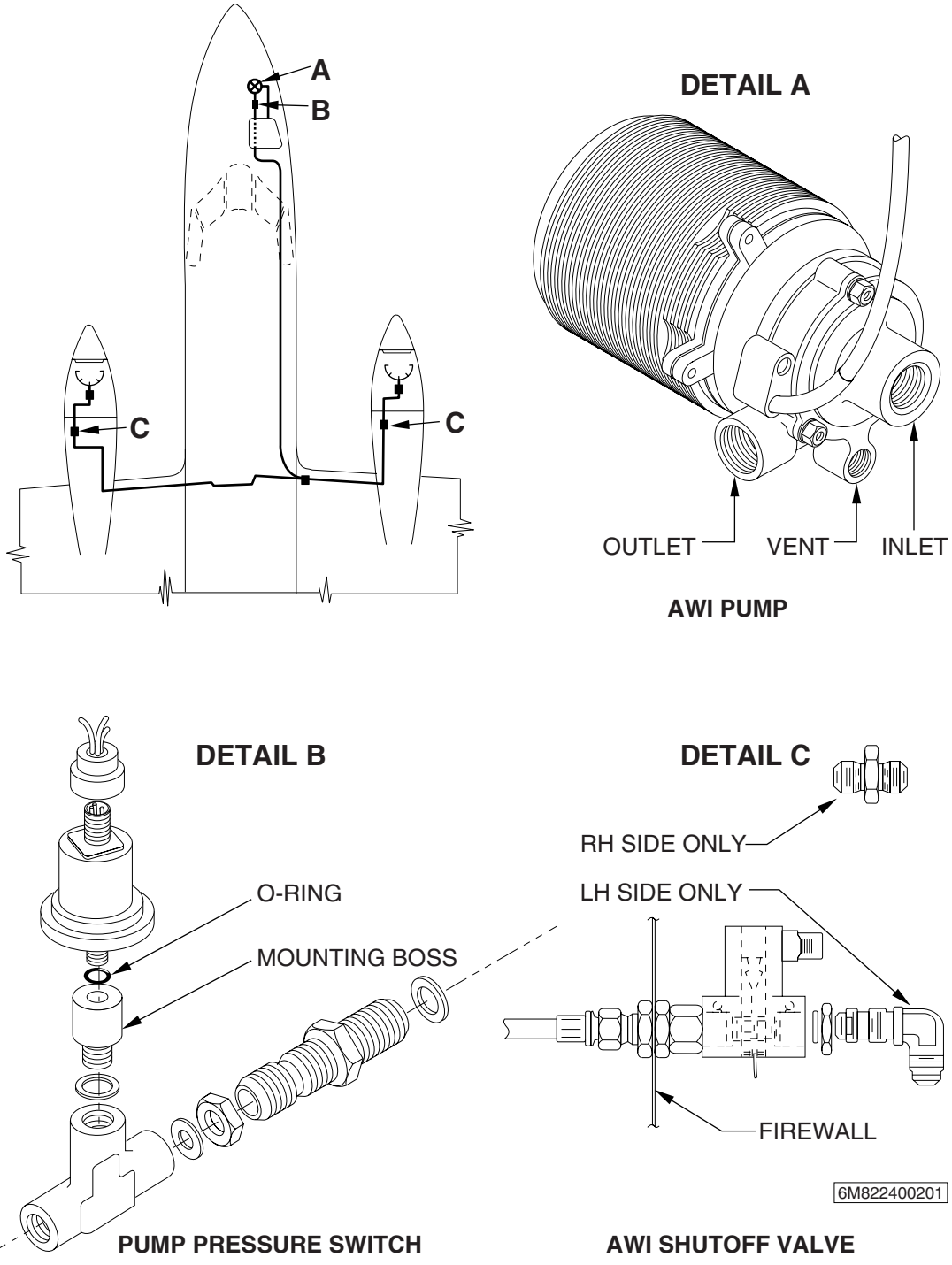


Distribution System Figure 201

1. Servicing – Distribution System

WARNING: METHANOL AND ITS VAPOR IS TOXIC AND FLAMMABLE. HANDLING MUST ALWAYS BE IN A WELL VENTILATED AREA. IN CASE OF ACCIDENTAL CONTACT FLUSH WITH WATER AT ONCE. DO NOT SMOKE, CREATE SPARKS, OR HANDLE NEAR AN OPEN FLAME.

A. Routine Filling

(1) Mix AWI fluid in accordance with the instructions in section 82-00-00.

(2) Remove the AWI filler cap, located inside the nose baggage compartment.

(3) Fill the tank to the desired level using the sight gauge to determine quantity. If desired, the tank may be filled to overflowing.

(4) Replace the fIller cap.

B. Inactive System

(1) When the system is to be left unused for extended periods of time (one month or longer):

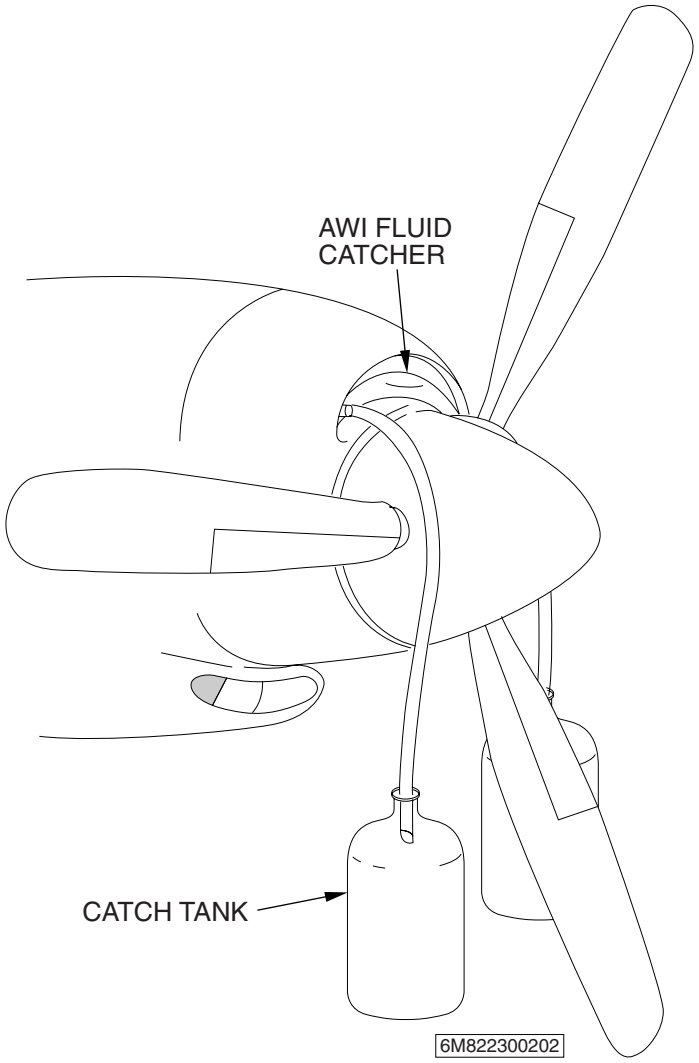
(a) Open the drain valve, located in the left hand wing-to-fuselage fairing.

(b) Open the drain valve on the right hand side of the nose compartment. The pump may be activated for quicker drainage.

(c) Close up the system and service with four gallons of distilled water.

(d) Move the aircraft to the run up area and fire the AWI into both engines for three minutes.

(e) After firing, open the drains in the nose section and left hand wing-to-fuselage fairing and allow the system to drain.



**Preparation
for Flow Check
Figure 202**

(f) Remove the left and right hand nose baggage compartment floorboards.

(g) Disconnect the pump inlet lines and allow to drain.

(h) Allow the system to air dry and then reassemble.

C. Reactivation of system

(1) Prior to placing the system into operation, perform the following steps:

(a) Service the tank with fluid and operate both pumps.

(b) Inspect the system and repair any leaks or defects.

(c) Move the aircraft to run up area and fire the AWI into each engine IAW the Airplane Flight Manual instructions.

(d) When the operation is satisfactory and the system is free from leaks, the system may be returned to service.

2. Inspection/Check – Distribution System

A. Visually inspect the system for chafing, corrosion, leakage, and security.

B. Every 9,000 hours, or each nine years, perform the following:

(1) Disassemble the system and perform a visual inspection of the "wet" components of the system IAW steps **2.**, B., (1), (a) through (d).

(a) Remove the left and right hand nose baggage compartment floorboards. Remove the outlet and inlet lines from each AWI pump. Inspect the lines and pumps for evidence of corrosion.

(b) Remove the right hand forward wing-to-fuselage fairing. Remove the AWI supply hose from the distribution "t" and inspect both fittings.

(c) Remove the AWI line from each engine's metering valve and inspect the line and valve for evidence of corrosion.

(d) Reassemble and service the system. Test the system IAW the Airplane Flight Manual instructions.

C. System Calibration

(1) The system is calibrated by flowing AWI fluid into each engine for a measured period of time and measuring the flow in fluid ounces. In order to accurately measure the flow of fluid into the engine it will be necessary to remove the engine nose cowl and install the AWI manifold over a bucket or other container for catching the fluid.

(2) As an alternate, section 82-SERVC provides a catcher that may be locally manufactured and used in the engine inlet to catch the AWI mixture and direct it into a bucket or other container.

Aircraft Serial Number _____ Date: _____

		LEFT	RIGHT	
DESIRED FLOW RATE (PPH)	MAXIMUM		950	950
	MINIMUM		780	780
BOTH ENGINES	FLOW	(ounces)		
	TIME	(seconds)		
	RATE	(pph)*		
LH ENGINE ONLY	FLOW	(ounces)		
	TIME	(seconds)		
	RATE	(pph)*		
RH ENGINE ONLY	FLOW	(ounces)		
	TIME	(seconds)		
	RATE	(pph*		

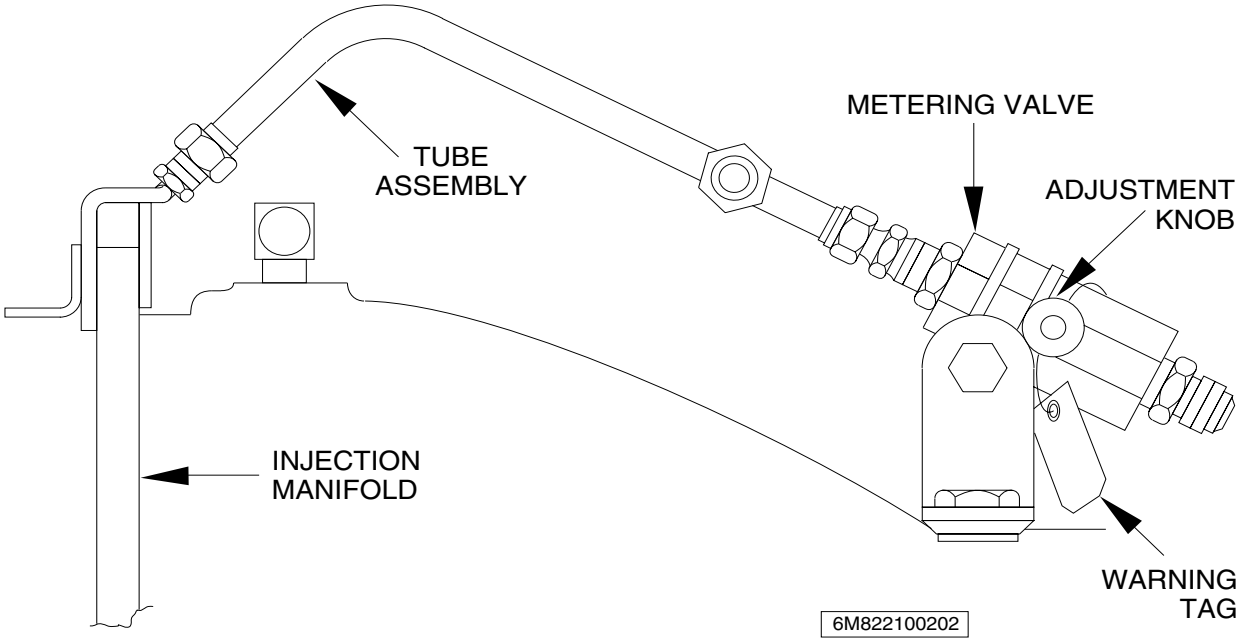
*Determine flow rate using the following formula:

$$\frac{225 \times \text{FLOW (ounces)}}{\text{TIME (seconds)}}$$

EXAMPLE: $\frac{225 \times 256}{65} = 886.1$

Claibration Data Sheet
Table 201

(3) System calibration and flow check will be necessary if either of the AWI metering valves, flow manifolds, or pumps are replaced. In the case of a metering valve or manifold, the new components must be adjusted and calibrated as necessary to achieve the required flow limits. If the pump is replaced, the flow through the system must be checked to see if the required limits are met. If the flow is within the required limits, no adjustment to the system will be necessary.



Metering Valve Assembly
Figure 203

D. Required Equipment

(1) Stop watch.

(2) Measuring equipment (graduated container of 32 fluid ounces minimum capacity).

(3) Source of 28 VDC power.

(4) Calibration data sheet (Table 201).

E. Preparation

(1) Install liquid catchers in the inlet of each engine as shown in Figure 202. Two five gallon bottles may be used to catch the mixture.

NOTE: In lieu of the liquid catcher, the nose cowling may be removed and the manifold directed into a bucket or other suitable container.

(2) Fill the tank with AWI fluid IAW instructions in section 82-00-00.

NOTE: While performing the flow check procedure, monitor the fluid level in the AWI tank. Replenish when the level reaches approximately one-half.

(3) Connect a well regulated source of 28 VDC electrical power to the aircraft. If a power source of sufficient capacity is not available, a smaller source may be connected directly to the pump relay. Voltage fluctuations, as are common with standard APU units, will cause the flow to vary and make precise adjustment difficult.

F. Flow Check Procedure

(1) Place the AWI selector switch in the ARM position.

(2) Observe the AWI pump lights are illuminated.

(3) Advance the both hand power lever until firing switch closes (switch can be heard to click). Start timer.

(4) After 60 seconds, return the AWI selector switch to the OFF position.

(5) Measure the amount of liquid collected from the left engine and compute the flow from the information supplied in Table 201. The flow rate must be 780 PPH minimum per engine.

(6) If the flow rate is less than 780 PPH, manifold calibration must be accomplished as outlined in **2.**, G. If the flow is 780 PPH or greater, proceed to step **2.**, F., (7).

(7) Retard both power levers and place the AWI selector switch in the ARM position. Observe the AWI pump lights illuminated.

(8) Advance the right hand power lever until the firing switch closes (switch can be heard to click). Start timer.

(9) After 60 seconds, return the AWI selector switch to the OFF position.

(10) Measure the amount of liquid collected from the left engine and compute the flow from the information supplied in Table 201. The flow rate must not be over 950 PPH.

(11) If the flow rate is over 950 PPH, manifold calibration must be accomplished as outlined in **2.**, G. If the flow rate is 950 PPH or less, proceed to step **2.**, G., (12).

(12) Retard the left hand power lever and place the AWI selector switch in the ARM position. Observe the AWI pump light illuminated.

(13) Advance right power lever until the firing switches close (switches can be heard to click). Start timer.

(14) After 60 seconds, return the AWI switch to the OFF position.

(15) Measure the amount of liquid collected from each engine and compute the flow rate from Table 201. The flow rate must not be over 950 PPH.

(16) If the flow rate is more than 950 PPH, manifold calibration must be accomplished as outlined in **2.**, G. If the rate is 950 or less, the system is within limits and no further action is required.

G. Manifold Calibration

(1) Place the AWI selector switch in the ARM position. Observe the AWI pump lights illuminated.

(2) Advance the power lever corresponding to the manifold to be calibrated. (left hand manifold, advance left hand power lever) until the firing switch closes (switch can be heard to click). Start timer.

(3) After 60 seconds, return the AWI selector switch to the OFF position.

(4) Measure the amount of liquid collected from the engine and compute the flow rate from the information supplied in Table 201.

(5) Adjust the engine installed metering valve open or closed as necessary to adjust the flow to near, but not over 950 PPH.

(6) Repeat steps **2.**, G. for the opposite engine.

(7) After both engines metering valves have been adjusted, move the AWI selector switch to ARM. Advance both power levers until the firing switches close. Start timer.

(8) After 60 seconds, return the AWI selector switch to the OFF position.

(9) Measure the amount of liquid collected from each engine and compute the flow from the information supplied in Table 201. The flow must be 780 PPH minimum per engine. If the flow is not met, then the adjustments made in steps **2.**, G., (1) through (5) must be repeated.

(10) After each engine's flow rate is within limits, tighten the jam nut on the valve stem. Pass a length of safety wire through the roll pin in the valve knob and warning tag. Wrap the safety wire around the body of the valve and twist.

AWI PUMP – DESCRIPTION AND OPERATION

1. General

A. The AWI pump is used to provide pressurized water-methanol for delivery to the engine's. The pump is installed in a mounting bracket located beneath the right hand nose baggage compartment floorboard. The pump is a centrifugal type with no set displacement. Output pressure and flow will vary with the output load applied to the pump. Pump pressure, running deadheaded at 28.5 volts, should be 43 to 48 psi.

AWI PUMP – FAULT ISOLATION

1. Troubleshooting – No. 1 AWI Pump

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pump is noisy	Defective pump	Replace pump.
Pump output is low as indicated by lack of AWI pump light illumination	Defective pump	Install a pressure gauge in the pressure switch mounting boss. If the pressure output is low, make sure no leaks exist in the system before replacing the pump.
Pump output is high, as indicated by difficulty during system calibration	Mounting bolts to pump bracket overtorqued	Loosen the three mounting bolts securing the pump to the bracket and retorque to no more than 25 inch-pounds. If the output remains high, replace the pump.
Pump inoperative	Defective pump relay	Troubleshoot relay according to section 82-24-03.
	Defective wiring	Troubleshoot wiring according to applicable wiring diagram.
	Defective pump	Replace pump.

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AWI PUMP – MAINTENANCE PRACTICES

1. Servicing – AWI Pump

A. No regularly scheduled maintenance is required of the AWI pump. During inspections, the pump should be checked for security of mounting, physical damage and/or evidence of leakage. The AWI pump is life limited in some installations; refer to section 5-50-82 for specified limits.

2. Removal/Installation – AWI Pump

A. Pump removal and replacement

CAUTION: IF THE AWI PUMP IS REPLACED, THE SYSTEM FLOW RATE MUST BE FLOW CHECKED AND CALIBRATED IF NECESSARY. REFER TO SECTION 82-24-00 FOR CALIBRATION PROCEDURES.

(1) Pull the AWI pump circuit breaker.

(2) Remove the right hand nose baggage compartment floorboard.

(3) Remove the supply, pressure, and vent connections to the pump. If the tank has not been drained, the supply line must be capped. If the pump is to be removed for any purpose other than immediate replacement, cap all lines to prevent the entry of foreign material into the system.

(4) Disconnect the electrical connections from the pump.

(5) Remove the two screws and one bolt securing the pump and bracket in place. Remove the pump and bracket.

(6) Remove the three bolts securing the pump to the bracket and remove the bracket.

(7) Remove the fittings from the pump and discard the O-ring packings.

(8) Install the pressure fitting and new O-ring (P/N MS28778-10) in the pump. Install the supply line fitting and new O-ring (P/N MS28778-10 in the pump. Install the vent line fitting and new O-ring (P/N MS28778-4 in the pump. Orient all fittings to their original positions; minor reorientation may be necessary on installation.

(9) Remove the three front plate bolts from the pump. Install the pump mounting bracket using the three front plate bolts. Torque the bolts to 25 inch-pounds and safety.

CAUTION: DO NOT OVERTORQUE NOR UNSYMMETRICALLY TORQUE THE BOLTS.

(10) Install the pump and bracket in the aircraft.

(11) Remove the caps from the vent, supply, and pressure lines and connect to the pump. If necessary, service the AWI tank.

(12) Hook the electrical connections to the pump. Reset the pump circuit breaker.

(13) Operate the pump and check for leaks. Make sure the AWI armed light illuminates (indicates pump pressure). It may be necessary to loosen the pressure and/or supply and vent lines to prime the pump.

(14) Reinstall the nose baggage compartment floorboard.

AWI PUMP PRESSURE SWITCH – DESCRIPTION AND OPERATION

1. General

A. An AWI pump pressure switch is installed in the outlet side of the AWI pump. Closure of the switch completes the path necessary to illuminate the AWI pump light. The switch is set to close at 30 ± 1 psi increasing pressure and open by 26 psi decreasing pressure.

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AWI PUMP PRESSURE SWITCH – FAULT ISOLATION

1. Troubleshooting – AWI Pump Pressure Switch

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No AWI pump light, pump is operating	Defective wiring between switch and cockpit	Remove electrical connector from switch, apply ground to pin B. If the annunciator AWI pump light does not illuminate, problems exist within the wiring or the annunciator panel; refer to the applicable section and troubleshoot as necessary. If the AWI pump light does illuminate, the problem is in the switch or pump output.
	Pump output is low	Install a direct reading pressure gauge in the switch mounting and read pump output pressure (43 to 48 psi with the pump deadheaded).
	Switch is defective	Replace switch.

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AWI PUMP PRESSURE SWITCH – MAINTENANCE PRACTICES

1. Servicing – AWI Pump Pressure Switch

A. No regularly scheduled maintenance is required of the pump pressure switch. During inspections, the pressure switch should be checked for security of mounting, physical damage and/or evidence of leakage.

2. Removal/Installation – AWI Pump Pressure Switch

A. Pressure Switch Removal and Replacement

(1) Gain access to the switch by removing the right hand nose baggage compartment floorboard.

(2) Remove the safety wiring from the electrical connector and remove the connector. Unscrew the switch from its mounting boss and remove the switch.

(3) Install a new O-ring (P/N MS28778-4) on the new switch and install in the boss. Torque the switch to 30 inch-pounds maximum.

(4) Install and safety the electrical connector to the switch.

(5) Operate the pump and check for proper operation of the annunciator light.

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AWI PUMP RELAY – DESCRIPTION AND OPERATION

1. General

A. The AWI pump relay is used to control the application of power to the AWI pump. Circuitry within the AWI control system opens and closes the relay. Failure of the relay to the open position will not allow the pump to operate. Failure to the closed position will cause the pump to operate anytime the battery switches are placed in the ON position.

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AWI PUMP RELAY – FAULT ISOLATION

1. Troubleshooting – AWI Pump Relay

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Pump will not operate	Defective wiring to pump relay	Battery switches ON; AWI selector in ARM; check for 28 volts at A1 and X1 of the relay. Remove the wire connected to X2 and check for 0 ohms resistance to ground.
	Defective relay or defective pump	Check for 28 volts at A2 of the relay; if voltage is not present, replace relay; if voltage is present, replace pump.

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AWI PUMP RELAY – MAINTENANCE PRACTICES

1. Servicing – AWI Pump Relay

A. No regularly scheduled maintenance is required on the pump relay. During inspections, the relay should be checked for security of mounting, physical damage and security of the electrical connections to the relay.

2. Removal/Installation – AWI Pump Relay

A. Pump Relay Removal and Replacement

(1) Remove the electrical connector to the right hand inverter and remove the inverter.

(2) Remove the two screws securing the pump relay to the inverter channel.

(3) Tag the electrical wires connected to the relay and disconnect the wires.

(4) Connect the electrical wires to the new relay.

(5) Install the relay in the inverter channel using the hardware removed in step 2.

(6) Reinstall the inverter. Install and safety the inverter electrical connector.

(7) Check the operation of the AWI pump and inverter system.

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AWI SHUTOFF VALVE – DESCRIPTION AND OPERATION

1. General

A. An AWI shutoff valve installed in each nacelle is used to control the flow of AWI fluid to the engine. The normally closed valve is opened by an electrical signal from the AWI control circuitry.

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AWI SHUTOFF VALVE – FAULT ISOLATION

1. Troubleshooting – AWI Shutoff Valve

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Valve will not open	Defective wiring between valve and cockpit	Place AWI selector switch in the ARM position; move the respective power lever past flight idle; 28 volts should be present at pin A of the valve connector.
	Defective valve	Replace valve

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AWI SHUTOFF VALVE – MAINTENANCE PRACTICES

1. Servicing – AWI Shutoff Valve

A. No regularly scheduled maintenance is required of the AWI shutoff valves. During inspections, the shutoff valves should be checked for security of mounting, physical damage and/or evidence of leakage.

2. Removal/Installation – AWI Shutoff Valve

A. Shutoff Valve Removal and Replacement

(1) Gain access to the shutoff valve through the access door on the left hand side of the nacelle just aft of the firewall.

(2) Remove the AWI hose from the firewall fitting on the forward side of the firewall. Remove the supply line from the aft side of the valve.

(3) Remove the nut from the firewall fitting on the forward side of the firewall. Withdraw the valve through the access door opening and remove the electrical connector.

(4) Remove the inlet and outlet fittings from the valve and discard the O-rings.

(5) Install the fittings into the new valve using two new O-rings (P/N MS28778-10). Torque the fittings enough to seat the O-rings (70 inch-pounds maximum).

(6) Install and safety the electrical connector to the valve; insert the valve through the access door opening.

(7) Connect the inlet hose to the valve and slide the outlet fitting through the firewall hole. Install the nut on the outlet fitting.

(8) Tighten the outlet fitting nut and inlet hose connection. Install and tighten the engine AWI hose to the outlet fitting.

(9) Place the AWI selector switch in ARM and allow the system to build pressure. Move the respective power lever past the flight idle gate. The shutoff valve should open.

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AWI TORQUE SWITCH – DESCRIPTION AND OPERATION

1. General

A. An AWI torque switch is installed on each engine to sense the power output of the engine and supply this information to the AWI control circuitry. The switch is set to close by 22 psi increasing pressure and open by 15 ± 2 psi decreasing pressure.

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AWI TORQUE SWITCH – FAULT ISOLATION

1. Troubleshooting – AWI Shutoff Valve

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
AWI will not fire	Defective wiring to torque switch	Place AWI selector switch in ARM; power levers forward of flight idle; 28 volts should be present at pin A of the torque switch connector; pin B to ground should be 0 ohms resistance. Move selector switch to cont pos-tion; 28 volts should be present at pin C of the torque switch connector; pin B to ground should be 0 ohms resistance.
	Defective torque switch	Replace switch.

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AWI TORQUE SWITCH – MAINTENANCE PRACTICES

1. Servicing – AWI Torque Switch

A. No regularly scheduled maintenance is required for the AWI torque switches. During inspections, the switches should be checked for security, leakage, and condition of the electrical connection to the switch.

2. Removal/Installation – AWI Torque Switch

A. Torque Switch Removal and Replacement

(1) Gain access to the switch by removing the right hand side cowl from the engine.

(2) Cut the safety wire and remove the electrical connector to the switch.

(3) Unscrew the switch from its mounting boss.

(4) Install a new O-ring (P/N MS29561-012) on the new torque switch. Install the new switch in its mounting boss and torque to 30 inch-pounds maximum.

(5) Install and safety the electrical connector to the switch.

(6) Operate and test the system in accordance with the Airplane Flight Manual instructions.

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ANNUNCIATOR PANEL – MAINTENANCE PRACTICES

1. Inspection/Check – Annunciator Panel

A. Fairchild Aircraft Corporation recommends the annunciator be replaced as an assembly. The following test will determine if the problem is with the aircraft wiring or within the annunciator panel.

(1) Remove the two plugs from the back of the annunciator panel. The small plug contains 37 pins; the large plug contains 50 pins.

(2) With the battery switches ON, measure 28 volts at pins 1, 35, and 29 of the small connector.

(3) Turn the battery switches OFF, and install the small connector.

(4) Turn the battery switches ON, and push the press-to-test button on the annunciator panel. Make sure the following lights illuminate.

(a) AWI PUMP

(b) AWI ARMED

(c) AWI LOW WATER

(5) Using a jumper, supply a ground to the following pins of the large receptacle at the back of the annunciator panel:

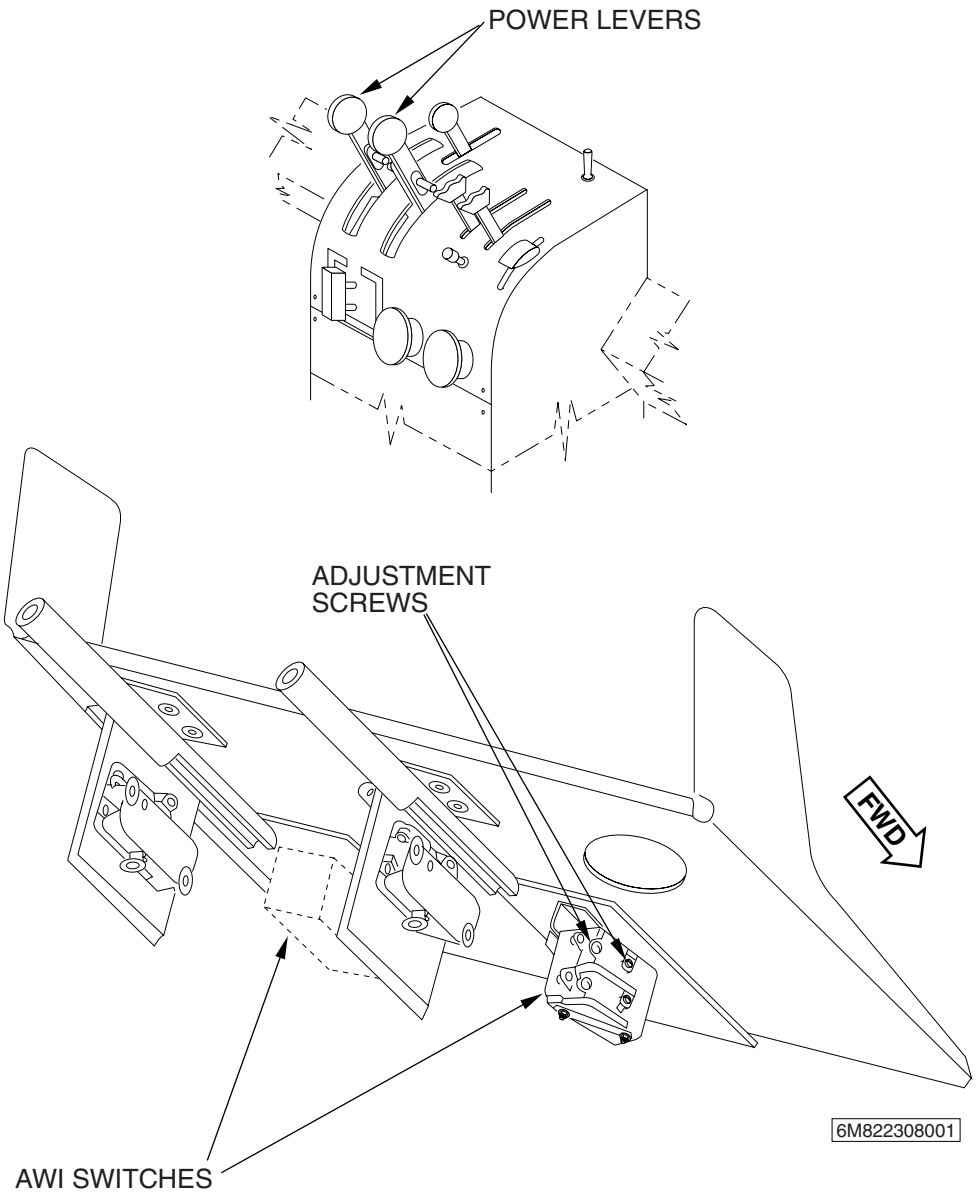
(a) Pin 40 to ground. AWI PUMP light illuminates.

(b) Pin 41 to ground. AWI ARMED light illuminates.

(c) Pin 39 to ground. AWI Low water illuminates.

NOTE: If the problem is within the annunciator, the annunciator assembly should be replaced.

POWER LEVER POSITION SWITCHES – DESCRIPTION AND OPERATION



ADJUST SWITCHES TO ACTUATE WHEN THE POWER LEVERS ARE 1/8 INCH BEYOND FLIGHT IDLE.

Power Lever Position Switch Adjustment Figure 1

POWER LEVER POSITION SWITCHES – DESCRIPTION AND OPERATION

1. General

A. Two normally open microswitches are installed in the cockpit pedestal. Each switch is actuated to the closed position by the respective power lever. A slotted mounting plate is installed to allow adjustment of the switches.

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POWER LEVER POSITION SWITCHES – FAULT ISOLATION

1. Troubleshooting – Power Lever Position Switches

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
AWI will not fire	Power lever switch not closing	Check for bleed air shut off with selector switch in ARM and respective power lever moved beyond flight idle.
AWI fires without power lever beyond flight idle	Shorted bleed air isolation diode	Move bleed air switch to OFF while AWI is firing; if AWI stops, diode is shorted.

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POWER LEVER POSITION SWITCHES – MAINTENANCE PRACTICES

1. Servicing – Power Lever Position Switches

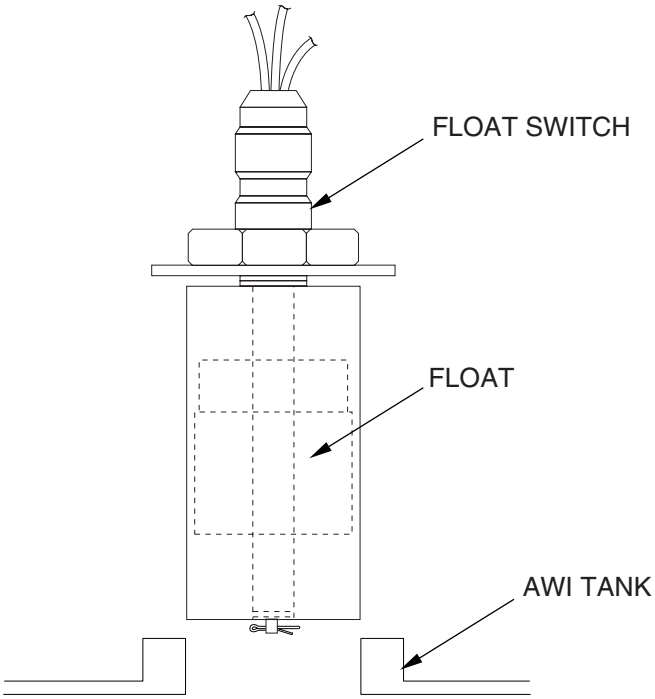
A. No regularly scheduled maintenance is required of the power lever switches. During routine inspections, the switches should be inspected for condition and security.

2. Adjustment/Test – Power Lever Position Switches

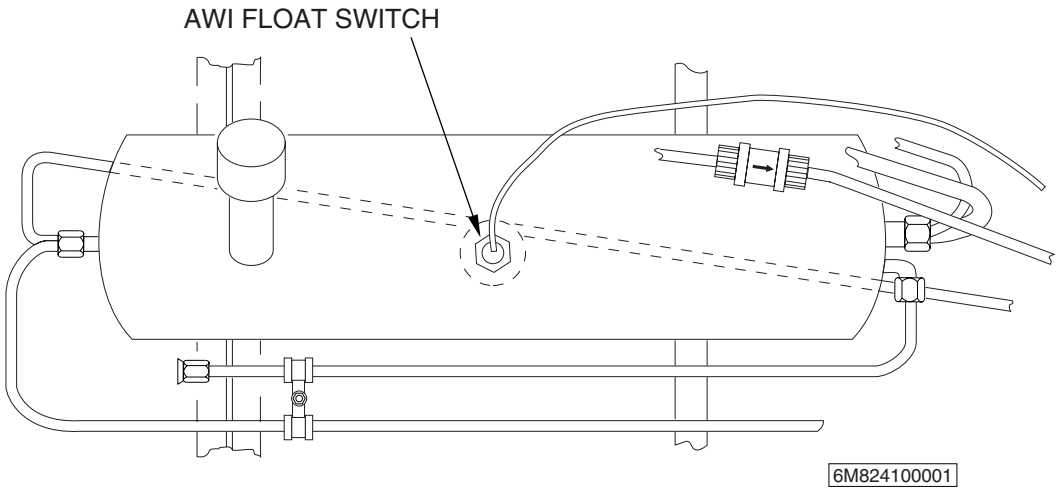
A. Power Lever Position Switch Adjustment

(1) Gain access to the switches either by removing the forward center pedestal access panel or by working from the back of the pedestal.

(2) Adjust the respective switch to actuate when the power lever is 1/8 inch beyond the FLIGHT IDLE gate.



FLOAT SWITCH - EXPLODED VIEW



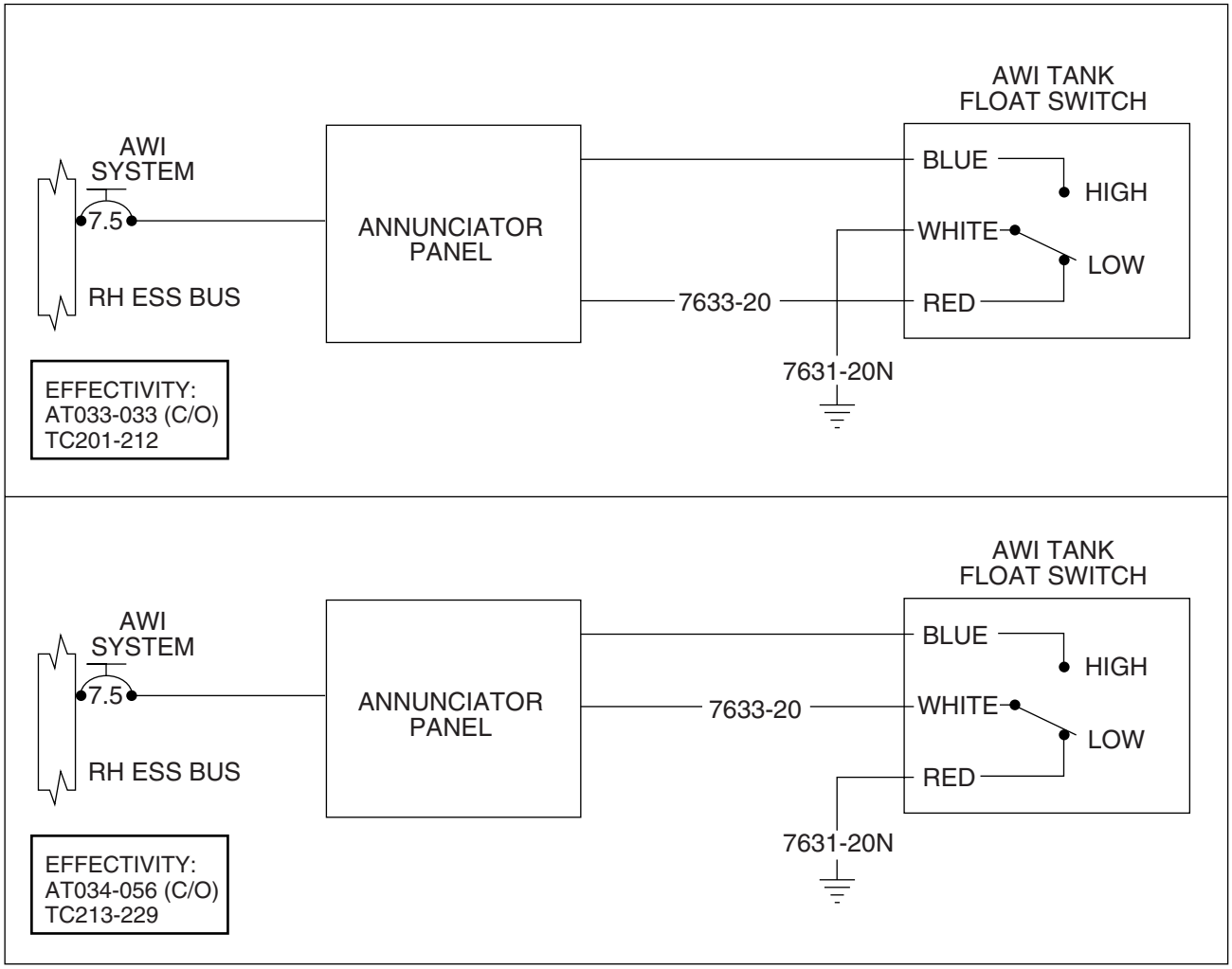
AWI Quantity Indicating System Figure 1

AWI QUANTITY INDICATING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The AWI quantity indicating system consists of a float switch and a cockpit installed warning light.

B. The low level switch is installed in the AWI tank. The switch closes and causes the annunciator LOW WATER light to illuminate when approximately 1.7 gallons of fluid remain in the AWI tank.



AWI Quantity Indicating System Schematic
Figure 100

AWI QUANTITY INDICATING SYSTEM – FAULT ISOLATION

1. Troubleshooting – AWI Quantity Indicating System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No LOW Water light (tank quantity is low)	Defective wiring between switch and cockpit	Gain access to switch electrical connections. Ground wire 7633-20. LOW WATER light should illuminate.
	No ground to switch	Check for zero ohms to ground at wire 7631-20N.
	Defective switch	Replace switch.

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AWI QUANTITY INDICATING SYSTEM – MAINTENANCE PRACTICES

1. Servicing – AWI Quantity Indicating System

A. No regularly scheduled maintenance is required of the AWI quantity indicating system. During routing inspections, all components of the system should be checked for security and condition.

B. System Calibration

(1) Remove the right hand aft fuselage-to-wing fairing.

(2) DC power must be used during the calibration; an APU will prevent excessive discharging of the aircraft batteries.

(3) Fill the AWI tank and observe the annunciator LOW WATER light off.

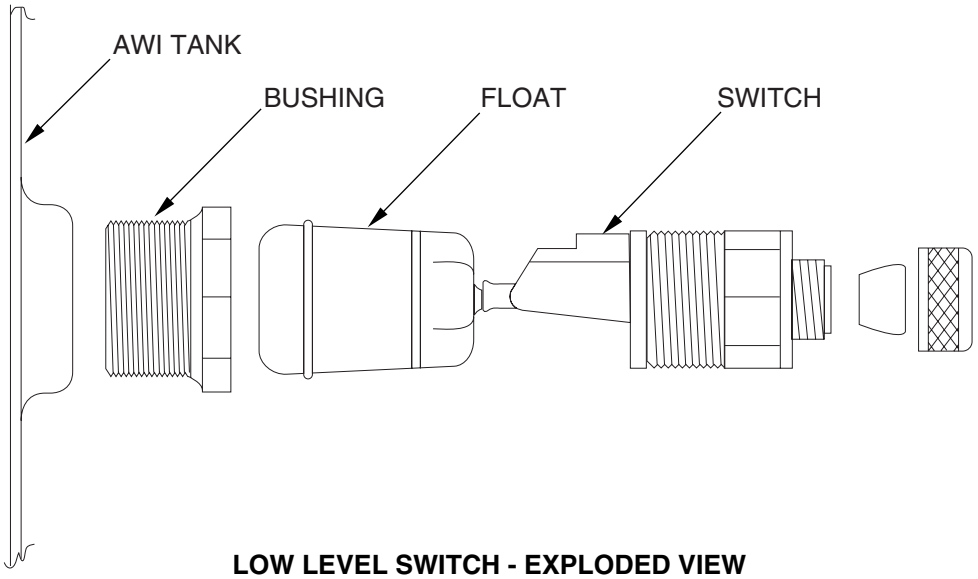
(4) Open the AWI tank drain and allow the tank to drain into a container until the LOW WATER light illuminates. An alternate method is to siphon AWI fluid out through the filler until the light illuminates.

(5) Measure the amount of fluid collected. This amount must be between 20 and 35 ounces.

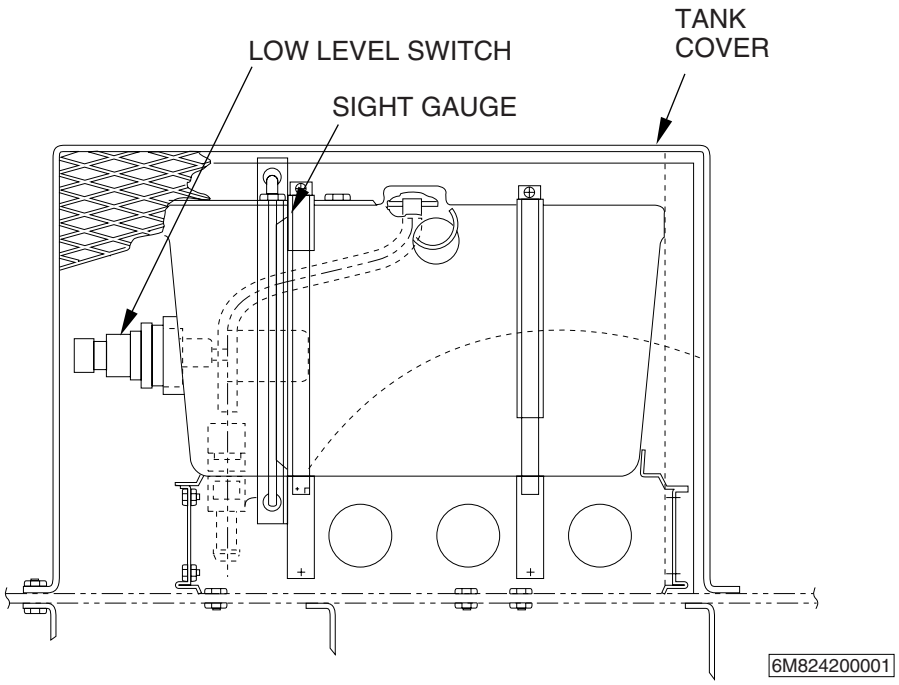
(6) If the amount is not correct, remove the float switch and adjust the height of the switch as necessary. Repeat steps 3, 4, and 5.

NOTE: The critical limit is 35 ounces maximum. If the light illuminates with less than this, the system is acceptable; however, if the light does not require approximately 20 or more ounces to illuminate, the LOW WATER light may blink during the system ground test and during climbs and descents.

(7) When the system tests within the limits specified, retorque all lines and fittings and reservice the tank. Reinstall the fuselage-to-wing fairing.



LOW LEVEL SWITCH - EXPLODED VIEW



AWI Quantity Indicating System Figure 1

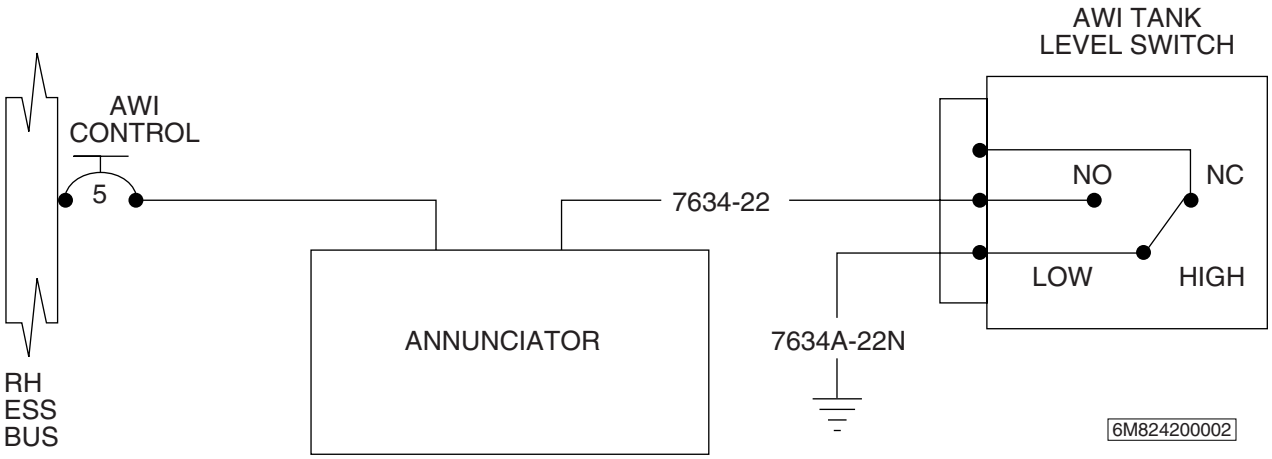
AWI QUANTITY INDICATING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The AWI quantity indicating system consists of tank level sight gauge and a low level switch with a cockpit installed warning light.

B. The low level switch is installed on the aft side of the AWI tank. The switch closes and causes the annunciator LOW WATER light to illuminate when two gallons of fluid remain in the AWI tank.

C. The level sight gauge consists of a clear plastic line routed from the bottom of the tank to the vent system. A metal scale, graduated in gallons, is attached to the outboard side of the tank assembly. The clear plastic line passes behind this scale and the quantity, in gallons, is read by comparing the level in the line to the scale graduations.



AWI Quantity Indicating System Figure 101

AWI QUANTITY INDICATING SYSTEM – FAULT ISOLATION

1. Troubleshooting – AWI Quantity Indicating System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No LOW Water light (tank quantity is low)	Defective wiring between switch and cockpit	Gain access to switch electrical connections. Ground wire 7634-22. LOW WATER light should illuminate.
	No ground to switch	Check for zero ohms to ground at wire 7634A-22N.
	Defective switch	Replace switch.

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AWI QUANTITY INDICATING SYSTEM – MAINTENANCE PRACTICES

1. Servicing – AWI Quantity Indicating System

A. No regularly scheduled maintenance is required of the AWI quantity indicating system. During routing inspections, all components of the system should be checked for security and condition. The tank level switch should be checked for evidence of leakage.

2. Removal /Installation – AWI Quantity Indicating System

A. Low Level Switch – Remove and Replace

(1) Pull the AWI system circuit breaker. Drain the AWI tank.

(2) Remove the tank shroud as necessary to gain access to the aft side of the tank

(3) Disconnect the handshake connections attaching the wires to the switch.

(4) Remove the low level switch. The switch and bushing may be removed as an assembly. As an alternate method, the access panel on top of the tank may be removed, the float unscrewed from the switch, and the switch unscrewed from the bushing. On replacing the switch, do not tightly screw the switch into the bushing without the bushing being installed in the tank. Teflon tape may be used on the threads of either the bushing or the switch.

(5) Connect the electrical leads to the switch.

(6) Check the operation of the switch by filling the tank until the LOW WATER light is extinguished. Drain fluid until the LOW WATER light illuminates. Drain and measure the remaining fluid. The amount should be two gallons.

(7) Reinstall the switch cover and the tank shroud.

3. Adjustment /Test – AWI Quantity Indicating System

A. AWI Sight Level Gauge Adjustment

(1) Fill the AWI tank until the LOW WATER light is extinguished.

(2) Drain the tank until the LOW WATER light illuminates. Stop as soon as the light illuminates.

(3) Loosen the two screws securing the calibration scale. Adjust the scale up or down until the mark corresponding to two gallons lines up with the liquid level in the clear sight tube. Tighten the screws securing the calibration scale.

(4) Service the tank as required.

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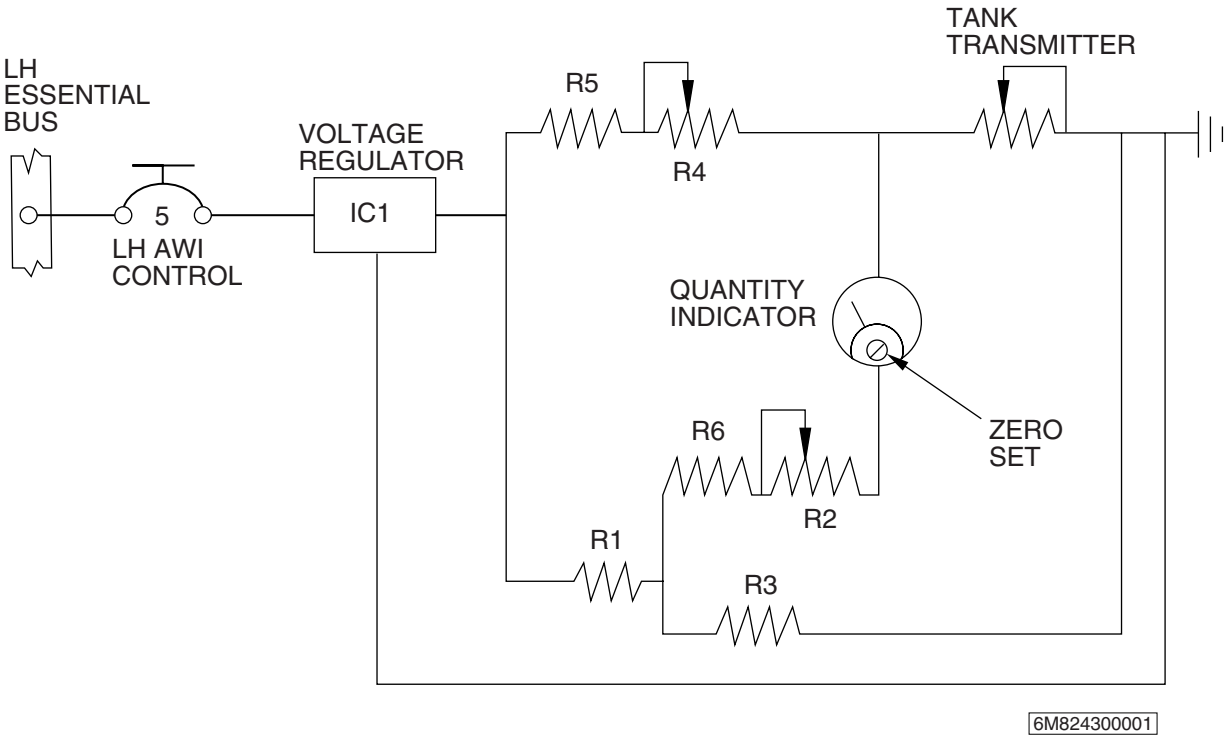


Electrical Quantity Indicating System Figure 1

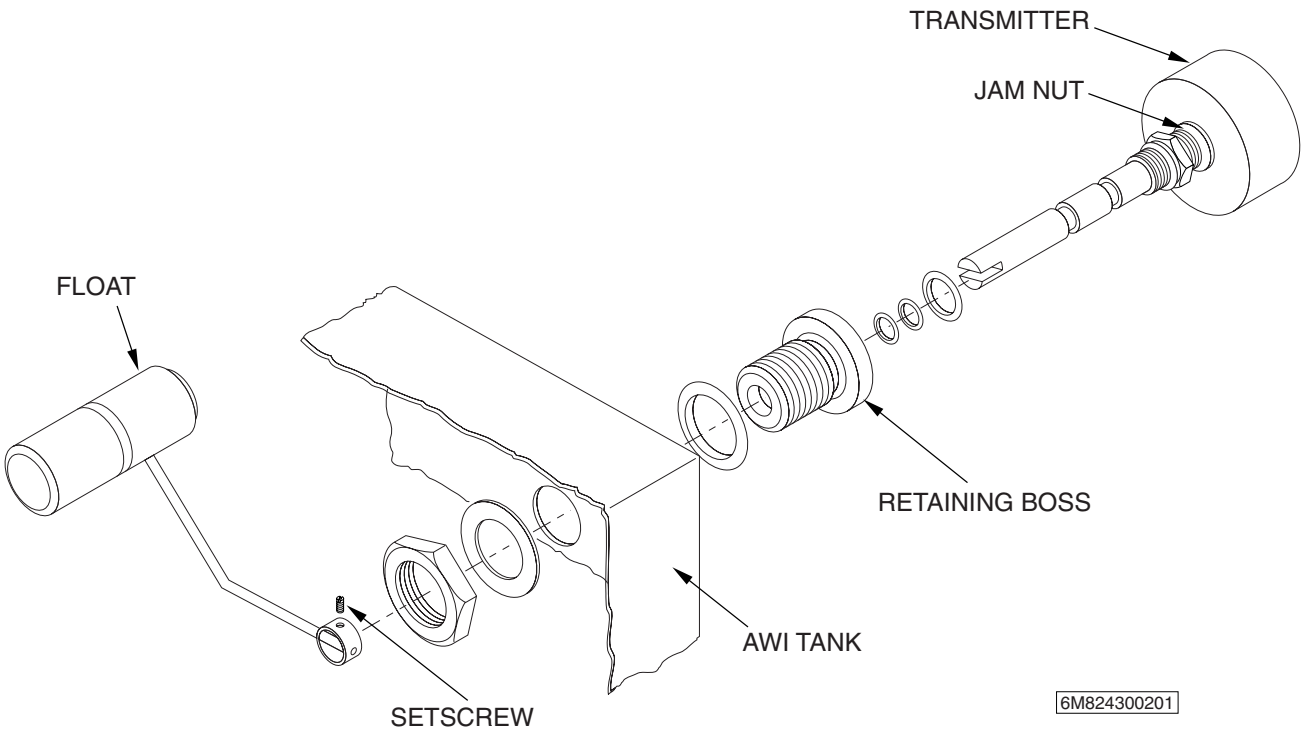
AWI QUANTITY INDICATING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The AWI quantity indicating system includes a sight gauge installed on the right hand side of the AWI tank and an electrical quantity indicating system.

B. The electrical quantity indicating system (see Figure 1) consists of a tank quantity sensing transmitter, circuitry within the AWI control box, and a cockpit installed quantity indicator. The tank transmitter is driven by a float that follows the level of the fluid in the tank. The float position is electrically converted to an indicated in gallons and displayed on the cockpit indicator. A voltage regulator is incorporated to insure that the accuracy of the indicating system will not be affected by changes in voltage.

C. The sight gauge consists of a clear plastic line routed from the bottom of the tank to the vent system. A metal scale, graduated in gallons, is attached to the right hand side of the tank. The clear plastic line passes behind this scale and the quantity, in gallons, is read by comparing the level in the line to the scale graduations.



Removal of
Quantity Transmitter
Figure 201

AWI QUANTITY INDICATING SYSTEM – MAINTENANCE PRACTICES

1. Servicing – AWI Quantity Indicating System

A. No regularly scheduled maintenance is required of the AWI quantity indicating system. During routing inspections, all components of the system should be checked for security and condition. The tank quantity transmitter should be checked for evidence of leakage.

2. Removal /Installation – AWI Quantity Indicating System

A. Tank Quantity Transmitter Removal and Replacement

(1) Drain the AWI tank. Pull the AWI system circuit breaker.

(2) Remove the shroud as necessary to gain access to the AWI tank. Remove the left hand top access cover on the AWI tank.

(3) Remove the setscrew securing the float to the transmitter (see Figure 201) and remove the float. Disconnect the handshake connectors on the electrical wiring to the transmitter.

(4) Loosen the jam nuts and unscrew the transmitter from the retaining boss.

(5) Unsolder the wires from the old transmitter and solder them to the new transmitter.

(6) Lubricate the shaft of the new transmitter with DC-4 silicone grease (DOW Chemical). Fill the cavity in the retaining boss with DC-4. Slide the transmitter through the retaining boss.

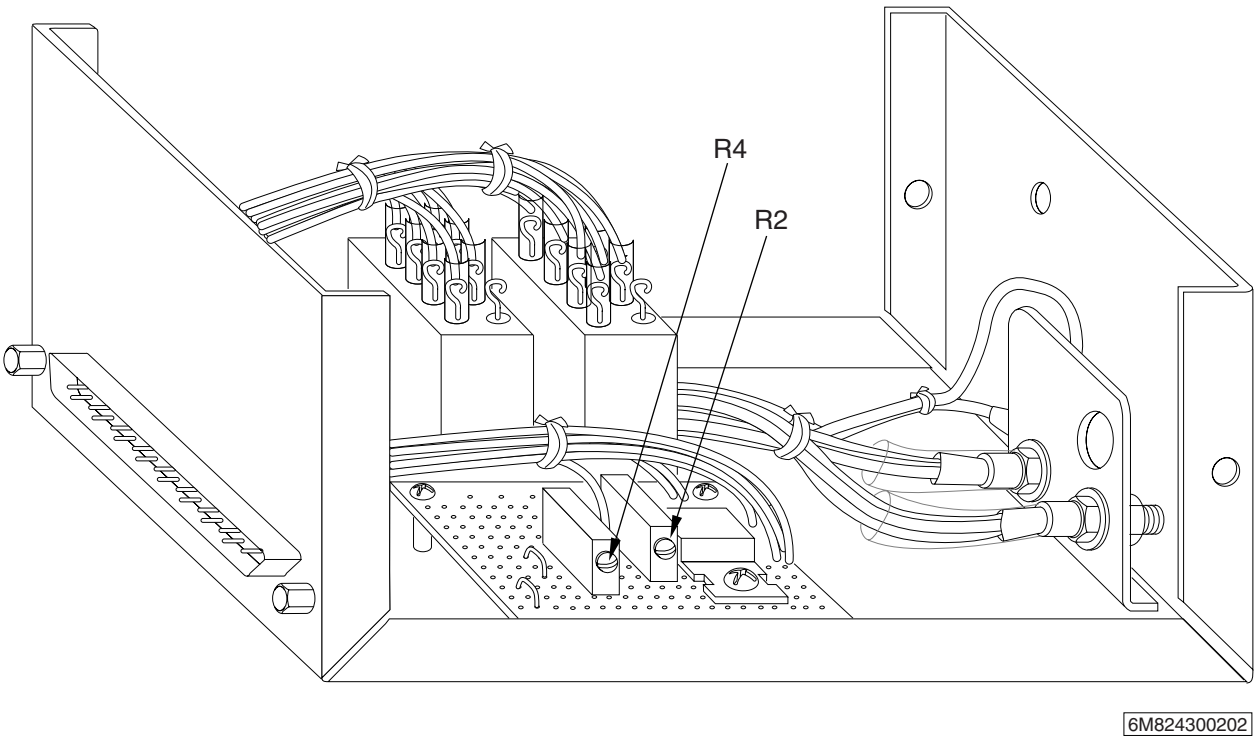
(7) Screw the transmitter into the retaining boss. Install the float on the transmitter and tighten the setscrew.

(8) Calibrate the AWI quantity indicating system.

3. Adjustment /Test – AWI Quantity Indicating System

A. System Calibration

(1) Level the aircraft



AWI Control Box
Figure 202

(2) Remove the inspection cover adjacent to the AWI tank.

(3) Place a quantity AWI fluid into the system to produce a level of 1/8 inch in the bottom of the AWI tank. If the tank has been removed or replaced, use the AWI fluid to determine if the tank is level. Minor shimming under the tank may be necessary.

(4) Place an additional 1.5 gallons of fluid in the tank. This is the zero quantity level.

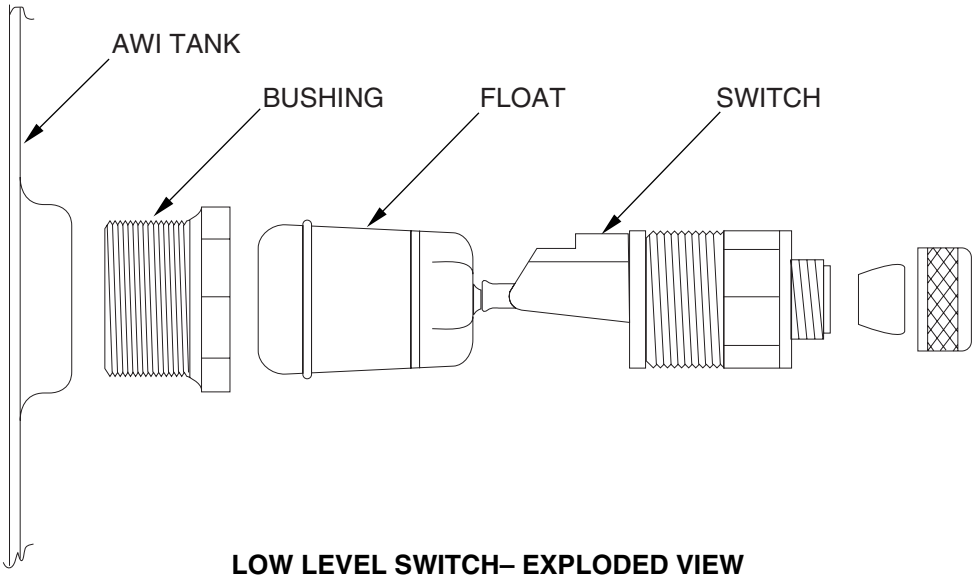
(5) Connect an ohmmeter to the transmitter. Loosen the jam nut and rotate the transmitter as necessary to achieve a resistance of 624 ohms. Secure the transmitter in place by tightening the jam nut.

(6) Check the sight gauge. If necessary, loosen the fasteners on the sight glass and line up the 0 indication mark with the level in the clear gauge tube. Tighten the fasteners to lock the sight glass in place.

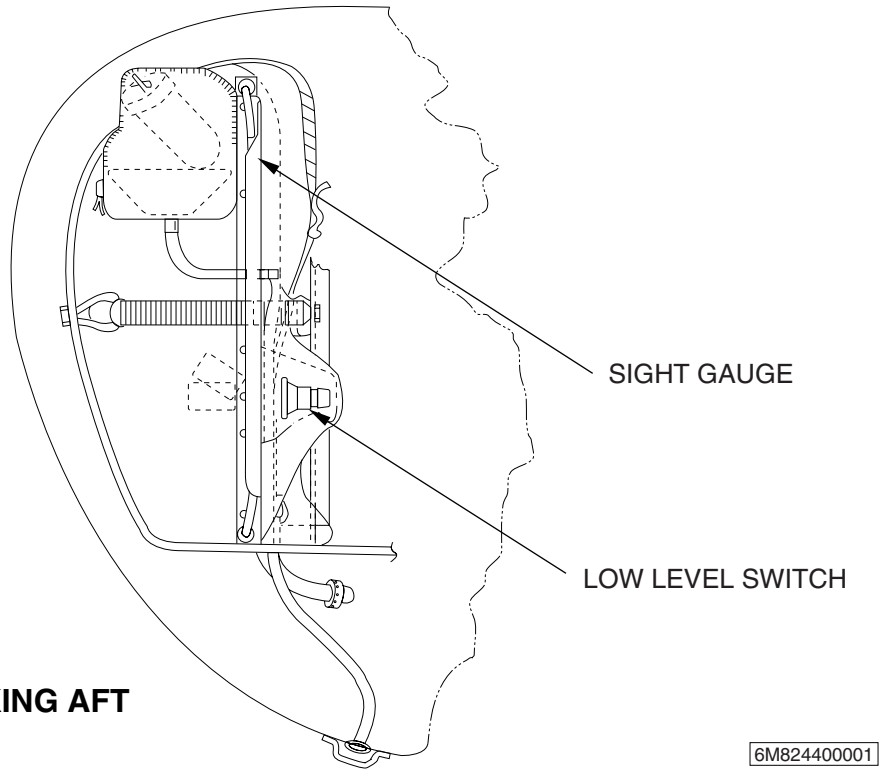
(7) Remove the ohmmeter and connect the transmitter wiring to the aircraft system. Adjust the zero set on the quantity indicator to cause the indicator to read 0 gallons.

(8) Fill the tank in two gallon increments to 14 gallons. Verify the quantity indicator readings at each two gallon increment. The reading should be within $\pm$ 0.25 gallons of the actual quantity. Minor adjustments of potentiometers R2 and R4 inside the AWI control box may be necessary (see Figure 202).

(9) Reinstall the inspection cover to the AWI tank. Torque the fasteners to 15 to 20 inch-pounds.



LOW LEVEL SWITCH– EXPLODED VIEW



AWI Quantity Indicating System Figure 1

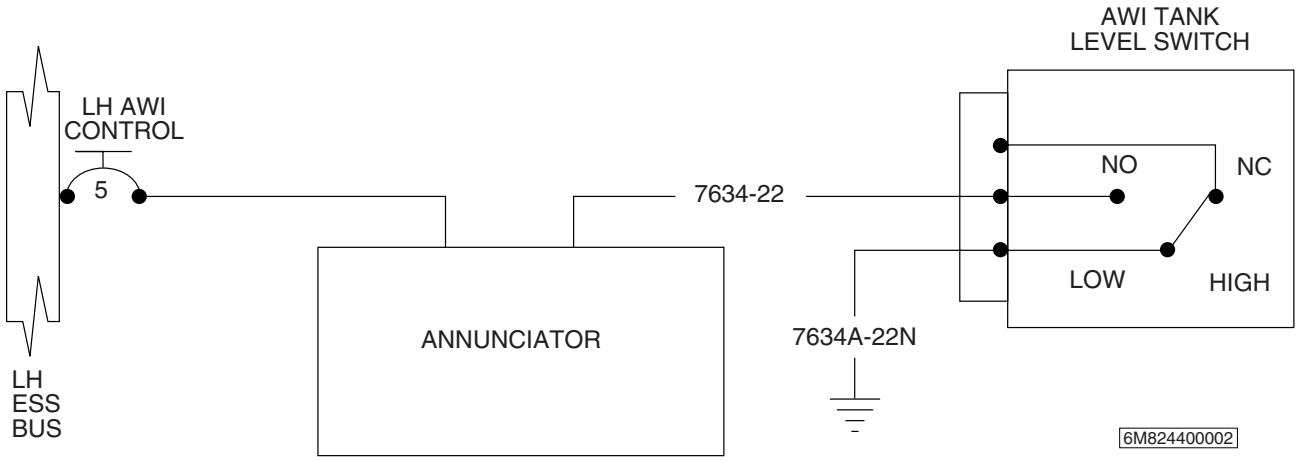
AWI QUANTITY INDICATING SYSTEM – DESCRIPTION AND OPERATION

1. General

A. The AWI quantity indicating system consists of a tank level sight gauge and a low level switch with a cockpit installed warning light.

B. The low level switch is installed on the inboard side of the AWI tank. The switch closes and causes the annunciator LOW WATER light to illuminate when four gallons of fluid remain in the AWI tank.

C. The level sight gauge consists of a clear plastic line routed from the bottom of the AWI tank to the vent system. A metal scale, graduated in gallons, is attached to the inboard side of the tank assembly. The clear plastic line passes behind this scale and the quantity, in gallons, is read by comparing the level in the line to the scale graduations.



AWI Quantity Indicating System Schematic
Figure 101

AWI QUANTITY INDICATING SYSTEM – FAULT ISOLATION

1. Troubleshooting – AWI Quantity Indicating System

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
No LOW Water light (tank quantity is low)	Defective wiring between switch and cockpit	Gain access to switch electrical connections. Ground wire 7634-22. LOW WATER light should illuminate.
	No ground to switch	Check for zero ohms to ground at wire 7634A-22N.
	Defective switch	Replace switch.

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AWI QUANTITY INDICATING SYSTEM – MAINTENANCE PRACTICES

1. Servicing – AWI Quantity Indicating System

A. No regularly scheduled maintenance is required of the AWI quantity indicating system. During routing inspections, all components of the system should be checked for security and condition. The tank quantity transmitter should be checked for evidence of leakage.

2. Removal /Installation – AWI Quantity Indicating System

A. Low Level Switch – Remove and Replace

(1) Pull the AWI system circuit breaker. Drain the AWI tank.

(2) Remove the tank shroud as necessary to gain access to the inboard side of the tank.

(3) Remove the protective cover over the level switch and disconnect the handshake connectors attaching the wires to the switch.

(4) Remove the low level switch. The switch and bushing may be removed as an assembly. As an alternate method, the access panel on top of the tank may be removed, the float unscrewed from the switch, and the switch unscrewed from the bushing. On replacing the switch, do not tightly screw the switch into the bushing without the bushing being installed in the tank. Teflon tape may be used on the threads of either the bushing or the switch.

(5) Connect the electrical leads to the switch.

(6) Check the operation of the switch by filling the tank until the LOW WATER light is extinguished. Drain fluid until the LOW WATER light illuminates. Drain and measure the remaining fluid. The amount should be four gallons.

(7) Reinstall the switch cover and tank shroud.

3. Adjustment /Test – AWI Quantity Indicating System

A. AWI Sight Level Gauge Adjustment

(1) Fill the AWI tank until the LOW WATER light is extinguished.

(2) Drain the tank until the LOW WATER light illuminates. Stop as soon as the light illuminates.

(3) Loosen the two screws securing the calibration scale. Adjust the scale up or down until the mark corresponding to four gallons lines up with the liquid level in the clear sight tube. Tighten the screws securing the calibration scale.

(4) Service the tank as required.

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MAINTENANCE MANUAL

CHAPTER 91 – CHARTS

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91-00-10	MP	203	APR 30/08
91-00-10	MP	204	APR 30/08
91-00-10	MP	205	APR 30/08
91-00-10	MP	206	APR 30/08
91-00-10	MP	207	APR 30/08
91-00-10	MP	208	APR 30/08
91-00-10	MP	209	APR 30/08
91-00-10	MP	210	APR 30/08
91-00-20	MP	201	APR 30/08
91-00-20	MP	202	APR 30/08
91-00-20	MP	203	APR 30/08
91-00-20	MP	204	APR 30/08
91-00-20	MP	205	APR 30/08
91-00-20	MP	206	APR 30/08
91-00-20	MP	207	APR 30/08
91-00-20	MP	208	APR 30/08
91-00-30	MP	201	APR 30/08
91-00-30	MP	202	APR 30/08
91-00-30	MP	203	APR 30/08
91-00-30	MP	204	APR 30/08
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CHAPTER 91 – CHARTS

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CHAPTER 91 – CHARTS

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CHAPTER 91 – CHARTS

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CHARTS – DESCRIPTION AND OPERATION

1. General

A. The requirements and procedures specified in this chapter apply to tightening of threaded fasteners and connections in all SA227 Series aircraft. Exceptions or modifications to these procedures are noted in the applicable chapter(s) of this Maintenance Manual. The information provided in this chapter applies to all fasteners and plumbing with the exception of threaded parts that are used for adjustment, such as turnbuckles or rod-ends. For these type items, refer to the specific Maintenance Manual chapters.

B. This chapter also provides a list of consumable materials generally used on SA227 Series aircraft as well as cleaning procedures and methods of applying bonding adhesives.

C. The final section of this chapter provides conversion tables to convert units of measurement used on SA227 Series aircraft into other standard measurements.

TORQUE PROCEDURES AND VALUES – MAINTENANCE PRACTICES

1. Procedures – Torque Procedures and Values

A. General Information

(1) To ensure components are securely installed without over-stressing nuts, bolts, screws, or pipe connections, the torque values given in the tables must be used.

CAUTION: BEFORE USING A TORQUE WRENCH, ENSURE THAT IT IS CORRECTLY CALIBRATED AND THAT THE PERMISSIBLE TIME BETWEEN BENCH CHECKS HAS NOT BEEN EXCEEDED.

(2) There is no satisfactory method of determining the torque previously applied to a threaded fastener. When re-torquing always back off approximately one quarter (1/4) turn or more before reapplying torque.

(3) Torque values shown in Tables 201 thru 204 are for clean non-lubricated parts. Do not use lubricants unless specified. When threads are lubricated, decrease torque by 10%.

(4) If threads are cadmium plated or lubricated with graphite, decrease torque by 15%.

CAUTION: CHLORINE BASED SOLVENTS MUST NOT COME IN CONTACT WITH TITANIUM AS IT CAUSES BRITTLENESS.

(5) Standard titanium bolts are coated with molybdenum disulfide to prevent corrosion or seizure and no extra treatment is required when fitting. Should the coating be damaged, change the bolt. If threads are coated with molybdenum disulfide, decrease torque by 15%.

(6) When using a pre-set torque wrench, apply the load gradually until the wrench gives its indicating snap. Load must be removed immediately as wrench snaps. The snapping of the wrench does not prevent additional torque from being applied if more pressure is applied to the handle.

(7) Torque accessible nuts on threaded screws to the values shown in Table 201. Slotted and Phillips head screws that do not have accessible nuts or are attached to nut plates, and screws with threads other than those shown in Table 201, need not be torqued.

(8) Whenever possible tighten nut when torquing. If necessary to torque a fastener from the head end, increase the maximum torque value by an amount equal to the shank friction. Measure shank friction with a torque wrench.

(9) Nuts requiring a cotter pin must initially be tightened to the minimum torque value shown in the tables. The torque may be increased to insert the cotter pin but must not exceed the maximum specified value.

(10) If parts are to be heated or cooled during assembly, nuts and bolts should not be torqued before the parts have returned to ambient temperature.

(11) Nuts or bolts that must be sequentially tightened are, unless otherwise stated, first to be hand tightened and then torqued in the sequence shown in Figure 202.

CAUTION: TORQUE WRENCHES MUST NOT BE USED TO LOOSEN NUTS OR BOLTS.

(12) Nuts and bolts fitted with flat seals or gaskets are to be initially torqued to slightly less than the required value. A brief pause will allow the seal or gasket to compress before final torquing.

(13) Install straight threaded connections, such as flared tubes and hose end fittings, and torque to the values listed in Table 202.

(14) Straight threaded connections using O-rings or gaskets for seal such as AN924 nut, AN6289 nut, and fittings conforming to MS33656 Style E, need not be torqued. Install them IAW AND 10064.

(15) When fitting self-locking nuts comply with the following instructions:

(a) Self-locking nuts are only to be fitted to screws or bolts that have a domed or chamfered start to the thread.

(b) Self-locking nuts are not to be used on bolts that have drillings in the threads to accept cotter pins.

(c) The re-tapping of a self-locking nut is not permitted.

(d) All metal self-locking nuts with thread cutters may only be used once.

(e) When fitting a self-locking nut check that the frictional resistance of the locking section is sufficient. If the frictional resistance can be overcome by hand, discard the nut.

(f) Do not lubricate the threads of the nut or bolt.

(g) Self-locking nuts with nylon-metallic inserts (NYLSTOP nuts) must not be used where temperatures exceed 248°F (120°C).

(h) When torquing self-locking nuts, add standard torque to the torque necessary to run the nut down to the material (tare value).

(16) Assemble threaded fasteners (bolts, screws, and nuts) at wrench torque values listed in Table 201 and Table 203.

(17) When an extension or adapter is used with a torque wrench the torque is altered. Use the chart in Table 207 or the formula in Figure 201 to calculate the correct setting.

(a) The adapter length is the distance between the torque centers of the wrench and the adapter.

(b) The torque arm is the approximate distance from the center of the hand grip of the wrench to its torque center.

NOTE: The use of an adapter extender at right angles to the torque arm DOES NOT affect the torque reading.

(18) If a universal (swivel) socket is required to torque inaccessible nuts or bolts, use the chart in Table 206 to determine the correct torque value.

(19) When an adapter and a universal socket extends the wrench length at an angle, the correct torque is obtained by consulting Figure 201 or Table 206 for the torque reading, then multiplying by a factor for the adapter obtained from Table 207 to yield the appropriate torque.

(20) Sheet metal screws need not be torqued but should have the heads drawn down tight.

(21) Specific torque is not applicable to bolts, nuts, and screws used in control systems or other installations where the torques required would cause binding or interference with proper operation of parts. On such applications the assembly should be firm, but not tight enough to cause binding.

(22) Tighten swagelok fittings by installing fitting until finger tight, and while holding the fitting body steady with a back up wrench, tighten the nut one and one quarter (1 1/4) turns, for 1/16, 1/8, and 3/16 size fitting, tighten only three quarter (3/4) turn from finger tight.

(23) Tighten nylo-seal fitting by installing fitting until finger tight. Tighten the nut two to two and one half (2 to 2 1/2) full turns. The nut may be marked at the six o'clock position to ensure the nut is tightened the required amount.

(24) Both the nylo-seal and swagelok may be disconnected and retightened. Retighten IAW **1.**, A., (22) and (23). After retightening swagelok fittings IAW **1.**, A., (16), tighten an additional one quarter (1/4) turn.

(25) Tighten screw type hose clamps finger tight plus one half (1/2) additional turn.

B. Torque Tables

Bolt/Nut Thread Size	Self Locking and Castellated Nuts			Self Locking External Wrenching Nuts					
	Part Number			Steel Shear			Steel Shear		
	NAS679 AN310 NAS1021 MS2104 MS21044			AN320			LH14367 160,00 PSI (11,248 kg/cm^2)		
	daNm	in. lbs	kpm	daNm	in. lbs	kpm	daNm	in. lbs	kpm
8 - 32	0.14 - 0.17	12 - 15	0.14 - 0.17	0.08 - 0.10	7 - 9	0.08 - 0.10	-- -- -- -- --	-- -- -- -- --	-- -- -- -- --
10 -32	0.23 - 0.28	20 - 25	0.23 - 0.29	0.14 - 0.17	12 - 15	0.14 - 0.17	-- -- -- -- --	-- -- -- -- --	-- -- -- -- --
1/4 - 28	0.57 - 0.79	50 - 70	0.58 - 0.81	0.34 - 0.45	30 - 40	0.35 - 0.46	0.73 - 1.02	65 - 90	0.55 - 1.04
5/16 - 24	1.13 - 1.16	100 - 140	1.15 - 1.61	0.90 - 1.00	60 - 85	0.69 - 1.00	1.47 - 2.03	130 - 180	1.50 - 2.07
3/8 - 24	1.76 - 2.15	160 - 190	1.76 - 2.19	1.07 - 1.24	95 - 110	1.09 - 1.26	2.50 - 4.07	220 - 360	2.53 - 4.15
7/16 - 20	5.08 - 5.65	450 - 500	5.08 - 5.70	3.05 - 3.38	270 - 300	3.11 - 3.45	4.18 - 6.89	370 - 610	4.25 - 7 03
1/2 - 20	5.42 - 7.80	480 - 690	5.42 - 7.95	3.28 - 4.63	290 - 410	3.34 - 4.72	7.12 - 10 73	630 - 950	7.26 - 10.95
9/16 - 18	9.04 - 11.30	800 - 1000	9.22 - 11.52	5.42 - 6.78	480 - 600	5.53 - 6.91	11.30 - 14.69	1000 - 1300	11.52 - 14.98
5/8 - 18	12.42 - 14.69	1100 - 1300	12.67 - 14.98	7.46 - 8.81	660 - 780	7.60 - 8.98	15.82 - 19.21	1400 - 1700	16.82 - 19.59
3/4 - 16	25.99 - 28.25	2300 - 2500	26.50 - 28.80	14.69 - 16.95	1300 - 1500	14.96 - 17.28	27.12 - 35 03	2400 - 3100	27.65 - 35 71
7/8 - 14	29.38 - 33.90	2600 - 3000	29.96 - 34.56	16.95 - 20.33	1500 - 1800	17.28 - 20.73	41.80 - 55.36	3700 - 4900	42.62 - 56.45
1-12 / 1-14	41.80 - 62.14	3700 - 5500	42.62 - 63.33	24.86 - 37.28	2200 - 3300	25.34 - 38.01	57.62 - 89.26	5100 - 7900	58.79 - 91.02
1 1/8 - 12	56.49 - 79.90	5000 - 7000	57.61 - 80.65	33.90 - 47.45	3000 - 4200	34.56 - 48.39	77.95 - 107.34	6900 - 9500	79.50 - 109.45
1 1/4 - 12	101.69 - 124.29	9000 - 11000	103.69 - 112.98	61.07 - 74.57	5400 - 6600	62.22 - 76.04	129.93 - 158 18	11500 - 14000	132.49 - 161.30
1 3/8 - 12				79.90 - 101,69	7000 - 9000	80.65 - 103 69			
1 1/2 - 12				112.99 - 135.58	10000 - 12000	115.21 - 138.25			

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Nut And Bolt Torques
Table 201

Tubing O.D. (Inches)	Aluminum Alloy Tubing - Flare MS33583 or MS33584			Steel Tubing Flare MS33584			Hose End Fittings and Hose Assemblies (Inch-Pounds) MS24587 or Similar Fitting End		
	daNm	in. lbs	kpm	daNm	in. lbs	kpm	daNm	in. lbs	kpm
1/4	0.45 - 0.68	40 - 60	0.46 - 0.69	0.57 - 1.02	50 - 90	0.58 - 1.04	0.57 - 1.02	50 - 90	0.58 - 1.04
5/16	0.68 - 0.90	60 - 80	0.69 - 0.92	0.79 - 1.37	70 - 120	0.81 - 1.38	0.79 - 1.37	80 - 120	0.81 - 1.38
3/8	0.85 - 1.41	75 - 125	0.86 - 1.44	1.02 - 1.70	90 - 150	1.04 - 1.73	1.02 - 1.70	90 - 150	1.04 - 1.73
1/2	1.70 - 2.82	150 - 250	1.73 - 2.88	1.75 - 2.82	155 - 250	1.79 - 2.88	1.75 - 2.82	155 - 250	1.79 - 2.88
5/8	2.26 - 3.96	200 - 350	2.30 - 4.03	3.39 - 4.52	300 - 400	3.47 - 4.61	3.39 - 4.52	300 - 400	3.47 - 4.61
3/4	3.39 - 5.65	300 - 500	3.47 - 5.76	4.86 - 6.50	430 - 575	4.96 - 6.58	4.86 - 6.50	430 - 575	4.96 - 6 58
1	5.65 - 7.91	500 - 700	5.76 - 8.07	6.21 - 8.47	550 - 750	6.34 - 8.64	6.21 - 8 47	550 - 750	6.34 - 8.64
1 1/4	6.78 - 10.17	600 - 900	6.91 - 10.37						

4M910010202

Coupling Nut Torques
Table 202

Torque values for screws and nuts with inch thread and a strength ot 125,000 psi (861.8 N/mm^2) and above. (NAS 464, NAS 1303 to NAS 1306, NAS 1308)						
Screw Size No.	Standard Nuts AN 310, NAS 1021			Flat Nuts AN 316, AN 320		
	daNm	in. lbs	kpm	daNm	in. lbs	kpm
10 -32	0.45	40	0.46	0.28	25	0.28
1/4 - 28	0.79 - 1.13	70 - 100	0.81 - 1.15	0.34 - 0.45	30 - 40	0.35 - 0.46
5/16 - 24	1.58 - 2.54	140 - 225	1.61 - 2.59	0.68 - 0.96	60 - 85	0.69 - 0.97
3/8 - 24	2.15 - 4.40	190 - 390	2.19 - 4.49	1.07 - 1.24	95 - 110	1.09 - 1.26
7/16 - 20	5.08 - 6.21	450 - 550	5.18 - 6.33	3.05 - 3.38	270 - 300	3.11 - 3.45
1/2 - 20	5.42 - 7.90	480 - 700	5.53 - 8.06	3.28 - 4.63	290 - 410	3.34 - 4.72
9/16 - 18	9.04 - 14.12	800 - 1250	9.22 - 14.40	5.42 - 6.78	480 - 600	5.53 - 6.91
5/8 - 18	12.42 - 18.08	1100 - 1600	12.67 - 18.43	7.46 - 8.81	660 - 780	7.60 - 8.98
3/4 - 16	25.99 - 33.90	2300 - 3000	26.50 - 34.56	14.69 - 16.95	1300 - 1500	14.98 - 17.28
7/8 - 14	28.25 - 47.46	2500 - 4200	28.80 - 48.38	16.95 - 20.33	1500 - 1800	17.28 - 20.73
1 - 12	41.80 - 67.80	3700 - 6000	42.62 - 69.12	24.86 - 37.28	22.00 - 3300	25.34 - 38.01

4M910010203

Screw And Nut Torques
Table 203

Screw Size No.	Torque values for screws and nuts with inch threads and with a minimum strength of 125,000 psi (861.8 N/mm^2)			Torque values for light alloy rod ends with inch thread and with a minimum strength of 62,580 psi (431.5 N/mm^2)		
	daNm	in. lbs	kpm	daNm	in. lbs	kpm
5/16 - 24	0.54 - 0.73	50 - 65	0.55 - 0.75	0.29 - 0.36	26 - 32	0.30 - 0.37
3/8 - 24	0.79 - 0.98	70 - 85	0.80 - 0.98	0.49 - 0.57	45 - 50	0.52 - 0.58
9/16 - 20	1.36 - 1.69	120 - 150	1.38 - 1.73	0.68 - 0.88	60 - 80	0.69 - 0.90
1/2 - 20	1.76 - 2.59	160 - 230	1.84 - 2.65	0.98 - 1.18	85 - 105	0.98 - 1.21
9/16 - 18	2.65 - 3.27	240 - 290	2.77 - 3.34	1.37 - 1.57	120 - 140	1.48 - 1.61
5/8 - 18	3.14 - 3.92	280 - 350	3.23 - 4.03	1.76 - 1.96	160 - 170	1.83 - 1.96

4M910010204

Steel And Light Alloy
Rod End Torques
Table 204

Torque values for hollow screws with Anti-Seize compound on threads				
Size	**daNm**	**in. lbs**	**ft. lbs**	**kpm**
1/8 - 27	0.68 - 0.92	60 - 80	5.0 - 6.6	0.69 - 0.92
1/4 - 18	1.47 - 1.70	130 - 150	10.9 - 12.5	1.50 - 1.73
3/8 - 18	2.09 - 2.43	185 - 215	15.4 - 18.0	2.13 - 2.47
1/2 - 14	2.88 - 3.22	255 - 285	21.1 - 23.8	2.94 - 3.29
3/4 - 14	3.50 - 3.96	310 - 350	25.8 - 29.2	3.57 - 4.03

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Hollow Screw Torques
Table 205

TORQUE PROCEDURES AND VALUES – MAINTENANCE PRACTICES

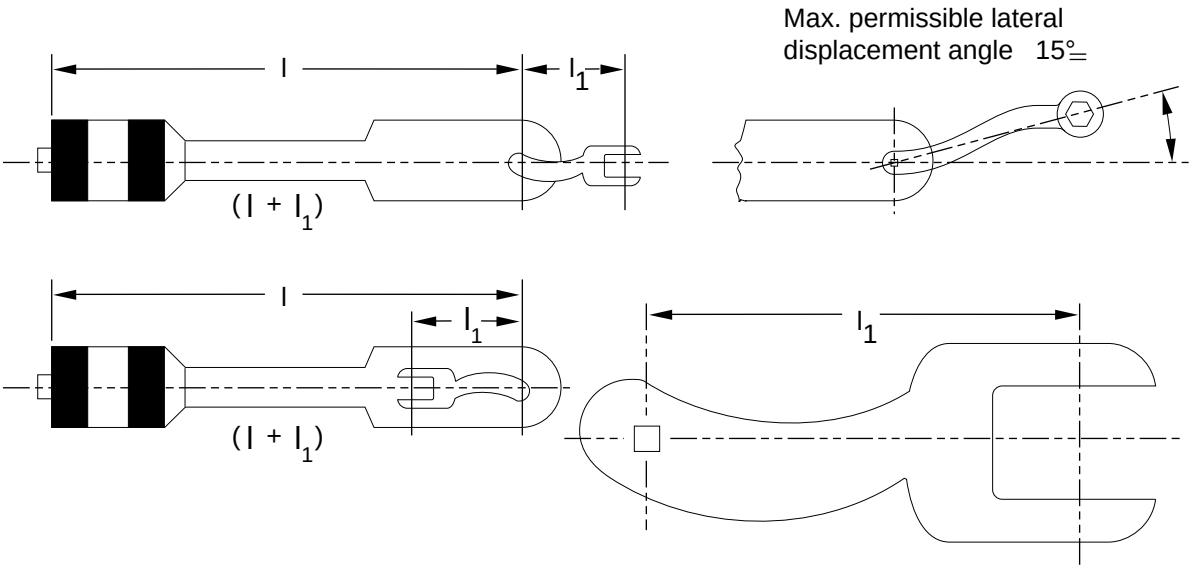
Formula: $Tx = \frac{Ta \times I}{I + I_1}$

Where: Ta = actual (required) torque value at end of extension (in. lbs or daNm)

Tx = Indicated or required value

I = length of arm (in. or m)

I_1 = effective length of extension (in. or m)



Torque Wrench Extensions
Figure 201

Conversion Torque Table for Use with Universal Sockets							
To Obtain a Torque of	Dial Must Read – With Universal Held at These Angles						
Inch or Ft. Lbs.	10°	20°	30°	40°	45°	50°	60°
10	10.2	10.7	11.5	13.1	14.1	15.6	20.0
20	20.4	21.4	23.0	26.2	28.2	31.2	40.0
30	30.6	32.1	34.5	39.3	42.3	46.9	60.0
40	40.8	42.8	46.0	52.4	56.4	62.4	80.0
60	61.2	64.2	69.0	78.6	84.6	93.6	120.0
80	81.6	85.6	92.0	104.8	112.8	124.8	160.0
100	102.0	107.0	115.0	131.0	141.0	156.0	200.0
120	122.4	128.4	138.0	157.2	169.2	187.2	240.0
140	142.8	149.8	161.0	183.4	197.4	218.4	280.0
180	183.6	192.6	207.0	235.8	253.8	280.8	360.0
200	204.0	214.0	230.0	262.0	282.0	312.0	400.0
220	224.4	235.5	243.0	288.2	310.2	343.2	440.0
240	244.8	256.8	276.0	314.4	338.4	374.4	480.0
260	265.2	278.2	299.0	340.6	366.6	405.6	520.0
280	385.6	299.6	312.0	366.8	394.8	436.8	560.0
300	306.0	321.0	345.0	393.0	423.0	468.0	600.0
320	326.4	342.4	368.0	419.2	451.2	499.2	640.0
340	346.8	363.8	391.0	445.4	479.4	530.4	680.0
360	367.2	385.2	414.0	471.6	507.6	561.6	720.0
380	387.6	406.6	437.0	497.8	535.8	592.8	760.0
400	408.0	428.0	460.0	524.0	564.0	624.0	800.0
420	428.4	449.4	483.0	550.2	602.2	655.0	840.0
440	448.8	470.8	506.0	576.4	620.4	686.4	880.0
460	469.2	492.2	529.0	593.6	648.6	717.6	920.0
480	489.6	512.6	552.0	628.8	676.8	748.8	960.0
500	510.0	535.0	575.0	655.0	705.0	780.0	1000.0

4M910010207

Universal Socket
Torque Conversion
Table 206

CORRECTION FACTORS FOR TORQUE WRENCHES USED WITH ADAPTORS WHICH EXTEND THE WRENCH							
Approx. Length of Torque Arm	Adaptor Lengths						
	1" thru 1 5/8"	Over 1 5/8" thru 2 3/8"	Over 2 3/8" thru 3 1/8"	Over 3 1/8" thru 3 7/8"	Over 3 7/8" thru 4 5/8"	Over 4 5/8" thru 5 3/8"	Over 5 3/8" thru 6"
7"	0.85	0.77	0.71	0.66	0.61	0.57	0.54
9 1/2"	0.89	0.83	0.78	0.74	0.69	0.67	0.64
13 1/2"	0.91	0.87	0.83	0.79	0.76	0.73	0.70
24"	0.95	0.92	0.90	0.87	0.85	0.83	0.81

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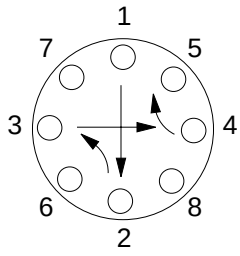
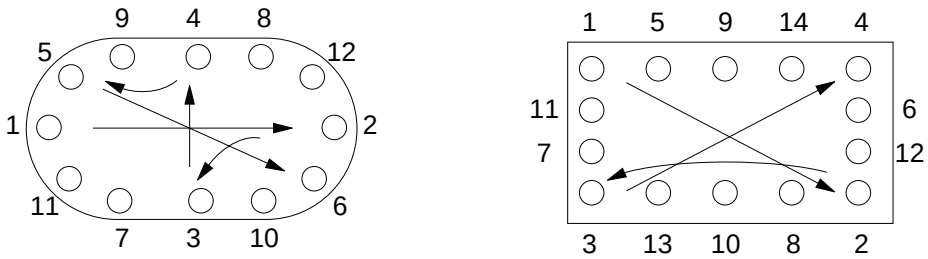
Extension/Adapter Correction Factors Table 207





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Torquing Sequence Figure 202

kpm	To: in. lbs 89.796236	To: daNm 0.9806665	in. lbs	To: kpm 0.0115212	To: daNm 0.0112985	daNm	To: kpm 1.019716	To: in. lbs 88.507535
1	86.795	0.981	1	0.012	0.011	1	1.020	88.508
2	173.592	1.961	2	0.023	0.023	2	2.039	177.015
3	260.389	2.942	3	0.035	0.034	3	3.059	265.523
4	347.185	3.923	4	0.046	0.045	4	4.079	354.030
5	433.981	4.903	5	0.058	0.056	5	5.099	442.538
6	520.777	5.884	6	0.069	0.068	6	6.118	531.045
7	607.574	6.865	7	0.081	0.079	7	7.138	619.553
8	694.370	7.845	8	0.092	0.090	8	8.158	708.060
9	781.166	8.826	9	0.104	0.102	9	9.177	796.568
10	867.962	9.807	10	0.115	0.113	10	10.197	885.075
11	954.759	10.787	11	0.127	0.124	11	11.217	973.583
12	1041.555	11.768	12	0.138	0.136	12	12.237	1062.090
13	1128.351	12.749	13	0.150	0.147	13	13.256	1150.598
14	1215.147	13.729	14	0.161	0.158	14	14.276	1239.105
15	1301.944	14.710	15	0.173	0.169	15	15.296	1327.613
16	1388.740	15.691	16	0.184	0.181	16	16.315	1416.121
17	1475.536	16.671	17	0.196	0.192	17	17.335	1504.628
18	1562.332	17.625	18	0.207	0.203	18	18.355	1593.136
19	1649.128	18.633	19	0.219	0.215	19	19.375	1681.643
20	1735.925	19.613	20	0.230	0.226	20	20.394	1770.151
21	1822.721	20.594	21	0.242	0.237	21	21.414	1858.658
22	1909.517	31.575	22	0.253	0.249	22	22.434	1947.166
23	1996.313	22.555	23	0.265	0.260	23	23.453	2035.673
24	2083.110	23.536	24	0.277	0.271	24	24.473	2124.181
25	2169.906	24.517	25	0.288	0.282	25	25.493	2212.688
30	2603.887	29.420	30	0.346	0.339	30	30.591	2655.226
35	3037.868	34.323	35	0.403	0.395	35	35.690	3097.764
40	3471.849	39.227	40	0.461	0.452	40	40.789	3540.301
45	3905.831	44.130	45	0.518	0.508	45	45.887	3982.830
50	4339.812	49.033	50	0.576	0.565	50	50.986	4425.377
60	5207.774	58.840	60	0.691	0.678	60	61.183	5310.452
70	6075.736	68.647	70	0.806	0.791	70	71.380	6195.527
80	6943.699	78.453	80	0.922	0.904	80	81.577	7080.603
90	7811.661	88.260	90	1.037	1.017	90	91.774	7965.678
100	8679.624	98.067	100	1.152	1.130	100	101.974	8850.753
			200	2.304	2.260			
			300	3.456	3.390			
			400	4.609	4.519			
			500	5.761	5.649			
			600	6.913	6.779			
			700	8.065	7.909			
			800	9.217	9.039			
			900	10.369	10.169			
			1000	11.521	11.299			
			2000	23.042	22.597			
			3000	34.564	33.896			
			4000	46.085	45.194			
			5000	57.606	54.493			
			6000	69.127	67.791			
			7000	80.648	79.090			
			8000	92.170	90.388			
			9000	103.691	101.687			4M910010210
			10000	115.212	112.985			

Torque Conversion
Table 208
(Sheet 1 of 2)

ft. lbs.	To: kpm	To: daNm	To: in. lbs.	kpm	To: ft. lbs.	daNm	To: ft. lbs.	in. lbs	To: ft. lbs.
	0.138255	0.135	12.00		7.233011		7.407		0.0833
1	0.138	0.135	12.00	1	7.233	1	7.407	1	0.083
2	0.276	0.270	24.00	2	14.466	2	14.814	2	0.166
3	0.414	0.405	36.00	3	21.699	3	22.221	3	0.249
4	0.553	0.540	48.00	4	28.932	4	29.628	4	0.333
5	0.691	0.675	60.00	5	36.165	5	37.035	5	0.416
6	0.829	0.810	72.00	6	43.398	6	44.442	6	0.499
7	0.967	0.945	84.00	7	50.631	7	51.849	7	0.583
8	1.106	1.080	96.00	8	57.864	8	59.256	8	0.666
9	1.244	1.215	108.00	9	65.097	9	66.663	9	0.749
10	1.382	1.350	120.00	10	72.330	10	74.070	10	0.833
15	2.073	2.025	180.00	15	108.495	15	111.105	15	1.249
20	2.765	2.700	240.00	20	144.660	20	148.140	20	1.666
25	3.456	3.375	300.00	25	180.885	25	185.175	25	2.082
30	4.147	4.050	360.00	30	216.990	30	222.210	30	2.499
35	4.838	4.725	420.00	35	253.155	35	259.245	35	2.915
40	5.530	5.400	480.00	40	289.320	40	296.280	40	3.332
45	6.221	6.075	540.00	45	325.485	45	333.315	45	3.748
50	6.912	6.750	600.00	50	361.650	50	370.350	50	4.165

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Torque Conversion
Table 208
(Sheet 2 of 2)

CONSUMABLE MATERIALS – MAINTENANCE PRACTICES

1. General

A. Cleaning – Procedures

(1) METALS – All metals shall be solvent cleaned or vapor-degreased until all grease, dirt, etc., is removed.

CAUTION: USE ONLY ALIPHATIC NAPTHA TO CLEAN ACRYLIC MATERIALS.

(2) ACRYLICS – Clean acrylic sheet by wiping faying surfaces with a clean cloth saturated with aliphatic naptha (TT-N-95, Type II) and immediately drying with a clean dry cloth.

(3) GLASS FIBER REINFORCED PLASTIC – Clean glass fiber reinforced plastic by wiping faying surfaces with a clean cloth saturated with trichlorethylene or similar solvent. Surfaces must be abraded for better adhesion.

(4) RUBBER – Prepare rubber materials by abrading or roughening the surface to remove the surface gloss, then clean with solvent.

(5) ABS (THERMOPLASTIC SHEET) – Clean surface by wiping faying surfaces with a clean cloth saturated with trichlorethylene or similar solvent.

B. Cementing and Bonding Instructions:

NOTE: The following requirements and procedures apply to all general purpose cementing and bonding accomplished on SA227 Series aircraft. Exceptions or modifications to these procedures are noted in the applicable chapter of this manual under Maintenance Practices.

(1) SCOTCH-WELD 1838 B/A: Scotch-Weld adhesive No. 1838 consists of two (2) parts. Mix thoroughly by volume five (5) parts of B (base) and six (6) parts of A (accelerator). Color should be a uniform green. Apply adhesive to both surfaces to be bonded. Join the adhesive coated surfaces and allow to cure at 60°F (15.6°C) or above until completely cured (approximately twelve [12] hours). Cure may be accelerated by application of heat up to 300°F (148.9°C). Contact pressure is sufficient; however, parts must remain stationary.

(2) RELIABOND No. 380: Reliabond is a two (2) part filled epoxy adhesive system. Mix thoroughly by weight one hundred (100) parts A (accelerator) to thirty three (33) parts B (base). Apply adhesive to both surfaces (approximately 0.0312 inch [0.792 mm]) to be bonded. Join the adhesive coated surfaces and allow to cure at 60°F (15.6°C) or above until completely cured (approximately twenty four [24] hours). Pot life (work life) is thirty (30) minutes. Alternate time/temperature cycles are for one (1) hour at 200°F (93.3°C), fourty (40) minutes at 300°F (148.9°C), or thirty (30) minutes at 350°F (176.7°C).

(3) SCOTCH-WELD 2216 B/A: A gray two (2) part, room temperature curing structural adhesive. Mix thoroughly, by weight, three (3) parts of B (base) and seven (7) parts of A (accelerator). Color should be a uniform grey. Pot life (work Life) is one (1) hour. Apply adhesive to both surfaces (approximately 0.0312 inch [0.792 mm]) to be bonded. Join the adhesive coated surfaces and allow to cure at 60°F (15.6°C) or above until completely cured (approximately twelve [12] hours). Keep parts stationary during cure.

(4) EC880, EC1300L, D-218, 5452: Apply a thin even coat of adhesive to each surface. Allow to dry until quite tacky, but no longer transfers to the finger when touched, (usually between five (5) and thirty (30) minutes). Porous surface may require two (2) coats. Place surfaces together and press firmly to ensure good contact. Good bond strength is obtained after twenty four (24) hours.

(5) RTV-102 SILICONE - SILASTIC RTV – 732: Apply one (1) coat to each clean faying surface. Place surfaces together and press to ensure positive contact. Allow bond line to cure twenty four (24) hours at room temperature before bond joint is subjected to any stresses. The part may be handled after four (4) hours of cure.

(6) SOLVENT BONDING: Many plastics may be successfully bonded to plastics by the use of solvents. The solvents may be thickened by adding small particles of the plastic to be bonded. Solvents may be applied to plastic surfaces by brushing, spraying, dipping, or felt pad, etc. Allow the solvent to remain on material until both surfaces soften, then immediately join surfaces while wet. The parts must be jigged so that they are in a positive close knit contact until the bond is firmly set.

(7) EASTMAN 910 – LOCTITE QUICK SET 404: Surfaces must be clean, dry, and free of oil and grease. Apply adhesive to only one (1) surface without exerting undue pressure on the adhesive film. Rub both surfaces together, very quickly one (1) or two (2) times, before final adjustments to ensure uniform contact. Good bond strength is obtained thirty (30) minutes after assembly.

(8) SOLVENT REACTIVATION: The mating surfaces must be clean, dry, and free of oil and grease. Using a clean, lint-free cloth or brush heavily moistened with the proper solvent, reactivate the surface or the cement. Avoid excess rubbing of the cement, which would remove it from the surface. Press surface firmly together to ensure good contact at all points. Bonds have considerable strength immediately after joining and develop ultimate strength in one (1) to three (3) days.

(9) URALANE 8089: Mix by adding one hundred (100) parts, by weight, of part A (accelerator) to fourty (40) parts, by weight, of part B (base). The pot life (work life) of mixed material is approximately fourty five (45) minutes at 77°F (25°C). Surfaces must be clean, dry, and free of oil and grease. Mixed materials should be placed between faying surfaces and surfaces squeezed together. No loads or stresses should be applied for twenty four (24) hours.

(10) Bonding of Pre-coated Velcro Fasteners:

NOTE: Velcro fasteners with a pre-coat of 0145A adhesive shall be bonded as follows:

(a) Surfaces shall be cleaned IAW Section A. Light sand very smooth surfaces to enhance adhesion.

(b) Apply one (1) coat of EC1300L to the surface. Allow to dry until adhesive will not transfer when touched lightly. Press the pre-coated velcro to the pre-coated surface press firmly together to ensure good contact at all points. Bonds will have considerable strength immediately after joining and will develop ultimate strength in twenty four (24) hours.

2. List of Consumable Materials

A. All consumable materials used on SA227 Series aircraft and specified in the Maintenance Manual are contained in this List of Consumable Materials. The materials are divided into categories as follows:

(1) Sealers, Fillers, and Cements

(a) Deice Boot – Leading Edge

1 EC-801 – Sealing Compound, 3M Co.

2 A-56-B – Cement, B. F. Goodrich

3 EC - 1403 – Cement, 3M Co.

4 EC - 1300L – Cement, 3M Co.

(b) Alcohol/Water Injection

1 Resin No. 393 Polyester – Comm. Resins Div., Interplastics Corp.

2 Catalyst, 60% MEKP – Norak Inc.

(c) Windows

1 MIL-S-8802 – Sealant – Class B (PR-1422, P/S 890 or equivalent).

2 MIL-S-8802 – Sealant – Class C (PR-1422, P/S 890 or equivalent).

(d) Fuel Tanks

1 MIL-C-27725 – Coating – Corrosion prevention, Integral fuel tanks.

2 MIL-C-5541 – Chemical Films & Materials for aluminum and aluminum alloys.

3 MIL-S-8802 – Sealing Compound – Temperature resistant, integral fuel tanks (PR-1422, P/S 890 or equivalent).

NOTE: PR-1826 may be used as an alternate sealer. PR-1826 must be used IAW manufacturers instructions and with manufacturers primer or adhesion promoter.

(e) Electrical Components

1 MIL-S-8516 Class 1 – Potting Compound – (PR-1201, P/S 727 or equivalent).

(2) Adhesives

(a) Class 1 – Epoxy Base Adhesives

NOTE: Used for bonding metal to metal, glass fiber reinforced plastic to metals, ABS to ABS, and thermoplastic to metals.

1 Scotch Weld - 1838 B/A

2 Reliabond No. 380

3 Scotch Weld - 2216 B/A

(b) Class 2 – Chloroprene Base Adhesives

NOTE: Used for bonding rubber, fabric, vinyl, glass batting, carpet, and leather; to each other, ABS or metals.

1 EC-880 3M Co.

2 EC-1300L 3M Co.

3 D218 Armstrong Cork Company

4 5452 Hadley Adhesive and Chemical Company

(c) Class 3 – Nitrile Rubber Base Adhesives

NOTE: Use for cementing gaskets and rubber in applications where resistance to fuel is necessary.

1 EC-826 3M Co.

2 EC-847 3M Co.

(d) Class 4 – Flexible Bond Adhesives

NOTE: An adhesive that makes a flexible bond, used only for bonding upholstery foam to upholstery foam.

1 3668 Hadley Adhesive and Chemical Co.

(e) Class 5 – Silicone Base Adhesive

NOTE: Used for bonding silicone to silicone, metals, or plastics.

1 RTV-012 Silicone Rubber Adhesive/Sealant, General Electric Silastic RTV-3145, Dow Corning.

(f) Class 6 – Solvent Bond

1 Cyclophexanone (Also MIBK, MPK, Acetone) for use on:

a ABS (Royalite, Cycolac)

b Cellulose Acetate Butyrate (Tenite Butyrate)

c Styrene

2 Methylene Chloride for use on:

a Acrylic (Plexiglas, Lucite, Acrylite)

b Polycarbonate (Lexan)

(g) Class 7 – Acrylic Monomer Adhesives

NOTE: Used for bonding glass, metal, rubber, cork, leather, acrylics, and vinyls to itself, or each other.

1 Eastman 910 – Eastman Chemical Products

2 Loctite Quick Set 404, Loctite Corporation

(h) Class 8 – Solvent Reactivated Adhesive

NOTE: Surfaces that have been prepared with an adhesive should be reactivated IAW **1.**, B., (8), with a solvent recommended by the manufacturer. The precoated systems should be used to bond only recommended surfaces.

(i) Class 9 – Urethane Base Adhesive

1 Uralane 8090 – Furnace Plastics Incorporated

(3) Lubricants

(a) Grease & Oil

1 MIL-G-81322 – Grease, Aircraft and Actuator screw.

2 MIL-G-3545 – Grease, bearing, high-temperature, Shell Oil Aerogrease No. 5, Texaco MIL-G-3545B.

3 Hartzell DG – Grease, Propeller – Hartzell Propeller Co.

4 MIL-L-7870 – Oil, General purpose, low temperature.

5 MIL-G-7187 – Grease, graphite.

6 MIL-H-83282 – Hydraulic Fluid.

7 903 Dri-Slide – Liquid Hydro-carbon Carrier, a Click-Clack lubricant.

(b) Thread Compound

1 MIL-T-5544 – Thread compound, Anti-seize – graphite (Fuel, engine hot bleed air).

2 MIL-T-5542 – Thread compound, Anti-seize (Oxygen).

(c) MIL-T-5542B – Thread compound, Anti-seize (Oxygen system male tapered threads).

(d) TT-A-580 – Thread Compound, Anti-seize, white lead base Deicer (air), Pitot & Static.

(4) Cleaners

(a) Miscellaneous

1 Methyl propyl ketone (MPK) – McKesson & Robbins, Inc.

2 Isopropyl Alcohol – R.N. Hughes Co.

3 Varsol No. 1 – Chevron Oil.

4 Toluene – Humble Oil.

5 P-D-680 Type I – Solvent, Dry Cleaning.

6 VV-K-211 – Solvent, Dry Cleaning.

(b) Aluminum

1 Aluminetch No. 2 – Turco Products.

2 Smut-Go No. 2 – Turco Products.

(c) Stainless

1 Turco Nitradd – Turco Products.

(d) Steel

1 Turco Prokleen – Turco Products

2 Turco De-scaler – Turco Products.

(e) Degreaser

1 Trichlorethylene – Kelite Co. (Fed. Spec. O-T-34)

(5) Expendable Items

(a) Miscellaneous

1 MIL-T-27730 – Teflon Tape (Oxygen System).

2 MIL-C-18767 – Compound, Window scratch removing, Wilco No. 35.

(b) Inspection Penetrant

1 MIL-I-68866B (ASG) – Penetrant, Inspection

2 MIL-I-6807A (ASG) – Penetrant, Inspection - Non-destructive.

3 MIL-STD-410 – Penetrant, Magnetic Particle.

4 MIL-I-25135 – Penetrant, Inspection Materials.

(c) Paint Primer

1 MIL-P-8585A-1 – Zinc Chromate Primer.

2 TT-T-266 – Thinner, dope & lacquer (Cellulose nitrate).

3 Prep-Sol 3919 – Cleaner, Dupont Co.

4 METHYL PROPYL KETONE (MPK) – McKesson & Robbins, Inc..

(d) Self – etch Prime Finishing

1 MIL-C-8514C – Mist Coat Primer, accelerator No. 166 Jones Blair Co.

2 Isopropyl Alcohol – R.N. Hughes Co.

(e) Aluminum Corrosion Treatment

1 MIL-C-5541 – Turcoat 4187-6, Turco Products

2 Turco W.O. No. 1, Tuco Products.

(f) Silver Paint Finish

1 Clear Lacquer – Exterior type, Dupont Co.

2 Aluminum Bronze Powder – 400 Mesh, Ohio Bronze Powder Co.

3 Lacquer Thinner – Dupont Co.

4 Prep-Sol No. 3919 – Cleaner, Dupont Co.

(g) Epoxy Base Primer

1 GP-208 – Glid-Plate Primer, Glidden Co.

2 GP-209 – Glid-Plate Mixing Liquid, Glidden Co.

3 Epoxy No. 583 – Primer, Dupont Co.

4 UA-83339586 or ENG 6425 – Activator, Dupont Co.

5 T-3871 or 31004 – Thinner, Dupont Co.

(h) Magnesium Protective Treatment (MIL-M-3171C)

1 Hydrofluoric Acid – Dow Chemical

2 Sodium Dichromate – Dow Chemical

3 Concentrated Nitric Acid – Dow Chemical

4 Calcium or Magnesium Fluoride – Dow Chemical

5 Aliphatic Naptha – Dow Chemical

6 Prokleen – Turco Products

7 METHYL PROPYL KETONE (TT-M-2616) – McKeeson & Robbins Inc.

(i) Rain Erosion Coating (MIL-C-7439C Type II)

(j) Cadmium Plating

1 Nav. Spec. S-72 – Sodium Cyanide

2 MIL-C-6151 – Cadmium Oxide

3 Fed Spec P-F-631B – Caustic Soda

4 Fed Spec O-C-303 – Chromic Acid

5 Fed Spec O-A-36 – Hydrochloric Acid

6 Fed Spec QQ-C-61 – Cadmium Anodes

(k) Pressure Leak and Detection – Fuel Tanks

1 Ammonia (28% to 33%) – Commercial or household

2 LD200 Leak Detector – Phillips Petro Co.

3 Automatic Liquid Dye -

4 Yellow No. 662.

5 Automatic Red B –Morton Chemical Co.

(l) Oxygen System

1 MIL-L-25567B – Leak detector fluid, Type I, Zep, Aero Sherlock.

(m) Approved engine fuels, engine oils, and alcohol/water injection mixture.

CONVERSION PROCEDURES AND VALUES – MAINTENANCE PRACTICES

1. Conversion Procedures

A. General Information

(1) The following tables contain information necessary to convert standard measurement units contained in SA227 Commuter Category Aircraft Maintenance Manuals to other measurements.

(a) Power Conversion Table

1 Table 201 converts DIN HP to kW and HP; HP to DIN HP and kW; and kW to DIN HP and HP.

2 Multiply values not included on the chart by conversion factors at the top of each column to obtain desired equivalent.

(b) Weight Conversion Table

1 Table 202 converts kg to lbs and daN; lbs to kg and daN; and daN to kg and lbs.

2 Multiply values not included on the chart by conversion factors at the top of each column to obtain desired equivalent.

(c) Pressure Conversion Table (liquid and Gas)

1 Table 203 Sheet 1 converts kg/cm^2 to psi and bar; psi to kg/cm^2 and bar; and bar to kg/cm^2 and bar. Sheet 2 converts in. Hg to psi and kg/cm^2; bar to in. Hg; kg/cm^2 to in. Hg; and psi to in. Hg.

2 Multiply values not included on the chart by conversion factor at the top of each column to obtain desired equivalent.

(d) Temperature Conversion Table

1 Table 204 converts Celsius temperatures to Fahrenheit temperatures and Fahrenheit temperatures to Celsius temperatures.

2 Find temperature to be converted in bold lettering. Fahrenheit to celsius equivalents appear tin the left hand column and Celsius to Fahrenheit in the right.

(e) Inch Conversion Table

1 Table 205 converts inches too millimeters.

2 Multiply decimal inches by a factor of 25.4 to obtain equivalent in millimeters for values not listed on chart.

(2) Conversion Charts

DIN HP	To: kw	To: HP	HP	To: DIN HP	To: kw	kw	To: DIN HP	To: HP
	0.735499	0.896321		1.013869	0.745699		1.359621	1.341024
1	0.736	0.986	1	1.014	0.746	1	1.360	1.341
2	1.471	1.973	2	2.028	1.491	2	2.719	2.682
3	2.206	2.959	3	3.042	2.237	3	4.079	4.023
4	2.942	3.945	4	4.055	2.983	4	5.438	5.364
5	3.677	4.932	5	5.069	3.728	5	6.798	6.705
6	4.413	5.918	6	6.083	4.474	6	8.158	8.046
7	5.148	6.904	7	7.097	5.220	7	9.517	9.387
8	5.844	7.891	8	8.111	5.966	8	10.877	10.728
9	6.619	8.877	9	9.125	6.712	9	12.237	12.069
10	7.355	9.863	10	10.139	7.457	10	13.596	13.410
11	8.090	10.850	11	11.153	8.203	11	14.956	14.751
12	8.826	11.836	12	12.166	8.948	12	16.315	16.092
13	9.561	12.822	13	13.180	9.694	13	17.675	17.433
14	10.297	13.808	14	14.194	10.440	14	19.035	18.774
15	11.032	14.795	15	15.208	11.185	15	20.394	20.115
16	11.768	15.781	16	16.222	11.931	16	21.754	21.456
17	12.503	16.767	17	17.236	12.677	17	23.114	22.797
18	13.239	17.754	18	18.250	13.423	18	24.473	24.138
19	13.974	18.740	19	19.264	14.168	19	25.883	25.479
20	14.710	19.726	20	20.277	14.913	20	27.192	26.280
21	15.445	20.713	21	21.291	15.660	21	28.552	28.161
22	16.181	21.699	22	22.305	16.405	22	29.912	29.503
23	16.916	22.685	23	23.319	17.151	23	31.271	30.884
24	17.652	23.672	24	24.333	17.897	24	32.631	32.185
25	18.387	24.658	25	25.347	18.642	25	33.991	33.526
30	22.065	29.590	30	30.416	22.371	30	40.789	40.231
35	25.742	34.521	35	35.485	26.099	35	47.587	46.986
40	29.420	39.453	40	40.555	29.828	40	54.385	53.641
45	33.097	44.384	45	45.624	33.556	45	61.183	60.346
50	36.775	49.316	50	50.694	37.285	50	67.981	67.051
60	44.130	59.179	60	60.832	44.742	60	81.577	80.461
70	51.485	69.043	70	70.971	52.199	70	95.174	93.872
80	58.840	78.906	80	80.111	59.656	80	108.770	107.282
90	66.195	88.769	90	81.248	67.113	90	122.366	120.692
100	73.550	98.632	100	101.387	74.570	100	135.962	134.102
200	147.100	197.264	200	202.774	149.140	200	271.924	268.205
300	220.650	295.896	300	304.161	223.710	300	407.886	402.307
400	294.200	394.528	400	405.548	298.280	400	543.849	536.410
500	367.750	493.161	500	506.935	372.850	500	679.811	670.512
600	441.299	591.793	600	608.321	447.419	600	815.773	804.614
700	514.849	690.425	700	709.708	521.989	700	951.735	938.717
800	588.399	789.057	800	811.095	596.559	800	1087.697	1072.819
900	661.949	887.689	900	912.482	671.129	900	1223.659	1206.921
1000	735.499	986.321	1000	1013.869	745.699	1000	1359.621	1341.024
2000	1470.998	1972.642	2000	2027.738	1491.398	2000	2719.242	2682.047
3000	2206.497	2958.963	3000	3041.607	2237.097	3000	4078.863	4023.071
4000	2941.966	3945.284	4000	4055.476	2982.796	4000	5438.485	5364.095
5000	3677.495	4931.605	5000	5069.345	3728.495	5000	6798.106	6705.119
6000	4412.994	5917.926	6000	6083.214	4474.194	6000	8157.727	8046.142
7000	5148.493	6904.247	7000	7097.083	5219.893	7000	9517.349	9387.166
8000	5883.992	7890.568	8000	8110.952	5965.592	8000	10876.969	10728.190
9000	6619.491	8876.889	9000	9124.821	6711.291	9000	12236.591	12069.213
10000	7345.990	9863.210	10000	10138.690	7459.990	10000	13596.212	13410.237

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Power Conversion
Table 201

kg	To: lbs	To: daN	lbs	To: kg	To: daN	daN	To: kg	To: lbs
	2.204622	0.980665		0.453592	0.444822		1.019716	2.248091
1	2.205	0.981	1	0.454	0.445	1	1.020	2.248
2	4.409	1.961	2	0.907	0.890	2	2.039	4.496
3	6.614	2.942	3	1.361	1.334	3	3.059	6.744
4	8.818	3.923	4	1.814	1.779	4	4.079	8.992
5	11.023	4.903	5	2.268	2.224	5	5.099	11.240
6	13.228	5.884	6	2.722	2.669	6	6.118	13.489
7	15.432	6.865	7	3.175	3.114	7	7.138	15.737
8	17.637	7.845	8	3.629	3.559	8	8.158	17.985
9	19.842	8.826	9	4.082	4.003	9	9.177	20.233
10	22.046	9.807	10	4.536	4.448	10	10.197	22.481
11	24.251	10.787	11	4.990	4.893	11	11.217	24.729
12	26.455	11.768	12	5.443	5.338	12	12.237	26.977
13	28.660	12.749	13	5.897	5.783	13	13.256	29.225
14	30.865	13.729	14	6.350	6.228	14	14.276	31.473
15	33.069	14.710	15	6.804	6.672	15	15.296	33.721
16	35.274	15.691	16	7.257	7.117	16	16.315	35.969
17	37.479	16.671	17	7.711	7.562	17	17.335	38.218
18	39.683	17.652	18	8.165	8.007	18	18.355	40.466
19	41.888	18.633	19	8.618	8.452	19	19.375	42.714
20	44.092	19.613	20	9.072	8.896	20	20.394	44.962
21	46.297	20.594	21	9.525	9.341	21	21.414	47.210
22	48.502	21.575	22	9.979	9.786	22	22.434	49.458
23	50.706	22.555	23	10.433	10.231	23	23.453	51.706
24	52.911	23.536	24	10.886	10.676	24	24.473	53.954
25	55.116	24.517	25	11.340	11.121	25	25.493	56.202
30	66.139	29.420	30	13.608	13.345	30	30.591	67.443
35	77.162	34.323	35	15.876	15.569	35	35.690	78.683
40	88.185	39.227	40	18.144	17.793	40	40.789	89.924
45	99.208	44.130	45	20.412	20.017	45	45.887	101.164
50	110.231	49.033	50	22.680	22.241	50	50.986	112.405
60	132.277	58.840	60	27.216	26.689	60	61.183	134.885
70	154.323	68.647	70	31.751	31.138	70	71.380	157.366
80	176.370	78.453	80	36.287	35.585	80	81.577	179.847
90	198.416	88.260	90	40.823	40.034	90	91.774	202.328
100	220.462	98.067	100	45.359	44.482	100	101.972	224.809
200	440.924	196.133	200	90.718	88.964	200	203.943	449.618
300	661.386	294.200	300	136.078	133.447	300	305.915	674.427
400	881.848	392.266	400	181.437	177.929	400	407.886	899.236
500	1102.310	490.333	500	226.796	222.411	500	509.858	1124.046
600	1322.772	588.399	600	272.155	266.893	600	611.830	1348.855
700	1543.234	686.466	700	317.514	311.375	700	713.801	1573.664
800	1763.696	784.532	800	3362.874	355.857	800	815.773	1798.473
900	1984.158	882.599	900	408.233	400.340	900	917.745	2023.282
1000	2204.622	980.685	1000	453.592	444.822	1000	1019.715	2248.091
2000	4409.244	1961.330	2000	907.184	889.644	2000	2039.432	4496.182
3000	6613.866	2941.995	3000	1360.776	1334.465	3000	3059.149	6744.273
4000	8818.488	3922.660	4000	1814.368	1779.287	4000	4078.865	8992.364
5000	11023.110	4903.325	5000	2267.960	2224.109	5000	5098.581	11240.455
6000	13227.732	5883.990	6000	2721.552	2668.931	6000	5118.297	13488.546
7000	15432.354	6864.655	7000	3175.144	3113.753	7000	7138.038	15736.637
8000	17636.976	7845.320	8000	3628.736	3558.574	8000	8157.730	17984.728
9000	19841.598	8825.985	9000	4082.328	4003.396	9000	9177.446	20232.819
10000	22046.220	9806.650	10000	4535.920	4448.218	10000	10197.162	22480.910

4M910030202

Weight Conversion
Table 202

kg/cm^2	To: psi	To: bar	psi	To: kg/cm^2	To: bar	bar	To: kg/cm^2	To: psi
	14.223335	0.980665		0.0703069	0.0689476		1.019716	14.503768
1	14.223	0.981	1	0.070	0.069	1	1.020	14.504
2	28.447	1.961	2	0.141	0.138	2	2.039	29.008
3	42.670	2.942	3	0.211	0.207	3	3.059	43.511
4	56.893	3.923	4	0.281	0.276	4	4.079	58.015
5	71.117	4.903	5	0.352	0.345	5	5.099	72.519
6	85.340	5.884	6	0.422	0.414	6	6.118	87.023
7	99.563	6.865	7	0.492	0.483	7	7.138	101.526
8	113.787	7.845	8	0.565	0.552	8	8.158	116.030
9	128.010	8.826	9	0.633	0.621	9	9.177	130.534
10	142.233	9.807	10	0.703	0.689	10	10.197	145.038
11	156.457	10.787	11	0.773	0.758	11	11.217	159.541
12	170.680	11.768	12	0.844	0.827	12	12.237	174.045
13	184.903	12.749	13	0.914	0.896	13	13.256	188.549
14	199.127	13.729	14	0.984	0.965	14	14.276	203.053
15	213.350	14.710	15	1.055	1.034	15	15.296	217.557
16	227.573	15.691	16	1.125	1.103	16	16.315	232.060
17	241.797	16.671	17	1.195	1.172	17	17.335	246.564
18	256.020	17.652	18	1.266	1.241	18	18.355	261.068
19	270.243	18.633	19	1.336	1.310	19	19.375	275.572
20	284.467	19.613	20	1.406	1.379	20	20.394	290.075
21	298.690	20.594	21	4.476	1.448	21	21.414	304.579
22	312.913	21.575	22	4.547	1.517	22	22.434	319.083
23	327.137	22.555	23	1.617	1.586	23	23.453	333.587
24	341.360	23.536	24	1.687	1.655	24	24.473	348.090
25	355.583	24.517	25	1.758	1.724	25	25.493	362.594
30	426.700	29.420	30	2.109	2.068	30	30.591	435.113
35	497.817	34.323	35	2.461	2.413	35	35.690	507.632
40	568.933	39.227	40	2.812	2.758	40	40.789	580.151
45	640.050	44.130	45	3.164	3.103	45	45.887	652.670
50	711.167	19.033	50	3.515	3.447	50	50.986	725.188
60	853.400	58.840	60	4.218	4.137	60	61.183	870.226
70	995.633	68.647	70	4.921	4.826	70	71.380	1015.264
80	1137.867	78.459	80	5.625	5.516	80	81.577	1160.301
90	1280.100	88.260	90	6.328	6.205	90	91.774	1305.339
100	1422.334	98.067	100	7.031	6.895	100	101.972	1450.377
200	2844.667	196.133	200	14.061	13.790	200	203.943	2900.754
300	4257.001	294.200	300	21.092	20.684	300	305.915	4351.130
400	5689.334	392.266	400	28.123	27.579	400	407.886	5801.507
500	7111.668	490.333	500	35.154	34.474	500	509.858	7251.884
600	8534.001	588.399	600	42.184	41.369	600	611.830	8702.261
700	9956.335	686.466	700	49.215	48.263	700	713.801	10152.638
800	11378.668	784.532	800	56.246	55.158	800	815.773	11603.014
900	12801.002	882.599	900	63.276	62.053	900	917.745	13053.391
1000	14223.335	980.665	1000	70.307	69.948	1000	1019.715	14503.768
2000	28446.670	1961.330	2000	140.614	137.895	2000	2039.432	29007.536
3000	42670.005	2941.995	3000	210.921	206.843	3000	3059.149	43511.304
4000	56893.340	3922.660	4000	281.228	275.790	4000	4078.865	58015.072
5000	71116.675	4903.325	5000	351.535	344.738	5000	5098.581	72518.840
6000	85340.010	5883.990	6000	421.841	413.686	6000	5118.297	87022.608
7000	99563.345	6864.655	7000	492.148	482.633	7000	7138.038	101526.376
8000	113786.680	7845.320	8000	542.455	551.581	8000	8157.730	116030.144
9000	128010.015	8825.985	9000	632.762	620.528	9000	9177.446	130533.912
10000	142233.350	9806.650	10000	703.069	689.476	10000	10197.162	145037.680

4M910030203

Pressure Conversion
Table 203
(Sheet 1 of 2)

in. Hg	To: psi	To: kg/cm	To: bar	bar	To: in. Hg	kg/cm	To: in. Hg	psi	To: in. Hg
	0.491152	0.0345314	0.0338638		29.530059		28.959150		2.036028
1	0.491	0.035	0.034	1	29.530	1	28.959	1	2.036
2	0.982	0.069	0.068	2	59.060	2	57.918	2	4.072
3	1.473	0.104	0.102	3	88.590	3	86.877	3	6.108
4	1.965	0.138	0.135	4	118.120	4	115.837	4	8.144
5	2.456	0.173	0.169	5	147.650	5	144.796	5	10.180
6	2.947	0.207	0.203	6	177.180	6	173.755	6	12.216
7	3.438	0.241	0.237	7	206.710	7	202.714	7	14.252
8	3.929	0.276	0.271	8	236.240	8	231.673	8	16.288
9	4.420	0.310	0.305	9	265.771	9	260.632	9	18.324
10	4.912	0.345	0.339	10	295.301	10	289.591	10	20.360
11	5.403	0.380	0.373	11	324.831	11	318.551	11	22.396
12	5.894	0.414	0.406	12	354.361	12	347.510	12	24.432
13	6.385	0.449	0.440	13	383.891	13	376.469	13	26.468
14	6.876	0.483	0.474	14	413.421	14	405.428	14	28.504
15	7.367	0.518	0.508	15	442.951	15	434.387	15	30.540
16	7.858	0.553	0.542	16	472.481	16	463.346	16	32.576
17	8.350	0.587	0.576	17	502.011	17	492.306	17	34.612
18	8.841	0.622	0.610	18	531.541	18	521.265	18	36.649
19	9.332	0.656	0.643	19	561.071	19	550.224	19	38.685
20	9.823	0.691	0.677	20	590.601	20	579.183	20	40.721
21	10.314	0.725	0.711	21	620.131	21	608.142	21	42.757
22	10.805	0.760	0.745	22	649.661	22	637.101	22	44.793
23	11.297	0.794	0.779	23	679.191	23	666.060	23	46.289
24	11.788	0.829	0.813	24	708.721	24	695.020	24	48.865
25	12.279	0.863	0.847	25	738.251	25	723.979	25	50.901
30	14.735	1.036	1.016	30	885.902	30	868.775	30	61.081
35	17.190	1.209	1.185	35	1033.552	35	1013.570	35	71.261
40	19.646	1.381	1.355	40	1181.202	40	1158.366	40	81.441
45	22.102	1.554	1.524	45	1328.853	45	1303.162	45	91.621
50	24.558	1.727	1.693	50	1476.503	50	1447.958	50	101.801
60	29.469	2.072	2.032	60	1771.804	60	1737.549	60	122.162
70	34.381	2.417	2.370	70	2067.104	70	2027.141	70	142.522
80	39.292	2.763	2.709	80	2362.405	80	2316.732	80	162.882
90	44.204	3.108	3.048	90	2657.705	90	2606.324	90	183.243
100	49.115	3.453	3.386	100	2953.006	100	2895.915	100	203.603

4M910030204

Pressure Conversion
Table 203
(Sheet 2 of 2)

C.	C. or F.	F.	C.	C. or F.	F.	C.	C. or F.	F.	C.	C. or F.	F.
-40	-40	-40	10.6	51	123.8	71	160	320	371	700	1292
-34.4	-30	-22	11.1	52	125.6	77	170	338	377	710	1310
-28.9	-20	-4	11.7	53	127.4	82	180	356	382	720	1328
-23.3	-10	14	12.2	54	129.2	88	190	374	388	730	1346
-17.8	0	32	12.8	55	131.0	93	200	392	393	740	1364
-17.2	1	33.8	13.3	56	132.8	99	210	410	399	750	1382
-16.7	2	35.6	13.9	57	134.6	100	212	413	404	760	1400
-16.1	3	37.4	14.4	58	136.4	104	220	428	410	770	1418
-15.6	4	39.2	15.0	59	138.2	110	230	446	416	780	1436
-15	5	41.0	15.6	60	140.0	116	240	464	421	790	1454
-14.4	6	42.8	16.1	61	141.8	121	250	482	427	800	1472
-13.9	7	44.6	16.7	62	143.6	127	260	500	432	810	1490
-13.3	8	46.4	17.2	63	145.4	132	270	518	438	820	1508
-12.8	9	48.2	17.8	64	147.2	138	280	536	443	830	1526
-12.2	10	50.0	18.3	65	149.0	143	290	554	449	840	1544
-11.7	11	51.8	18.9	66	150.8	149	300	572	454	850	1562
-11.1	12	53.6	19.4	67	152.6	154	310	590	460	860	1580
-10.6	13	55.4	20.0	68	154.4	160	320	608	466	870	1598
-10.0	14	57.2	20.6	69	156.2	166	330	626	471	880	1616
-9.44	15	59.0	21.1	70	158.0	171	340	644	477	890	1634
-8.89	16	60.8	21.7	71	159.8	177	350	662	482	900	1652
-8.33	17	62.6	22.2	72	161.6	182	360	680	488	910	1670
-7.78	18	64.4	22.8	73	163.4	188	370	698	493	920	1688
-7.22	19	66.2	23.3	74	165.2	193	380	716	499	930	1706
-6.67	20	68.0	23.9	75	167.0	199	390	734	504	940	1724
-6.11	21	69.8	24.4	76	168.8	204	400	752	510	950	1742
-5.56	22	71.6	25.0	77	170.6	210	410	770	516	960	1760
-5.00	23	73.4	25.6	78	172.4	216	420	788	521	970	1778
-4.44	24	75.2	26.1	79	174.2	221	430	806	527	980	1796
-3.89	25	77.0	26.7	80	176.0	227	440	824	532	990	1814
-3.33	26	78.8	27.2	81	177.8	232	450	842	538	1000	1832
-2.78	27	80.6	27.8	82	179.6	238	460	860	543	1010	1850
-2.22	28	82.4	28.3	83	181.4	243	470	878	549	1020	1868
-1.67	29	84.2	28.9	84	183.2	249	480	896	554	1030	1886
-1.11	30	86.0	29.4	85	185.0	254	490	914	560	1040	1904
-0.56	31	87.8	30.0	86	186.8	260	500	932	566	1050	1922
0	32	89.6	30.6	87	188.5	266	510	950	571	1060	1940
0.56	33	91.4	31.1	88	190.4	271	520	968	577	1070	1958
1.11	34	93.2	31.7	89	192.2	277	530	986	582	1080	1976
1.67	35	95.0	32.2	90	194.0	282	540	1004	588	1090	1994
2.22	36	96.8	32.8	91	195.8	288	550	1022	593	1100	2012
2.78	37	98.6	33.3	92	197.6	293	560	1040	599	1110	2030
3.33	38	100.4	33.9	93	199.4	299	570	1058	604	1120	2048
3.89	39	102.2	34.4	94	201.2	304	580	1076	610	1130	2066
4.44	40	104.0	35.0	95	203.0	310	590	1094	616	1140	2084
5.00	41	105.8	35.6	96	204.8	316	600	1112	621	1150	2102
5.56	42	107.6	36.1	97	206.6	321	610	1130	627	1160	2120
6.11	43	109.4	36.7	98	208.4	327	620	1148	632	1170	2138
6.67	44	111.2	37.2	99	210.0	332	630	1166	636	1180	2156
7.22	45	113.0	37.8	100	212.0	338	640	1184	643	1190	2174
7.78	46	114.8	43	110	230	343	650	1202	649	1200	2192
8.33	47	116.6	49	120	248	349	660	1220	654	1210	2210
8.89	48	118.4	54	130	266	354	670	1238	660	1220	2228
9.44	49	120.2	60	140	284	360	680	1256	666	1230	2246
10.00	50	122.0	66	150	302	366	690	1274	671	1240	2264

4M910030205

Temperature Conversion
Table 204

in	mm	in	mm	in	mm	in	mm	in	mm
0.0001	0.0025	0.001	0.0254	0.01	0.254	0.10	2.540	1.00	25.40
0.0002	0.0050	0.002	0.0508	0.02	0.508	0.20	5.080	2.00	50.80
0.0003	0.0076	0.003	0.0762	0.03	0.762	0.30	7.620	3.00	76.20
0.0004	0.0101	0.004	0.1015	0.04	1.015	0.40	10.160	4.00	101.60
0.0005	0.0127	0.005	0.1270	0.05	1.270	0.50	12.700	5.00	127.00
0.0006	0.0152	0.006	0.1524	0.06	1.524	0.60	15.240	6.00	152.40
0.0007	0.0177	0.007	0.1778	0.07	1.778	0.70	17.780	7.00	177.80
0.0008	0.0203	0.008	0.2032	0.08	2.032	0.80	20.320	8.00	203.20
0.0009	0.0228	0.009	0.2286	0.09	2.286	0.90	22.860	9.00	228.60

4M910030206

Inch Conversion
Table 205

TO: OPERATORS OF SA226–TC AIRCRAFT

Enclosed are Chapter 99 Optional Equipment additions to your SA226 Series Maintenance Manual, Illustrated Parts Catalog, and Wiring Manual.

– –

HIGHLIGHTS

- The list of Effective Pages and Table of Contents for your Chapter 99 manual additions apply only to your aircraft.

- The information in the Chapter 99 additions apply only to your aircraft.

SWEARINGEN AVIATION CORPORATION
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MAINTENANCE MANUAL

CHAPTER 99– OPTIONAL EQUIPMENT

LIST OF EFFECTIVE PAGES

SUBJECT	PAGE	DATE	SUBJECT	PAGE	DATE
99-EFF-PGS	1	JUN 30/07	99-24-04	22	JUN 15/79
99-EFF-PGS	2	JUN 30/07	99-24-04	23	JUN 15/79
99-CONTENTS	1/2	JUN 15/79	99-24-04	24	JUN 15/79
99-CONTENTS	3	JUN 15/79	99-24-04	25	JUN 15/79
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99-24-04	2	JUN 15/79	99-52-02	1	JUN 15/79
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99-24-04	5	JUN 15/79	99-52-02	4	JUN 15/79
99-24-04	6	JUN 15/79	99-52-02	5	JUN 15/79
99-24-04	7	JUN 15/79	99-52-02	6	JUN 15/79
99-24-04	8	JUN 15/79	99-52-02	7	JUN 15/79
99-24-04	9	JUN 15/79	99-52-02	8	JUN 15/79
99-24-04	10	JUN 15/79	99-52-02	9	JUN 15/79
99-24-04	11	JUN 15/79	99-52-02	10	JUN 15/79
99-24-04	12	JUN 15/79	99-52-02	11	JUN 15/79
99-24-04	13	JUN 15/79	99-52-02	12	JUN 15/79
99-24-04	14	JUN 15/79	99-52-02	13	JUN 15/79
99-24-04	15	JUN 15/79	99-52-02	14/15	JUN 15/79
99-24-04	16	JUN 15/79	99-52-02	16	JUN 15/79
99-24-04	17	JUN 15/79	99-53-02	1	JUN 15/79
99-24-04	18	JUN 15/79	99-53-02	2	JUN 15/79
99-24-04	19	JUN 15/79	99-53-02	3	JUN 15/79
99-24-04	20	JUN 15/79	99-53-02	4/5	JUN 30/07
99-24-04	21	JUN 15/79			

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MAINTENANCE MANUAL

CHAPTER 99 – OPTIONAL EQUIPMENT

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			AT	C/O
			TC	C/O
TEST EQUIPMENT DC INPUT		4	T	- - -
THREE-PHASE INVERTER DC INPUT AND MONITORING		6	T	- - -
			AT	C/O
			TC	C/O
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MAINTENANCE PRACTICES			TC	C/O
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MAINTENANCE MANUAL

CHAPTER 99 – OPTIONAL EQUIPMENT

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INSTALLATION				
REMOVAL OF CAMERA GLASS				
AND PLUG		4		
INSTALLATION				
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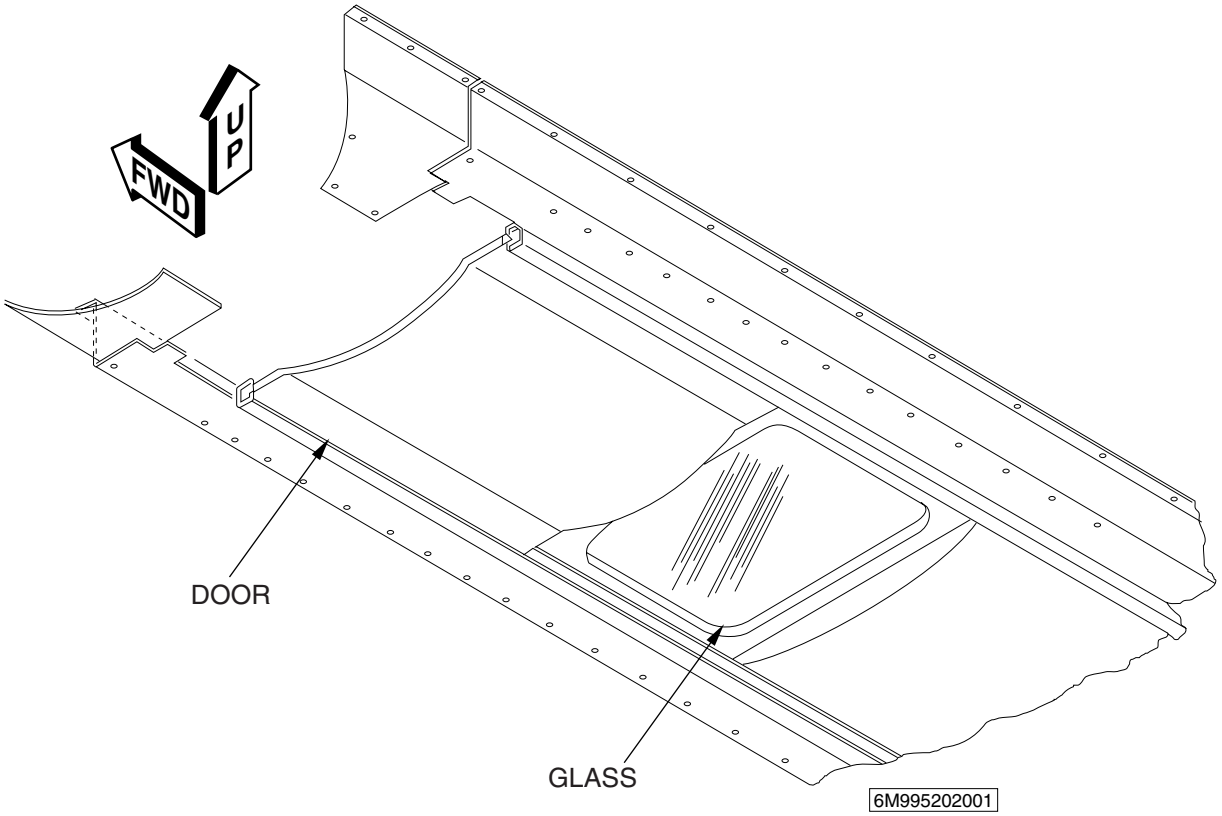
SINGLE CAMERA DOOR



Single Camera Door
Figure 1

SINGLE CAMERA DOOR

A. DESCRIPTION AND OPERATION – SINGLE CAMERA DOOR

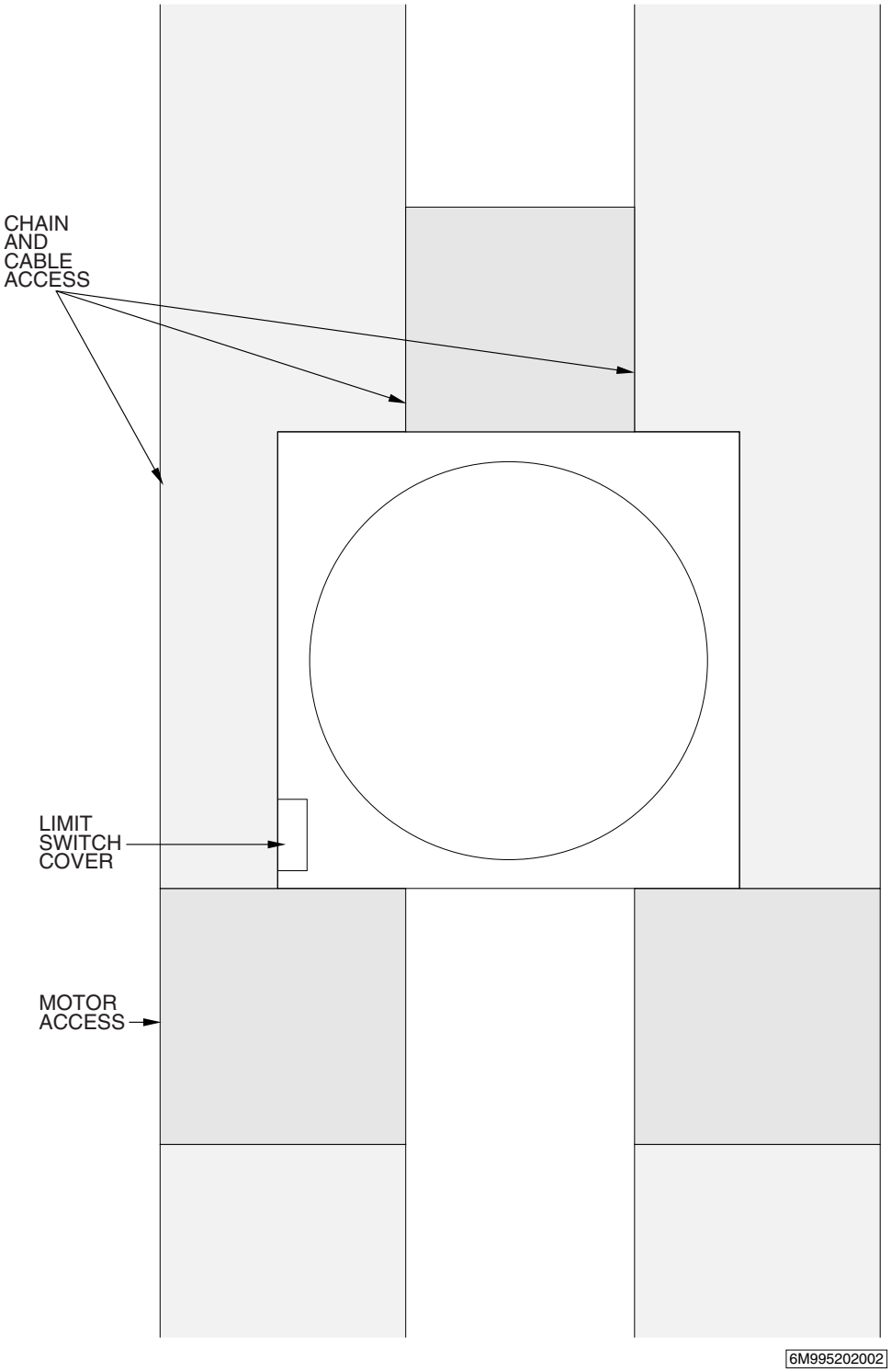
A track mounted electrically powered door is installed to cover the camera glass opening in the aft belly of the aircraft. When the door is completely forward, the entire camera glass opening is exposed (see Figure 1). An electric motor is used to power the door.

The electric motor is located behind the left aft end of the camera opening, under the cabin floorboard. Power to the motor is controlled by a switch in the cockpit and by limit switches. The limit switches stop the motor at each end of the doors' travel and are actuated by a cam geared to the door drive. The door drive consists of a chain driven by a sprocket on the motor shaft and a cable arrangement connected to the chain and each end of the door. A light on the annunciator panel (CAMR DOOR) will illuminate when the door is travelling.

B. MAINTENANCE PRACTICES

Each 200 hours or as part of Letter Check C:

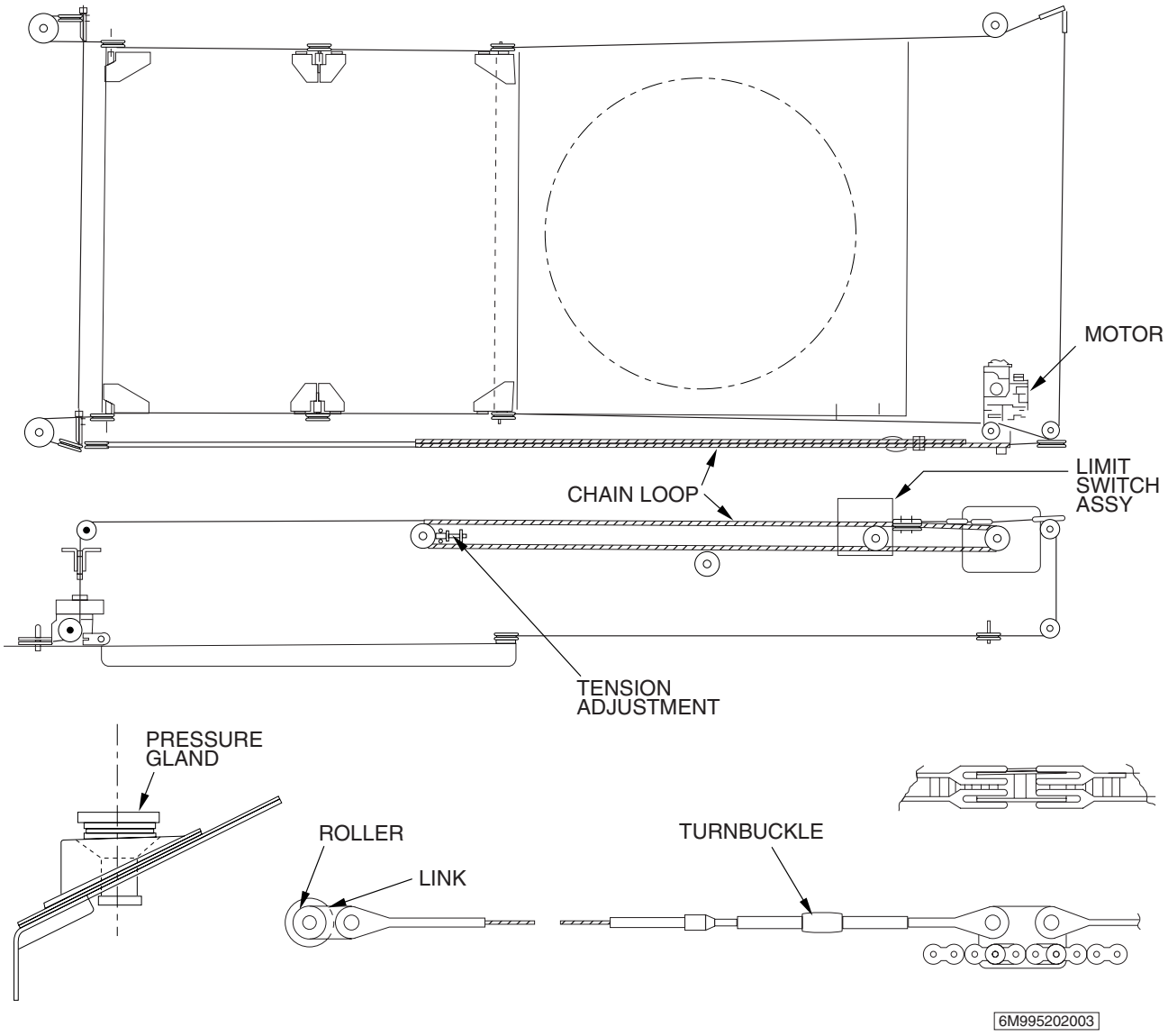
1. Remove the floorboards as indicated in figure 2.
2. Remove cover and inspect the limit switches for security and condition, and secure electrical connection.
3. Run the door to the open position. Check that the cam is contacting and completely depressing the actuator of the open limit switch.
4. Examine chain, all sprockets, pulleys, and cables for condition.
5. Move the door switch to the CLOSE position. As the door moves aft, observe the cable/chain movement for binding or interference. When the door stops it should completely cover the camera glass opening.
6. Check that the cam is contacting and completely depressing the actuator at the close limit switch.
7. Install floorboards and switch cover.
8. During the steps listed above, check the inside of the door tracks for cleanliness. The tracks may be washed with an oil based solvent (Varsol or equivalent) and blown dry with compressed air.



Camera Door Drive Access
Figure 2

9. The chain loop, sprockets, and gears may be lubricated using a light weight oil (10 weight motor oil or equivalent).

10. The cable assemblies may be coated with a corrosion preventive compound (Paralketone or equivalent). The type cable used in this system is pre-coated and does not normally require any type of protective coating; however, frequent washing of the tracks may necessitate coating the cables.



Camera Door Drive System
Figure 3

C. DESCRIPTION AND OPERATION – DOOR DRIVE SYSTEM

The door drive system (see figure 3) consists of a chain loop, a cable system, and an electric motor. The motor drives a sprocket that is the aft end of the chain loop. The forward end of the chain loop runs over an idler sprocket that is used for chain tension adjustment. Another idler sprocket in the middle of the chain loop is used to hold the chain against the sprocket that drives the limit switch assembly. Two sets of cables are attached to the chain at a common point. Turnbuckles are installed at this point to adjust cable tension. One set of cables runs forward, turn down and pass through pressure glands, run aft down the door tracks, and attach to the front of the door. The other set of cables run aft, turn down and pass through pressure glands, run forward down the door tracks, and attach to the aft end of the door.

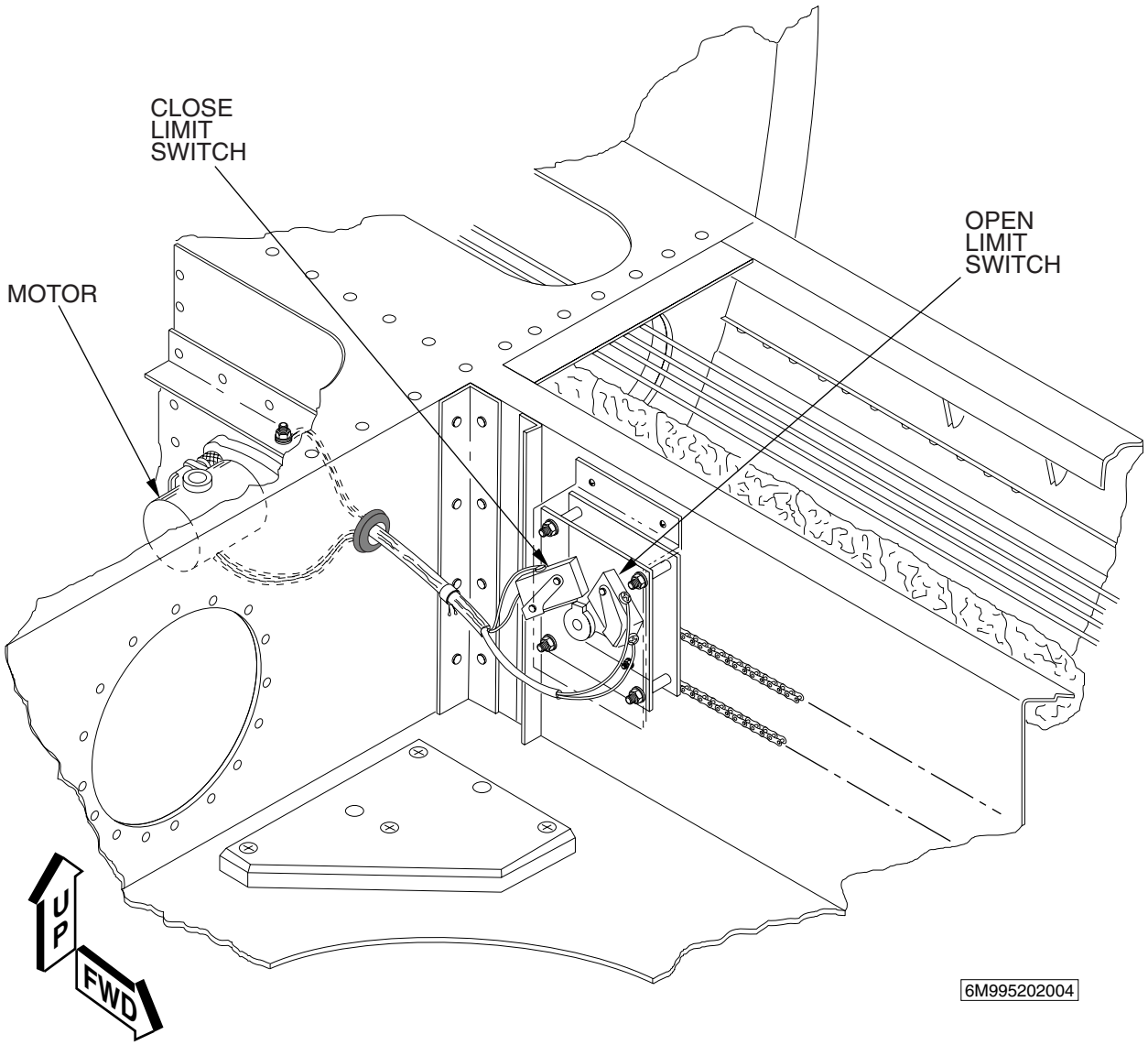
D. MAINTENANCE PRACTICES

1. When connecting forward and aft cables to doors (see figure 2). install pin through clevis, using AN960-10L washers as required to ensure that no end play exists.

2. Adjust the turnbuckles (see figure 3) until the cable tension is 20 to 25 pounds.

3. Operate the door open and closed. Check for cable or cable end interference with pulleys or structure. If any of the cable terminal devices contact pulleys or other guiding devices, the fittings attaching the aft cables to the chain loop may be adjusted in conjunction with the turnbuckles in the forward cables to eliminate cable interference.

4. Loosen the locknut securing the tensioning bolt (see figure 3). Adjust the bolt until the top loop of the chain can just be made to touch the lower loop of the chain with moderate finger pressure (three to five pounds). Tighten the jam nut.



Camera Door Limit Switch
Figure 4

E. DESCRIPTION AND OPERATION – ELECTRICAL SYSTEM

The electrical system consists of a CAMERA DOOR control switch (see figure 4) aft on the left console, a CAMR DOOR light on the annunciator panel, open and close limit switches, and a drive motor. DC power is supplied through the CAMR DOOR circuit breaker on the left essential bus (see 99-52-02 Wiring Diagram Manual). Power is routed from the CAMERA DOOR switch to the open or close connection on the motor through the limit switches. When the door is fully opened or closed, the appropriate limit switch opens the connection to the motor. When the door is in transit, the CAMR DOOR light is illuminated.

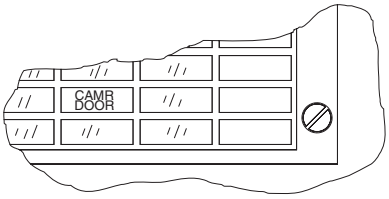
F. MAINTENANCE PRACTICES

CAUTION: DAMAGE TO THE DRIVE SYSTEM OR MOTOR CAN OCCUR IF THE DOOR IS ALLOWED TO OVER TRAVEL IN EITHER DIRECTION.

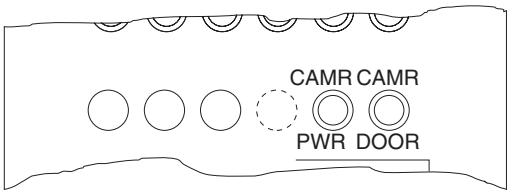
1. Remove the four bolts securing the motor and remove the motor.

CAUTION: DO NOT ALLOW THE MOTOR TO HANG BY THE ELECTRICAL WIRING.

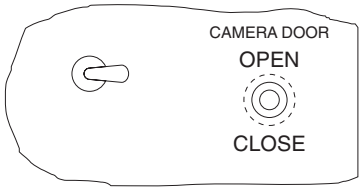
2. Slide the door completely aft and remove the cover on the limit switch assembly and:
   a. Adjust the close limit switch to the middle of the slot.
   b. Adjust the cam to completely depress the actuator of the close limit switch.
3. Slide the door completely forward. Adjust the open limit switch so that the cam completely depresses the actuator.
4. Turn the battery switches on and move the CAMERA DOOR switch to OPEN. Check that the motor is not running and the CAMR DOOR light on the annunciator is illuminated.
5. Slide the door aft approximately half way. Check that the motor is running.
6. Move the CAMERA DOOR switch to CLOSE. Check that the motor is running and the CAMR DOOR light extinguished.



CAMERA DOOR LIGHT ON
THE ANNUNCIATOR PANEL



LEFT ESSENTIAL BUS



CAMERA DOOR CONTROL
SWITCH ON THE REAR
LEFT CONSOLE

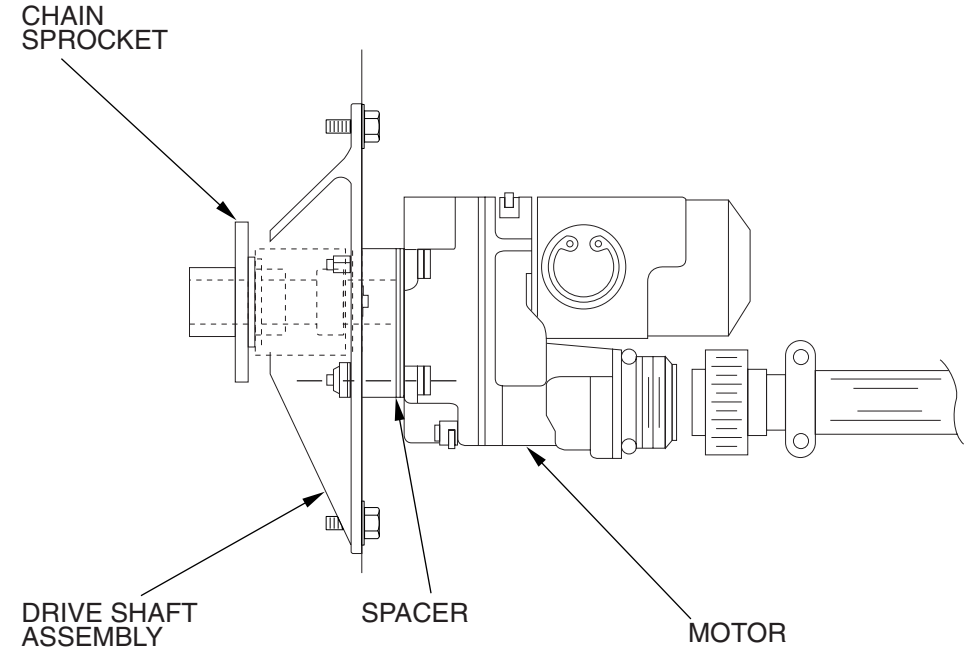
6M995202005

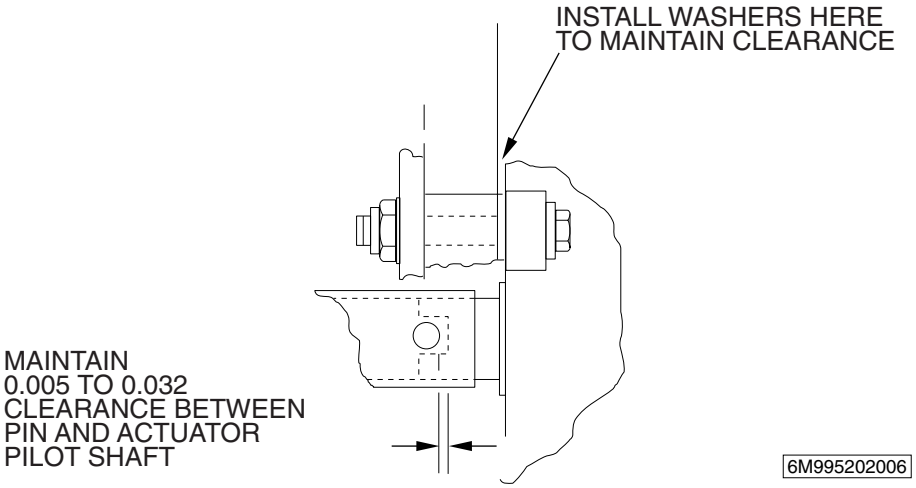
Camera Door Cockpit Electrical
Figure 5

7. Slide door completely aft. Check that the motor is not running.
8. Install motor. Check door operation through full travel.

G. TROUBLESHOOTING

DEFECT	PROBABLE CAUSE	CORRECTIVE ACTION
Circuit breaker trips.	Stalled system or electrical short.	DO NOT ATTEMPT TO RESET BREAKER. 1. Remove electrical connector to motor. 2. Move CAMERA DOOR switch through OPEN and CLOSE . If breaker trips, the electrical system is shorted and must be repaired. If breaker does not trip, proceed to step 3. 3. Remove motor and install electrical connector. Move CAMERA DOOR switch through OPEN and CLOSE. If breaker trips, replace motor. If breaker does not trip, proceed to step 4. 4. Manually move door from full open to full closed, checking for binding or catching and limit switches are fully actuated.





Camera Door Actuator
Figure 6

H. DESCRIPTION AND OPERATION – DRIVE MOTOR

An electric motor P/N MA11A1182 (see figure 6), is used to power the door open and closed. The motor has an output of 55 inch-pounds and a duty cycle of one minute on, four minutes off. The motor is secured by four mounting bolts attaching the base to a bracket under the cabin floor aft and to the left of camera opening.

I. MAINTENANCE PRACTICES

REMOVAL

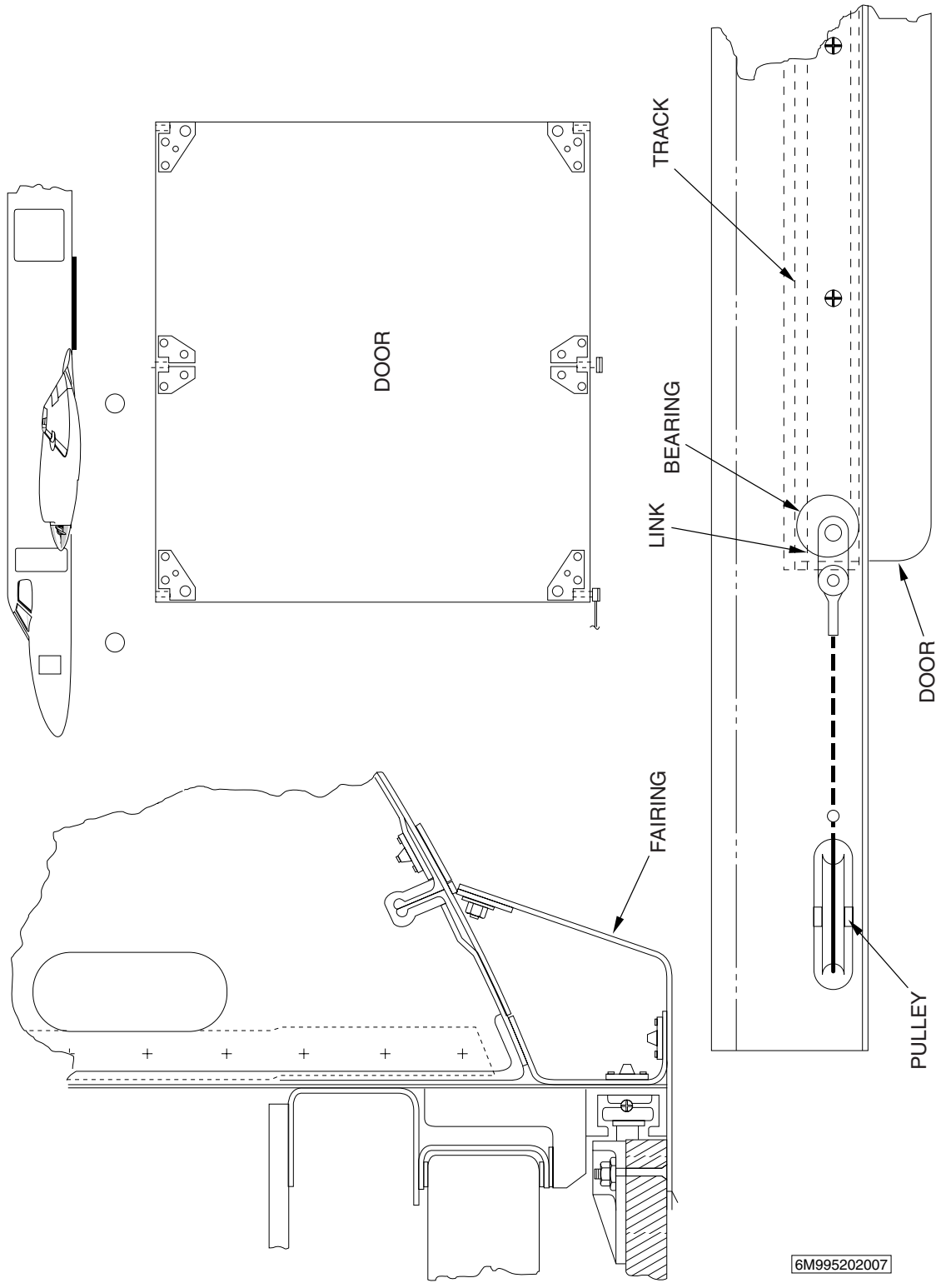
1. Open the floorboard above the motor (see figure 2).
2. Remove the electrical connector on the motor. Remove the four bolts holding the motor in place (see figure 6).
3. Remove motor.

INSTALLATION

1. Open the door approximately halfway.
2. Install the motor into position. It may be necessary to move the door slightly to line up the slot in the motor and the pin in the drive gear. Using the previously removed hardware, bolt the motor in place. Install the electrical connector.
3. Move the CAMERA DOOR switch to the OPEN position. The door should begin moving forward.

   CAUTION: IF THE DOOR BEGINS MOVING AFT, THE CONNECTIONS WITHIN THE MOTOR, OR TO THE MOTOR HAVE BEEN REVERSED. STOP THE DOOR IMMEDIATELY BY MOVING THE DOOR SWITCH TO THE OFF POSITION. DETERMINE THE CAUSE OF THE PROBLEM BEFORE PROCEEDING.

4. Safety the electrical connector to the motor. Install floorboard.



Door and Track Assemblies
Figure 7

J. DESCRIPTION AND OPERATION – DOOR AND TRACK ASSEMBLIES

The camera door is manufactured of honeycomb fiberglass for high strength and light weight. The door is supported by a roller bearing at each of the four corners on center of each side. The bearings roll in tracks that guide the door open or closed (see figure 7).

K. MAINTENANCE PRACTICES

DOOR AND BEARING REMOVAL

1. Remove left and right side fairings.
2. Release cable tension.
3. Remove cotter pins, washers, and clevis pins that attach cable ends to links forward and aft ends of door.

   NOTE: Keep screws, bearings, links, and spacers separated as to their position on door.
4. Slide door forward until bearings come off ends of tracks.
5. Remove screws through links, bearings, and spacers. Remove links, bearings, and spacers.

   NOTE: Observe order of links, bearings, and spacers for installation.
6. Slide forward until center bearings come off end of tracks.

   NOTE: Ensure door is supported.
7. Remove screws through center bearings and spacers. Remove bearings and spacers.
8. Slide door forward and off tracks.
9. Remove aft bearings if they are to be replaced. Remove pulleys at forward end of tracks if necessary.

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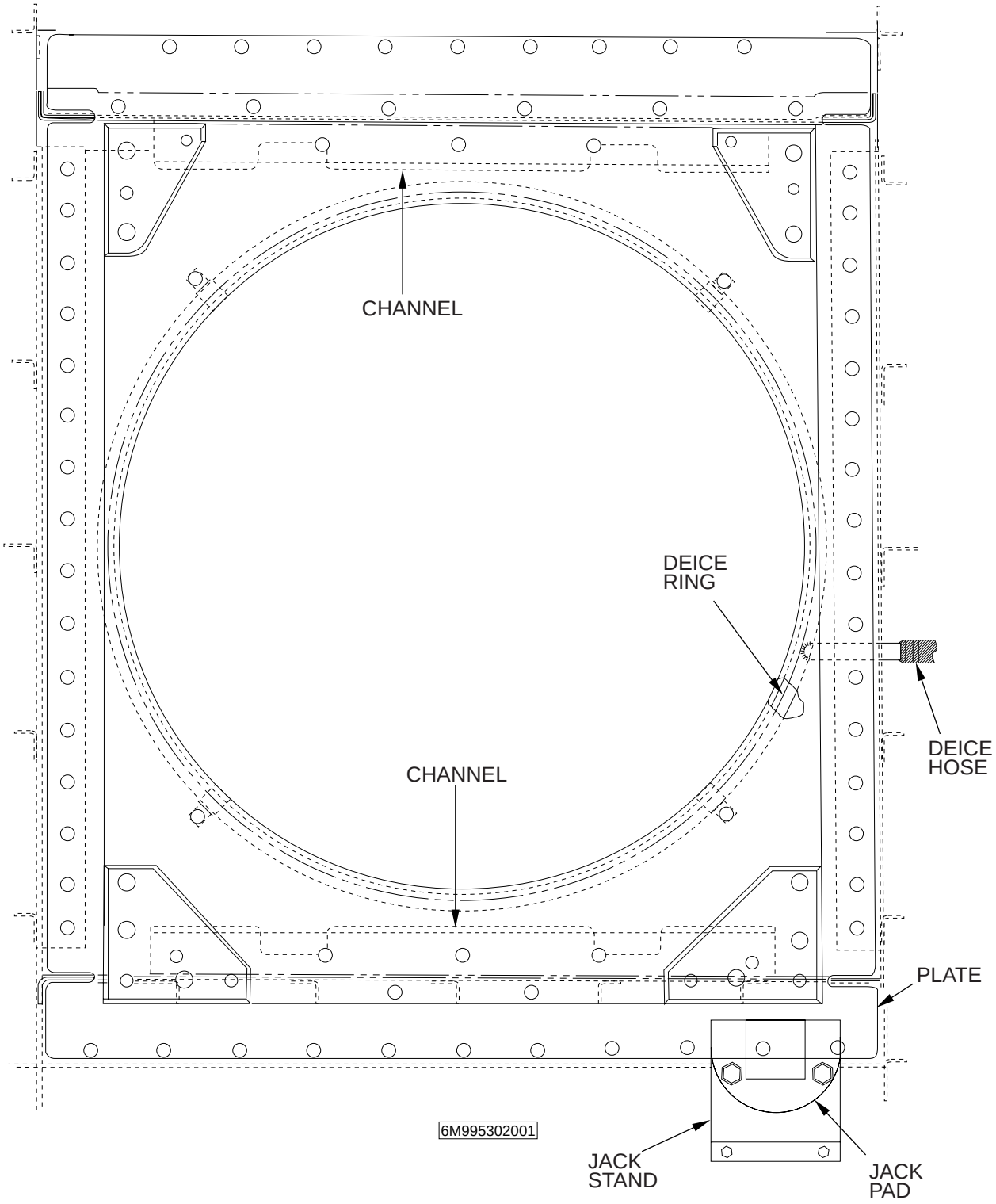
Installation

1. If aft bearings were replaced, ensure that link, bearing, and spacer are installed in correct order.
2. Position door, point links, aft, and slide bearing into tracks.
3. Slide door aft to center door support and install bearings, spacers, and screws in same order as removed.
4. Slide door aft to forward door support and install links, bearings, and spacers in same order as removed.
5. Point links forward and slide door aft.
6. Install pulleys if removed.
7. Install clevis pins through cable ends and links. Install washers and new cotter pins.
8. Set tension on cables.
9. Cycle door through full open and closed check for binding and chafing.
10. Install side fairings.



Camera Mount Plate Assembly
Figure 1

CAMERA PROVISIONS

MAINTENANCE PRACTICES

A. REMOVAL OF CAMERA ADAPTER MOUNT AND PLATE

NOTE: Care should be taken in all operations around or near camera glass to avoid chipping or scratching glass.

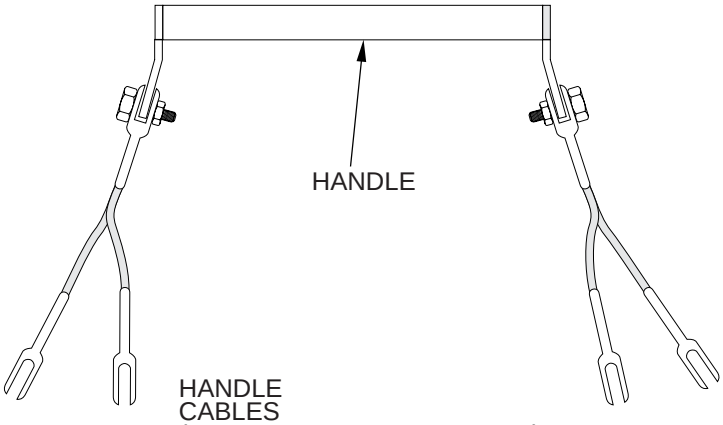
1. Remove electrical connector from camera.
2. Remove bolts holding camera base to adapter mount.
3. Install jack stand and jack.
4. Jack camera up and remove from aircraft.

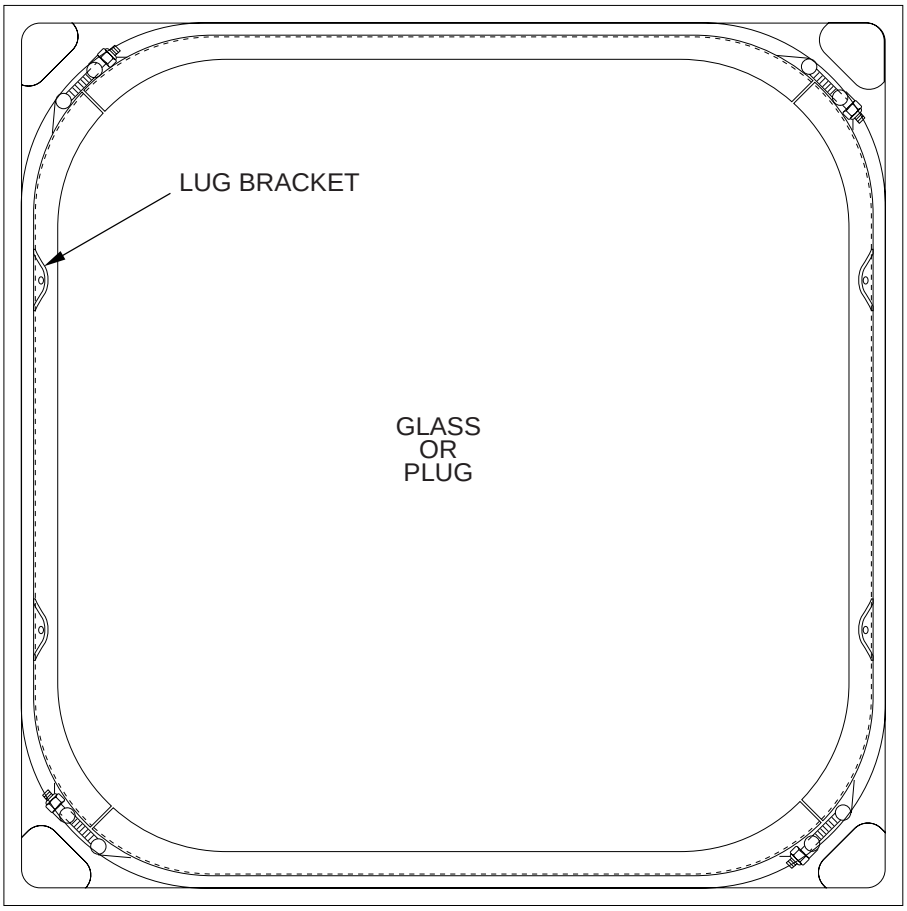
   NOTE: Care should be taken as camera, adapter mount, and plate pass door limit switch assembly.

5. Remove jack and jack stand.
6. Remove bolts holding adapter mount to plate and remove mount.
7. Remove screws attaching plate to channels.
8. Pull plate up and disconnect deice hose and cap hose.

B. INSTALLATION

1. Position plate and install deice hose to deice ring.
2. Push plate down and install screws holding plate to channels.
3. Position adapter mount and install bolts attaching mount to plate.
4. Install jack stand and jack.
5. Install bolts holding camera base to adapter mount.
6. Install electrical connector.





Camera Glass or Plug and Handle
Figure 2

C. REMOVAL – CAMERA GLASS AND PLUG

NOTE: Care should be taken in all operations around or near camera glass to avoid chipping or scratching glass.

1. Remove camera, adapter mount, and plate.
2. Remove screws in four channels holding glass down in position.
3. Position four handle cables on lug brackets on sides of glass and install four bolts.
4. Pull camera glass out and place in protected area.
5. Removal of plug same as steps 3 and 4.

D. INSTALLATION

1. Position four handle cables on lug brackets on sides of glass and install four bolts.
2. Position glass and lower down in place.
3. Installation of plug same as steps 1 and 2.
4. Remove four bolts and remove handle.
5. Position four channels and install screws.
6. Install plate, adapter mount, and camera.

E. CAMERA GLASS CLEANING

1. Remove all excessive amounts of dirt and other substances from the glass surface with clean water.
2. Wash with mild soap and water or a 50/50 solution of isopropanol and water.

NOTE: Do not use any abrasive material, strong acids or bases, MEK, MPK, or Varsol.

3. Clean the surface with a soft cloth or sponge. Wipe with a straight rubbing motion.

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