Beechcraft

Maintenance Manual

P/N 99-590015-1C4

Model 99 Series
U-1 and after

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 05 - TIME LIMITS/MAINTENANCE CHECKS

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TIME LIMITS
OVERHAUL AND REPLACEMENT
GENERAL

1. TIME LIMITS

A. Overhaul and Replacement

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gears are down and locked prior to removing the airplane from the jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

The first overhaul and replacement must be performed not later than the recommended period. The condition of the item at the end of the first period can be used as a criterion for determining subsequent periods applicable to the individual airplane or fleet operation, provided the operator has an approved monitoring system.

The time periods for inspections noted in this manual are based on average usage and average environmental conditions.

NOTE: The recommended periods do not constitute a guarantee the item will reach the period without malfunction as the aforementioned factors cannot be controlled by the manufacturer.

2. SPECIAL CONDITIONS CAUTIONARY NOTICE

WARNING: Prior to performing maintenance on an engine or the airframe, ALWAYS pull the starter control circuit breakers and the landing gear circuit breaker. This will kill power to the starter control as well as the ignitor power relay and the landing gear control relay.

Airplanes operated for air taxi, or other than normal operation, and airplanes operated in humid tropics, or cold and damp climates, etc., may need more frequent inspections for wear, corrosion and/or lack of lubrication. In these areas, periodic inspections should be performed until the operator can set his own inspection periods based on experience.

NOTE: The date noted on the “STANDARD AIRWORTHINESS CERTIFICATE”, FAA Form No. 8100-2, which is issued with each new airplane, is to be used as the basis for all TBO or replacement components listed in the following schedule.

An engine cycle is defined as the period of time from the initial start to shutdown of the engine. This encompasses startup, increase to full power or partial power (as required during a flight regime) and back to complete engine shutdown. Normal operation results in the number of landings being equivalent to engine cycles.

*A flight cycle is defined as: Engine startup and increase to full or partial power (as required during a normal flight), one landing gear retraction and extension and a complete shutdown.

Items not listed are to be overhauled or replaced "On Condition". Conditions for these items are if they are worn, inoperative, inaccurate, intermittent and not repairable through normal maintenance. Only items that are calendar, cycle or hour limited are included in the following list.

3. OVERHAUL AND REPLACEMENT SCHEDULE

A. Landing Gear

ITEM	Overhaul or Replace
Main and Nose Gear Upper Brace Assemblies	Disassemble and inspect the internal surfaces of the upper brace assemblies every 5 years of service. Replace any assembly that has a significant amount of pitting or build up of corrosion.
Main and Nose Gear Shock Strut Assemblies	Every 7,500 cycles* or 5 years.
Main and Nose Gear Torque Knee and Drag Brace Assemblies	Every 7,500 cycles*.
Accumulator	On Condition.
Landing Gear Power Pack	On Condition.
Landing Gear Power Pack Pressure Switch	On Condition.
Shimmy Damper	On Condition. Overhaul or replace if leaks or improper operation are apparent.
Main Gear Wheels and Tires	Refer to Chapter 32 of the Model 99 Airliner Series Component Maintenance Manual.
Nose Gear Wheel and Tire	Refer to Chapter 32 of the Model 99 Airliner Series Component Maintenance Manual.
Main Gear Brakes	Refer to Chapter 32 of the Model 99 Airliner Series Component Maintenance Manual.
Hydraulic Landing Gear Service Valve	On Condition.
Hydraulic Landing Gear Hand Pump	On Condition, if leaks are apparent.
Master Cylinder	On Condition, if leaks are apparent.
All Landing Gear Hoses	Replace when conditions warrant; when cracks or leaks are apparent.
Hydraulic Power Pack Motor (Brushes)	Every 2,500 cycles* or on condition.
Retract Motor, Clutch and Gearbox (Mechanical Gear)	Every 10,000 cycles*.
Landing Gear Motor Controller (Mechanical Gear)	At first 10,000 hours of time in service and every 5,000 hours thereafter. Inspect in accordance with Manual P/N 98-37609B, Test and Adjustment Procedure with IPB for Hartman Motor Controller Model Procedure with IPB for Hartman Motor Controller Model MC815AS1. Repair or replace if required. Motor controller may be replaced an alternate to the above inspection if test facilities are not available.

ITEM	Overhaul or Replace
Actuators (Mechanical Gear)	Every 1,000 cycles* or 30 months, whichever occurs first, check for screw end play as indicated in Chapter 32-30-00 NOSE AND MAIN (MECHANICAL) LANDING GEAR ACTUATOR INSPECTION. Inspect the clevis of each main gear actuator for excessive wear at 1,000 cycles*. If the maximum diameter of clevis hole exceeds 0.512 inch in any direction, the clevis must be replaced.
Actuators (Hydraulic)	Overhaul or replace every 10,000 hours. Check shuttle valve operation every 5,000 cycles* (Ref. Chapter 32-31-00).

B. Power Plant

ITEM	Overhaul or Replace
Starter Generator (Ref. Chapter 24)	Overhaul or replace every 1,000 hours.
Engine TBO	Refer to Pratt and Whitney Service Bulletin 1003 or subsequent for PT6A-20, -27 and -28 engines, or Pratt and Whitney Service Bulletin 1403 or subsequent for PT6A-36 engines. **NOTE** A TBO (time between overhaul) recommendation is in no way to be construed as a warranty or engine life proration basis. The TBO recommendation is based on the projected time for most advantageous initial overhaul. The individual operator's experience may indicate a departure in either direction from the recommended TBO for the particular operation.
Pedestal Power Lever Detent Pin	Every 1,000 hours or when wear exceeds 0.03 inch, check for wear as detailed in Chapter 76-00-00.
Engine Vibration Isolator Mounts	Every 4,000 flight hours. Replace when rubber is deteriorated.
Exhaust System	On Condition. Replace if cracked.
Oil Cooler	On Condition. Replace when contaminated.
All Hoses	On Condition. Replace when cracks or leaks are apparent.

C. Propeller

ITEM	Overhaul or Replace
Propeller	Hub TBO every 3,000 hours or 5 years whichever occurs first.
Propeller Overspeed Governor	At Engine Overhaul, every 4,000 flight hours.
Propeller Governor	At Engine Governor, every 4,000 flight hours.

D. Fuel System

ITEM	Overhaul or Replace
Fuel Cells	On Condition. If leaks are apparent, refer to Chapter 28.
Jet Transfer Pump Motive Valve	On Condition. Overhaul or replace if leaking or malfunctioning
Firewall Shutoff Valve	On Condition or when leakage rate reaches 2.0 cc per minute at 60 psi.
Electric Fuel Boost Pumps	On Condition. Overhaul if leaking or malfunctioning.
Engine Driven Fuel Boost Pump	On Condition. Overhaul if leaking or malfunctioning.
Fuel Drain Collector Pump	On Condition. Overhaul if inoperative or leaking.

E. Instruments

ITEM	Overhaul or Replace
Altimeter	Every 24 Months per FAA Directive.
All Hoses (Instruments)	On condition; when cracks or leaks are apparent.

F. Electrical System

ITEM	Overhaul or Replace
Battery (Emergency Locator Transmitter)	At 50% of useful life as stated on the battery or any time the transmitter is used more than one cumulative hour.

G. Flight Controls

ITEM	Overhaul or Replace
Aileron Tab Actuator	On Condition. Check end play per Chapter 27 of the Model 99 Airliner Series Component Maintenance Manual.
Flap Motor	Every 10,000 cycles*.
Flap Gearbox	Every 10,000 cycles*.
Flap Actuators and 90° Drives	Every 10,000 cycles*.
Flight Controls	On Condition. Any bulges, splits or damages in any pushrod is cause for replacement unless otherwise specified. Any bent or cracked turnbuckle or end fitting is cause for replacement. Any crack in a casting or control system component is cause for replacement. Any broken cable strands or evidence of corrosion in any control system component is cause for replacement unless otherwise specified.
Flap Flexible Shaft	Every 10,000 cycles*.
Rudder Tab Actuator	On Condition. Check end play per Chapter 27 of the Model 99 Airliner Series Component Maintenance Manual.
Horizontal Stabilizer Actuator	After the first 2,000 hours of actuator service and every 2,000 hours thereafter, the actuators must be load tested per Chapter 27-40-00. Overhaul at 20,000 hours.

H. Utility Systems

ITEM	Overhaul or Replace
Cabin Heater, Fuel (U-1 thru U-164, except U-50)	Pressure test every 500 hours of heater operation or every 1,000 hours of airplane operation and overhaul every 3,000 hours of airplane operation.
Heater Ignition	Switch points every 1,000 hours and replace every 2,000 hours of heater operation.
Heater Fuel Shutoff Valve	Overhaul or replace if malfunctioning or leaking.
Heater Fuel Pump (Output Pressure)	Pressure test every 500 hours of heater operation or every 1,000 hours of airplane operation.
Bleed Air Pressure Regulator/Shutoff Valve (U-50, U-165 and After)	Every 5,000 hours or perform bleed air temperature and pressure check. Refer to troubleshooting (Ref. Chapter 21-41-00).

ITEM	Overhaul or Replace
Bleed Air Modulating Valve (U-50, U-165 and After)	Every 5,000 hours or perform bleed air temperature and pressure check. Refer to troubleshooting (Ref. Chapter 21-41-00).
Air Conditioner Condenser Blower Brushes	Every 600 hours of blower operation.
Surface Deicer Boots	On Condition. Inspect for erosion or splits. Repair per vendor instructions.
Deicer Distributor Valve	On Condition. Replace if malfunction occurs.
Windshield Wiper Blade Assembly	On Condition. Replace when blade does not clear wipe area.
Cabin Fire Extinguisher	Hydrostatically test every 12 years.
Engine Fire Extinguisher Bottle	Hydrostatically test every 5 years in accordance with Title 49 CFR Chapter 1, Section 180.209 or replace with a new bottle.
Engine Fire Extinguisher Cartridge (Squib)	Replace squib P/N 13083-5 in conjunction with hydrostatic test of the Engine Fire Extinguisher Bottle (destroy old squib through detonation).
Oxygen Cylinder (Standard Weight) DOT 3AA 1800	Hydrostatically test every 5 years (DOT regulation), service life is unlimited.
Oxygen Regulator (Attached to Standard Weight Oxygen Cylinder)	5 Years.
Oxygen Cylinder (Lightweight) DOT 3HT 1850	Hydrostatically test every 3 years (DOT regulation); replace after 24 years or 4,380 refills.
Oxygen Regulator (Attached to Lightweight Oxygen Cylinder)	3 years.
Oxygen Regulator Valve (Attached to the pilot's sidepanel) (99, 99A, A99, A99A and B99 only)	Every 5,000 hours.
Emergency Exit Signs (Self Illuminating)	Replace seven years from the date of manufacture as stated on the sign, or inspect annually, beginning at five years and replace before the brightness value drops below 100 microlamberts.

I. Structure

ITEM	Overhaul or Replace
Horizontal Stabilizer	
a. Attachment Point Check - Inspect Horizontal Stabilizer Attachment point for wear.	300 hours.

ITEM	Overhaul or Replace
Wing and associated structure fatigue safe life.	**NOTE** Refer to the applicable Airworthiness Directive for safe life of wing and associated structure if Kit No. 99-4023-1 has not been installed.
a. Model 99, 99A, A99, A99A and B99 airplane serials with incorporation of Kit No. 99-4023-1 spar modification (Service Instruction No. 0986)	20,000 hours.
b. Model 99, 99A, A99, A99A and B99 airplane serials with incorporation of Kit No. 99-4023-3 after installation of Kit No. 99-4023-1 spar modification.	An additional 20,000 hours.
c. C99 airplane serials U-50, U-165 and After	20,000 hours.
d. C99 airplane serials U-50, U-165 and After with Kit No. 99-4028-1 spar modification incorporated.	An additional 20,000 hours.
e. Nacelle Splice Plates, Wing Spars, and Wing Attach Bolts.	Inspect and replace as outlined in the Structural Inspection and Repair Manual, P/N 98-39006B or subsequent.

J. Windshield

ITEM	Overhaul or Replace
Windshield	If cracking occurs in only the outer ply of the windshield, flight may be conducted provided the following procedures and restrictions are observed: a. Cracking must not significantly impair visibility. b. Cracking must not interfere with use of windshield wipers. c. Heating elements must be operative. If cracking of inner ply is determined to exist, the windshield should be replaced prior to the next flight. **NOTE** Inner ply cracking is not a structural consideration for replacement prior to next flight, but one of possible glass flaking which could interfere with pilot's vision.

TIME LIMITS
SCHEDULED MAINTENANCE CHECKS
MAINTENANCE PRACTICES

1. GENERAL

NOTE: For a listing of alternative inspection programs available for use with this airplane, refer to the current Publications Price List CD ROM (P/N 994-32808). For information on these publications contact the Technical Manual Distribution Center (TMDC) at 1-800-796-2665, fax (316) 676-4824 or E-mail TMDC@hawkerbeechcraft.com.

The time periods for the inspections noted in this schedule are based on normal usage under average environmental conditions. Airplanes operated in humid tropics, or in cold, damp climates, etc., may need more frequent inspections for wear, corrosion, lubrication, and/or lack of maintenance. Under these adverse conditions, perform periodic inspections in compliance with this guide at more frequent intervals until the owner or operator can set his own inspection periods based on the contingencies of field experience. Airplanes operated less than 100 hours a year must have a 100-Hour Inspection performed no later than 12 months following the date of the preceding 100-Hour Inspection. The 100-hour interval between performance of the procedures specified herein should NEVER be exceeded by more than 10%, which can be used only if the additional time is required to reach a place where the inspection can be satisfactorily accomplished. However, any extension of a 100-hour interval must be subtracted from the following 100-hour interval. For example, if an inspection is done at 110 hours, the next inspection is due 90 hours later.

NOTE: Any time an airplane is repainted or touched up, inspect all placards and decals to assure they are not covered with paint, are easily readable, and are securely attached. Replace any placards that have been inadvertently defaced or removed.

This inspection program in accordance with 14 CFR Part 91.169 consist of, but is not limited to, inspection items listed in the Inspection Guide, any applicable Airworthiness Directives issued against the airframe or any equipment installed therein and conformity to Type Certificate Data Sheet and Maintenance Manual Airworthiness Limitations Chapter (Chapter 4) as applicable.

The owner or operator is responsible for maintaining the airplane in an airworthy condition, including compliance with all applicable Airworthiness Directives as specified in Part 39 of Title 14 of the Code of Federal Regulations (14 CFR). It is further the responsibility of the owner or operator to ensure that the airplane is inspected in conformity with the requirements covered in 14 CFR Parts 43 and 91. These 14 CFR Parts cover the requirements concerning the Inspection Guide. Hawker Beechcraft Corporation has prepared this inspection guide to assist the owner or operator in meeting the foregoing responsibilities. This inspection guide is not intended to be all-inclusive, for no such guide can replace the good judgement of a certified airframe and power plant mechanic in the performance of his duties. As the one primarily responsible for the airworthiness of the airplane, the owner or operator should select only qualified personnel to maintain the airplane.

While the inspection guide may be used as an outline, detailed information of the many systems and components in the airplane will be found in the various sections/chapters of this maintenance manual and the pertinent supplier publications. It is also recommended that reference be made to the applicable Maintenance Handbooks, Service Instructions, Hawker Beechcraft Corporation Service Bulletins, applicable FAA Regulations and Publications, suppliers bulletins and specifications for torque values, clearances, settings, tolerances, and other requirements. It is the responsibility of the owner/operator to ensure the airframe and power plant mechanic inspecting the airplane has access to previously noted documents as well as to this inspection guide.

Hawker Beechcraft Corporation issues service information for the benefit of owners and operators in the form of two classes of Service Bulletins. MANDATORY (Red Border) Service Bulletins are changes, inspections or modifications that could affect safety. The factory considers compliance with these Service Bulletins mandatory. OPTIONAL (No Border) Service Bulletins cover changes, modifications, improvements or inspections which may benefit the owner. Due to the wide range of information covered by the OPTIONAL Service Bulletin, each owner or operator is responsible for conducting a thorough review of each OPTIONAL Service Bulletin to determine if compliance is required based on the applicability of the OPTIONAL Service Bulletin to their particular set of operating conditions. It is the responsibility of the owner or operator to ensure that all Hawker Beechcraft Corporation Service Bulletins which are pertinent to their particular operation are complied with.

NOTE: In addition to the inspections prescribed by this schedule, the altimeter system and all ATC transponders MUST be tested and inspected at 24-month intervals in compliance with the requirements specified in 14 CFR Parts 91.171 and 91.172.

Owners or operators wishing to utilize Inspection Intervals less than 100 hours may do so without obtaining Hawker Beechcraft Corporation concurrence. Refer to the latest revision of this form for inspection procedure details to meet 14 CFR 91.169 (f) (4).

Permission is granted for maintenance personnel to reprint the inspection form. Reprinting for resale is prohibited.

A. Wings

2. 100 HOUR INSPECTION PROCEDURE

A. Power Plant

Complete Steps 1 thru 26.	**Mechanic**		**Inspector**
	LH	**RH**	
(1) COWLING - Check adjustment of latches. Remove upper cowling and clean all cowling. Inspect for cracks and repair as necessary. Inspect all latches for condition.			
(2) ENGINE OIL FILTER - Inspect the oil filter for metal particles as described in the Pratt and Whitney Aircraft of Canada Ltd. engine maintenance data. Replace as required.			
(3) DRAIN PLUGS - Check all drain plugs for security. Check the magnetic chip detector as described in Chapter 12-10-00 for metal particles. Place a steel bar across the magnetic poles and check that the light in the annunciator panel illuminates. Make sure the light on the annunciator panel goes out when the bar is removed.			
(4) FUEL FILTERS AND SCREENS - Inspect the filter for evidence of foreign matter, corrosion, or microbiological growth in the fuel system. Inspect and clean the engine fuel pump screen and filter and the firewall filter as described in Chapter 28-00-00. Inspect transfer line fuel filter for condition and cleanliness.			

Complete Steps 1 thru 26.	Mechanic		Inspector
	LH	RH	
(5) IGNITION GLOW PLUGS - Inspect the glow plugs as described in the engine maintenance data. If necessary, clean them as instructed in the Engine Maintenance Manual. IGNITER PLUGS - Inspect the plugs as described in Chapter 12-10-00. If necessary, replace them as instructed in the Engine Maintenance Manual.			
(6) ENGINE EXHAUST SYSTEM - Remove exhaust stacks. Examine the exhaust section, turning vanes, and visible portion of the power turbine for burning, distortion, damage and cracks. If any are found, refer to the Pratt and Whitney Aircraft of Canada Ltd. engine maintenance data for corrective action.			
(7) INDUCTION SYSTEM - Check the air intake duct and engine inlet screen for obstruction and damage. Remove the air inlet screen; inspect compressor inlet area, struts and first stage blades for dirt deposits, corrosion, erosion, and foreign object damage.			
(8) OIL COOLER - Check cooler and plumbing for condition and attachment.			
(9) PLUMBING - Inspect plumbing and associated equipment and accessories for condition and attachment.			
(10) ENGINE AND PROPELLER CONTROLS - Check controls and associated equipment for condition, attachment and rig, and inspect power control seals for deterioration.			
(11) ENGINE ACCESSORIES - Inspect all accessories for condition, attachment, and leakage.			
(12) ENGINE MOUNTS - Inspect for condition and attachment. Replace or repair the vibration isolator mount as required. All mounts on an engine must be of the same manufacture. The mount may be rebuilt by replacing the rubber cushion. Refer to Chapter 71-20-00 of the Model 99 Airliner Series Component Maintenance Manual.			
(13) ELECTRICAL WIRING AND EQUIPMENT - Inspect wiring and associated equipment and accessories for condition and attachment.			
(14) FUEL CONTROL HEATERS - Check operation, condition and security of attachment.			
(15) INERTIAL ANTI-ICER - Check vane for freedom of movement and proper travel.			

Complete Steps 1 thru 26.		Mechanic		Inspector
		LH	RH	
(16)	DEICER BOOTS - Check propeller and air inlet boots for condition and security of attachment; check propeller deice slip rings for gouges, cracks, burned or discolored areas, and deposits of grease, oil or dirt. Clean as necessary.			
(17)	PROPELLERS - Check for condition, attachment and operation. Check condition of mechanical feedback ring, stop rods and springs. Inspect the carbon block pin for freedom of movement. Check for no metal-to-metal contact between the brass ring and the reverse lever. Check the reversing linkage for proper adjustment, operation, evidence of binding and security of attachment. Check for proper operation of all pedestal controls and switches. Check the tips of the blades for evidence of lightning strikes. If there is evidence of lightning strikes, check the area of entering or leaving the blade for hardness. The inside and outside shank area should be checked for arcing. In all cases of lightning strikes, the thrust bearings should be replaced, soft blades rejected, and all steel parts demagnetized. For U-1 thru U-164, except U-50 - Check the secondary low pitch stop switch assembly for attachment and condition on airplanes equipped with PT6A-20 engines or on airplane serials prior to U-165 (except U-50) equipped with PT6A-27 or PT6A-28 engines not in compliance with Service Instruction No. 0844-247.			
(18)	PROPELLER SYNCHRONIZER - For U-1 thru U-164, except U-50 - Check drive unit for condition, security and electrical connection. Check flexible shaft and trimmer for security of attachment and rigging. Check rod end bearings for lubrication.			
(19)	PROPELLER SYNCHROPHASER - For U-50, U-165 and After - Inspect the propeller governor for condition, security, electrical connection, wear and rigging. Check rod end bearings for condition and lubrication. Check for proper gap between sensor and target as instructed in Chapter 61-23-00.			
(20)	AUTOFEATHER RELAYS, DUMP SOLENOIDS AND PRESSURE SWITCHES - Check for condition, security of attachment and proper electrical connection.			
(21)	AUTO-IGNITION PRESSURE SWITCHES - Check for condition, security of attachment and proper electrical connection.			
(22)	ENGINE - Perform hot section inspection, when required. Refer to the appropriate Pratt and Whitney Engine Maintenance data.			

Complete Steps 1 thru 26.	Mechanic		Inspector
	LH	RH	
(23) STARTER GENERATOR - Inspect one set of brushes for indication of excessive wear or damage (determine wear by observing diagonal groove on brush). Inspect inlet duct and blast cap for cracks or obstruction. Inspect QAD attach flange. For U-1 thru U-164, except U-50 and airplanes in compliance with Pratt & Whitney Service Bulletin No. 1386 - At 200 hours remove generator and lubricate drive splines.			
(24) FUEL DRAIN COLLECTOR SYSTEM - Remove the drain collector tank and clean once a year. Check for security of attachment of tank and pump. Check hoses for cracks, deterioration and proper attachment. Clean and check the pump screen per Chapter 71-70-00 every third inspection.			
(25) WATER/METHANOL SPRAY RING - Check for condition, security of attachment and restricted orifices. Check overboard vent for obstructions. Check and clean or replace inline filter annually. Check all plumbing fittings for leaks.			
(26) FIRE EXTINGUISHER - Check for presence of activation voltage to the cartridge/squib. Perform the EXTINGUISHER ACTIVATION CHECK procedure in Chapter 26-20-00.			

B. Air Conditioning

Complete Steps 1 thru 9.	Mechanic	Inspector
(1) COMPRESSOR - Check for condition and attachment, and oil leaks. Check clutch for proper gap of 0.022 to 0.057 inch.		
(2) COMPRESSOR DRIVE BELTS - Check for condition. Check adjustment as outlined in Chapter 21-50-00.		
(3) REFRIGERANT LINES AND SERVICE VALVES - Inspect lines and valves for condition and attachment.		
(4) AIR FILTERS - Replace recirculation filter and the evaporator filter in accordance with the Service Schedule in Chapter 12-20-00.		
(5) BLEED AIR CONTROL UNIT - Check for proper operation per Chapter 21-41-00.		
(6) HEATERS (FUEL) - For U-1 thru U-164, except U-50 - Check the fuel plumbing, pumps and regulator for condition and attachment. Clean and inspect the two fuel filters, one located at the inlet port of the fuel control valve in the top of the left side of the center section and the other in the left main landing gear wheel well. Check the heater for proper operation in all modes. Inspect the drain lines for obstruction.		

Complete Steps 1 thru 9.	Mechanic	Inspector
(7) HEATERS (ELECTRIC) - For U-50, U-165 and After - Inspect heater, heating elements and associated wiring and ducting for condition and attachment.		
(8) HEATER FUSE - Examine fuse for evidence of deterioration. Replace if deterioration is evident. Check for continuity.		
(9) COMPRESSOR DRIVE QUILL SHAFT - Check for wear. Lubricate the splines on both ends with lubricant (111, Table 6, 91-00-00) every 600 hours of engine operation (Airplanes complying with Pratt & Whitney SB 1354 lubricate aft spline only).		

C. Nose Section

Complete Steps 1 thru 10.	Mechanic	Inspector
(1) SKIN - Inspect skin and structure for condition and loose or missing rivets.		
(2) STRUCTURE - Check for cracks, loose rivets and concealed damage.		
(3) RADIO EQUIPMENT - Inspect radio rack structure; check security of units in their mounts. Check radio controls for security and proper operation.		
(4) BRAKE RESERVOIR - Inspect reservoir and fill as needed.		
(5) DOOR FASTENER AND SEAL - Inspect seal for deterioration, doors, and latches for proper adjustment.		
(6) BATTERY - (nose installation) Remove the battery, inspect the general condition of the battery, battery box and cables. Service the battery in accordance with the applicable Battery Maintenance Manual. Refer to Chapter 12 of this manual or to Chapter 24 of the Model 99 Airliner Component Maintenance Manual P/N 99-590015-5 or subsequent. (Do not use any device which has been used for lead acid batteries).		
(7) STATIC PORTS - Check and clean as necessary. For U-50, U-165 and After - Check for proper plumbing per baggage pod installation in Chapter 34-10-00.		
(8) PITOT SYSTEM - Check and clean as necessary. Drain pitot line.		
(9) INSTRUMENT AIR FILTER - For U-50, U-165 and After - Inspect for condition. Replace every fourth inspection or sooner if conditions warrant, such as operation in heavy smoke or dust.		
(10) OXYGEN SYSTEM - Check condition of oxygen system and check the oxygen masks for cleanliness and stowage.		

D. Pilot's Compartment

Complete Steps 1 thru 14.	Mechanic	Inspector
(1) CABLES AND PULLEYS - Check control cables, pulleys and associated equipment for condition, attachment, alignment, clearance and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter. INSPECT AND RECORD AILERON AND AILERON TAB CABLE TENSIONS: TEMPERATURE:_____degrees F. Control Column Interconnect cable tension:_____ 1/8 in. Aileron Cable Tension: LEFT_____RIGHT_____ Aileron Tab Cable Tension:_____		
(2) BRAKE SYSTEM - Check brake system components for leakage. Inspect brake master cylinder for proper operation, brake line plumbing for condition and attachment, brake pedals and linkage for condition, attachment and proper operation. Check for proper pedal travel and parking brake release.		
(3) RUDDER PEDALS - Check rudder pedal condition, clearance and attachment. Remove the two pivot bolts of each pedal and thoroughly examine the pivot holes and adjacent area for wear per the Model 99 Airliner Continued Airworthiness Program Inspection Manual.		
(4) INSTRUMENT PLUMBING AND WIRING - Inspect instrument panel, subpanel, placards, shock mounts, instruments and instrument plumbing and wiring for condition, attachment, chafing, etc.		
(5) CONTROL COLUMN - Check condition, attachment and operation. Check flight control lock for condition, positive locking and alignment. Check extension cord and control wheel switches. Every fifth inspection, inspect control wheel adapter for cracks.		
(6) HEATER - Check cabin heat shut-off valve and defroster valve for proper operation. Check all ducts for condition.		

Complete Steps 1 thru 14.	Mechanic	Inspector
(7) ELECTRICAL EQUIPMENT - Check the operation of all pilot compartment and panel lights. At each 12th inspection, remove and inspect the edgelighted panels as outlined in Chapter 33-11-00. Check switches, knobs and circuit breakers for looseness and operation. Check all wiring for chafing, security, etc. Prior to U-165 except U-50, each fourth inspection, check the dual bus feeder diodes for open or shorted diodes. Refer to Chapter 24 or to the Model 99 Airliner Series Wiring Diagram Manual, P/N 99-590028 or subsequent. Check AC bus switching relays.		
(8) EMERGENCY STATIC AIR SOURCE - Check for security and operation of control.		
(9) ENGINE AND PROPELLER CONTROLS - Check for freedom of movement, full travel and proper friction lock operation.		
(10) SEATS, SEAT BELTS AND SEAT TRACKS - Inspect seats and seat belts for condition, proper operation and attachment. Inspect seat tracks for wear. Refer to Chapter 25-10-00.		
(11) WINDOWS - Inspect windows and seal for general condition.		
(12) WINDSHIELDS - Inspect electrically heated windshields for antistatic coating and tab bonding. Refer to Chapter 56-10-00.		
(a) Inspect windshield fillet seals for deterioration and voids.		
(13) PILOTS COMPARTMENT HATCH - (optional) Check release handle and latch assembly for proper operation. Check that the hatch moves out freely. Check the complete hatch assembly for condition and all moving parts for proper operation. With hatch closed, check for proper latching and seal.		
(14) PITCH TRIM - Examine pitch trim wires for chafing, proper separation or protection from other wires.		

E. Cabin Section

Complete Steps 1 thru 20.	Mechanic	Inspector
(1) SKIN - Inspect skin for condition and loose or missing rivets. If damage is found, check adjacent structure.		
(2) STRUCTURE - Check for cracks, loose or missing rivets and concealed damage.		
(3) CABLES AND PULLEYS - Check control cables, pulleys and associated equipment for condition, attachment, alignment, clearance and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter.		
(4) LANDING GEAR GEARBOX (Mechanical Gear Only) - Check for security and leakage. Lubricate as necessary.		
(5) LANDING GEAR POWER PACK (Hydraulic Gear Only) - Check for security and leakage. Inspect and replace filters and screens as instructed in Chapter 12-10-00.		
(6) FLAP MOTOR AND DRIVES - Inspect for condition and attachment, lubricate as necessary.		
(7) AILERON QUADRANT - Check for security, attachment, operation and proper travel.		
(8) WINDOWS - Inspect windows and window seals for general condition.		
(9) CABIN ENTRANCE DOOR - Inspect door, seals, latching mechanism, and cables for condition and security of attachment. Check that folding steps do not engage too soon and that they fold properly without interference.		
(10) CARGO DOOR - (Optional) Inspect door, seals and latching mechanism for condition and security of attachment.		
(11) ELECTRICAL WIRING AND EQUIPMENT - Check cabin and compartment lights for condition and attachment, replace all bulbs as necessary. Inspect wiring and equipment for condition and attachment at every fifth inspection.		
(12) HEATER DUCTS - Check cabin hot and cold outlets and valves for condition, obstruction and proper operation. Inspect heating and cooling ducts for condition and attachment.		
(13) EJECTOR AND DEICER DISTRIBUTOR VALVE - Check equipment and plumbing for security and condition.		

Complete Steps 1 thru 20.	Mechanic	Inspector
(14) INSTRUMENT AIR FILTER - For U-1 thru U-151, except U-50 - Replace every fourth inspection or sooner if conditions warrant, such as operation in heavy dust or smoke.		
(15) SEATS, SEAT BELTS AND SEAT TRACKS - Check seats and seat belts for condition, proper operation and attachment. Check seat tracks for wear as described in Chapter 25-20-00.		
(16) ACCESS DOORS - Check for fit and attachment.		
(17) AUTOPILOT - Inspect and replace air filters if dirty. Check the components for damaged cases, bent or broken pins and electrical connectors, damaged seals, poor solder connections, and damaged insulation on wiring. Check the plumbing for condition and security of attachment.		
(18) EMERGENCY EXIT HATCH - Check the emergency release handle and latch assembly for proper operation. Check that the hatch moves out freely. Check complete hatch assembly for condition and all moving parts for proper operation. Check for proper latching and seals of closed hatch.		
(19) BELLY DRAIN HOLES - Inspect for possible obstruction.		
(20) RADIO ANTENNAS - Check for cracks and proper attachment.		

F. Rear Fuselage and Empennage

Complete Steps 1 thru 17.	Mechanic	Inspector
(1) SKIN - Inspect skin for condition and loose or missing rivets. If damage is found, check adjacent structure.		
(2) STRUCTURE - Check for cracks, loose rivets and concealed damage.		
(3) CABLES AND PULLEYS - Check control cables, pulleys and associated equipment for condition, attachment, alignment, clearance and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter. INSPECT AND RECORD ELEVATOR, RUDDER AND RUDDER TAB CABLE TENSIONS: TEMPERATURE:_____degrees F. Elevator cable tension: UP_____DOWN_____ Rudder cable tension: LEFT_____RIGHT_____ Rudder tab cable tension:_____		
(4) EMPENNAGE SURFACES - Inspect skin for condition and loose or missing rivets. Check surfaces for proper attachment and freedom of movement. Check trim actuators and motors for proper operation, smoothness, and attachment.		
(5) EMPENNAGE STRUCTURE - Check for cracks, skin panels for distortion, loose rivets and/or concealed damage.		
(6) RUDDER AND RUDDER TAB - Check for freedom of movement and attachment. Check rudder tab free play as instructed in Chapter 27-20-00.		
(7) ELEVATOR - Check for freedom of movement and attachment. Check aft bell crank for cracks and loose taper pins per Chapter 27-30-00. For U-1 thru U-148, except U-50 - Check for elongation of holes or cracks in the elevator horn casting and attachment flange and replace if conditions are noted per Service Instruction No. 0157-133 Rev 3 or subsequent.		
(8) PLUMBING - Inspect for condition and attachment.		
(9) DEICE BOOTS - Check for condition and attachment.		
(10) NAVIGATION LIGHTS AND ROTATING BEACONS - Check for cracked or broken lenses. Replace bulbs as necessary.		

Complete Steps 1 thru 17.	Mechanic	Inspector
(11) ACCESS DOORS - Check fit and attachment.		
(12) VENTRAL FIN AND AFT FUSELAGE DRAINS - Check the drain holes in the ventral fin for obstructions. Remove the door and check for obstructions of the drain holes in the base of the stringers on each side of the door at the point of juncture with the aft bulkhead. Check for obstruction of the drain hole in the base of the drain tube adjacent to where the relief hose attaches to the drain tube. Check bayonet-type drain for obstruction.		
(13) HORIZONTAL STABILIZER - Check for freedom of movement and attachment. For U-1 thru U-145, except U-50 and U-132 - Check the horizontal stabilizer elevator hinge fitting assembly every fifth inspection as indicated in Service Instruction No. 0437-132. For U-1 thru U-157, except U-50 - Inspect wiper brush as instructed in Service Instruction No. 0841.		
(14) HORIZONTAL STABILIZER TRIM ACTUATOR - Load test the horizontal stabilizer trim actuator every 2,000 hours of actuator operation as instructed in Chapter 27-40-00.		
(15) ELEVATOR MAIN SPAR LEADING EDGE RIB FLANGE - For U-1 thru U-164, except U-50 - Check for cracks per Service Instruction No. 0799-133 or subsequent every fourth inspection.		
(16) ELEVATOR CONTROL PUSH RODS - Check for condition as instructed in Chapter 27-30-00.		
(17) ELEVATOR TORQUE TUBE COLLAR - Inspect each torque tube assembly collar for cracks and loose taper pins as instructed in Chapter 27-30-00.		

G. Wings

Complete Steps 1 thru 16.	Mechanic		Inspector
	LH	RH	
(1) SKIN - Inspect skin for condition and loose or missing rivets. If damage is found, check adjacent structure.			
(2) STRUCTURE - Check for cracks, loose rivets and concealed damage. Inspect the wing attach fittings, spar caps and structure as outlined in the latest issue of the Structural Inspection and Repair Manual (P/N 98-39006).			
(3) FRONT SPAR CAP - Check for corrosion after 5 years of service and every year thereafter. Refer to Chapter 57-00-00 of the Structural Inspection and Repair Manual (P/N 98-39006).			
(4) ACCESS DOORS - Check for fit and attachment.			

Complete Steps 1 thru 16.	Mechanic		Inspector
	LH	RH	
(5) CABLES AND PULLEYS - Check control cables, pulleys and associated equipment for condition, attachment, alignment, clearance and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter. INSPECT AND RECORD AILERON AND AILERON TAB CABLE TENSIONS: TEMPERATURE:_____degrees F. Aileron Cable Tension: LEFT_____RIGHT_____ Aileron Tab Cable Tension:_____			
(6) AILERON AND TAB - Inspect skin for condition and loose or missing rivets. Check surfaces for proper attachment and freedom of movement. Check trim tab actuator for proper operation, smoothness and attachment. Check trim tab free play as instructed in Chapter 27-10-00.			
(7) FLAPS AND ACTUATORS - Inspect flap drive and actuator for condition and attachment. Check skin and structure for condition and loose or missing rivets.			
(8) LIGHTS - Check navigation and landing lights for broken lenses and replace bulbs as necessary.			
(9) FUEL CELLS AND VENTS - Inspect fuel cells for leaks and vents for obstruction. Perform the FUEL CELLS INSPECTION procedure in Chapter 28-10-00.			
(10) FUEL PROBES - Check for leaks at point of attachment.			
(11) WING BOLTS - Refer to Chapter 57-00-00 for wing bolt, nut, and fitting inspection criterion and frequency. Check drain holes in upper wing bolt fittings for possible obstruction and clear as required.			
(12) PLUMBING - Check for leaks, chafing or damage and proper attachment.			
(13) ELECTRICAL WIRING AND EQUIPMENT - Check for security, chafing, damage and attachment every fifth inspection.			
(14) DEICER BOOTS - Check for condition and attachment.			
(15) LIFT TRANSDUCER - Check for proper heating and aural operation.			
(16) STROBE LIGHTS - Check for proper operation, broken or cracked lenses and replace flash tube as necessary.			

H. Wing Center Section

Complete Steps 1 thru 14.	Mechanic		Inspector
	LH	RH	
(1) SKIN - Inspect for condition and loose or missing rivets. If damage is found, check adjacent structure.			
(2) STRUCTURE - Check for cracks, loose rivets and concealed damage. Inspect the wing attachment fittings, spar caps and structure as outlined in the latest issue of the Structural Inspection and Repair Manual.			
(3) FUEL CELLS, VENTS, VALVES AND PUMPS - Inspect the fuel cells for leaks and plugged vents. Check the pumps, drain valves, firewall shutoff valves and the crossfeed valve for proper operation, pressure leaks and security of attachment.			
(4) FUEL PROBES - Check for leaks at point of attachment. Inspect nacelle fuel probes for contamination and micro sludge buildup every 2,500 hours or 30 months. Clean probes thoroughly before reinstallation.			
(5) FLAPS AND ACTUATORS - Inspect flap drive and actuator for condition and attachment. Check skin and structure for condition and loose or missing rivets.			
(6) LEADING EDGE AND NACELLE PLUMBING AND WIRING - Check the plumbing for damage, security of attachment and leaks. Every fifth inspection, check the wiring for chafing, security of attachment and electrical connections.			
(7) BATTERY (Wing installed) - Remove the battery, inspect the general condition of the battery, battery box and cables. Service the battery in accordance with the applicable Battery Maintenance Manual. Refer to Chapter 24-31-00 of this manual. (Do not use any device which has been used for lead acid type batteries.)			
(8) CABLES AND PULLEYS - Check control cables, pulleys, and associated equipment for condition, attachment, alignment, clearance, and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter.			
(9) FIRE DETECTORS - Check all detectors for discrimination and sensitivity. Check the control amplifier circuits per Chapter 26-10-00.			

Complete Steps 1 thru 14.	Mechanic		Inspector
	LH	RH	
(10) LANDING GEAR TORQUE SHAFTS (Mechanical Gear Only) - Inspect shafts and universal joints for condition, attachment and visible damage.			
(11) LANDING GEAR PLUMBING (Hydraulic Only) - Check for security and leakage.			
(12) ACCESS DOORS - Check for fit and attachment.			
(13) ENGINE FIRE EXTINGUISHER - Check the pressure of the supply cylinder. Inspect the plumbing for leakage and security of attachment.			
(14) BLEED AIR PRESSURE REGULATORS - Check for security and inspect and clean as necessary.			

I. Main Gear and Brakes

Complete Steps 1 thru 9.	Mechanic		Inspector
	LH	RH	
(1) BRAKES - Check brake disc, and linings and lines for wear, damage, corrosion and security of all units.			
(2) WHEELS - Inspect wheels as instructed in manuals P/N 98-35161, or 98-33818 or subsequent. Refer to Chapter 32 of the Model 99 Airliner Series Component Maintenance Manual P/N 99-590015-5.			
(3) TIRES - Check tires for condition, wear and proper inflation.			
(4) ACTUATOR (Mechanical) - Check actuator upper cover and bottom housing assembly for condition and support brackets for cracks at mounting lugs.			
(5) ACTUATOR (Hydraulic) - Check for leakage and support brackets for cracks at mounting lugs. Check for seepage from the end cap. If seepage is evident, inspect the end cap for possible cracks using a 10 power magnifying glass and flashlight.			
(6) LANDING GEAR STRUT - Inspect shock strut and components for condition, attachment, proper inflation and leakage. Deflate and check fluid level if signs of leakage are apparent. Remove the attaching bolt at the torque knee joint and inspect the connection joint of the torque knee halves as instructed in Chapter 32-10-00.			
(7) ELECTRICAL - Check for proper operation of switches. Clean dirt from the terminals and connectors. Check wiring for damage.			

Complete Steps 1 thru 9.	Mechanic		Inspector
	LH	RH	
(8) BRAKE DEICER (If Installed) - Check the plumbing and hose for leakage and security of attachment. Check the distributor manifold for condition and security of attachment. Check that the orifices in the manifold are clean and unobstructed.			
(9) DRAG LEG - Check security of attach fittings. Check downlock bolts for security, proper safety and possible overtorque at the downlock causing drag leg to bind. Inspect for binding of the drag leg caused by possible overtorque at the downlock. Inspect drag leg downlock hook attach hole as instructed in the Model 99 Airliner Series Component Maintenance Manual 99-590015-5.			

J. Nose Gear

Complete Steps 1 thru 10.	Mechanic	Inspector
(1) NOSE STEERING STOP - Check stop for distortion; check for damage to steering linkage if distortion is found. Remove stop on nose gear casting and check lug for cracks, reinstall per Chapter 32 of Model 99 Airliner Series Component Maintenance Manual P/N 99-590015-5.		
(2) WHEEL - Inspect wheel as instructed in P/N 98-33818 or 98-35161 or subsequent, in Chapter 32 of the Model 99 Airliner Series Component Maintenance Manual P/N 99-590015-5.		
(3) TIRE - Check tire for condition, wear and proper inflation.		
(4) SHIMMY DAMPER - Inspect for condition and attachment. Refill if necessary as outlined in Chapter 12-20-00.		
(5) ACTUATOR (Mechanical) - Check actuator upper cover and bottom housing assembly for condition and support brackets for cracks at mounting lugs.		
(6) ACTUATOR (Hydraulic) - Check for cracks in the end cap and at the support bracket lugs. Check for leaks at the shaft seals and end caps.		
(7) STEERING LINKAGE - Check nose steering mechanism for condition, attachment and correct adjustment per Chapter 32-50-00.		
(8) NOSE GEAR STRUT - Inspect strut and components for condition and attachment. At each 100 hour inspection following the initial inspection at 1,000 hours, inspect the lower shock absorber assembly per Service Bulletin No. 2102 or subsequent.		
(9) STRUT FLUID LEVEL - Inspect strut for proper inflation and leakage. Deflate and check fluid level if signs of leakage are apparent per Chapter 12-20-00.		

Complete Steps 1 thru 10.	Mechanic	Inspector
(10) ELECTRICAL - Check for proper operation of switches, clean dirt from terminals and connectors as required; check wiring for damage.		

K. Landing Gear Retraction

Complete Steps 1 thru 11.	Mechanic	Inspector
NOTE The Landing Gear Retraction System on all new airplanes should be inspected at the first 100 hours of flight and at 200-hour intervals thereafter. ***CAUTION*** *Since battery voltage is not sufficient to properly cycle the landing gear, use only an external power source capable of delivering and maintaining 28.25 ± 0.25 volts throughout the extension and retraction cycles when performing the landing gear retraction inspection. Jack the airplane before starting the landing retraction inspection.*		
(1) RETRACT MECHANISM (Mechanical and Hydraulic) - Check the retraction system for proper operation of all components through at least two complete cycles. Check for unusual noise and evidence of binding. Check for security of attachment, and leakage of all plumbing lines (hydraulic gear only). Measure the amount of wear in the lower drag leg attach lug holes on the nose gear brace. (If it exceeds 0.635 inch diameter, refer to LANDING GEAR CLEANING AND PARTS REPLACEMENT in Chapter 32-20-00 of the Model 99 Airliner Series Component Maintenance Manual P/N 99-590015-5.) The DU bushing installation in the drag leg legs should be checked for wear at 1,500 cycles. A fully run-in bushing will have a surface appearance similar to that of a bronze bearing. A grey-green bushing that sheds feathery particles is not fully run-in. A bushing with its surface coating worn down to the steel backing should be replaced immediately, for it has reached the point of maximum wear. On serials U-50, U-149, U-152 and After and those earlier airplanes in compliance with Service Instruction No. 0516-200 Rev 2 or subsequent, the DU bushings have been replaced with steel bushings and lubrication fittings.		
(2) DOORS AND LINKAGE - Check door operation, fit, fair and rig.		
(3) POSITION INDICATORS - Check for security and adjustment of switches, loose wires and proper indication.		
(4) WARNING HORN - Check for proper operation.		

Complete Steps 1 thru 11.	Mechanic	Inspector
(5) DOWNLOCKS - Check locking mechanism for positive engagement in extended position. Check the down position indicator switch for proper adjustment. Check the downlock bolts for the possibility of overtorque at the downlock assembly.		
(6) SAFETY SWITCH - Check for security and proper operation.		
(7) ACTUATORS (Mechanical System Only) - Check for noise, binding and proper operation.		
(8) ACTUATORS (Hydraulic Gear Only) - Check for leakage and proper rigging. Check for seepage from the end cap. If seepage is indicated, inspect the end cap for possible cracks using a 10 power magnifying glass and a flashlight.		
(9) LIMIT SWITCHES - Check for proper adjustment and attachment. ***CAUTION*** *Do not continue operation after receiving a gear down indication. Further movement of the handle could damage the drive mechanism and prevent subsequent electrical gear retraction. The landing gear cannot be retracted manually.*		
(10) EMERGENCY EXTENSION (Mechanical) - Check system for freedom of operation and positive engagement of downlocks. EMERGENCY EXTENSION (Hydraulic) - Check system for freedom of movement and proper operation as instructed in Chapter 32-31-01.		
(11) NOSE GEAR RETRACT CHAIN (Mechanical Gear) - Inspect for broken links, corrosion, dirt, rust and proper tension. Clean and lubricate as outlined in Chapter 32-30-00.		

L. Operational Inspection

Complete Steps 1 thru 36.	Mechanic	Inspector
NOTE The applicable Pilot's Operating Manual or Pilot's Operating Handbook should be used to assist during the following inspection procedures. The following Operational Inspection procedures are to be applied during the start and run of the engine:		
(1) ENGINE CONTROLS - Check for freedom of movement, full travel, and proper friction lock.		
(2) FUEL TRANSFER JET PUMPS - Check for proper operation.		
(3) STARTERS - Check for proper operation.		
(4) IGNITION - Check for annunciator panel light illumination.		

Complete Steps 1 thru 36.	Mechanic	Inspector
(5) OIL - Check for proper pressure and temperature limits.		
(6) FUEL QUANTITY GAGES - Check for proper operation.		
(7) FUEL CROSSFEED VALVE - Check for proper operation.		
(8) BOOST PUMPS, CROSSFEED VALVE, FIREWALL SHUTOFF VALVES, AND THRESHOLD LIGHT - Check for proper operation. If any unit fails to operate, check the fuses.		
(9) INTERSTAGE TURBINE TEMPERATURE - Check for proper limits on engine start.		
(10) VACUUM SYSTEM - Check for proper limits.		
(11) GYRO INSTRUMENTS - Check for erratic or noisy operation.		
(12) PROPELLERS - Check for proper operation. Perform flight idle torque checks. Perform secondary low pitch stop checks on airplanes equipped with PT6A-20 engines or on airplane serials prior to U-165, except U-50 equipped with PT6A-27 or PT6A-28 engines not in compliance with Service Instructions No. 0844-247 or subsequent. Refer to Chapter 61.		
(13) AUTOFEATHER - Check for instructions in Chapter 61-21-00.		
(14) PROPELLER SYNCHRONIZER - Check for proper operation.		
(15) PROPELLER SYNCHROPHASER - Check for proper operation.		
(16) PROPELLER GOVERNOR - Check for proper governor operation and feathering.		
(17) POWER CHECK (Without Jetcal) - Check in accordance with Chapter 77 for the procedure covering this check.		
(18) IDLE RPM - Check for correct rpm (both high and low rpm).		
(19) DEICER (Surface, windshield and brake, if installed) - Check for proper operation and cycling.		
(20) AIR CONDITIONING SYSTEM - Check for proper operation in manual heat, manual cool and all automatic modes. Check operation of all outlets and ease of all controls.		
(21) FLIGHT CONTROLLER - (Optional) Check for proper operation.		
(22) REFRIGERANT LEVEL - Check for proper level.		
(23) GENERATOR - Check the output.		
(24) PNEUMATIC SYSTEM - Check for proper limits.		
(25) CONDITION LEVER - Check for clean shut-down at IDLE CUT-OFF.		
(26) WATER/METHANOL INJECTION SYSTEM - Check for proper operation.		

Complete Steps 1 thru 36.	Mechanic	Inspector
NOTE The following Operational Inspection procedures are to be made with the engine shut down:		
(27) BRAKES (Toe and parking) - Check for proper travel, ease of operation and proper release of the parking brake.		
(28) PROPELLER DEICER - Check for proper operation and cycling. Refer to Chapter 30-60-00.		
(29) ENGINE INERTIAL ANTI-ICER - Check for proper operation and rig per Chapter 30-21-00.		
(30) ENGINE AIR INLET DEICER - Check for proper operation. Refer to Chapter 30-20-00.		
(31) PITOT HEAT - Check amperage drawn on ammeter and for proper heating at the unit.		
(32) ALL LIGHTS (Including annunciator panel lights) - Check for proper operation and replace defective bulbs as necessary.		
(33) STALL WARNING - Check for proper operation. Refer to Chapter 27-31-00.		
(34) AURAL TRIM WARNING SYSTEM - Check for proper operation. Refer to Chapter 27-40-00.		
(35) FLAPS - Check for noisy or erratic operation.		
(36) EMERGENCY LOCATOR TRANSMITTER - Check for proper operation and assure ELT is ARMED before returning airplane to service.		

M. General Service Items

Complete Steps 1 thru 5.	Mechanic	Inspector
(1) AIRPLANE - Clean and service as required.		
(2) AIRPLANE LUBRICATION - Lubricate as necessary. Refer to the Lubrication Schedule in Chapter 12-20-00.		
(3) ENGINES - Inspect after ground run-up or flight test. Check for oil leaks, security and attachment of all components.		
(4) PLACARDS - Verify that all appropriate exterior and interior placards are in place and legible.		
(5) Airworthiness Directives and Service Bulletins reviewed and complied with as required.		

3. SPECIAL INSPECTIONS

A. Propeller Deicer System Inspection

The various components of the propeller deicer system should be inspected every 50 hours for the appearance of defects. The following inspection may provide a means of detecting and correcting such defects before they render the deicer system inoperative.

B. Electric Propeller Deicer (50-Hour Guide)

(1) Lock the brakes and operate the engines at near take-off power. Turn the deicer system switch ON and observe the ammeter for at least 2 minutes (U-1 thru U-164, except U-50), or 3 minutes (U-50, U-165 and After). If the ammeter needle does not rest within the shaded band, except for a flicker at 30 second (U-1 thru U-164, except U-50), or 90 second (U-50, U-165 and After) intervals when the step switch of the timer cycles, refer to the troubleshooting chart for the probable sources of trouble.

WARNING: Prior to performing maintenance on an engine, ALWAYS pull the starter control circuit breakers. This will kill power to the starter control as well as the igniter power relay. Always stand clear of the propeller blade path when moving the propeller.

(2) With the engines shut off, turn the deicer switch ON and, while rotating the propellers, feel the deicer boots for the proper sequence of heater operation (at least four cycles should be allowed to occur during this procedure). The presence of local hot spots indicate service damage to the deicer heaters, which should be repaired before more serious damage develops.

(3) Remove the spinner dome and open all access doors pertaining to the wiring and components of the deicer system. Turn the deicer switch ON and station an assistant in the cockpit to observe the system ammeter. Flex all accessible wiring, particularly the lead straps, leads from the slip ring assembly, and the firewall electrical connectors and their wiring. Any movement of the ammeter, other than the cycling flicker that occurs at 30 or 90 second intervals, indicates a short or open circuit that must be located and corrected.

(4) To extend the life of the lead strap between the hub clamp and clip, reposition the bend in the strap at a point at least 1/2 inch from the existing location of the bend.

(5) Check for damaged brush rods or springs and for worn or damaged brushes.

C. Electric Propeller Deicer (100-Hour Guide)

(1) Repeat the 50-Hour inspection.

(2) Check for radio noise or radio compass interference by operating the engine at near take-off power with the radio gear turned ON. If, under these conditions, noise or interference occurs when the deicer switch is ON and disappears when the switch is OFF, refer to the troubleshooting chart for the probable source of trouble.

WARNING: Prior to performing maintenance on an engine, ALWAYS pull the starter control circuit breakers. This will kill power to the starter control as well as the igniter power relay. Always stand clear of the propeller blade path when moving the propeller.

(3) Check all clamps, clips, mounting, electrical connections, and connectors for tightness and electrical soundness. Check also for loose, broken or missing safety wire.

(4) Closely check the deicer boots for wrinkled, loose, or torn areas, particularly around the outboard end and at the point where the strap passes under the clips on the hub clamp. Look for abrasion or cuts along the leading edge and the flat or thrust face. If the heater wires are exposed in damaged areas or if the rubber is found to be tacky, swollen, or deteriorated (as from contact with oil or solvent fluids), replace the damaged deicer boot.

(5) Check that the clips of the hub clamps are tight. Inspect for cracks or other damage. Check to see that the cushioning material is not missing or damaged in the area under the clip on the hub clamp or on the edge of the spinner dome. Manually operate the propeller from low pitch to feathering while checking that the deicer lead straps do not come under tension.

(6) Check the slip rings for gouges, roughened surface, cracks, burned or discolored areas, and for deposits of oil, grease, or dirt. Clean greasy or contaminated slip rings with solvent (30, Table 6, 91-00-00). After such a cleaning, a run-in time of 5 hours of engine operation must be allowed before the deicer system is turned ON.

(7) If uneven wear or wobble is detected, check the alignment of the slip rings on the prop shaft with a dial indicator. While turning the propeller to check the slip ring alignment, push in on the propeller to eliminate play in the propeller thrust bearing. If the runout over 360° of rotation is over 0.005 inch or exceeds 0.002 inch in any 4 inch arc, refer to the paragraph on Slip Ring Alignment in Chapter 30.

(8) Examine the brush mounting brackets and housings for cracks, deformation, or other indications of damage. Check for tight connections and that the leads are not chafed or binding.

CAUTION: *While following the instructions of Step i., move the propeller back and forth to prevent arcing between the brushes and the slip ring.*

(9) Check to see that each brush rides fully on its slip ring over 360° of rotation. If the brush is not properly aligned, add shims under the brush block or elongate the holes in the mounting brackets to raise or lower the brush block to the proper position. If the brushes ride BOTH high and low with respect to the slip rings in 360° of rotation, the slip ring assembly is eccentrically mounted and the spinner bulkhead must be replaced.

(10) Check for proper spacing between the brush block and slip rings. If this distance is not within the specified limits, loosen the mounting screws and reposition them in the elongated holes until the block is properly positioned. If necessary, shims can be added between the thrust bearing plate and mounting bracket until the brush block is properly located.

(11) Estimate the angle of the brush block in relation to the slip rings. If this angle is not approximately 2°, loosen the mounting bolts and reposition the brush block until the proper angle exists between the brush block and slip rings. It should be noted that the spacing established in Step i. must also be maintained after the proper contact angle is established.

(12) With the deicer system operating and a man in the cockpit observing the ammeter, visually inspect and physically flex the wiring from the brush blocks to each component of the deicer system and to the airplane power supply. Jumps of the ammeter needle, other than the momentary flicker that occurs when the timer switches at 30 or 90 second intervals, indicate loose or broken wiring in the area under examination at the moment. In such instances, continue to flex the wiring in the area that first indicated trouble while checking the continuity through the individual wires of the affected

harness until the source of trouble is located. Refer to the Model 99 Airliner Series Wiring Diagram Manual, P/N 99-590028 or C99 Airliner Wiring Diagram Manual P/N 99-590030-5 for the circuitry of the deicer system.

TIME LIMITS
400-HOUR CONTINUOUS INSPECTION
MAINTENANCE PRACTICES

1. GENERAL

NOTE: Hawker Beechcraft's Recommended Inspection Program in accordance with Title 14 Code of Federal Regulations (14 CFR) 91.169 (f) (4) consists of, but is not limited to Mandatory Service Bulletins, previously issued Service Instructions, all Maintenance Manual Chapter 05 Time Limits-Overhaul or Replacement Items, all other inspection items listed in this Inspection Guide and any applicable Airworthiness Directives issued against the airframe or any equipment installed therein.

When an airplane is delivered new from the factory, an operator may commence inspections immediately in accordance with the program outlined in the Continuing Care Guide. However, if the airplane has been in service under a different inspection program, Hawker Beechcraft recommends that a complete 100 Hour Long Form Inspection be performed before the Continuing Care Inspection Guide is initiated.

The owner or operator is primarily responsible for maintaining the airplane in an airworthy condition, including compliance with all applicable Airworthiness Directives as specified in 14 CFR Part 39. It is further the responsibility of the owner or operator to ensure that the airplane is inspected in conformity with the requirements covered in 14 CFR Part 91.169 (f) (4). These 14 CFR Parts cover the requirements concerning the Continuous Inspection Procedures Program and approved airplane inspection programs. Hawker Beechcraft Corporation has prepared this Continuous Inspection Procedures Section to assist the owner or operator in meeting the foregoing responsibilities. This inspection guide is not intended to be all inclusive, for no such guide can replace the good judgement of a certified airframe and power plant mechanic in the performance of his duties. As the one primarily responsible for the airworthiness of the airplane, the owner or operator should select only qualified personnel to maintain the airplane.

NOTE: Material contained in this Inspection, including the inspection intervals, may be changed at any time by the owner/operator, with prior notification and approval of the local FAA General Aviation District Office, when warranted by service experience or engineering recommendations. Information contained herein is applicable to all Model 99 Airliner Series airplanes except where differences are indicated by serial effectivity.

The first overhaul and replacement must be performed not later than the required period. The condition of the item at the end of the first period can be used as a criterion for determining subsequent periods applicable to the individual airplane or fleet operation, providing the operator has an approved monitoring system.

While this inspection guide may be used as an outline, detailed information of the many systems and components in the airplane will be found in the various chapters/sections of the maintenance manual and the pertinent supplier publications. It is also recommended that reference be made to the applicable maintenance handbooks, service bulletins, applicable FAA Regulations, Publications, and supplier's specifications for torque values, clearances, settings, tolerances, and other requirements.

NOTE: Verify that all interior and exterior placards are legible and in place. Replace any placards that have been defaced by paint or other repairs.

Hawker Beechcraft highly recommends compliance with Optional Service Bulletins in that they will usually benefit the operator and in some cases be a necessity. All applicable Optional Service Bulletins not complied with as of the date of this inspection should be reviewed by the owner/operator and the agency complying with this inspection for necessity of compliance based on the operator's particular operating environment, conditions area, climate and any other items which might be pertinent to the continued operation of the airplane.

In the final analysis, it is the responsibility of the owner or operator to ensure that the airframe and power plant mechanic inspecting the airplane has access to the previously noted documents as well as to this inspection guide.

NOTE: In addition to the inspections prescribed by this schedule, the altimeter instrument and static system and all ATC transponders MUST be tested and inspected at 24 month intervals in compliance with the requirements specified in Parts 91.171 and 91.172 under Title 14 of the code of Federal Regulations.

The time periods for the inspections noted in this inspection guide are set up to enable the owner or operator to perform a complete inspection (Routine Inspections every 50 hours and all four 100 Hour Interval Inspections) of the airplane each 12 calendar months. A complete inspection of the airplane must be accomplished within each 12 calendar months for compliance with the Federal Aviation Regulations. The time periods for inspections stated in this inspection guide should NEVER be exceeded by more than 10 hours, and then only if the additional time is required to reach a place where the inspections can be satisfactorily accomplished. The time used to reach an inspection facility must be deducted from the next inspection time. If 10 hours were used to reach the inspection facility, the next 100 hour interval inspection would be due in 90 hours.

2. SPECIAL CONDITIONS CAUTIONARY NOTICE

The time periods for the inspections noted in this schedule are based on normal usage under average environmental conditions. Airplanes operated in extremely humid tropics, or in exceptionally cold, damp climates, etc., may need more frequent inspections for wear, corrosion, lubrication, and/or lack of maintenance. Under these adverse conditions, perform periodic inspections in compliance with this guide at more frequent intervals until the owner or operator can set his own inspection periods based on the contingencies of the field experience.

NOTE: The recommended periods do not constitute a guarantee the item will reach the period without malfunctions as the aforementioned factors cannot be controlled by the manufacturer.

WARNING: Use only genuine Hawker Beechcraft Corporation or Hawker Beechcraft Corporation approved parts obtained from Hawker Beechcraft Corporation approved sources, in connection with the maintenance and repair of Hawker Beechcraft Corporation airplanes.

Genuine Hawker Beechcraft Corporation parts are produced and inspected under rigorous procedures to insure airworthiness and suitability for use in Hawker Beechcraft Corporation airplane applications. Parts purchased from sources other than those approved by Hawker Beechcraft Corporation, even though outwardly identical in appearance, may not have had the required tests and inspections performed, may be different in fabrication techniques and materials, and may be dangerous when installed in an airplane.

Hawker Beechcraft Corporation expressly disclaims any responsibility for malfunctions, failures, damage or injury caused by use of parts not authorized by the Hawker Beechcraft Corporation.

Salvaged airplane parts, reworked parts obtained from sources not approved by the Hawker Beechcraft Corporation or parts, components or structural assemblies, the service history of which is unknown or cannot be authenticated, may have been subjected to unacceptable stresses or temperatures or have other hidden damage, not discernible through routine visual or usual nondestructive testing techniques. This may render the part, component or structural assembly, even though originally manufactured by Hawker Beechcraft Corporation, unsuitable and unsafe for airplane use.

Section 1

3. CONTINUOUS INSPECTION GENERAL INFORMATION

A. Purpose and Use

The Continuous Inspection Procedure is programmed to maintain the Model 99 Airliner Series airplanes that are utilized by owners and operators or by Hawker Beechcraft Corporation for airline transportation purposes.

The complete inspection program for each airplane is divided into several parts consisting of a Routine Inspection of the airplane every 50 hours of service time and a Detailed Inspection of a portion of the airplane every 100 hours of service time, thus providing a complete inspection of the airplane every 400 hours. This method will provide greater availability of the airplane during normal operating hours without sacrificing the quality desired during maintenance and inspection periods.

A complete inspection cycle (all Routine Inspections and all Detailed Inspections) must be accomplished within 12 calendar months from the date the inspection cycle was started. Any part of the inspection which has not been accomplished will become due immediately.

B. Definitions

The terminology pertaining to the inspection procedures and their use as explained in this manual are in accordance with 14 CFR Parts 1, 43, and 91, issued by the Federal Aviation Administration. These terms are defined as follows:

Continuous Inspection - A continuous inspection is a continuing airworthiness inspection of an airplane and its various components and systems at scheduled intervals in accordance with procedures prescribed by the Administrator of the Federal Aviation Administration.

Detailed Inspection - Detailed inspection consists of a thorough examination of the appliances, the airplane and components and systems with such disassembly as necessary.

Flight Time - Flight time shall mean the total time from the moment the airplane first moves under its own power for the purpose of flight until the moment it comes to rest at the next point of landing, (Block to block time).

Maintenance - Means inspection, overhaul, repair, preservations, and the replacement of parts, but excludes preventive maintenance.

Pilot in Command - Pilot in command shall mean the pilot responsible for the operation and safety of the airplane during the time defined as flight time.

Preventive Maintenance - Means simple or minor preservative operations and the replacement of small standard parts not involving complex assembly operations.

Routine Inspection - Routine inspection consists of visual examination or check of the appliances, the airplane and its components and systems insofar as is practicable without disassembly.

Time in Service - Time in service, as used in computing maintenance and inspection time records, is the time from the moment the airplane leaves the ground until it touches the ground at the end of the flight.

C. Forms and Records

NOTE: Permission is granted for maintenance personnel to reprint the forms below. Reprinting for resale is prohibited. Examples of the Continuous Inspection Work Sheets, In-Flight Work Sheets and Overhaul and Replacement Worksheets are provided at the end of this subchapter.

The forms and records used with the Continuous Inspection Program consist of a Routine Inspection form, four separate 100 Hour Interval Detailed Inspection forms, Continuous Inspection Work Sheets, In-Flight Work Sheets and Overhaul and Replacement Worksheets.

Routine Inspection Work Sheet - This work sheet lists the airplane components which are to be checked at each 50 flight hour interval of service time. At each 100 Hour Interval Inspection, this form will be completed in addition to the 100 Hour Interval Detailed Inspection form.

100 Hour Interval Detailed Inspection Form - Four separate and individual 100 Hour Interval Detailed Inspection forms are used with the Continuous Inspection Program. Each form covers only certain portions of the airplane and is designated as 1st, 2nd, 3rd, or 4th 100 Hour Interval Detailed Inspections. Completion of the 1st, 2nd, 3rd, and 4th 100 Hour Interval Detailed Inspection will provide a complete airworthiness inspection of the airplane.

Continuous Inspection Work Sheet - This form is used in conjunction with each of the inspection forms to provide a list of all discrepancies which are found during the inspection and their corrective action.

In-Flight Work Sheet - Copies of this work sheet are to be kept in the airplane and will be used by the pilot in command to list any discrepancy which occurs during a flight. When the flight is completed, this form will then be forwarded to the Maintenance Shop for proper disposition.

Overhaul and Replacement Work Sheet - This form is used to list all components which are removed from the airplane for overhaul or replacement and the Part Number and Serial Number of the component which was installed.

If the Continuous Inspection Program is discontinued, written notification must be sent to the local FAA General Aviation District Office.

4. INSPECTION PROCEDURES

A. Routine Inspections (50 Hour Intervals)

A Routine Inspection of the airplane shall be conducted each 50 hours of time the airplane is in service. This inspection consists mainly of a visual inspection of the major components of the airplane. The only parts of the airplane which are opened are the engine cowlings so that the general condition of the engine and propeller can be checked. This inspection may be conducted by persons qualified to do preventative maintenance. Complete instructions for conducting a routine inspection are contained in Section 2 of this Guide.

B. Detailed Inspections (100 Hour Intervals)

Four separate individual detailed inspections of the airplane are required to accomplish one complete inspection. Only a portion of the airplane’s components or systems are inspected at each 100 hour interval, thus accomplishing a complete inspection of the airplane once every 400 hours of time in service. Items requiring attention at periods of less than 400 hours are duplicated on the appropriate Detailed Inspection form. These inspections are to be conducted by the Crew Chief in Charge of

Maintenance, or by a properly qualified mechanic under his supervision. Complete instructions for conducting the 1st, 2nd, 3rd and 4th Detailed Inspections are contained in Sections 3, 4, 5, and 6 of this guide respectively.

C. Discrepancies

Discrepancies found on the airplane during an inspection will be listed on the Continuous Inspection Work Sheet. Two lines are provided on the work sheet for each item. The discrepancy will be entered on the top line of the entry space, and the corrective action which is taken will be noted on the bottom line. If more than one line is required to state the discrepancy, as many entry spaces as necessary may be used. The same method should be used for corrective action explanations which require more than one line, except the bottom lines of the extra spaces will be used. Each separate entry on the sheet will be numbered in the ITEM block 1, 2, 3, 4, etc. As many work sheets as necessary will be used to list all discrepancies with the entry numbers in the ITEM block continuing in sequence on each of the individual pages. All discrepancies listed must be corrected before the work sheet is routed to the airplane file.

Discrepancies that affect the airworthiness of the airplane will require the necessary corrective action to be accomplished before the airplane is returned to service.

Discrepancies that do not affect the airworthiness of the airplane may, at the discretion of the maintenance crew chief, be carried over to the next inspection period. All discrepancies thus carried over will be retained in the Shop File until corrected, and will also be reflected on the Shop Status and Scheduling Board.

Discrepancies which occur during a flight will be entered on the In-Flight Work Sheet by the pilot in command or another responsible person. At the end of the flight this work sheet is then submitted to the responsible person in the Transportation Maintenance Department.

D. Away From Station Requirements

Away From Station Inspection - If the airplane is to be away from the home location at the time an inspection is due, the pilot in command of the fight will take with him all forms which will be required for the inspection, and a copy of this manual. The detailed inspection will be conducted or supervised by one of the following:

- A certified airframe repair station.
- An appropriately rated certified mechanic with inspection authorization.

The results of the inspection will be noted on the proper forms which are then brought back to the home location. The pilot will be responsible for all inspection form and work sheet entries with the inspector’s and/or mechanic’s signature and identification.

Away From Station Discrepancies - Discrepancies affecting the airworthiness of the airplane, when the airplane is away from the local station, will be corrected by one of the following:

- A certificated airframe repair station.
- An appropriately rated certified mechanic.

The discrepancy and the corrective action taken is to be listed on the In-Flight Work Sheet. The pilot will be responsible for all work sheet entries with the mechanic’s and/or inspector’s signature for identification.

E. Inspection Time Limitations

Inspection intervals called out in the inspection schedule (the four detailed inspections outlined in the CONTINUOUS SCHEDULE OUTLINE) shall not be exceeded by more than 10 hours.

A complete detailed inspection (all four 100 Hour Interval Inspections) must be accomplished within a 12 calendar month period. Under an approved continuous maintenance program, this time frame may be extended. Any part of the inspection which has not been accomplished will become due immediately. Completion of the 1st, 2nd, and 3rd 100 Hour Detailed Inspections and a Routine Inspection shall not be considered as a complete inspection.

F. Continuous Inspection Schedule Outline

A/C TIME (HOURS)	TYPE OF INSPECTION	COMPONENTS INSPECTED
50	Routine Inspection.	Entire Airplane.
100	1st Detailed Inspection and Routine Inspection.	Wings, Wing Center Section, Rear Fuselage and Empennage, and General Service Items.
150	Routine Inspection.	Entire Airplane.
200	2nd Detailed Inspection and Routine Inspection.	Power Plant, Air Conditioning, Rear Fuselage and Empennage, Wings, Landing Gear, Operational and General Service Items.
250	Routine Inspection.	Entire Airplane.
300	3rd Detailed Inspection and Routine Inspection.	Flight Compartment, Cabin Section, Rear Fuselage and Empennage, Wings and General Service Items.
350	Routine Inspection.	Entire Airplane.
400	4th Detailed Inspection and Routine Inspection.	Nose Section, Main Gear and Brakes, Nose Gear, Landing Gear Retraction, Rear Fuselage and Empennage, Wings, and General Service Items.

NOTE: Any item with an inspection requirement of more than 400 hours must be inspected on an individual basis.

5. RESPONSIBILITIES

NOTE: The titled positions listed below may be changed by individual companies as desired; however, the detailed responsibilities must be maintained to ensure that proper inspections and maintenance functions are performed.

A. Transportation Manager

It shall be the responsibility of the Transportation Manager:

- To ensure that all flight time and In-Flight Work Sheet discrepancies are properly recorded and coordinated with maintenance.

- To ensure the airplane is scheduled into maintenance in accordance with the Continuous Inspection Program.
- To maintain individual airplane files of all pertinent airframe, power plant, and component data such as installation dates, total airframe time of component installation, component part and serial numbers, etc.

B. Foreman of Transportation Maintenance

It shall be the responsibility of the Foreman of Transportation Maintenance:

- To see that all forms are properly executed by the responsible party per the instructions on each form.
- To provide the necessary technical data, including overhaul manuals, service instructions, A.D. Notes, etc., for the airframe, engines, and components undergoing inspection.
- To provide necessary training and assistance to his subordinates to ensure that the proper procedures, methods and practices are employed during inspection and maintenance.

C. Crew Chief In Charge

It shall be the responsibility of the Crew Chief in Charge:

- To supervise and/or conduct the Routine and Detailed Inspection and Special Inspections so as to maintain all aircraft in an airworthy condition.
- To ensure operating time, serial numbers, airworthiness directives, etc. are properly recorded.
- To prepare Routine and Detailed Inspection forms as inspections become due.
- To assign to an able assistant, with proper notification and approval by the Foreman of Maintenance, the duties of Crew Chief during periods of absence due to illness, vacation, etc.

D. Pilot in Command

It shall be the responsibility of the Pilot in Command of all flights:

- To ensure all in flight discrepancies are listed on the In-Flight Work Sheet.
- To ensure that all forms and manuals required for the inspection are aboard the airplane for away from station inspections.
- To ensure that all forms are properly filled out with the mechanic’s and inspector’s signatures and identification following away from station inspections or discrepancy corrective action measure.
- To ensure that all forms are expedited to the shop upon his return to the home location.

6. CONTINUOUS INSPECTION SCHEDULE

A. General Information

The Continuous Inspection Schedule Chart on the following pages lists the major components of the airplane which require periodic inspections. For overhaul and replacement schedule (Ref. 05-10-00). When an item is removed for overhaul, the part number of the item which is being installed will be recorded in the place provided in the Inspection Work Sheets.

B. Continuous Inspection Schedule Chart

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
WING AND CENTER SECTION GROUP			
Wing bolts	400		1
Wing skin	50	X	1
Stall fences	50	X	1
Wing structure	400		1
Aileron and tab skin	50	X	1
Aileron and tab structure	100		1, 2, 3, 4
Aileron tab actuator	400		1
Flap skin	50	X	1
Flap structure	400		1
Flap motor	400		3
Flap gearbox	400		3
Flap actuators	400		1
Flap flexible shafts	400		1
Fuel cells	400		1
Nacelle tank fuel capacitance probe	400		1
Wing tank fuel capacitance probe	400		1
Fuel cell drain valves	400		1
Fuel system check valves	400		1
Fuel crossfeed valve	400		1
Firewall shutoff valve	400		1
Primary boost pumps	400		1
Secondary boost pumps	400		1
Fuel transfer (Jet) pump	400		1
Inverters	400		1
Voltage regulator	400		1
Battery (if wing installed)	200		1, 3
Battery master relay	200		1, 3
Battery charge diode	200		1, 3

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
WING AND CENTER SECTION GROUP (Continued)			
Battery monitoring system	200		1, 3
Starter relay	400		1
External power relay	400		1
External power reverse polarity diode	400		1
Landing gear control relay (Mechanical system only)	400		4
Landing gear power relay (Hydraulic system only)	400		4
Power pack filter and screens	400		3
All other relays	400		4
Navigation lights	50	X	2
Ice lights	50	X	2
Landing lights	50	X	2
Access doors	50	X	1
Control cables and pulleys	400		1
Plumbing	400		1
Strobe lights	400		1
Deice boots	400		1
Electrical wiring and equipment	400		1
Fuel tanks and vents	50	X	1
FUSELAGE AND EMPENNAGE GROUP			
Fuselage skin	50	X	1
Fuselage structure	400		1
Stabilizer skin	50	X	1
Stabilizer structure	400		1
Vertical fin skin	50	X	1
Vertical fin structure	400		1
Rudder and tab skin	50	X	1
Rudder and tab structure	400		1
Elevator skin	50	X	1

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
FUSELAGE AND EMPENNAGE GROUP (Continued)			
Elevator structure	400		1
Rudder tab actuator	400		1
Stabilizer trim motors and actuator - Main	400		1
Stabilizer trim motors and actuator - Standby	400		1
Stabilizer trim motors and actuator - Wiring	400		1
Rotating beacons	50	X	1
Access doors	50	X	2
Pitot heaters and drains	50	X	2
Emergency locator transmitter	50	X	1
Control surface cables and pulleys	400		1
Plumbing	400		1
Deicer boots	400		1
Navigation lights	400	X	1
Autopilot	400		3
Elevator hinge fittings and torque tube collar	100		1, 2, 3, 4
POWER PLANT GROUP			
Engine	50	X	1, 2
Engine mounts	400		2
Engine cowling	50	X	2
Engine accessories	400		2
Starter-generator	200		2, 4
Brushes - (Inspect and replace as specified in Chapter 24-30-00)	100		1, 2, 3, 4
Fuel filters and screens (Inspect and replace as specified in Chapter 12-10-00)	200		2,4
Fuel System	400		1
Fuel drain collector system	200		2,4
Oil system	400		1

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
POWER PLANT GROUP (Continued)			
Fire and smoke detectors	400		1
Fire extinguisher	400		1, 2
Engine controls	400	X	2
Ignition system	400		2
Igniter plugs	400		2
Engine oil cooler (Replace when contaminated)	400	X	2
Engine oil filter (Inspect and replace as specified in Chapter 12-10-00)	200		2,4
Exhaust system	400		2
Propeller	50	X	2
Autofeather system	400		2
Propeller controls	400		2
Propeller governor	400		2
Propeller overspeed governor	400		2
Fuel topping governor (Power turbine governor)	400		2
All hose assemblies	400		2
Propeller deicer brushes and slip ring	50	X	2
Access Covers	50	X	2
Engine bleed air pressure regulators	400		2
Drain plugs	200		2
Induction system	400		2
Electrical wiring and equipment	400		2
Inertial anti-icer	400		2
Propeller deicer boots	400		2
Propeller synchronizer	400		2
Propeller synchrophaser	400		2

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
AIR CONDITIONING (OPTIONAL)			
Refrigerant level	200		2,4
Refrigerant compressor	200		2,4
Refrigerant compressor belt	200		2,4
Refrigerant lines and service valves	400		2
Air conditioning air filters	400		2
Bleed air control unit	400		2
HEATING AND VENTILATING			
NOTE			
Any item with an inspection requirement of more than 400 hours must be inspected on an individual basis.			
Ventilation blower(s)	400		2
Cabin combustion heater (Pressure test every 500 hours of heater operation or 1000 hours of airplane operation)	400		2
Combustion heater igniter and plugs	400		2
Combustion heater vibrator	Change to second set of points at 1,000 hours. Replace at 2,000 hours of heater operation.		2
Combustion heater fuel shutoff valve	200 hours of heater operation.		2
Combustion heater fuel filters	100 hours of heater operation.		2
Heater fuse	400		2
FLIGHT COMPARTMENT			
Flight controls	50	X	3
Flight controls - Control wheel switches and wire	400		3
Flight controls - Control column extension wire (cord)	400		3
Engine controls	400	X	3

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
FLIGHT COMPARTMENT (Continued)			
Pneumatic and vacuum system	400		3
Electrical system	400		3
Electrical system - Bus feeder diodes	400		3
Electrical system - AC bus and switching relays	400		3
Pitch trim circuit wires behind subpanel	400		3
Stall warning and landing gear warning horn circuit	400		2
Brake System	50	X	4
Brake master cylinders	400		4
Parking brake valve	400		2
Brake shuttle valve	400		2
Brake hoses	400		4
Gyro horizon indicator	400		2
Directional gyro indicator	400		2
RMI indicator	400		3
Turn and Bank indicator	400		3
Airspeed indicator	400		3
Rate of Climb indicator	400		3
Altimeter and static system	400		3
Clock	400		3
Magnetic compass	400		3
Volt/load meter	400		3
Fuel quantity indicators	400		2
Propeller/inlet anti-icer ammeter	400		2
Instrument pressure gage	400		3
Fuel flow indicators	400		3
Interstage turbine temperature gages	400		2
Gas generator tachometers	400		3
Torquemeters	400		3
Propeller tachometer	400		3
Oil pressure indicator	400		3

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
FLIGHT COMPARTMENT (Continued)			
Oil temperature indicator	400		3
Flap position indicator	400		3
Propeller synchroscope	400		3
Deicer pressure gage	400		3
Free air temperature indicator	400		3
Pitch trim indicator	400		3
Instrument hose assemblies	400		3
Oxygen regulator	400		3
Cockpit lights	50	X	2
Warning lights	50	X	2
Stall warning or safe flight indicator	50	X	2
Emergency static air source	400		2
Seats and seat belts	400		4
Windows	400		3
Access doors	400		3
Cabin entrance door	400		3
Electrical wiring and equipment	400		3
Heater ducts	400		3
Ejector and deicer distributor valve	400		3
CABIN SECTION			
Instrument air filter	200		3, 4
Emergency exits	400		3
Cabin skin	400		3
Cabin structure	50	X	3
Flap motor and drives	400		3
Landing gear gearbox and clutch assemblies (mechanical gear only)	400		3
Landing gear power pack (hydraulic gear only)	400		3

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
CABIN SECTION (Continued)			
Landing gear retract motor	400		4
Cabin lights	50	X	3
Control cables and pulleys	400		3
Windows	400		3
Belly drain holes	400		3
NOSE SECTION			
Battery (if nose installed)	50		3, 4
Instrument Air Filters	200		3, 4
Nose skin	50	X	4
Nose structure	400		4
Oxygen cylinder (22.0 cu. ft. standard weight) ICC 3AA 1800. Hydrostatically test every 5 years (ICC Regulation).	400		4
Oxygen cylinder (49.1 and 65.6 cu. ft. light weight) ICC 3HT 1850. Hydrostatically test every 3 years (ICC Regulation). Replace after 24 years or 4,380 refills.	400		4
Radio equipment	400		4
Radio antennas	400		3
Radio control	400		4
Baggage compartment lights	50	X	2
Static and pitot system	400		4
Brake reservoir	400		4
Door, fastener and seal	400		4
LANDING GEAR AND BRAKES			
Main landing gear	50	X	4
Nose landing gear	50	X	4
Landing gear actuators (Inspect and replace as specified in Chapter 32-30-00). Inspect clevis bolt holes for elongation.	400		4

ITEM	INSPECTION TIME LIMITS	ROUTINE INSPECTION	DETAILED INSPECTION
LANDING GEAR AND BRAKES (Continued)			
Wheel brake assemblies	50	X	2,4
Shimmy damper	200		2,4
Shock struts	50	X	2,4
Wheel	400		4
Tires	50	X	4
Landing gear torque shafts (mechanical gear only)	400		1
Landing gear plumbing (hydraulic gear only)	400		1
Aileron quadrant	400		3
Drag leg and downlock bolts for proper torque	400		4
Nose gear steering and linkage	400		4
Landing gear electrical system	400		4
Landing gear retraction system	400		4
Brake deicer	200		2,4
EMERGENCY AND SURVIVAL EQUIPMENT			
First aid kit	50	X	
Emergency procedure card	50	X	
Fire extinguisher	50	X	
Over water flight equipment	50	X	
Lighting	50	X	

SECTION 2

7. ROUTINE INSPECTION PROCEDURE

A. Forms Required

(1) Model 99 Airliner Series Routine Inspection.

(2) Continuous Inspection Work Sheet.

NOTE: The Model 99 Airliner Series Routine Inspection form will be used for all Routine Inspections whether the Inspections are conducted at 50 hour intervals or in conjunction with a 100 hour Detailed Inspection.

B. Reference Material

(1) Model 99 Airliner Series Maintenance Manual

(2) Model 99 Airliner Series Component Maintenance Manual

(3) Model 99 Airliner Series Wiring Diagram Manual

C. Inspection Procedures

(1) Fill out the heading on each form in its entirety.

(2) As each item is inspected, the responsible person will make entries as required and will initial in the space provided in the right hand column.

(3) List all discrepancies found on the Continuous Inspect Work Sheet.

(4) When the inspection is complete, the person making the inspection will sign the form in the space provided.

NOTE: Any repairs made during the Routine Inspection will be noted on the Continuous Inspection Work Sheet and attached to the completed Routine Inspection Form (Ref. Step No. 3). Maintenance, other than preventive maintenance, will be signed off by the mechanic and Quality Control Inspector. Only the inspection mechanic's initials will be required on work classified as preventive maintenance.

8. ROUTINE INSPECTION

A. Model 99 Airliner Series Routine Inspection

A/C NUMBER _____ **A/C TIME** _____ **DATE** _____

B. Airframe

Complete Steps 1 thru 18.	MECHANIC	
	LH	RH
(1) Check fuselage, empennage and wings for visual damage.		
(2) Check all control surfaces for security and visual damage.		
(3) Check all access plates for being in place and secure.		
(4) Check main gear and wheel well area for leaks, security of components and general condition.		
(5) Check brakes for thickness of discs and leaks.		
(6) Check struts for proper inflation.		
(7) Check tires for condition and inflation.		
(8) Check nose gear and wheel well area for leaks, security of components and general condition.		
(9) Check navigation lights, beacon, landing lights and ice lights for operation.		
(10) Check all flight compartment lights, including annunciator warning lights for operation.		
(11) Check all cabin and baggage compartment lights for operation.		
(12) Check pitot heaters and drain pitot lines.		
(13) Check stall warning or safe flight indicator.		
(14) Check all flight control and tabs for freedom of operation.		
(15) Check cabin and flight compartment for cleanliness and general condition.		
(16) All In-Flight Work Sheet discrepancies cleared.		
(17) Check Service Instructions and Service Bulletins for conformity.		
(18) EMERGENCY LOCATOR TRANSMITTER - Check for proper Operation. Check to assure ELT is in the armed position when airplane is returned to service.		

C. Engines

Complete Steps 1 thru 9.	**MECHANIC**	
	LH	**RH**
(1) Check propellers for nicks and general condition.		
(2) Check deicer brushes and slip rings for general condition.		
(3) Check engines through front cowl, cowl doors and access openings for fuel, oil and exhaust leaks and general condition.		
(4) Check oil coolers.		
(5) Check all engine controls for freedom of operation.		
(6) Check engine cowling and access plates for security and general condition.		
(7) Engine oil tanks serviced to proper level.		
(8) All InFlight Work Sheet discrepancies cleared.		
(9) Check fuel heaters for leaks and security of attachment.		

D. Emergency and Survival Equipment

Complete Steps 1 thru 5.	**MECHANIC**	
	LH	**RH**
(1) Check first aid kit to make sure it is on board and complete.		
(2) Check emergency procedure cards for condition and that one is available to each passenger.		
(3) Check cabin fire extinguisher for condition, mounting and proper charge.		
(4) Check over water flight equipment as required per FAR regulations.		
(5) Check emergency lighting equipment per FAR regulations.		

I certify that a Routine Inspection was performed in accordance with the Continuous Inspection Program and that the airplane is approved for return to service.

INSPECTION COMPLETED BY:_____

SECTION 3

9. FIRST 100 HOUR INTERVAL DETAILED INSPECTION

A. Forms Required

(1) First 100 Hour Interval Detailed Inspection.

(2) Model 99 Airliner Series Routine Inspection.

(3) Continuous Inspection Work Sheet.

NOTE: A Routine Inspection must be conducted in conjunction with each Detailed Inspection to comply with Continuous Inspection Regulations.

B. Reference Material

(1) Model 99 Airliner Series Maintenance Manual

(2) Model 99 Airliner Series Component Maintenance Manual

(3) Model 99 Airliner Series Wiring Diagram Manual

C. Inspection Procedures

(1) Fill out the heading on each form in its entirety.

(2) The mechanic checks each item on the inspection form and initials the form in the spaces provided.

(3) List all discrepancies found on the Continuous Inspect Work Sheet.

NOTE: Check all “In-Flight Work Sheets” turned in since the last inspection for discrepancies that have not yet been worked off.

(4) Each discrepancy is to be signed off by the mechanic, crew chief, and a quality control inspector when the discrepancy has been corrected.

(5) In the spaces provided on the Overhaul and Replacement Worksheet, the mechanic is to list all components which are removed from the airplane for overhaul or replacement and the Part Number and Serial Number of the component which was installed.

(6) The Quality Control Inspector will stamp off each item on the inspection form to complete the inspection.

(7) When the inspection has been completed, the mechanic, crew chief and Quality Control Inspector will sign the “INSPECTION COMPLETED” block at the end of the inspection sheet.

10. DETAILED INSPECTION PROCEDURE

Inspection Date_____

Airplane Serial _____ Airframe Hours _____ Cycles _____

LH Engine Hrs _____ Cycles _____ RH Engine Hrs _____ Cycles _____

LH Engine Power Module S/N_____

RH Engine Power Module S/N_____

LH Engine Gas Generator S/N_____

RH Engine Gas Generator S/N_____

NOTE: For access to inspection areas, (Ref. Chapter 12) in this maintenance manual for locations of applicable access plates and doors.

Certain items must be inspected or replaced at more frequent intervals when the airplane is initially placed in service. Refer to the Overhaul and Replacement Schedule (Ref. 05-10-00) for these items.

A. Wings

First 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 16.	**MECH**		**INSP**
	LH	**RH**	
(1) SKIN - Inspect skin for condition and loose or missing rivets. If damage is found, check adjacent structure.			
(2) STRUCTURE - Check for cracks, loose rivets and concealed damage. Inspect the wing attachment fittings, spar caps and structure as outlined in the Structural Inspection and Repair Manual (P/N 98-39006).			
(3) FRONT SPAR CAP - Check for corrosion after 5 years of service and every year thereafter. Refer to Chapter 57-15-00 of the Structural Inspection and Repair Manual (P/N 98-39006).			

First 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 16.	MECH		INSP
	LH	RH	
(4) CABLES AND PULLEYS - Check control cables, pulleys and associated equipment for condition, attachment, alignment, clearance and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter. INSPECT AND RECORD AILERON AND AILERON TAB CABLE TENSIONS: TEMPERATURE_____degrees F. Aileron Cable Tension: LEFT_____RIGHT_____ Aileron Tab Cable Tension:_____			
(5) AILERON AND TAB - Inspect skin for condition and loose or missing rivets. Check surfaces for proper attachment and freedom of movement. Check trim tab actuator for proper operation, smoothness, and attachment. Check tab free play as instructed in Chapter 27-10-00.			
(6) FLAPS AND ACTUATORS - Inspect flap drive and actuator for condition and attachment. Check skin and structure for condition and loose or missing rivets.			
(7) LIGHTS - Check navigation and landing lights for broken lenses and replace bulbs as necessary.			
(8) FUEL CELLS AND VENTS - Inspect fuel cells for leaks and vents for obstruction.			
(9) FUEL PROBES - Check for leaks at points of attachment.			
(10) ELECTRICAL WIRING AND EQUIPMENT - Check for security, chafing, damage and attachment.			
(11) PLUMBING - Check for leaks, chafing or damage, and proper attachment.			
(12) WING BOLTS - (Ref. Chapter 57-00-00) for wing bolt, nut, and fitting inspection criterion and frequency. Check drain holes in upper wing bolt fittings for possible obstructions and clear as required.			
(13) DEICER BOOTS - Check for condition and attachment.			

First 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 16.	MECH		INSP
	LH	RH	
(14) ACCESS DOORS - Check for fit and attachment.			
(15) LIFT TRANSDUCER - Check for proper heating and aural operation.			
(16) STROBE LIGHTS - Check for proper operation, broken or cracked lenses and replace flashtubes as necessary.			

B. Wing Center Section

First 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 14.	MECH		INSP
(1) SKIN - Inspect for condition and loose or missing rivets. If damage is found, check adjacent structure.			
(2) STRUCTURE - Check for cracks, loose rivets and concealed damage. Inspect the wing attachment fittings, spar caps and structure as outlined in the Structural Inspection and Repair Manual (P/N 98-39006).			
(3) FUEL CELLS, VENTS, VALVES AND PUMPS - Inspect the fuel cells for leaks and plugged vents. Check the pumps, drain valves, firewall shutoff valves, and the crossfeed valve for proper operation, pressure leaks and security of attachment.			
(4) FUEL PROBES - Check for leaks at points of attachment. Inspect nacelle fuel probes for contamination and micro sludge buildup every 2,500 hours or 30 months. Clean probes thoroughly before installation.			
(5) FLAPS AND ACTUATORS - Inspect flap drive and actuator for condition and attachment. Check skin and structure for condition and loose or missing rivets.			
(6) LEADING EDGE AND NACELLE PLUMBING AND WIRING - Check plumbing for damage, security and leaks. Inspect wiring for chafing, security of attachment and electrical connections.			

First 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 14.	MECH		INSP
(7) CABLES AND PULLEYS - Check control cables, pulleys, and associated equipment for condition, attachment, alignment, clearance, and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter.			
(8) BATTERY (Wing Installed) - Remove battery, inspect battery box, cables, and vent tubes for general condition. Check the external power relay for proper operation. With the battery in a fully charged state, check the water level. (DO NOT use any device which has been used for lead acid type batteries.) Refer to applicable Battery Service and Maintenance Manual and Servicing Instructions (Ref. Chapter 24-31-00).			
(9) FIRE DETECTORS - Check all detectors for discrimination and sensitivity. Check the control amplifier circuits (Ref. Chapter 26-20-00).			
(10) LANDING GEAR TORQUE SHAFTS (Mechanical Gear) - Inspect shafts and universal joints for condition, attachment and visible damage.			
(11) LANDING GEAR PLUMBING (Hydraulic Gear) - Check for security and leakage.			
(12) ACCESS DOORS - Check for fit and attachment.			
(13) ENGINE FIRE EXTINGUISHER - Check the pressure of the supply cylinder. Inspect the plumbing for leakage and security of attachment.			
(14) BLEED AIR PRESSURE REGULATORS - Check for security and inspect and clean as necessary.			

C. Rear Fuselage and Empennage

First 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 17.	MECH	INSP
(1) SKIN - Inspect skin for condition and loose or missing rivets. If damage is found, check adjacent structure.		
(2) STRUCTURE - Check for cracks, loose rivets and concealed damage.		

First 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 17.	MECH	INSP
(3) CABLES AND PULLEYS - Check control cables, pulleys and associated equipment for condition, attachment, alignment, clearance and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter. INSPECT AND RECORD ELEVATOR, RUDDER AND RUDDER TAB CABLE TENSIONS: TEMPERATURE_____degrees F. Elevator cable tension: UP_____DOWN_____ Rudder cable tension: LEFT_____RIGHT_____ Rudder tab cable tension:_____		
(4) EMPENNAGE SURFACES - Inspect skin for condition and loose or missing rivets. Check surfaces for proper attachment and freedom of movement. Check trim actuator and motors for proper operation, smoothness, and attachment.		
(5) PLUMBING - Inspect plumbing for condition and attachment.		
(6) DEICER BOOTS - Check for condition and attachment.		
(7) NAVIGATION LIGHTS AND ROTATING BEACONS - Check for cracked or broken lenses. Replace bulbs as necessary.		
(8) ACCESS DOORS - Check for fit and attachment.		
(9) RUDDER AND RUDDER TAB - Check for freedom of movement and attachment. Check rudder tab free play as instructed (Ref. Chapter 27-20-00).		
(10) VENTRAL FIN AND AFT FUSELAGE DRAINS - Check the drain holes in the ventral fin for obstructions. Remove the door and check for obstructions of the drain holes in the base of the stringers on each side of the door at the point of juncture with the aft bulkhead. Check for obstruction of the drain hole in the base of the drain tube adjacent to where the relief hose attaches to the drain tube. Check bayonet type drain for obstruction.		

First 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 17.	MECH	INSP
(11) HORIZONTAL STABILIZER - For U-1 thru U-145, except U-50 and U-132 - Check the horizontal stabilizer elevator hinge fitting assemblies every fifth inspection as instructed in Service Instructions No. 0437-132 or subsequent. For U-1 thru U-157, except U-50 - Inspect the wiper brush as instructed in Service Instructions No. 0841 or subsequent.		
(12) HORIZONTAL STABILIZER TRIM ACTUATOR - Load test the horizontal stabilizer trim actuator every 2,000 hours of actuator operation as instructed (Ref. Chapter 27-40-00).		
(13) ELEVATOR MAIN SPAR AND LEADING EDGE RIB FLANGE - For U-1 thru U-164, except U-50 - Check for cracks per Service Instructions No. 0799-133, or subsequent, every fourth inspection.		
(14) ELEVATOR CONTROL PUSH RODS - Check for condition as instructed (Ref. Chapter 27-30-00).		
(15) ELEVATOR TORQUE TUBE COLLAR - Inspect each elevator torque tube assembly collar for cracks and loose taper pins as instructed (Ref. Chapter 27-30-00).		
(16) ELEVATOR - Check for freedom of movement. Check aft bell crank for cracks and loose taper pins (Ref. Chapter 27-30-00). For U-1 thru U-148 - Check for elongation of holes or cracks in the elevator horn casting and attachment flange and replace if conditions are noted per Service Instructions No. 0157-133 Rev 3 or subsequent.		
(17) EMPENNAGE STRUCTURE - Check for cracks, skin panels for distortion, loose or missing rivets and concealed damage.		

D. Power Plant

First 100 Hour Interval Detailed Inspection

Complete Step 1.	MECH		INSP
	LH	RH	
(1) STARTER GENERATOR - Inspect one set of brushes for indication of excessive wear or damage (determine wear by observing diagonal groove on brush). Inspect inlet duct and blast cap for cracks or obstruction. Inspect QAD attachment flange.			

E. Landing Gear

First 100 Hour Interval Detailed Inspection

Complete Step 1.	MECH	INSP
(1) NOSE GEAR STRUT - Inspect strut and components for condition and attachment. Inspect strut for proper inflation and leakage. Deflate and check fluid level, if signs of leakage are apparent. At each 100 hour inspection following the initial inspection at 1,000 hours, inspect the lower shock absorber assembly per Service Instructions No. 2102 or subsequent.		

F. General Service Items

First 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 5.	MECH	INSP
(1) AIRPLANE - Clean and service as required.		
(2) AIRPLANE LUBRICATION - Lubricate as required by the lubrication schedule (Ref. Chapter 12-20-00).		
(3) ENGINES - Inspect after ground run up or flight test. Check for oil leaks, security and attachment of all components.		
(4) PLACARDS - Verify that all appropriate exterior and interior placards are in place and legible.		
(5) Airworthiness Directives and Service Bulletins reviewed and complied with as required.		

Inspection Completed

I certify that a Detailed Inspection was performed in accordance with the Continuous Inspection Program and that the airplane is approved for return to service.

CREW CHIEF _____ **MECHANIC** _____

QUALITY CONTROL INSPECTOR _____

SECTION 4

11. SECOND 100 HOUR INTERVAL DETAILED INSPECTION

A. Forms Required

(1) Second 100 Hour Interval Detailed Inspection.

(2) Model 99 Airliner Series Routine Inspection.

(3) Continuous Inspection Work Sheet.

NOTE: A Routine Inspection must be conducted in conjunction with each Detailed Inspection to comply with Continuous Inspection Regulations.

B. Reference Material

(1) Model 99 Airliner Series Maintenance Manual

(2) Model 99 Airliner Series Component Maintenance Manual

(3) Model 99 Airliner Series Wiring Diagram Manual

C. Inspection Procedures

(1) Fill out the heading on each form in its entirety.

(2) The mechanic checks each item on the inspection form and initials the form in the spaces provided.

(3) List all discrepancies found on the Continuous Inspect Work Sheet.

NOTE: Check all “In-Flight Work Sheets” turned in since the last inspection for discrepancies that have not yet been worked off.

(4) Each discrepancy is to be signed off by the mechanic, crew chief, and a quality control inspector when the discrepancy has been corrected.

(5) In the spaces provided on the Overhaul and Replacement Worksheet, the mechanic is to list all components which are removed from the airplane for overhaul or replacement and the Part Number and Serial Number of the component which was installed.

(6) The Quality Control Inspector will stamp off each item on the inspection form to complete the inspection.

(7) When the inspection has been completed, the mechanic, crew chief and Quality Control Inspector will sign the “INSPECTION COMPLETED” block at the end of the inspection sheet.

12. DETAILED INSPECTION PROCEDURE

Inspection Date_____

Airplane Serial _____ Airframe Hours _____ Cycles _____

LH Engine Hrs _____ Cycles _____ RH Engine Hrs _____ Cycles _____

LH Engine Power Module S/N_____

RH Engine Power Module S/N_____

LH Engine Gas Generator S/N_____

RH Engine Gas Generator S/N_____

NOTE: For access to inspection areas, (Ref. Chapter 12) in this maintenance manual for locations of applicable access plates and doors.

Certain items must be inspected or replaced at more frequent intervals when the airplane is initially placed in service. Refer to the Overhaul and Replacement Schedule (Ref. 05-10-00) for these items.

A. Power Plant

Second 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 27.	**MECH**		**INSP**
	LH	**RH**	
(1) COWLING - Check adjustment of latches. Remove upper cowling and clean all cowlings. Inspect for cracks and repair as necessary. Inspect all latches for condition.			
(2) ENGINE OIL FILTER - Inspect the oil filter for metal particles as described in the Pratt and Whitney Aircraft of Canada Ltd. Engine Maintenance Manual. Replace as required.			
(3) DRAIN PLUGS - Check all drain plugs for security. Check the Magnetic Chip Detector for metal particles (Ref. Chapter 12-10-00). Place a steel bar across the magnetic poles and check that the light in the annunciator panel illuminates. Make sure the light on the annunciator panel goes out when the bar is removed.			
(4) FUEL FILTERS AND SCREENS - Inspect the filters and screens for evidence of foreign matter, corrosion, or microbiological growth in the fuel system. Clean the firewall filter and fuel pump screen and filter (Ref. Chapter 12-10-00). Inspect transfer line fuel filter for condition and cleanliness.			

Second 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 27.	MECH		INSP
	LH	RH	
(5) IGNITION GLOW PLUGS - Inspect the glow plugs as described in the Engine Maintenance Manual; if necessary, clean them as instructed in the manual.			
(6) IGNITER PLUGS - Inspect the plugs as described in Chapter 12-10-00 and the Engine Maintenance Manual. If necessary, replace them as instructed in the applicable Engine Maintenance Manual.			
(7) ENGINE EXHAUST SYSTEM - Remove exhaust stacks. Examine the exhaust section, turning vanes and visible portions of the power turbine for burning, distortion, damage and cracks. If any are found, reference the Pratt and Whitney Aircraft of Canada Ltd. Engine Maintenance Manual for corrective action.			
(8) INDUCTION SYSTEM - Check the inertial separator, air intake duct and engine inlet screen for obstruction and damage. Remove air inlet screen; inspect compressor inlet area and struts, first stage blades for dirt deposits, corrosion, erosion, and foreign object damage.			
(9) OIL COOLER - Check cooler and plumbing for condition and attachment.			
(10) PLUMBING - Inspect plumbing and associated equipment and accessories for condition and attachment.			
(11) ENGINE AND PROPELLER CONTROLS - Check controls and associated equipment for condition, attachment, and proper rigging. Check power control seals for deterioration.			
(12) ENGINE ACCESSORIES - Inspect all accessories for condition, attachment, and leakage.			
(13) ENGINE MOUNTS - Inspect for condition and attachment. Replace or repair the vibration isolator mount as required. All mounts on an engine must be of the same manufacturer. The mount may be rebuilt by replacing the rubber cushion (Ref. Chapter 71-20-00).			
(14) ELECTRICAL WIRING AND EQUIPMENT - Inspect wiring and associated equipment and accessories for condition and attachment.			
(15) FUEL CONTROL HEATERS - Check for leaks and security of attachment.			
(16) INERTIAL ANTIICER - Check vane for freedom of movement and proper travel.			

Second 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 27.	MECH		INSP
	LH	RH	
(17) DEICER BOOTS - Check propeller and air inlet boots for condition and security of attachment; check propeller deicer slip rings for gouges, cracks, burned or discolored areas, and deposits of grease, oil or dirt. Clean as necessary.			
(18) PROPELLERS - Check for condition, attachment and operation. Check condition of mechanical feedback ring, stop rods and spring. Inspect the carbon block pin for freedom of movement. Check for no metal to metal contact between the brass ring and the reversing lever. Check the reversing linkage for proper adjustment, operation, evidence of binding and security of attachment. Check for proper operation of all pedestal controls and switches. Check the tip of the blades for evidence of lightning strikes. If there is evidence of lightning strikes, check the area of entering or leaving the blade for hardness. The inside and outside shank area should be checked for arcing. In all cases of lightning strikes, the thrust bearings should be replaced, soft blades rejected, and all steel parts demagnetized. For U-1 thru U-164, except U-50 - Check the secondary low pitch stop switch assembly for attachment and condition on airplanes equipped with PT6A-20 engines or on airplane serials prior to U-165, except U-50, equipped with PT6A-27 or PT6A-28 engines NOT in compliance with Service Instructions No. 0844-247.			
(19) PROPELLER SYNCHRONIZER - For U-1 thru U-164, except U-50 - Check drive unit for condition, security and electrical connection. Check flexible shaft and trimmer for security of attachment and rigging. Check rod end bearings for lubrication.			
(20) PROPELLER SYNCHROPHASER - For U-50, U-165 and After Inspect the propeller governor for condition, security, electrical connection, wear and rigging. Check rod end bearings for condition and lubrication. Check for proper gap between sensors and slip rings (Ref. Chapter 61-23-00).			
(21) AUTOFEATHER RELAYS, DUMP SOLENOIDS, AND PRESSURE SWITCHES - Check for condition, security of attachment and proper electrical connections.			
(22) AUTOIGNITION PRESSURE SWITCHES - Check for condition, security of attachment and proper electrical connection.			

Second 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 27.	MECH		INSP
	LH	RH	
(23) ENGINE - Perform hot section inspection when required. (Refer to the appropriate Pratt and Whitney Engine Maintenance data).			
(24) STARTER GENERATOR - Inspect one set of brushes for indication of excessive wear or damage (determine wear by observing diagonal groove on brush). Inspect inlet duct and blast cap for cracks or obstruction. Inspect QAD attachment flange. For U-1 thru U-164, except U-50 and airplanes in compliance with Pratt & Whitney Service Bulletin No. 1386 - Remove the generator and lubricate drive spline per the Lubrication Schedule (Ref. Chapter 12-20-00).			
(25) FUEL DRAIN COLLECTOR SYSTEM - Remove the drain collector tank and clean. Check for security of attachment of tank and pump. Check hoses for cracks, deterioration and proper attachment. Clean and check the pump screen (Ref. Chapter 71-70-00).			
(26) WATER/METHANOL SPRAY RINGS - Check for condition, security of attachment and restricted orifices. Check overboard vents for restrictions. Check and clean or replace inline filter annually. Check all plumbing connections for leaks.			
(27) FIRE EXTINGUISHER - Check for presence of activation voltage to the cartridge/squib. Perform the EXTINGUISHER ACTIVATION CHECK (Ref. Chapter 26-20-00).			

B. Air Conditioning

Second 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 9.	MECH	INSP
(1) COMPRESSOR - Check for condition and attachment, and oil leaks. Check clutch for proper gap of 0.022 to 0.057 inch.		
(2) COMPRESSOR DRIVE BELTS - Check for condition. Check belt tension (Ref. Chapter 21-50-00).		
(3) REFRIGERANT LINES AND SERVICE VALVES - Inspect lines and valves for condition and attachment.		
(4) AIR FILTERS - Replace recirculation filter and the evaporator filter in accordance with the Service Schedule (Ref. Chapter 12-20-00).		

Second 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 9.	MECH	INSP
(5) BLEED AIR CONTROL UNIT - Check for proper operation (Ref. Chapter 21-40-00).		
(6) COMPRESSOR DRIVE QUILL SHAFT - Check for wear. Lubricate the splines on both ends with lubricant (111, Table 6, 91-00-00) every 600 hours of engine operation. (Airplanes complying with Pratt & Whitney SB 1354 lubricate aft spline only).		
(7) HEATER (Fuel) - For U-1 thru U-164, except U-50 - Check the fuel plumbing, pumps and regulator for condition and attachment. Clean and inspect the two fuel filters, one located at the inlet port of the fuel control valve in the top of the LH side of the center section and the other in the LH main landing gear wheel well. Check the heater for proper operation in all modes. Inspect the drain lines for obstruction.		
(8) HEATER (Electrical) - For U-50, U-165 and After - Inspect heater, heating elements and associated wiring and ducting for condition and attachment.		
(9) HEATER FUSE - Examine fuse for evidence of deterioration. Replace if deterioration is evident. Check for continuity.		

C. Rear Fuselage and Empennage

Second 100 Hour Interval Detailed Inspection

Complete Steps 1 and 2.	MECH	INSP
(1) ELEVATOR TORQUE TUBE COLLAR - Inspect each torque tube assembly collar for cracks and loose taper pins as instructed (Ref. Chapter 27-30-00).		
(2) RUDDER TRIM TAB - Check rudder tab free play as instructed (Ref. Chapter 27-20-00).		

D. Wings

Second 100 Hour Interval Detailed Inspection

Complete Step 1.	MECH		INSP
	LH	RH	
(1) AILERON TRIM TAB - Check aileron trim tab free play as instructed (Ref. Chapter 27-10-00).			

E. Landing Gear

Second 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 3.	MECH	INSP
(1) MAIN LANDING GEAR STRUT - Inspect shock strut and components for condition, attachment, proper inflation and leakage. Deflate and check fluid level if signs of leakage are apparent. Remove the attaching bolt at the torque knee joint and inspect the connection point of the torque knee halves.		
(2) NOSE GEAR SHIMMY DAMPER - Inspect for condition and attachment. Refill if necessary as outlined (Ref. Chapter 32-20-00).		
(3) NOSE GEAR STRUT - Inspect strut and components for condition and attachment. Inspect strut for proper inflation and leakage. Deflate and check fluid level, if signs of leakage are apparent. At each 100 hour inspection following the initial inspection at 1,000 hours, inspect the lower shock absorber assembly per Service Instructions No. 2102 or subsequent.		

F. Operational Inspection

Second 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 35.	MECH	INSP
NOTE The applicable Pilot's Operating Manual or Pilot's Operating Handbook should be used to assist during the following inspection procedures. The following Operational Inspection procedures are to be applied during the start and run of the engine:		
(1) ENGINE CONTROLS - Check for freedom of movement, full travel, and proper friction lock.		
(2) FUEL TRANSFER JET PUMPS - Check for proper operation.		
(3) STARTERS - Check for proper operation.		
(4) IGNITION - Check for annunciator panel light illumination.		
(5) OIL - Check for proper pressure and temperature limits.		
(6) FUEL QUANTITY GAGES - Check for proper operation.		
(7) FUEL CROSSFEED VALVE - Check for proper operation.		

Second 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 35.	MECH	INSP
(8) BOOST PUMPS, FIREWALL SHUTOFF VALVES, AND THRESHOLD LIGHT - Check for proper operation; if any unit fails to operate, check the fuses.		
(9) INTERSTAGE TURBINE TEMPERATURE - Check for proper limits on engine start.		
(10) VACUUM SYSTEM - Check for proper limits.		
(11) GYRO INSTRUMENTS - Check for erratic or noisy operation.		
(12) PROPELLERS - Check for proper operation. Perform flight idle torque checks. Perform secondary low pitch stop checks on airplanes equipped with PT6A-20 engines or on airplane serials prior to U-165, except U-50, equipped with PT6A-27 or PT6A-28 engines NOT in compliance with Service Instructions No. 0844-247 (Ref. Chapter 61-20-00).		
(13) AUTOFEATHERING - Check per instructions (Ref. Chapter 61-21-00).		
(14) PROPELLER SYNCHRONIZER OR SYNCHROPHASER - Check for proper operation.		
(15) PROPELLER GOVERNOR - Check for proper governor operation and feathering.		
(16) POWER CHECK (Without Jetcal) - Check in accordance with Chapter 77-10-00 for the procedure covering this check.		
(17) IDLE RPM - Check for correct rpm (both high and low rpm).		
(18) DEICER (Surface, windshield, and brake, if installed) - Check for proper operation and cycling.		
(19) AIR CONDITIONING SYSTEM - Check for proper operation in manual heat, manual cool and all automatic modes. Check operation of all outlets and ease of all controls.		
(20) FLIGHT CONTROLLER - (Optional) Check for proper operation.		
(21) REFRIGERANT LEVEL - Check for proper level.		
(22) GENERATOR - Check the output.		
(23) PNEUMATIC SYSTEM - Check for proper limits.		
(24) CONDITION LEVER - Check for clean shutdown at IDLE CUT OFF.		

Second 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 35.	MECH	INSP
(25) WATER/METHANOL INJECTION SYSTEM - Check for proper operation.		
NOTE The following Operational Inspection procedures are to be made with the engine shut down:		
(26) BRAKES (Toe and parking) - Check for proper travel, ease of operation and proper release of the parking brake.		
(27) PROPELLER DEICER - Perform the PROPELLER DEICER SYSTEM INSPECTION (Ref. Chapter 30-60-00).		
(28) ENGINE INERTIAL ANTIICER - Check for proper operation and rig (Ref. Chapter 30-21-00).		
(29) ENGINE AIR INLET DEICER - Check for proper operation (Ref. Chapter 30-20-00).		
(30) PITOT HEAT - Check amperage drawn on ammeter and for proper heating at the unit.		
(31) ALL LIGHTS (Including annunciator panel lights) - Check for proper operation and replace defective bulbs as necessary.		
(32) STALL WARNING - Check for proper operation.		
(33) AURAL TRIM WARNING SYSTEM - Check for proper operation as indicated (Ref. Chapter 27-40-00).		
(34) FLAPS - Check for noisy or erratic operation.		
(35) EMERGENCY LOCATOR TRANSMITTER - Check for proper operation and assure ELT is ARMED before returning airplane to service.		

G. General Service Items

Second 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 5.	MECH	INSP
(1) AIRPLANE - Clean and service as required.		
(2) AIRPLANE LUBRICATION - Lubricate as required by the lubrication schedule (Ref. Chapter 12-20-00).		
(3) ENGINES - Inspect after ground run up or flight test. Check for oil leaks, security and attachment of all components.		
(4) PLACARDS - Verify that all appropriate exterior and interior placards are in place and legible.		
(5) Airworthiness Directives and Service Bulletins reviewed and complied with as required.		

Inspection Completed

I certify that a Detailed Inspection was performed in accordance with the Continuous Inspection Program and that the airplane is approved for return to service.

CREW CHIEF _____ **MECHANIC** _____

QUALITY CONTROL INSPECTOR _____

SECTION 5

13. THIRD 100 HOUR INTERVAL DETAILED INSPECTION

A. Forms Required

(1) Third 100 Hour Interval Detailed Inspection.

(2) Model 99 Airliner Series Routine Inspection.

(3) Continuous Inspection Work Sheet.

NOTE: A Routine Inspection must be conducted in conjunction with each Detailed Inspection to comply with Continuous Inspection Regulations.

B. Reference Material

(1) Model 99 Airliner Series Maintenance Manual

(2) Model 99 Airliner Series Component Maintenance Manual

(3) Model 99 Airliner Series Wiring Diagram Manual

C. Inspection Procedures

(1) Fill out the heading on each form in its entirety.

(2) The mechanic checks each item on the inspection form and initials the form in the spaces provided.

(3) List all discrepancies found on the Continuous Inspect Work Sheet.

NOTE: Check all “In-Flight Work Sheets” turned in since the last inspection for discrepancies that have not yet been worked off.

(4) Each discrepancy is to be signed off by the mechanic, crew chief, and a quality control inspector when the discrepancy has been corrected.

(5) In the spaces provided on the Overhaul and Replacement Worksheet, the mechanic is to list all components which are removed from the airplane for overhaul or replacement and the Part Number and Serial Number of the component which was installed.

(6) The Quality Control Inspector will stamp off each item on the inspection form to complete the inspection.

(7) When the inspection has been completed, the mechanic, crew chief and Quality Control Inspector will sign the “INSPECTION COMPLETED” block at the end of the inspection sheet.

14. DETAILED INSPECTION

Inspection Date_____

Airplane Serial _____ Airframe Hours _____ Cycles _____

LH Engine Hrs _____ Cycles _____ RH Engine Hrs _____ Cycles _____

LH Engine Power Module S/N_____

RH Engine Power Module S/N_____

LH Engine Gas Generator S/N_____

RH Engine Gas Generator S/N_____

NOTE: For access to inspection areas, (Ref. Chapter 12) in this maintenance manual for locations of applicable access plates and doors.

Certain items must be inspected or replaced at more frequent intervals when the airplane is initially placed in service. (Ref. 05-10-00) the Overhaul and Replacement Schedule for these items.

A. Flight Compartment

Third 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 15.	MECH	INSP
(1) FLIGHT CONTROL COMPONENTS, CABLES AND PULLEYS - Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Check control cables, pulleys and associated equipment for condition, attachment, alignment, clearance, and proper operation. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter. INSPECT AND RECORD AILERON AND AILERON TAB CABLE TENSIONS: TEMPERATURE_____degrees F. Control Column Interconnect cable tension:_____ 1/8” Aileron Cable Tension: LEFT_____RIGHT_____ Aileron Tab Cable Tension:_____		

Third 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 15.	MECH	INSP
(2) BRAKE SYSTEM - Check brake system components for leakage. Inspect brake master cylinder for proper operation, brake line plumbing for condition and attachment, brake pedals and linkage for condition, attachment and proper operation. Check for proper pedal travel and parking brake release.		
(3) RUDDER PEDALS - Check rudder pedals for condition, clearance and attachment. Remove the two pivot bolts of each pedal and thoroughly examine the pivot holes and adjacent area for wear per the 99 Airliner Continued Airworthiness Program Inspection Manual.		
(4) INSTRUMENT PLUMBING AND WIRING - Inspect instrument panel, subpanels, placards, shock mounts, instruments, and instrument plumbing and wiring for condition, attachment, chafing, etc.		
(5) CONTROL COLUMN - Check condition, attachment and operation. Check flight control lock for condition, positive locking and alignment. Check extension cord, and control wheel switches. Inspect control wheel adapters for cracks every second inspection.		
(6) HEATER - Check cabin heat shut off valve and defroster valve for proper operation. Check all ducts for condition.		
(7) ELECTRICAL EQUIPMENT - Check the operation of all pilot compartment and panel lights. At each 4th inspection or every 1,200 hours, remove and inspect the edgelighted panels (Ref. Chapter 33-11-00). Check switches, knobs and circuit breakers for looseness and operation. Check all wiring for chafing, security, etc. Prior to U-165, except U-50, check the dual bus feeder diodes for open or shorted diodes as outlined in the 99 Airliner Series Wiring Diagram Manual, P/N 99-590028 or C99 Wiring Diagram Manual P/N 99-590030-5. Check AC bus switching relays.		
(8) EMERGENCY STATIC AIR SOURCE - Check for security and operation of the control.		
(9) ENGINE AND PROPELLER CONTROLS - Check for freedom of movement, full travel and proper friction lock operation.		
(10) SEATS, SEAT BELTS AND SEAT TRACKS - Inspect seats for proper operation and security, determine that seat belts are satisfactory. Inspect seat tracks (Ref. Chapter 25-10-00).		

Third 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 15.	MECH	INSP
(11) WINDOWS - Inspect windows and seal for general condition.		
(12) PILOT FUEL CONTROL PANEL/COPILOT CIRCUIT BREAKER PANEL- Remove and visually inspect the pilots fuel control panel assembly, lower edgelighted panel assembly and the copilots edgelighted circuit breaker panel assemblies for burned out bulbs, delamination, corrosion, and evidence of moisture. If evidence of moisture exists the flight compartment side window seal should be replaced and the latch adjusted to prevent leakage.		
(13) WINDSHIELDS - Inspect electrically heated windshields for antistatic coating and tab bonding (Ref. Chapter 56-10-00).		
(14) ACCESS DOORS - Check for fit and attachment.		
(15) PITCH TRIM - Examine pitch trim wires for chafing, proper separation or protection from other wires.		

B. Cabin Section

Third 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 21.	MECH	INSP
(1) SKIN - Inspect skin for condition and loose or missing rivets. If damage is found, check adjacent structure.		
(2) STRUCTURE - Check for cracks, loose or missing rivets, and concealed damage.		
(3) CABLES AND PULLEYS - Check control cables, pulleys and associated equipment for condition, attachment, alignment, clearance and proper operation. Check turnbuckles for cracks and proper safety. Replace control system components (push rods, turnbuckles, end fittings, castings, etc.) that have bulges, splits, bends, or cracks. Replace cables that have three (3) or more broken strands along any three (3) foot segment or evidence of corrosion anywhere on the cable. Check cable tension at the first inspection and at every 12 months thereafter.		
(4) LANDING GEAR GEARBOX (Mechanical Gear) - Check for security and leakage; lubricate as necessary.		
(5) LANDING GEAR POWER PACK (Hydraulic Gear) - Check for security and leakage. Inspect and replace filters and screens (Ref. Chapter 12-10-00).		

Third 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 21.	MECH	INSP
(6) FLAP MOTOR AND DRIVES - Inspect for condition and attachment, lubricate as necessary.		
(7) AILERON QUADRANT - Check for security, attachment, operation and proper travel.		
(8) WINDOWS - Inspect windows and window seals for general condition.		
(9) CABIN ENTRANCE DOOR - Inspect door, seals, latching mechanism, and cables for condition and security of attachment. Check that folding steps do not engage too soon and that they fold properly without interference.		
(10) CARGO DOOR - (Optional) Inspect door, seals and latching mechanism for condition and security of attachment.		
(11) EMERGENCY EXIT DOOR - Inspect door, seal and latching mechanism for condition and security of attachment. Remove and install the emergency exit door, to ensure proper operation.		
(12) ELECTRICAL WIRING AND EQUIPMENT - Check cabin and compartment lights for condition and attachment, replace all bulbs as necessary. Inspect wiring and equipment for condition and attachment at every fifth inspection.		
(13) HEATER DUCTS - Check cabin hot and cold air outlets and valves for condition, obstructions and proper operation. Inspect heating and cooling ducts for condition and attachment.		
(14) EJECTOR AND DEICER DISTRIBUTOR VALVE - Check equipment and plumbing for security and condition.		
(15) INSTRUMENT AIR FILTER - For U-1 thru U-151, except U-50 - Replace every fourth inspection, or sooner if conditions warrant, such as operation in heavy dust or smoke.		
(16) SEATS, SEAT BELTS AND SEAT TRACKS - Check seats and seat belts for condition and attachment. Inspect seat tracks (Ref. Chapter 25-20-00).		
(17) ACCESS DOORS - Check for fit and attachment.		
(18) RADIO ANTENNA - Check for cracks and proper attachment.		

Third 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 21.	MECH	INSP
(19) AUTOPILOT - Inspect and replace the air filter if dirty. Check the components for damaged cases, bent or broken pins and electrical connectors, damaged seals, poor solder connections, and damaged insulation on the wiring. Check the plumbing for condition and security of attachment.		
(20) EMERGENCY EXIT HATCH - Check the emergency release handle and latch assembly for proper operation. Check that the hatch moves out freely. Check complete hatch assembly for condition and all moving parts for proper operation. Check for proper latching and seal of closed hatch.		
(21) BELLY DRAIN HOLES - Inspect for possible obstructions.		

C. Rear Fuselage and Empennage

Third 100 Hour Interval Detailed Inspection

Complete Steps 1 and 2.	MECH	INSP
(1) ELEVATOR TORQUE TUBE COLLAR - Inspect each torque tube assembly collar for cracks and loose taper pins (Ref. Chapter 27-30-00).		
(2) RUDDER TRIM TAB - Check rudder tab free play (Ref. Chapter 27-20-00).		

D. Wings

Third 100 Hour Interval Detailed Inspection

Complete Step 1.	MECH		INSP
	LH	RH	
(1) AILERON TRIM TAB - Check aileron trim tab free play (Ref. Chapter 27-10-00).			

E. Power Plant

Third 100 Hour Interval Detailed Inspection

Complete Step 1.	MECH		INSP
	LH	RH	
(1) STARTER GENERATOR - Inspect one set of brushes for indication of excessive wear or damage (determine wear by observing diagonal groove on brush). Inspect inlet duct and blast cap for cracks or obstruction. Inspect QAD attachment flange.			

F. Landing Gear

Third 100 Hour Interval Detailed Inspection

Complete Step 1.	MECH	INSP
(1) NOSE GEAR STRUT - Inspect strut and components for condition and attachment. Inspect strut for proper inflation and leakage. Deflate and check fluid level, if signs of leakage are apparent. At each 100 hour inspection following the initial inspection at 1,000 hours, inspect the lower shock absorber assembly per Service Instructions No. 2102 or subsequent.		

G. General Service Items

Third 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 8.	MECH	INSP
(1) AIRPLANE - Clean and service as required.		
(2) AIRPLANE LUBRICATION - Lubricate as required by the lubrication schedule (Ref. Chapter 12-20-00).		
(3) ENGINES - Inspect after ground run-up or flight test. Check for oil leaks, security and attachment of all components.		
(4) PLACARDS - Verify that all appropriate exterior and interior placards are in place and legible.		
(5) BATTERY - Remove the battery. Inspect the battery box, cables and vent tubes for general condition. Check the external power relay for proper operation. With the battery in a full charged condition, check the water level (DO NOT use any device which has been used for lead acid type batteries). Refer to the applicable Battery Service and Maintenance Manual and (Ref. Chapter 24-31-00).		

Third 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 8.	MECH	INSP
(6) FUEL DRAIN COLLECTOR SYSTEM - Check for security of attachment of tank and pump. Check hoses for cracks, deterioration and proper attachment. Inspect pump screen for condition, then clean.		
(7) BATTERY MONITORING SYSTEM - Check for proper operation and security and attachment of all components.		
(8) Airworthiness Directives and Service Bulletins reviewed and complied with as required.		

Inspection Completed

I certify that a Detailed Inspection was performed in accordance with the Continuous Inspection Program and that the airplane is approved for return to service.

CREW CHIEF _____ **MECHANIC** _____

QUALITY CONTROL INSPECTOR _____

SECTION 6

15. FOURTH 100 HOUR INTERVAL DETAILED INSPECTION

A. Forms Required

(1) Fourth 100 Hour Interval Detailed Inspection.

(2) Model 99 Airliner Series Routine Inspection.

(3) Continuous Inspection Work Sheet.

NOTE: A Routine Inspection must be conducted in conjunction with each Detailed Inspection to comply with Continuous Inspection Regulations.

B. Reference Material

(1) Model 99 Airliner Series Maintenance Manual

(2) Model 99 Airliner Series Component Maintenance Manual

(3) Model 99 Airliner Series Wiring Diagram Manual

C. Inspection Procedures

(1) Fill out the heading on each form in its entirety.

(2) The mechanic checks each item on the inspection form and initials the form in the spaces provided.

(3) List all discrepancies found on the Continuous Inspect Work Sheet.

NOTE: Check all “In-Flight Work Sheets” turned in since the last inspection for discrepancies that have not yet been worked off.

(4) Each discrepancy is to be signed off by the mechanic, crew chief, and a quality control inspector when the discrepancy has been corrected.

(5) In the spaces provided on the Overhaul and Replacement Worksheet, the mechanic is to list all components which are removed from the airplane for overhaul or replacement and the Part Number and Serial Number of the component which was reinstalled.

(6) The Quality Control Inspector will stamp off each item on the inspection form to complete the inspection.

(7) When the inspection has been completed, the mechanic, crew chief and Quality Control Inspector will sign the “INSPECTION COMPLETED” block at the end of the inspection sheet.

16. DETAILED INSPECTION

Inspection Date_____

Airplane Serial _____ Airframe Hours _____ Cycles _____

LH Engine Hrs _____ Cycles _____ RH Engine Hrs _____ Cycles _____

LH Engine Power Module S/N_____

RH Engine Power Module S/N_____

LH Engine Gas Generator S/N_____

RH Engine Gas Generator S/N_____

NOTE: For access to inspection areas, (Ref. Chapter 12) in this maintenance manual for locations of applicable access plates and doors.

Certain items must be inspected or replaced at more frequent intervals when the airplane is initially placed in service. Refer to the Overhaul and Replacement Schedule (Ref. 05-10-00) for these items.

A. Nose Section

Fourth 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 10.	MECH	INSP
(1) SKIN - Inspect skin and structure for condition and loose or missing rivets.		
(2) STRUCTURE - Check for cracks, loose rivets and concealed damage.		
(3) RADIO EQUIPMENT - Inspect radio rack structure; check security of units in their mounts. Check radio controls for security of attachment and proper operation.		
(4) BRAKE RESERVOIR - Inspect reservoir and fill as needed.		
(5) DOOR, FASTENERS, AND SEAL - Inspect seal for deterioration, and doors, and latches for proper adjustment.		
(6) BATTERY - (Nose installed) Remove the battery, inspect the general condition of the battery, battery box and cables. Service the battery in accordance with the applicable Battery Maintenance Manual. (Ref. Chapter 24-31-00) of this maintenance manual or to the 99 Airliner Series Wiring Diagram and Electrical Systems Manual P/N 99-590028 or 99 Airliner Series Component Maintenance Manual P/N 99-590015-5. DO NOT use any device which has been used for lead acid type batteries.		

Fourth 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 10.	MECH	INSP
(7) STATIC PORTS - Check and clean as necessary. For C99 airplanes U-50, U-165 and After - Check for proper plumbing per pod installation.		
(8) PITOT SYSTEM - Check and clean as necessary. Drain pitot line.		
(9) INSTRUMENT AIR FILTER - For U-50, U-152 and After - Inspect for condition. Replace every fourth inspection or sooner if conditions warrant, such as operation in heavy smoke or dust.		
(10) OXYGEN SYSTEM - Check condition of the oxygen system and check the oxygen mask for cleanliness and stowage.		

B. Main Gear and Brakes

Fourth 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 9.	MECH		INSP
	LH	RH	
(1) BRAKES - Check brake discs, linings and lines for wear, damage, corrosion and security of all units. Check all hoses for condition.			
(2) WHEELS - Inspect wheels as instructed in manuals P/N 98-33818 or 98-35161, in Chapter 32 of the 99 Airliner Series Component Maintenance Manual P/N 99-590015-5.			
(3) TIRES - Check tires for condition, wear and proper inflation.			
(4) ACTUATOR (Mechanical) - Check actuator cover and bottom assembly for condition and support brackets for cracks at mounting lugs. Inspect clevis bolt holes for possible elongation.			
(5) ACTUATOR (Hydraulic) - Check for leakage and support brackets for cracks at mounting lugs. Check for possible seepage from the end cap. If seepage is indicated, inspect the end cap for possible crack using a 10 power magnifying glass and a flashlight.			
(6) LANDING GEAR STRUT - Inspect shock strut and components for condition, attachment, proper inflation and leakage. Deflate and check fluid level if signs of leakage are apparent. Remove the attaching bolt at the torque knee joint and dye penetrant inspect the connection point of the torque knee halves.			

Fourth 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 9.	MECH		INSP
	LH	RH	
(7) ELECTRICAL - Check for proper operation of switches. Clean dirt from the terminals and connectors. Check wiring for damage.			
(8) BRAKE DEICER (If Installed) - Check the plumbing and hose for leakage and security of attachment. Check the distributor manifold for condition and security of attachment. Check that the orifices in the manifold are clean and unobstructed.			
(9) DRAG LEG - Check security of attach fittings. Check downlock bolts for security and proper safety. Check for possible binding of the drag leg caused by possible overtorque at the downlock. Inspect all drag leg components for wear. Inspect drag leg downlock hook hole as instructed in the Model 99 Airliner Series Component Maintenance Manual P/N 99-590015-5.			

C. Nose Gear

Fourth 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 10.	MECH	INSP
(1) WHEEL - Inspect wheel as instructed in manuals P/N 98-33818 or 98-35161, in Chapter 32 of the Model 99 Series Airliner Series Component Maintenance Manual P/N 99-590015-5.		
(2) TIRE - Check tire for condition, wear and proper inflation.		
(3) SHIMMY DAMPER - Inspect for condition and attachment. Refill if necessary (Ref. Chapter 12-20-00).		
(4) ACTUATOR (Mechanical) - Check actuator upper cover and bottom housing assembly for condition and support brackets for cracks at mounting lugs.		
(5) ACTUATOR (Hydraulic) - Check for leakage and support brackets for cracks at mounting lugs.		
(6) NOSE STEERING STOP - Check stop for distortion; check for damage to steering linkage if distortion is found. Remove stop on nose gear casting and check lug for cracks, reinstall per Chapter 32 of Model 99 Airliner Series Component Maintenance Manual.		
(7) STEERING LINKAGE - Check nose steering mechanism for condition, attachment and correct adjustment (Ref. Chapter 32-50-00).		

Fourth 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 10.	MECH	INSP
(8) NOSE GEAR STRUT - Inspect strut and components for condition and attachment. At each 100 hour inspection following the initial inspection at 1000 hours, inspect the lower shock absorber assembly per Service Bulletin No. 2102 or subsequent.		
(9) STRUT FLUID LEVEL - Inspect strut for proper inflation and leakage. Deflate and check fluid level if signs of leakage are apparent (Ref. Chapter 12-20-00).		
(10) ELECTRICAL - Check for proper operation of switches; clean dirt from terminals and connectors, as required; check wiring for damage.		

D. Landing Gear Retraction

Fourth 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 11.	MECH	INSP
CAUTION *Since battery voltage is not sufficient to properly cycle the landing gear, use only an external power source capable of delivering and maintaining 28.25 ± 0.25 volts throughout the extension and retraction cycles when performing the landing gear retraction inspection. Jack the airplane before starting the landing gear retraction inspection.*		
(1) RETRACT MECHANISM (Mechanical and Hydraulic Gear) - Check retraction system for proper operation of all components through at least two complete cycles. Check for unusual noises and evidence of binding. Check for security, attachment, and leakage of all plumbing lines (Hydraulic Gear Only). Measure the amount of wear in the lower drag leg attach lug hole on the nose gear brace. If it exceeds 0.635 inch diameter, refer to NOSE LANDING GEAR CLEANING AND PARTS REPLACEMENT in Chapter 32-20-00 of the 99 Airliner Series Component Maintenance Manual P/N 99-590015-5. The DU bushing installation in the drag legs should be checked for wear at 1,500 landings. A fully run in bushing will have a surface appearance similar to that of a bronze bearing. A grey green bushing that sheds feathering particles is not fully run in. A bushing with its surface coating worn down to the steel backing should be replaced immediately, for it has reached the point of maximum wear. On serials U-149, U-152 and After and those earlier airplanes in compliance with Service Instructions No. 0516-200, Rev. 2 or subsequent, the DU bushings have been replaced with steel bushings and lubrication fittings.		

Fourth 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 11.	MECH	INSP
(2) DOORS AND LINKAGE - Check door operation, fit, fair and rigging.		
(3) POSITION INDICATORS - Check for security and adjustment of switches, loose wires and proper indication.		
(4) WARNING HORNS - Check for proper operation.		
(5) DOWNLOCKS - Check locking mechanism for positive engagement in extended position. Check the down position indicator switch for proper adjustment. Check the downlock bolts for the possibility of over torque at the downlock assembly.		
(6) SAFETY SWITCH - Check for security and proper operation.		
(7) ACTUATORS (Mechanical System) - Check for noise, binding and proper rigging.		
(8) ACTUATORS (Hydraulic Gear) - Check for leakage and proper rigging. Check for possible seepage from the end cap. If seepage is indicated, inspect the end cap for possible cracks using a 10 power magnifying glass and a flashlight.		
(9) LIMIT SWITCHES - Check for proper adjustment and attachment.		
(10) EMERGENCY EXTENSION - Check system for freedom of operation and positive engagement of downlocks.		
CAUTION *Do not continue operation after receiving a gear down indication. Further movement of the handle could damage the drive mechanism and prevent subsequent electrical gear retraction. The landing gear cannot be retracted manually.*		
(11) NOSE GEAR RETRACT CHAIN (Mechanical Gear) - Inspect for broken links, corrosion, dirt, rust and proper tension. Clean and lubricate (Ref. Chapter 32-30-00).		

E. Rear Fuselage and Empennage

Fourth 100 Hour Interval Detailed Inspection

Complete Steps 1 and 2.	MECH	INSP
(1) ELEVATOR TORQUE TUBE COLLAR - Inspect each elevator torque tube assembly collar for cracks and loose taper pins (Ref. Chapter 27-30-00).		

Fourth 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 and 2.	MECH	INSP
(2) RUDDER TRIM TAB - Check rudder tab free play (Ref. Chapter 27-30-00).		

F. Wings

Fourth 100 Hour Interval Detailed Inspection

Complete Step 1.	MECH		INSP
	LH	RH	
(1) AILERON TRIM TAB - Check aileron trim tab free play (Ref. Chapter 27-10-00).			

G. Power Plant

Fourth 100 Hour Interval Detailed Inspection

Complete Step 1.	MECH		INSP
	LH	RH	
(1) STARTER GENERATOR - Inspect one set of brushes for indication of excessive wear or damage (determine wear by observing diagonal groove on brush). Inspect inlet duct and blast cap for cracks or obstruction. Inspect QAD attachment flange. On airplanes U-1 thru U-164 except U-50 and airplanes in compliance with Pratt & Whitney Service Bulletin No. 1386 the starter must be removed and lubricated per the Lubrication Schedule in Chapter 12-20-00.			

H. General Service Items

Fourth 100 Hour Interval Detailed Inspection

Complete Steps 1 thru 11.	MECH	INSP
(1) AIRPLANE - Clean and service as required.		
(2) AIRPLANE LUBRICATION - Lubricate as required by the lubrication schedule (Ref. Chapter 12-20-00).		
(3) ENGINE OIL FILTER - Inspect the oil filter for metal particles as described in the Pratt and Whitney Aircraft of Canada Ltd. Engine Maintenance Manual. Replace as required.		

Fourth 100 Hour Interval Detailed Inspection (Continued)

Complete Steps 1 thru 11.	MECH	INSP
(4) FUEL FILTER AND SCREENS - Inspect the filters and screens for evidence of foreign matter, corrosion, or microbiological growth in the fuel system. Clean the firewall filter and fuel pump screen (Ref. Chapter 28-00-00).		
(5) REFRIGERANT COMPRESSOR AND REFRIGERANT LEVEL - Check for condition, attachment and oil leaks. Check refrigerant for proper level. Check clutch for proper gap of 0.022 to 0.057 inch.		
(6) COMPRESSOR DRIVE BELTS - Check for condition. Check adjustment (Ref. Chapter 21-50-00).		
(7) COMPRESSOR DRIVE QUILL SHAFT - Check for wear. Lubricate the splines on both ends with lubricant (111, Table 6, 91-00-00) every 600 hours of engine operation. Airplanes complying with Pratt & Whitney SB 1354 lubricate aft spline only.		
(8) ENGINES - Inspect after ground run-up or flight test. Check for oil leaks, security and attachment of all components.		
(9) FUEL DRAIN COLLECTOR - Check for security of attachment of tank and pump. Check hoses for cracks, deterioration and proper attachment. Inspect pump screen for condition and then clean.		
(10) PLACARDS - Verify that all appropriate exterior and interior placards are in place and legible.		
(11) Airworthiness Directives and Service Bulletins reviewed and complied with as required.		

Inspection Completed

I certify that a Detailed Inspection was performed in accordance with the Continuous Inspection Program and that the airplane is approved for return to service.

CREW CHIEF _____ **MECHANIC** _____

QUALITY CONTROL INSPECTOR _____

Overhaul and Replacement Worksheet

COMPONENT	DATE	REASON FOR REPLACEMENT	REPLACEMENT Part No. Serial No.	NEXT O. H. A/C Hours or Cycles Date

Continuous Inspection Worksheet

AC SERIAL NUMBER _____ DATE _____

MODEL _____ INSPECTION TYPE _____ INSPECTION NUMBER _____

LH ENG. TIME_____ CYCLES _____

RH ENG. TIME_____ CYCLES _____

AC TIME _____ CYCLES _____

		ACCEPTANCE		
ITEM	**DESCRIPTION**	**MECH**	**CREW CHIEF**	**Q.C.**

In-flight Worksheet

(Page 1 of 2)

S/N _____ **N NUMBER** _____ **DATE** _____ **NO.** _____

FLT NO	PILOT	COPILOT	TO	TIME	LND	TIME	NO LND	FLT TM
						Today Total		
						Previous Total		
						Total		

FUEL		ENG. OIL			CRUISE CONDITION		
TYPE	GAL	LH	RH		WEATHER	LH	RH
					O. A. T.		
					PA		
					IAS		
					ITT		
					TORQUE		
					PROP RPM		
					N_1		
					FUEL FLOW		
					FUEL PRESS		
					OIL PRESS		
					OIL TEMP		

In-flight Worksheet (Continued)

(Page 2 of 2)

DISCREPANCY WORKSHEET

S/N _____ N NUMBER _____ DATE _____ NO._____

A/C HRS	DATE	DISCREPANCY	CORRECTIVE ACTION	DATE

TIME LIMITS
UNSCHEDULED MAINTENANCE CHECKS
MAINTENANCE PRACTICES

1. GENERAL

This subchapter is assembled in Table form to allow a technician to perform checks for damage after operating the airplane in conditions which could require unscheduled maintenance. Specific conditions, such as lightning strikes, turbulent air penetration or hard landings, etc., are included. Inspection instructions are included for each of the conditions listed.

A. Operation In Atmospheres of High Salt Content

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Engine Compressor Wash	Engines operated in a salty atmosphere should be washed with demineralized water as instructed in the engine maintenance manual. Inspect for corrosion.	Daily.

B. Operation In Atmospheres of High Industrial Air Pollution

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Engine Compressor Wash	When a loss of engine performance attributed to ingestion of air pollution deposits, wash engine in accordance with instructions in the engine maintenance manual. Inspect for corrosion.	As required.

C. When Operating In Areas of High Dust Content

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Nose Landing Gear Shock Strut	Clean off and wipe dry exposed polished surfaces.	Routine.
(2) Instrument Air Filters	Replace instrument line supply filters at or before 100 hours under extremely dusty conditions.	As noted.
CAUTION *To avoid damaging the barometric sensor, disconnect the autopilot sensor line prior to applying reverse air pressure to the pitot and static lines.*		
(3) Pitot and Static Lines	Check for obstructions by applying reverse air pressure (not to exceed 20 psi.) to the ends of the pitot and static lines with them disconnected from the instruments.	200 hours or as required.

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(4) Environmental Air Filter	Inspect for obstruction of air flow. Replace if necessary.	As required.

D. When Operating In Areas of High Humidity

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Floor Structure	Check structure under the floor for corrosion by removing a floor panel and inspection the structure, especially the channel sections.	At a scheduled inspection, but not more than 6 months apart.
(2) Aft Cabin	Remove aft cabin access covers and inspect for corrosion, especially aft of bulkhead points.	At a scheduled inspection, but not more than 6 months apart.
(3) Wing	Remove wing and center section access covers and check for corrosion.	At a scheduled inspection, but not more than 6 months apart.
(4) Empennage	Remove aft fuselage access covers and check for corrosion.	At a scheduled inspection, but not more than 6 months apart.

E. Operating From Very Soft or Unusual Terrain

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
NOTE The high flotation gear is recommended for use on very soft terrain.		
LANDING GEAR		
(1) Main Landing Gear	Check strut inflation (Ref. Chapter 32-10-00).	Routine.
a. Wheels	1. Check for obvious damage.	Routine.
	2. Remove and clean; inspect for abrasions, cracks and chipped rims, bearing for wear, corrosion, fretting and bluing; check seals for distortion, deterioration, and proper fit and security.	Every 100 hours.
b. Brake Units	1. Check cylinders and associated lines for damage and leaks.	Routine.
	2. Check for evidence of overheating.	Every 100 hours.
	3. Check discs for scoring, distortion, damaged plating and evidence of overheating.	Every 200 hours.

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
c. Shock Strut	Check surfaces for cleanliness, free from oil or grease deterioration. Check strut inflation (Ref. Chapter 32-10-00). Clean exposed surface of shock strut piston with clean cloth moistened with hydraulic fluid.	Every 100 hours.
d. Wheel Wells	Clean foreign material (dirt, etc.) from wheel wells. Inspect supports between main and aft spars in upper wheel well and the lift leg attach bracket at the main spar for deformation, cracks, etc.	As required.
e. Tires	Visually check for cuts, wear, deterioration and inflation (visual).	Routine.
(2) Nose Landing Gear	Visually check for obvious damage.	Routine.
a. Wheel	Remove and clean. Inspect for abrasions, cracks and chipped rims, bearings for wear, corrosion, fretting and bluing; check seals for distortion, deterioration, proper fit and security.	Every 100 hours.
b. Shock Strut	1. Check for obvious damage and leaks. Clean exposed surface of shock strut piston with clean cloth moistened with hydraulic fluid. Check strut inflation (Ref. Chapter 32-20-00).	Routine.
	2. Check for correct extension (Ref. Chapter 32-20-00).	Every 100 hours.
	3. Thoroughly clean and inspect for leaks, damage and security; service as necessary.	Every 100 hours.
c. Fork Assembly	Check for cleanliness and obvious damage.	Routine.
d. Nose Wheel Steering (Mechanical)	Check for obvious damage, associated rods and connections for damage.	Every 100 hours.
e. Actuator Linkage	Check for excessive play, safety and security.	Every 100 hours.
f. Shimmy Damper (Mechanical Steering)	Inspect for condition and attachment. Refill if necessary as noted in Chapter 32.	Every 100 hours.
g. Actuator	Check actuator and support brackets for visible damage and condition. Inspect bracket for loose or missing rivets. Inspect cover and bottom assembly of actuator for cracks at the mounting lug.	Every 100 hours.

2. INSPECTION AFTER HARD LANDING

NOTE: This inspection should be carried out after a hard landing and before the airplane is certified as ready for further flight. The inspections are conducted at two levels. The first level consists of determining if any external damage has occurred and looking for evidence of internal structural failure. The second level is concerned with a more detailed inspection of any damaged areas which were indicated in the findings of the first level inspection. If it is determined by the determined by the first level inspection that there is no damage to the airplane, it is not necessary to proceed to the second level inspection.

WARNING: Even though wrinkles in the wing or fuselage skin surface may be slight enough to be considered as negligible, a close inspection of the internal supporting structure may reveal serious damage.

A. First Level Inspection

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) General Appearance	(a) Determine that the airframe components (nacelles, wings, fuselage) are in their normal configuration.	After Hard Landing
(2) Landing Gear	(a) Inspect tires for excessive wear, splits in the tread, bottoming out or folding over the sidewalls.	
	(b) Check the wheels (rims) for flat spots or cracked castings.	
	(c) Check shock struts and attachment lugs for cracks.	
	(d) Inspect hydraulic brake lines for leaks.	
	(e) Inspect downlock, drag link and gear door retract linkage for damage.	
	(f) Inspect landing gear actuator attachment lugs. Inspect supports between main and aft spars in upper wheel well and the lift leg attach bracket at the main spar for deformation, cracks, etc.	
	(g) Inspect area around landing gear attach points.	

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
	(h) Inspect the main landing gear drag brace support structure as follows: 1 Place the airplane on jacks. 2 Disconnect the upper drag brace from the airplane structure in both LH and RH wheel wells. 3 Using a flashlight and a mirror, inspect all of the drag brace support structure for possible cracks, particularly at the lower radius of the U-channel where it attaches to the main spar. 4 If cracks are suspected but are not clearly defined, the suspect area should be dye penetrant inspected, using procedures as outlined in AC43.13-1A. 5 If cracks are found, contact Hawker Beechcraft Technical Support, Hawker Beechcraft Corporation, Wichita, Kansas, 67201 and report the findings for evaluation. 6 If no cracks are found, reinstall the drag brace.	After Hard Landing
Hawker Beechcraft Corporation recommends that airplanes having experienced severe or hard landings or other abnormal landing incidents which may have placed undue stress on the landing gears, be inspected within the first 100 service hours after such hard landing and at each 500 service hours thereafter. Airplanes having received repairs in this area, upon Hawker Beechcraft Corporation recommendations, are exempt from this inspection except in the event of a future hard or abnormal landing incident.		
(3) Nacelles	(a) Inspect external skin surfaces for distortion, loose or missing rivets.	After Hard Landing
	(b) Check cowling attachment fittings for alignment or damage.	

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
	(c) Inspect engine control cables for smooth operation and check plumbing and wiring for security and attachment.	After Hard Landing
	(d) Inspect engine support mounts for cracks or structural failure.	
	(e) Check tips of propellers for damage.	
	(f) Check propeller spinner and backplate for evidence of interference with cowling.	
	(g) Inspect wheel well structure for damage or cracks. Check area surrounding the landing gear attachment points.	
(4) Wing Center Section	(a) Inspect external skin surface (upper and lower) for cracks, abnormal wrinkles and loose or missing rivets.	
	(b) Check wing attachment fittings for cracks.	
	(c) Inspect plumbing, wiring and actuators for damage and security of attachment.	
	(d) Check keel, front and rear spar on the lower side of fuselage for damage and alignment.	
(5) Outboard Wing Panels	(a) Inspect external wing surface skin for cracks, abnormal wrinkles and loose or missing rivets.	
	(b) Check wing attachment fittings for cracks.	
	(c) Inspect internal structure and fuel cells through access panels.	
	(d) Inspect plumbing and wiring for security of attachment.	
(6) Fuselage, Nose	(a) Check external skin surface for cracks, abnormal wrinkles and loose or missing rivets.	
	(b) Check wheel well structure and area surrounding gear attach point for damage.	

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
	(c) Inspect avionics, radar antenna, wiring and plumbing for security and attachment.	After Hard Landing
	(d) Check baggage door alignment.	
(7) Fuselage Center Section	(a) Inspect external skin surface for cracks, abnormal wrinkles and loose or missing rivets.	
	(b) Inspect around cabin windows for structural cracks.	
(8) Fuselage, Aft	(a) Check external skin surface the entire length for cracks, abnormal wrinkles and loose or missing rivets.	
	(b) Inspect empennage and control surfaces for freedom of movement.	

B. Second Level Inspection

NOTE: Because shock loading may be transmitted along one structural member to another, carefully inspect the surrounding and supporting structure in any damaged area found in the first level inspection.

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Landing Gear	(a) Place airplane on jacks and check shock strut for free up and down movement.	After Hard Landing
	(b) Remove tires and inspect internally for cuts or broken areas.	
	(c) Disassemble and examine wheels (rims) for cracks or distortion.	
	(d) Visually inspect axle with 10-power glass. If suspect, dye check or magnaflux.	
	(e) Remove and replace or magnaflux the landing gear attach bolts, check bolt holes for cracks or elongation.	
	(f) Remove and replace or magnaflux drag link bolts and supports.	
	(g) Perform landing gear retraction test.	
(2) Nacelles	(a) If tips of propeller have been damaged, refer to the Engine Maintenance Manual for engine inspection procedure for prop strike.	

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
	(b) Inspect areas surrounding the engine mounts.	After Hard Landing
	(c) Check the internal structure of the wheel well for cracks or damage.	
	(d) Test plumbing and wiring for proper operation.	
(3) Wing Center Section	(a) Dye check wing attachment fittings; examine (magnaflux or replace) attachment bolts and check bolt holes for alignment and correct dimensions.	
	(b) Remove floorboards and access plates and inspect the front and rear spar, and keel structure for evidence of deformation or structural failure.	
	(c) Test plumbing, wiring, flaps, control cables, pulley mounts, and any other system found in this area for proper operation.	
(4) Outboard Wing Panels	(a) Dye check wing attachment fittings; examine (magnaflux or replace) attachment bolts and check bolt holes for alignment and correct dimensions.	
	(b) Test plumbing and wiring for proper operation.	
	(c) Inspect fuel cells and lines for leakage and damage.	
(5) Fuselage, Nose Section	(a) Remove baggage compartment floor boards and inspect the keel structure and supporting members for damage.	
	(b) Inspect wheel well structure and surrounding areas for signs of structural failure.	
	(c) Test avionics, radar antenna, plumbing and wiring for proper operation.	
NOTE Because shock loading may be transmitted along one structural member to another, carefully inspect the surrounding and supporting structure in any damaged area found in the first level inspection.		

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(6) Fuselage, Center and Aft Section	(a) Examine stringers, frames and sidewalls for deformation structural failure.	After Hard Landing
	(b) Test plumbing and wiring for proper operation.	
	(c) Inspect heating and air-conditioning ducts for damage.	
	(d) Examine the control cables and pulley mountings and check for clearance from structure at pass-through locations. Ensure a smooth operation.	
REPAIR OF DAMAGE		
Due to the variety and degree of structural damage which may be involved, the best repair or replacement procedure must be based on the findings of the individual airplane. If the hard landing inspection indicates that serious structural damage has occurred, contact Hawker Beechcraft Technical Support, Hawker Beechcraft Corporation, Wichita, Kansas, 67201 for assistance.		
LOG BOOK ENTRY		
Following a hard landing inspection, an entry covering the extent of inspection, the damage and the repair (if applicable) must be noted in the airplane permanent records.		

3. INSPECTION AFTER ENCOUNTERING TURBULENT AIR

NOTE: This inspection should be carried out after the airplane has been subjected to high G loading while flying through turbulent air and before the airplane is returned to service. The inspection is conducted on two levels. The first level consists of determining if any external damage has occurred and looking for evidence of internal structural failure. The second level is concerned with a more detailed inspection of damaged areas which were indicated in the findings of the first level inspection. If it is determined by the first inspection that there is no damage to the airplane, it is not necessary to proceed to the second level inspection.

WARNING: Even though wrinkles in the wing or fuselage skin surface may be slight enough to be considered as negligible, a close inspection of the internal supporting structure may reveal serious damage.

A. First Level Inspection

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) General Appearance	(a) Determine that the airframe components (nacelles, wings, fuselage and empennage) are in their normal configuration.	After Encountering Turbulent Air.

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(2) Wing Center Section	(a) Inspect the external skin surface (upper and lower) for cracks, abnormal stress wrinkles and loose or missing rivets.	After Encountering Turbulent Air.
	(b) Inspect plumbing, wiring and actuators for damage and security of attachment.	
	(c) Check the keel and the front and rear spar on the lower side of the fuselage for damage and alignment.	
(3) Nacelles	(a) Inspect the external skin surfaces for wrinkles and loose or missing rivets.	
	(b) Check cowling attachment fittings for alignment or damage.	
	(c) Inspect the engine support mounts for cracks or deformation or structural failure.	
	(d) Inspect engine control cables for smooth operation and check plumbing and wiring for security and attachment.	
	(e) Inspect structure in wheel well for damage or cracks.	
(4) Outboard Wing Panels	(a) Inspect the top and bottom wing surface for cracks, wrinkles and loose or missing rivets.	
	(b) Inspect wing attachment fittings for cracks.	
	(c) Inspect aileron, aileron tab and flaps for wrinkles or cracks.	
	(d) Inspect internal structure and fuel cells through access panels.	
	(e) Inspect plumbing and wiring for security of attachment.	
(5) Fuselage, Nose	(a) Check external skin surface for cracks, wrinkles and loose or missing rivets.	
	(b) Inspect area forward of windshield for evidence of structural deformation or failure.	

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
	(c) Inspect avionics, antenna and components for security of attachment.	After Encountering Turbulent Air.
(6) Fuselage, Center Section	(a) Inspect external skin surface for cracks, abnormal wrinkles and loose or missing rivets.	
(7) Fuselage, Aft	(a) Inspect the entire length of the external skin surface for cracks, stress wrinkles and loose or missing rivets.	
	(b) Check the empennage surfaces for damage and free movement.	
	(c) Inspect for skin wrinkles at the juncture of the fuselage and empennage.	
	(d) Check controls for freedom of movement.	

B. Second Level Inspection

NOTE: Because G loading may be transmitted along one structural member to another, carefully inspect the surrounding and supporting structure in any damaged area found in the first level inspection.

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Wing Center Section	(a) Dye check wing attachment fittings, examine (magnaflux or replace) attachment bolts and check bolt holes for alignment and correct dimension.	After Encountering Turbulent Air.
	(b) Remove floorboards and access plates and inspect the front and rear spar and keel structure for evidence of deformation or structural failure.	
	(c) Operational test plumbing, wiring, flaps, control cables, pulley mounts and any other system found in this area.	
(2) Nacelles	(a) Inspect areas surrounding the engine mounts.	
	(b) Inspect internal structure for cracks or damage.	
	(c) Operational test plumbing and wiring.	
(3) Outboard Wing Panels	(a) Dye check wing attachment fittings, examine (magnaflux or replace) attachment bolts and check bolt holes for alignment and correct dimension.	
	(b) If there is evidence of damage to the fuel cells or fuel lines, remove the cells and inspect the fuel cell liners and liner support structure.	
	(c) Operational test the plumbing and wiring, flap actuator, aileron and tab control cables and pulley mounting.	
(4) Fuselage Nose	(a) Remove floorboards and inspect the keel structure and supporting members for damage.	
	(b) Examine any fixed equipment for loose, broken or cracked mountings.	

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
	(c) Operational test avionics, radar antenna, plumbing and wiring.	After Encountering Turbulent Air.
(5) Fuselage Center Section	(a) Examine stringers, frames and sidewalls for deformation or structural failure.	
	(b) Examine heating and air-conditioning ducts for damage.	
	(c) Operational test plumbing and wiring.	
	(d) Examine the control cables, pulley mountings and the cable clearance at areas the cables pass through the structure. Ensure a smooth, normal operation.	
(6) Empennage	(a) Inspect elevator pushrods, torque tubes and bellcrank for damage.	
	(b) Inspect the attachment of the vertical stabilizer spars to the top of the fuselage for evidence of damage.	
	(c) Inspect skin surfaces for condition and loose or missing rivets.	
	(d) Check structure for cracks, loose rivets and/or concealed damage.	
	(e) Check rudder for freedom of movement and attachment.	
	(f) Check elevator for freedom of movement and attachment.	
	(g) Check trim tab actuators for smoothness of operation and attachment. Check the wiring of the electrical trim tab actuator for connection, security of attachment and condition. Check the electrical trim tab actuator for full travel and security of attachment.	

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
REPAIR OF DAMAGE		
Due to the variety and degree of structural damage which may be involved, the best repair or replacement procedure must be based on the inspection findings of the individual airplane. If the turbulent air inspection indicates that serious structural damage has occurred, contact Hawker Beechcraft Technical Support, Hawker Beechcraft Corporation, Wichita, Kansas 67201, for assistance.		
LOG BOOK ENTRY		
Following a turbulent air inspection, an entry covering the extent of inspection, the damage and the repair (if applicable) must be noted in the permanent records.		

4. INSPECTION AFTER LIGHTNING STRIKE

CAUTION: *Propellers must be inspected and/or replaced utilizing the data provided in the manufacturers manuals prior to returning the airplane to service following any lightning strikes or other impact damage.*

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Propeller	(a) Inspect the propeller for signs of arcing damage. If damage is evident, remove propeller. Return propeller and power output section to an approved facility. Most lightning strikes show evidence of entering or leaving the tips of the blades. In some instances the area is dark in color and erosion is present. Generally, the material around the area is annealed. The other area that is damaged is the thrust bearings for the blades. Arcing sometimes occurs and is evident on the balls and race of the bearings. In every case, all steel parts are magnetized due to the flow of the current. The blades having evidence of a strike must be checked for hardness around the point of entering or leaving. Also the shank area outside and inside should be checked for arcing. Soft blades and blades with evidence of arcing in the shank area should be rejected. In all cases, the thrust bearing should be replaced and all steel parts should be demagnetized.	After Lightning Strike
(2) Propeller Governor	(a) Refer to Woodward Service Bulletin 33574 or subsequent.	

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(3) Engine	(a) Inspect as instructed in the appropriate Engine Maintenance Manual.	After Lightning Strike
(4) Fuselage Empennage and Wing Surfaces	(a) Carefully inspect the exterior of the airplane. Evidence of a strike will usually appear as a burned hole or as a series of burned holes in metallic 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 the exit points. Antennas are frequently an entry point of lightning and should be carefully inspected for evidence of arcing, sooting or pitting.	
	(b) From the point of entry, the strike usually spreads aft in a series of small holes or burn marks. After the points 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 pitting. Cables, pulleys, bearings, bolts and all bonding jumpers in the area should be inspected 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. Normal structural repair methods should be used for metal surfaces damaged by lighting. Nonmetallic 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.	

5. ENGINE INSPECTION AFTER SUDDEN STOPPAGE

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Engine	(a) Inspect as instructed in the appropriate Engine Maintenance Manual.	After sudden engine stoppage
(2) Propeller Governor	(a) Refer to Woodward Service Bulletin 33574 or subsequent.	
(3) Propeller	(a) The propeller must be overhauled or replaced.	

6. SPECIAL INSPECTIONS

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Fuel Probes	(a) Inspect for microbiological sludge.	Every 2,500 hours or 30 months.
(2) Engines	(a) Perform hot section inspection as instructed in the engine maintenance manual.	As established by the current issue of the appropriate engine manual.
(3) Fuel Purge System	(a) Remove filter, check for rust or corrosion and clean with shop air. Remove tank and flush with solvent (38, Table 1, 91-00-00).	Every 12 months.
(4) Control Wheel Adapters	(a) Inspect for cracks in weld area.	Every 200 hours.

7. INSPECTION IN THE EVENT OF DEFORMED STEERING STOP

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
(1) Nose landing gear	(a) Inspect the lower portion of the nose landing gear for evidence of fluid leaks. (b) If an external leak is visible from the side of the upper nose landing gear brace assembly, the upper brace must be replaced. (c) If no evidence of fluid leakage is found, remove the steering stop. (d) Visually inspect the inner and outer sides of both stop support lugs on the aft side of the nose landing gear upper brace for cracks. (e) If a crack is visible, the upper brace must be replaced.	After steering stop is deformed

ITEM	INSPECTION REQUIREMENT	INSPECTION INTERVAL
	(f) If no evidence of cracks are detected visually, inspect the inner and outer sides of both stop support lug areas using fluorescent penetrant. (g) If evidence of a crack is detected, replace the stop support lugs (upper brace assembly). (h) If no evidence of cracks is detected using fluorescent penetrant, replace the steering stop. (i) When installing the steering stop, peel shim as required to maintain a total clearance of 0.000 to 0.002 inch between the stop and brace. Install shim on each side of the steering stop to center the stop between the lugs. (j) When installing nuts and bolts, snug (finger tight) using a minimum of one washer to allow installation of cotter pin. Do not overtorque nuts.	After steering stop is deformed

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 06 - DIMEMSIONS AND AREAS

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AIRPLANE ACCESS PANELS

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DIMENSIONS AND AREAS
GENERAL INFORMATION
DESCRIPTION AND OPERATION

1. GENERAL

This chapter provides dimensions, stations and zone illustrations to aid maintenance personnel in locating various components on the airplane as referenced elsewhere in this manual.

This Chapter is broken down in several sub-system/sections. The following is a list of the information contained in this Chapter.

06-10-00 - Airplane Dimensions

06-30-00 - Station Locations

06-50-00 - Fuselage Access Panels

DIMENSIONS AND AREAS
AIRPLANE DIMENSIONS
DESCRIPTION AND OPERATION

1. GENERAL

The major dimensions of the airplane are shown in Figure 1.

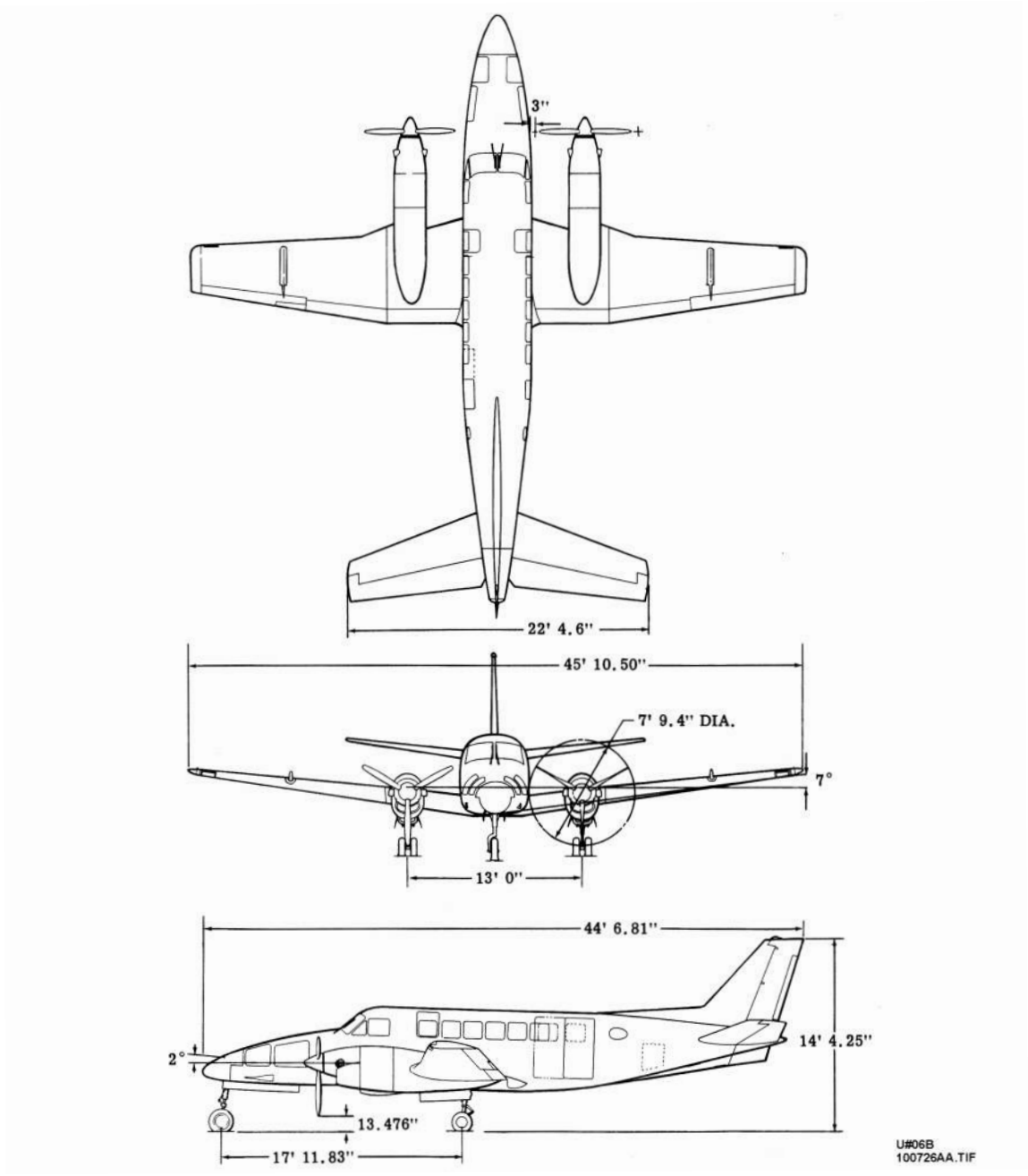

Figure 1
Airplane Dimensions

DIMENSIONS AND AREAS
AIRPLANE STATIONS
DESCRIPTION AND OPERATION

1. GENERAL

This chapter provides airplane station illustration to aid maintenance personnel in locating various components on the airplane as referenced elsewhere in this manual.

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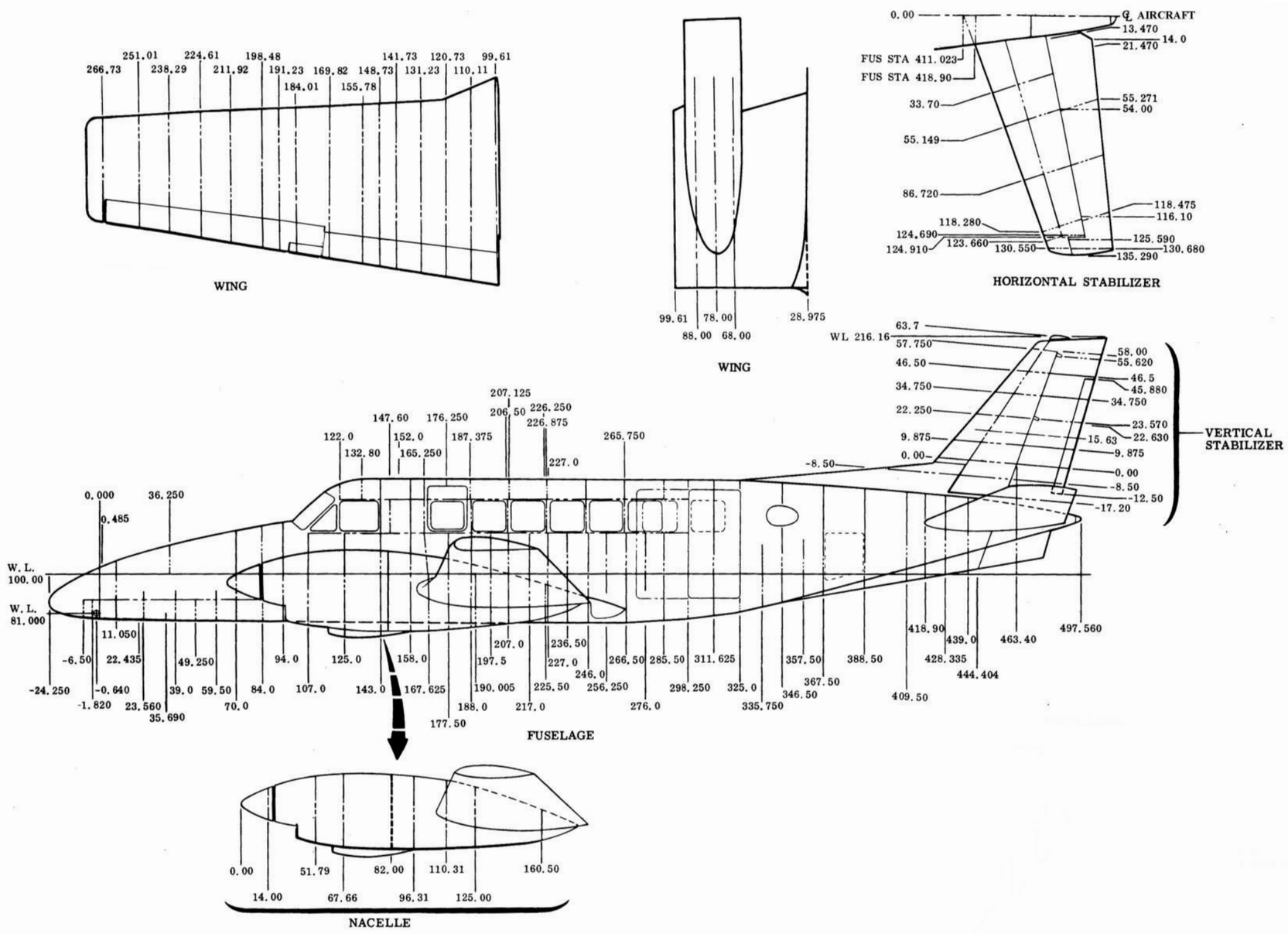
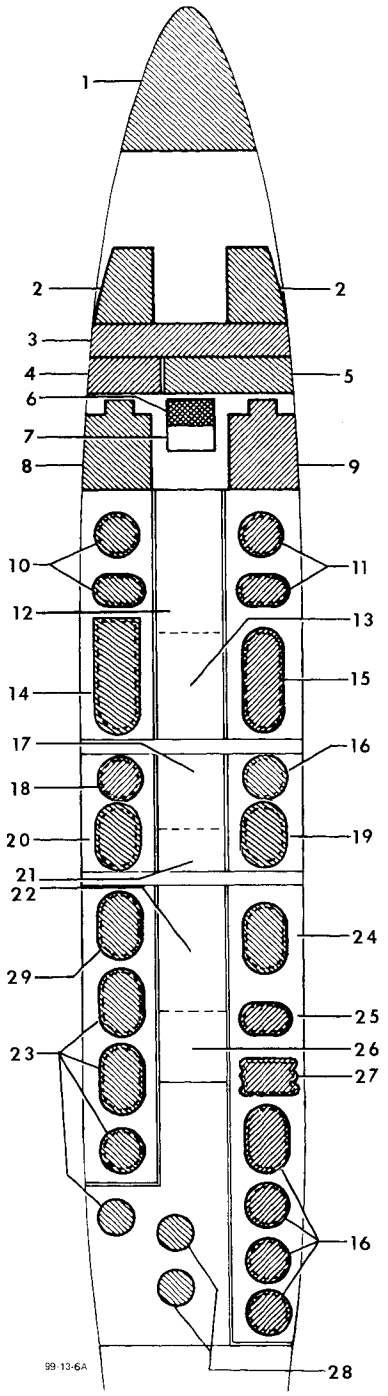


Figure 1
Stations Diagram

DIMENSIONS AND AREAS
AIRPLANE ACCESS PANELS
DESCRIPTION AND OPERATION

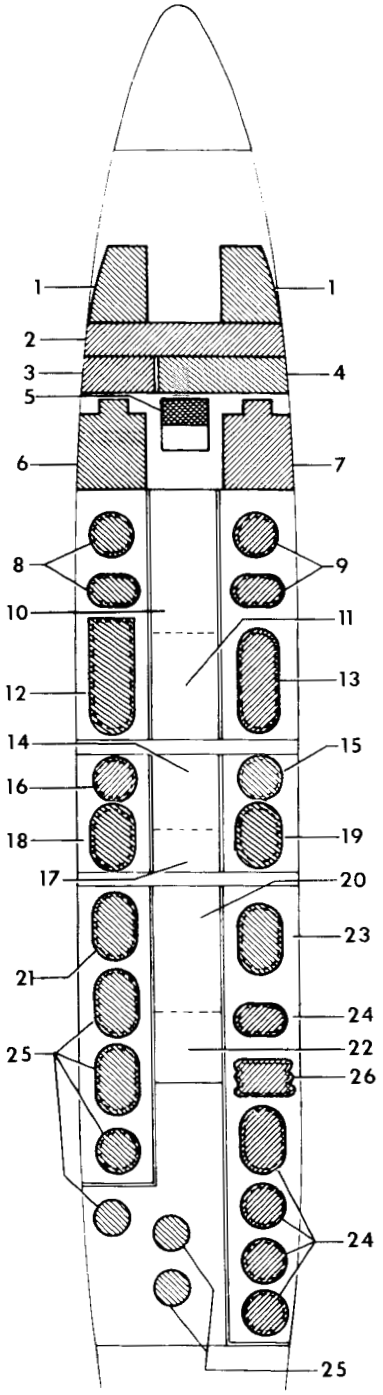
1. ACCESS OPENINGS

Figures 201 thru 206 provide illustrations of the access openings of the airplane.



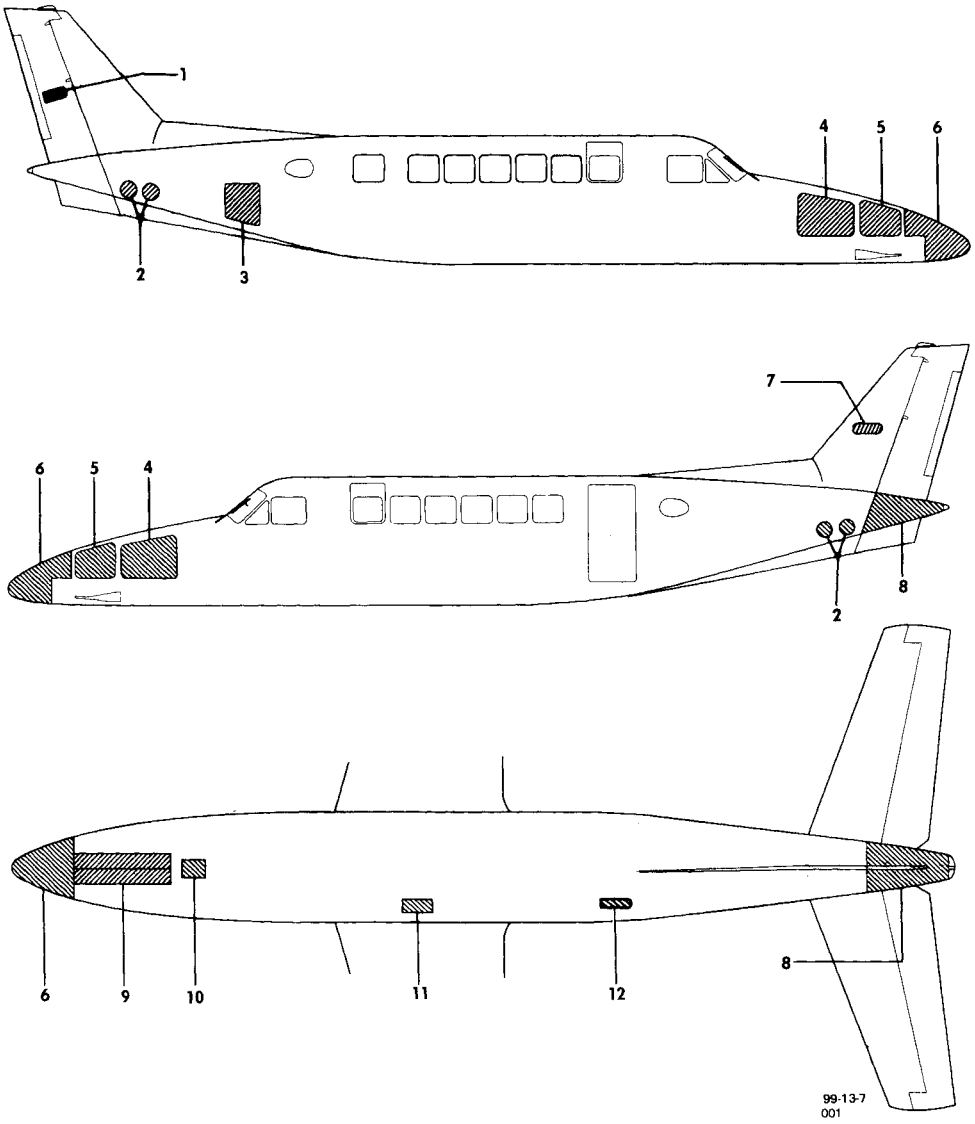
1. Battery and/or Avionics, Remove Cone To Gain Access To Nose Gear Hinge Bolts
2. Oxygen Bottles, Receiver Dryer, Overload Switch, Condenser Blower
3. Heater Combustion Blower, Condenser Coil
4. Evaporator Coil, Expansion Valve, Vent Blower, Solenoid Cycling Valve, Fresh Air Valve
5. Combustion Blower Differential Pressure Switch, Overtemp Switch, Heater Control Switch, Vent Blower Differential Pressure Switch
6. Return Air Screen
7. Floorboard Access Plate
8. Rudder, Pedal Shaft, Parking Brake Valve, Master Cylinder Attachment, Nose Gear Steering Connections, Trim Tab Cables, Heater Fuel Lines, Air Conditioner Lines
9. Copilots Rudder Pedal Shaft, Master-Cylinder Attachments, Shuttle Valve and Air Conditioner Plumbing, Rudder Pedal, Cabin Air Valve, Defroster, Heat and Vent Ducts, Annunciator Control Box
10. Control Cables and Pulleys
11. Heat and Vent Ducts
12. Landing Gear Emergency Extension Switch (Mechanical System), Heat and Vent Ducts, POWER DISTRIBUTOR PANEL: Generator Grounding Fault Transformers (L.H. & R.H.), Generator Out Relays (L.H. & R.H.), Paralleling Relays (L.H. & R.H.), Starter Ground Fault Relays (L.H. & R.H.), Battery Ground Fault Transformer, Battery Ground Fault Relay and Arc Suppression Diode, Battery Bus Diodes, External Power Ground Fault Relay, Ammeter Shunt, Printed Circuit Breaker Current Limiters, Landing Gear Remote Circuit Breaker Relay (Mechanical System), Landing Gear Power Relay (Hydraulic System), and Landing Gear Remote Circuit Breaker (Mechanical and Hydraulic System)
13. LANDING GEAR (Mechanical System): Motor and Gearbox, Limit Switches, Torque Shaft, Overload Clutch, Emergency Extension Clutch, and Retract Chain. LANDING GEAR (Hydraulic System): Power Pack, Motor, Pressure Switch, Service Valve, and Accumulator. UPPER ELECTRICAL EQUIPMENT PANEL: Voltage Regulators (L.H. & R.H.), Overvoltage Relays (L.H. & R.H.), Paralleling Rheostats, Generator Field Relays (L.H. & R.H.), Ground Fault Control Relay, Anti-Cycle Relay, Flap Micropositioner, Air Conditioner SCR Shutoff Capacitor, and Heat Sink Assembly. LOWER ELECTRICAL EQUIPMENT PANEL: Flap Dynamic Brake Relay, Inverter Power Relays (2), Landing Gear Motor Controller (Mechanical System Only), Landing Gear Time Delay Relay (Hydraulic System Only), Windshield Anti-Ice Control Relay, Heater Control Relay, Vent Blower Relay, Speed Control Vent Blower Relay, Engine Inlet Anti-Ice Control Relays (L.H. & R.H.), Engine Inlet Anti-Ice Shunts and Fuses, and Engine Inlet Heater Test Relays (2)
14. Avionics Power Relays (2), Avionics Cooling Relay, Warning Horn Silence Relay, Air Scoop Relay, Air Conditioner Time Delay Relay, Cool Lockout Relay, Evaporator Valve Lockout Relay, Heater Lockout Relay, Heater Cycle Relay, Combustion Air Time Delay Relay, Heater Fuse and Spares, Cycling Time Delay Relay, Surface Deice Relay, and Deicer Time Delay Relay
15. Landing Gear System Filter and Gear Select Solenoid Valve (Hydraulic System Only); Heat and Vent Ducts
16. Heat and Vent Ducts
17. Cabin Temperature Control Box, Flap Down Mode Relay, Fire Detector Amplifier, Smoke Detector Amplifier, Aileron Servo, Propeller Deice Timer, Shunt, and Fuses
18. Control Cables
19. Heat and Vent Ducts, Aileron and Trim Tab Pulleys
20. Control Surfaces Cables and Pulleys
21. Pressure Switch, Trim Tab Pulleys and Seals, Deicer Lines, Flap Motor and Gear Box, Flap Flexible Drives
22. Flight Hour Meter, Deice Lines, Instrument Air Ejector (U-152 and after), Trim Tab Pulleys and Seals, Bleed Air Regulator Valve (U-152 and after), Heater Ducts, Deicer Ejector, Distributor Valve
23. Control Surface Cables
24. Air Return Ducts, Heat and Vent Ducts, Air Conditioning Lines
25. (Optional Blower), Heat and Vent Ducts
26. Heater Ducting, Air Conditioner Evaporator Hardware, Deice Lines
27. Expansion Valve, Air Conditioner Evaporator and Hardware
28. Deice Lines
29. Instrument Air Filter (U-1 thru U-151)

Figure 1
Floorboard Access Openings (U-1 thru U-164, except U-50)



1. Oxygen Bottles
2. Air Conditioner Condenser Section, Condenser Blower, etc.
3. Evaporator Bypass Sensor Switch
4. Electric Heater, Overtemp Warning Switch, Overtemp Limit Switch, Heater Control Switch, Vent Blower Differential Pressure Switch
5. Return Air Screen
6. Rudder Pedal Shaft, Parking Brake Valve, Master Cylinder Attachment, Nose Gear Steering Connections, Trim Tab Cables, Heater Liner, Air Conditioner Lines
7. Rudder Pedal Shaft, Master Cylinder Attachment, Air Conditioner Plumbing, Cabin Air Valve, Defroster, Heat and Vent Ducts, Annunciator Control Box
8. Control Cables and Pulleys, Bleed Air Plumbing
9. Heat and Vent Ducts
10. Printed Circuit Board Rack Assembly, Fire Detection Control Amplifiers, No. 1 Avionics Bus Relay, No. 2 Avionics Bus Relay, Generator Line Contactors, Forward Evaporator Blower Speed Control Relay, Condenser Blower Relay, Power Isolator Limiters, Max Heat Power Limiter, Normal Heat Power Limiter, Forward Vent Blower Motor Relay
11. Generator Control Panels, External Power Relay, Landing Gear Time Delay, Landing Gear Power Relay, Lip Deice Shunts, Propeller Deice Shunt, Lip Deice Control Relays, Lip Deice Test Assembly, PCB Relay Assembly, Propeller Deice Manual Override Relay, Propeller Deice Timer, Deice Ammeter Positive CB, Deice Ammeter Negative CB, Propeller Deice CBs, Hydraulic Plumbing, Fuel Plumbing
12. Hydraulic Plumbing, Bleed Air Fail Warning Switches, Bleed Air Plumbing, Flight Control Cables
13. Heat and Vent Ducts, Hydraulic Plumbing
14. Hydraulic Power Pack, Service Valve, Uplock Pressure Switch, Accumulator, Gear Select Solenoid, Liquid Level Sensor, Landing Gear Motor, Hydraulic Plumbing
15. Heat and Vent Ducts
16. Flight Control Cables
17. Flap Motor and Gear Box, Flap Flexible Drives, Deice Lines, Flight Hour Meter Switch
18. Flight Control Cables
19. Deice Pressure Regulator and Relief Valve, Deice Lines, Instrument Air Ejector, Distribution Valve
20. Blank
21. Flight Control Cables
22. Air Conditioner Blower and Evaporator, Air Conditioner Lines
23. Heat and Vent Ducts, Air Conditioning Lines
24. Heat and Vent Ducts
25. Flight Control Cables and Pulleys
26. Expansion Valve, Heat and Vent Ducts

Figure 2
Floorboard Access Openings (U-50, U-165 and After)



. Rudder Tab Actuator
. Elevator Bell Crank and Stops
. Aft Fuselage Access and Baggage Door
. Baggage Access,Instrument Air Filter
. Avionics Equipment Access
. Radar Antenna (Optional)
7. Antenna Access
8. Empennage Control Linkage and Rudder Stops
9. Nose Landing Gear
10. Not in Use
11. External Power Receptacle

Figure 3
Fuselage Access Openings

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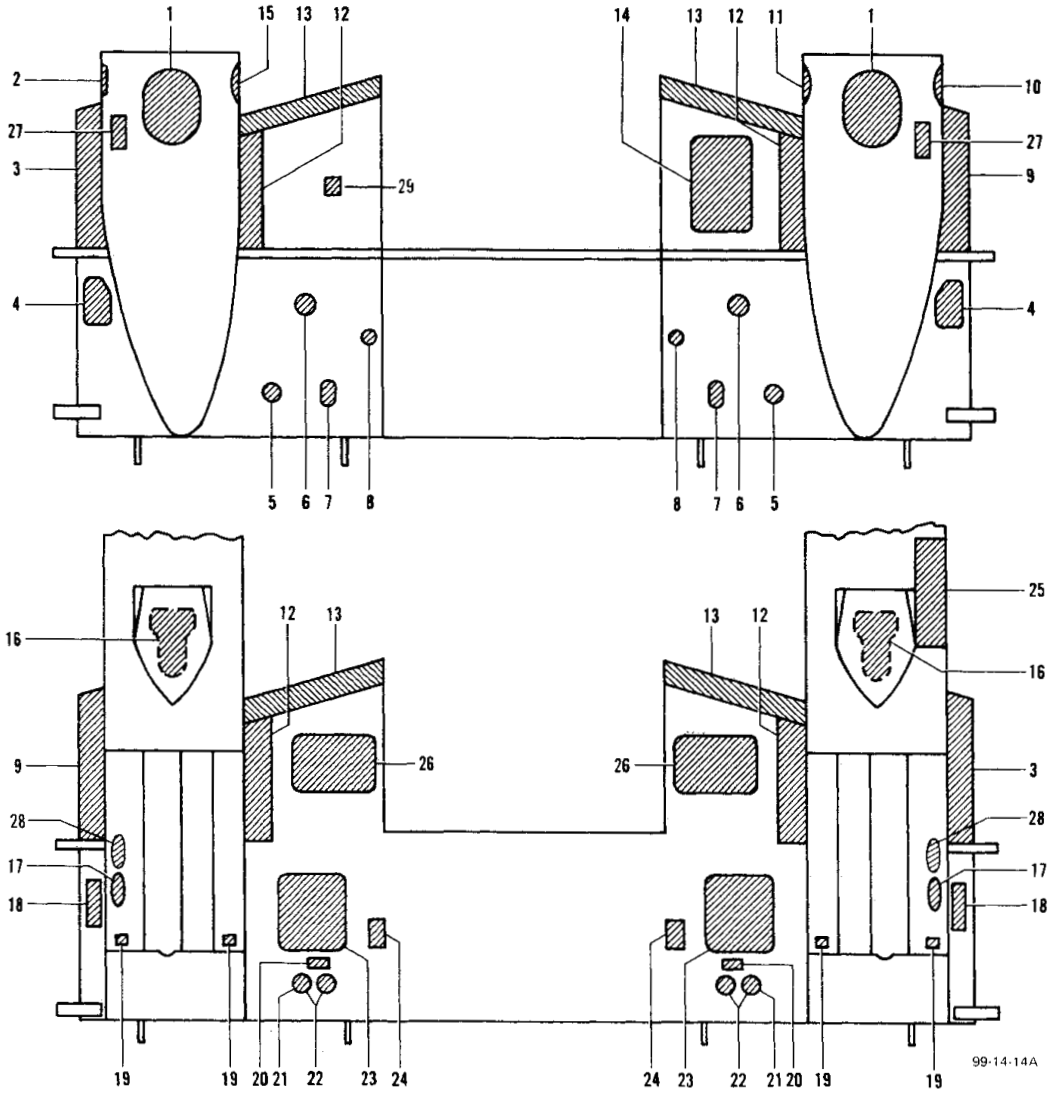


Figure 4 (Sheet 1 of 2)
Wing Center Section Access Openings (U-1 thru U-164, except U-50)

A. Wing Center Section Access (U-1 thru U-164, except U-50)

1. Nacelle Fuel Cell Filler Neck and Tank Unit (Not on A99A)
2. Fuel Line
3. Fire Extinguisher Lines, Fuel Tank Connector Lines and Wing Electrical Connectors
4. Inverter
5. Aileron and Aileron Tab Cable
6. 44 Gallon Fuel Cell
7. Inboard Flap Actuator
8. Fuel Tank Unit
9. External Power Receptacle (Prior to U-89), Fuel Line Check Valve and Wing Electrical Connectors
10. Fuel Line, Fuel Purge Valve, Fire Extinguisher Lines and Engine Control Cables
11. Bleed Air Lines and Firewall Shutoff Valve
12. Starter Relays, External Power Relay (RH Only), Crossfeed Valve (LH Only), Fuel Supply and Return Lines and Fuel System Drain Tee (LH Only)
13. Electrical Wiring, Hydraulic Lines, Engine Control Cables and ITT Balance Resistor
14. Battery (Optional Location)
15. Deice Lines, Electrical Lines, Firewall Fuel Shutoff Valve, Fuel Purge Valve and Engine Control Cables
16. Primary and Secondary Boost Pumps, Drain Manifold and Fuel Cell
17. Rear Nacelle
18. Inverter
19. Main Landing Gear Points
20. Fuel Line Connections
21. Inboard Flap Actuators
22. Aileron Cables
23. 44 Gallon Fuel Cell
24. Fuel Strainer and Drain
25. Engine Access
26. Landing Gear Retract Universal Bearings. RH Side Only - Emergency Fuel Fuse Panel. LH Side Only - Heater Fuel Pump, Fuel Shutoff Valve, Pressure Regulator and Strobe Light Power Unit
27. Wing Ice Light (U-89 and After)
28. Spar Inspection and Inverter Vent
29. Hydraulic Landing Gear Fill Can

Wing Center Section Access Openings (U-1 thru U-164, except U-50)
Figure 4 (Sheet 2 of 2)

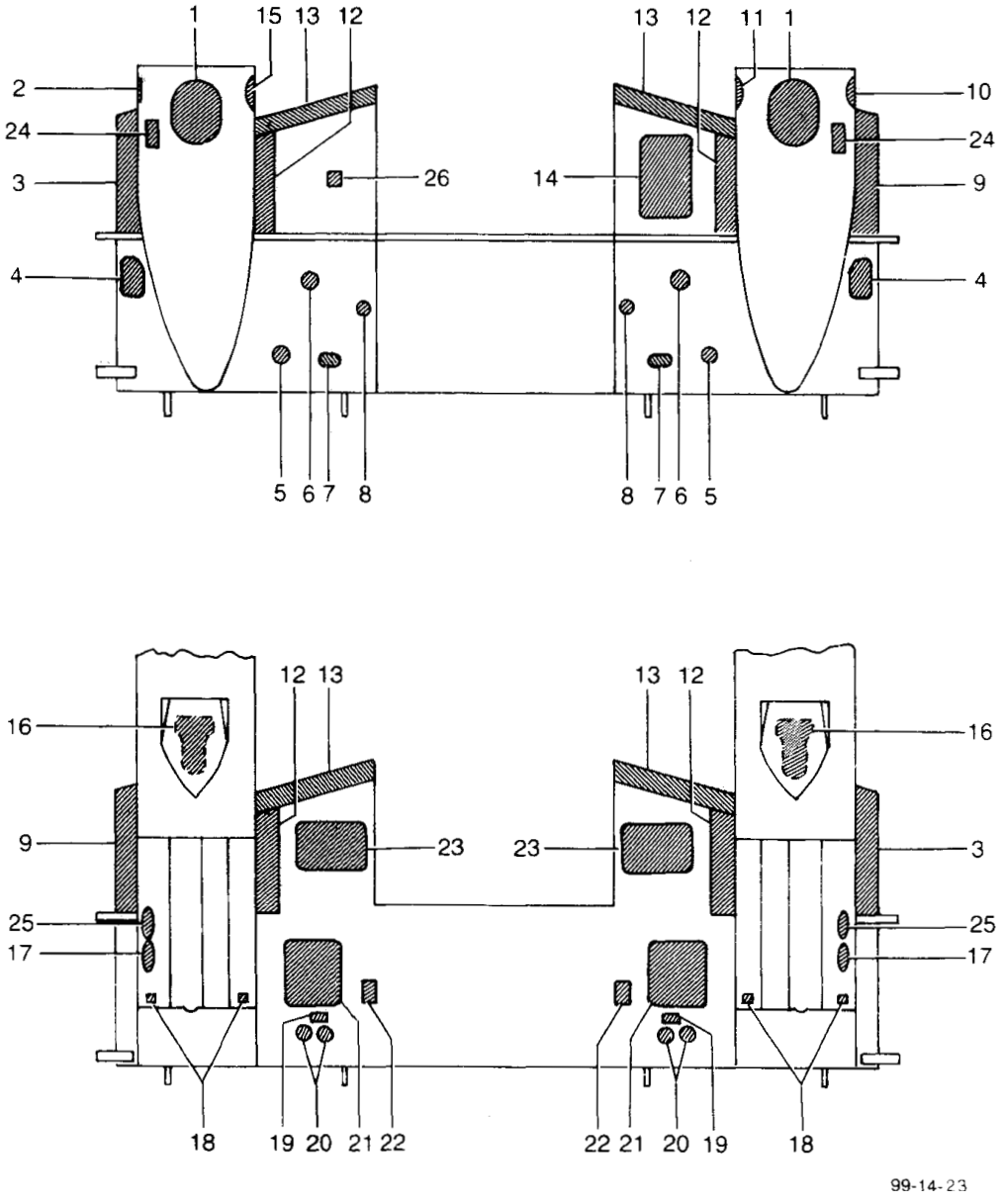
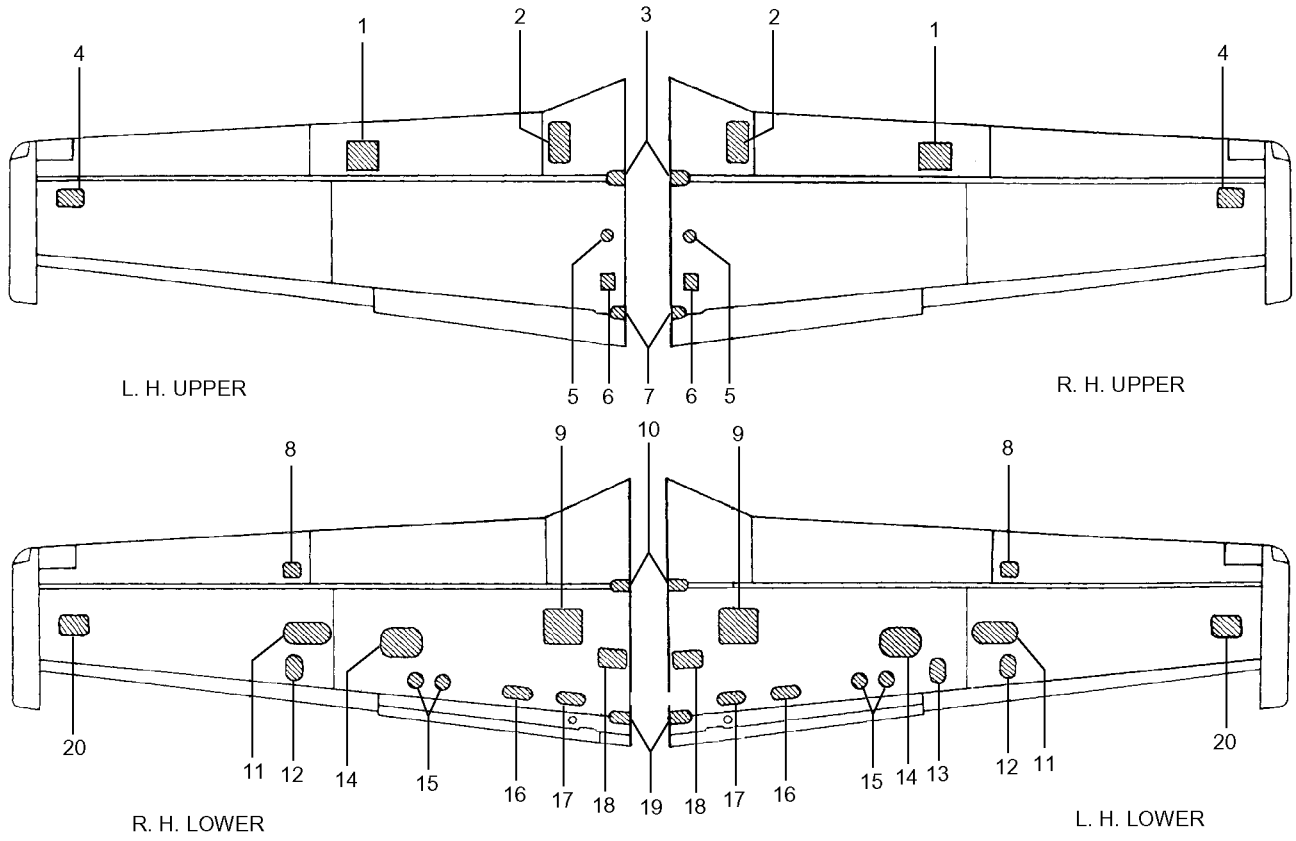


Figure 5 (Sheet 1 of 2)
Wing Center Section Access Openings (U-50, U-165 and After)

B. Wing Center Section Access (U-50, U-165 and After)

1. Nacelle Fuel Cell Filler Neck, Check Valve and Tank Unit
2. Fuel Lines, Electrical Wiring, Fuel Purge Valve and Motive Flow Valve
3. Fire Extinguisher Lines, Fuel Tank Connector Lines and Wing Electrical Connectors
4. Inverter
5. Aileron and Aileron Tab Cable
6. 44 Gallon Fuel Cell Filler Cap
7. Inboard Flap Actuator
8. Fuel Tank Unit
9. Fuel Line Check Valve and Wing Electrical Connectors
10. Fuel Line, Fuel Purge Valve, Fire Extinguisher Lines, Engine Control Cables and Motive Flow Valve
11. Bleed Air Shutoff Valve, Firewall Fuel Shutoff Valve and Wiring
12. Starter Relays, External Power Relay (RH Only), Crossfeed Valve (LH Only), Fuel Supply and Return Lines and Fuel System Drain Tee (LH Only)
13. Electrical Wiring, Hydraulic Lines, Engine Control Cables and ITT Balance Resistor
14. Battery
15. Bleed Air Shutoff Valve, Engine Wiring, Firewall Fuel Shutoff Valve and Engine Control Cables
16. Secondary Boost Pump, Drain Manifold and Fuel Cell
17. Rear Nacelle
18. Main Landing Gear Points
19. Fuel Line Connections
20. Inboard Flap Actuators and Aileron Cables
21. 44 Gallon Fuel Cell
22. Fuel Strainer and Drain
23. Landing Gear Retract Universal Bearings. RH Side Only - Emergency Fuel Fuse Panel. LH Side Only - Pressure Regulator and Strobe Light Power Unit
24. Wing Ice Light
25. Inverter Vent
26. Hydraulic Landing Gear Fill Can

Wing Center Section Access Openings (U-50, U-165 and After)
Figure 5 (Sheet 2 of 2)



1. LEADING EDGE FUEL CELL OUTBOARD QUANTITY TANK UNT AND FILLER NECK
2. LEADING EDGE FUEL CELL INBOARD QUANTITY TANK UNIT
3. UPPER FORWARD WING BOLT
4. REMOTE COMPASS
5. 23 GALLON FUEL CELL QUANTITY TANK UNIT
6. WING PLUMBING
7. UPPER AFT WING BOLT
8. DEICER, FUEL VENT
9. 23 GALLON FUEL CELL
10. LOWER FORWARD WING BOLT
11. DEICER LINES
12. AILERON BELLCRANKS AND STOPS
13. TRIM TAB ACTUATOR
14. 25 GALLON FUEL CELL
15. AILERON PULLEY AND FAIRLEAD
16. AILERON PULLEY
17. OUTBOARD FLAP ACTUATOR
18. FUEL VENT
19. LOWER AFT WING BOLT
20. WING TIP ATTACHMENT

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Figure 6
Wing Section Access Openings

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 07 - LIFTING AND SHORING

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JACKING

07-10-00

Maintenance Practices 1
Procedures 1
Three-Point Jacking 1
Lowering the Airplane After Three-point Jacking 2
Nose Jacking 2
Single-Point Jacking (For Wheel, Tire and Brake Maintenance Only) 3
Lowering the Airplane After Single-Point Jacking 3

SHORING

07-20-00

Maintenance Practices 1
Procedures 1
Hoisting 1

LIFTING AND SHORING
GENERAL INFORMATION
DESCRIPTION AND OPERATION

1. LIFTING

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gears are down and locked prior to removing the airplane from the jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

A. Hoisting

The airplane may be hoisted for maintenance or parts replacement. A minimum overhead clearance of 17 feet and a hoist of at least 10,000-pound capacity are required for hoisting.

If it is necessary to hoist the airplane with one or both engines removed, use a sling under the tail. The tail mooring hole can be used to hold the sling in position.

The 99-590029-1 hoisting sling (Ref. SPECIAL TOOLS, Chapter 12-20-00, 201), or its equivalent, is required to hoist the airplane. The hoisting sling is constructed of two broad-banded-nylon straps supported by a structural steel crossbar with wire ropes for attachment to an overhead crane or hoist. The sling is designed to support approximately 33,000 pounds to ensure a high safety factor during hoisting. The straps of the sling encircle the nose and midsection of the fuselage when the airplane is being hoisted as follows:

B. Jacking

The airplane has provisions for a three-point jack-pad system. Two pads are located on the center section rear spar inboard of each nacelle and one is near the aft edge of the nose wheel well, to the left of the fuselage centerline.

Each of the three jacks used to lift the airplane should have a lifting capacity of at least 10,000 pounds.

LIFTING AND SHORING
JACKING
MAINTENANCE PRACTICES

1. PROCEDURES

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gears are down and locked prior to removing the airplane from the jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

A. Three-Point Jacking

WARNING: When jacking the airplane in an unsheltered area where winds in excess of 35 kts will be encountered, never jack more than one gear at a time clear of the ground.

CAUTION: *Position airplane on a flat, hard, oil free surface prior to any jacking operations.*

Do not attempt any jacking operations outdoors during strong, gusty winds. If the airplane must be lifted outdoors, always face the airplane into the wind.

Ensure area is clear of obstacles that could damage the airplane during raising operations.

Before jacking the airplane, the battery switch, and the engine inlet deicer control must be OFF.

NOTE: The airplane has provisions for a three-point jack-pad system. Two pads are located on the center section rear spar inboard of each nacelle and one is near the aft edge of the nose wheel well, to the left of the fuselage centerline.

If tail stand is installed, remove tail stand during jacking operations.

Each of the three jacks used to lift the airplane should have a lifting capacity of at least 10,000 pounds.

(1) Install forward jacking point adapter (7) (Ref. Chapter 12-20-00, SPECIAL TOOLS) for the airplane nose jack (3) (Ref. Figure 201).

(2) Position a tripod jack (3) at the nose of the airplane directly under the jack pad with two of the jack legs (4) perpendicular to the fuselage (Ref. Figure 201).

(3) Position a tripod jack (3) at each wing of the airplane directly under each jack pad with two of the jack legs (4) parallel to the fuselage.

(4) Extend jack extension (1) of each jack to contact securely with its associated jack pad.

WARNING: When jacking the airplane in an unsheltered area where winds in excess of 35kts will be encountered, never jack more than one gear at a time clear of the ground.

CAUTION: *One person should be positioned forward of the airplane to ensure that the airplane remains level during three-point jacking.*

Keep the jack follower nut within two inches of the jack shoulder at all times during three-point jacking and lowering.

(5) Operate the wing and nose jacks together until the clearance between the tires and the ground is at least 2 inches (51 mm).

(6) Tighten the follower nut (2) of each jack (3) against the jack shoulder and release pressure from the jack pump cylinder (6).

(7) Install a tail stand at the aft fuselage mooring point of the airplane and bring it into secure contact with the ground.

B. Lowering the Airplane After Three-point Jacking

WARNING: Ensure that the landing gear is in the fully down and locked position. Failure to do so may result in injury to personnel and damage to the airplane.

CAUTION: *Ensure area is clear of obstacles that could damage the airplane during lowering operations.*

Keep the jack follower nut within two inches of the jack shoulder at all times during three-point jacking and lowering.

(1) Ensure that the landing gear is in the fully down and locked position.

(2) Wipe down the extended surface of each landing gear strut using a shop rag dampened with hydraulic fluid.

CAUTION: *One person should be positioned forward of the airplane to ensure that the airplane remains level during the lowering operation.*

(3) Ensure that the airplane weight is off the tail stand, then remove the tail stand.

CAUTION: *Keep the jack follower nut within two inches of the jack shoulder at all times during three-point jacking and lowering.*

(4) Operate jacks to loosen the jack follower nuts (2), then slowly release pressure from the jack pump cylinder (6) to lower the airplane slowly to the ground (Ref. Figure 201).

CAUTION: *Ensure that all jacks are fully retracted and that the airplane has settled prior to jack removal. Damage to the airplane may occur.*

(5) Remove the jacks (3) from under the airplane.

(6) Remove the forward jacking point adapter (7) from the airplane.

C. Nose Jacking

CAUTION: *When jacking the nose wheel, use an automobile-type jack with a capacity of at least 4,000 pounds.*

A jack point on the nose landing gear enables the wheel to be raised for changing the tires and wheel repair.

D. Single-Point Jacking (For Wheel, Tire and Brake Maintenance Only)

WARNING: This procedure is intended for wheel and brake maintenance only. Never perform any other landing gear maintenance procedures using the Single-Point Jacking Procedure.

Never jack the airplane in an unsheltered area where winds in excess of 35 knots will be encountered.

NOTE: Any jack used to lift the airplane should have a lifting capacity of at least 10,000 pounds.

CAUTION: *Position airplane on a flat, hard, oil free surface prior to any jacking operations.*

Do not attempt any jacking operations outdoors during strong, gusty winds. If the airplane must be lifted outdoors, always face the airplane into the wind.

Ensure area is clear of obstacles that could damage the airplane during raising operations.

Before jacking the airplane, the battery switch, and the engine inlet deicer control must be OFF.

NOTE: If tail stand is installed, remove tail stand during jacking operations.

(1) Place airplane chocks forward and aft of the landing gear wheels that are not being jacked.

NOTE: In ice or snow conditions, ice-grip wheel chocks are preferred. Sandbags may be used if ice-grip chocks are not available, or if the airplane is parked on a steel mat.

(2) Position a tripod jack (3) at the wing of the airplane directly under the jack pad with two of the jack legs (4) parallel to the fuselage (Ref. Figure 201).

(3) Extend jack extension (1) of the jack to contact securely with its associated jack pad.

CAUTION: *Keep the jack follower nut within two inches of the jack shoulder at all times during jacking and lowering procedures.*

(4) Install a strut limiter to the landing gear strut being worked (Ref. Chapter 12-20-00, SPECIAL TOOLS).

(5) Operate the jack as required to raise the affected wheel until the clearance between the tires and the ground is at least 2 inches (51 mm).

(6) Tighten the follower nut (2) of the jack (3) against the jack shoulder and release pressure from the jack pump cylinder (Ref. Figure 201).

E. Lowering the Airplane After Single-Point Jacking

CAUTION: *Ensure area is clear of obstacles that could damage the airplane during lowering operations.*

(1) Wipe down the extended surface of the landing gear strut using a shop rag dampened with hydraulic fluid.

CAUTION: *Keep the jack follower nut within two inches of the jack shoulder at all times during jacking and lowering procedures.*

(2) Operate jack to loosen the jack follower nut (2), then slowly release pressure from the jack pump cylinder to lower the airplane slowly to the ground (Ref. Figure 201).

CAUTION: *Ensure that the jack is fully retracted and that the airplane has settled prior to jack removal. Damage to the airplane may occur.*

(3) Remove the jack (3) from under the airplane.

(4) Remove the strut limiter from the landing gear strut.

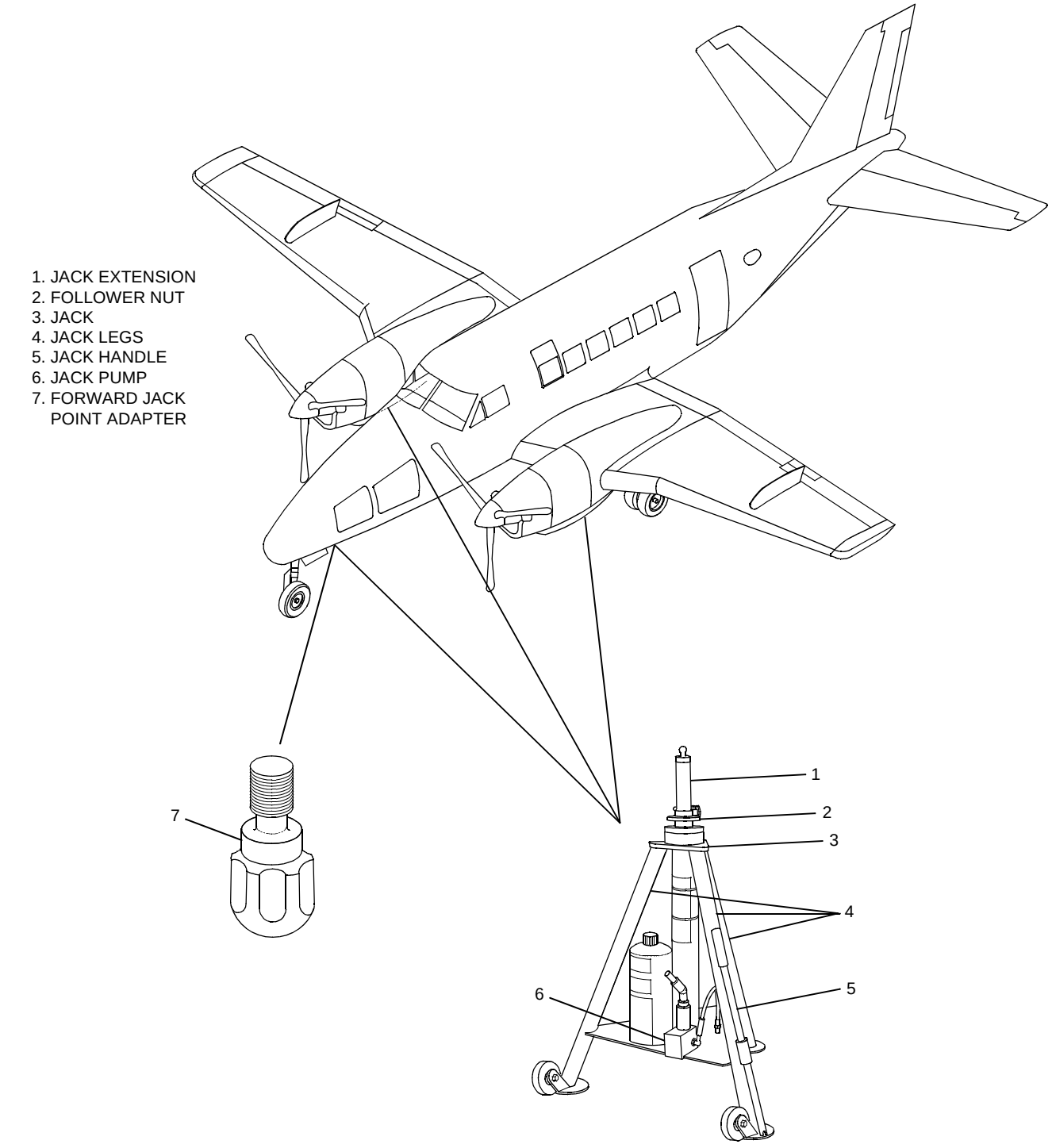

Figure 201
Landing Gear Jacking Points

LIFTING AND SHORING
SHORING
MAINTENANCE PRACTICES

1. PROCEDURES

A. Hoisting

WARNING: Prior to attaching the sling to the airplane, inspect the sling for damage or wear. Any defect could diminish the strength of the sling.

CAUTION: *Before hoisting the airplane, the fuel tanks must be drained and the airplane should be empty.*

The airplane may be hoisted for maintenance or parts replacement. A minimum overhead clearance of 17 feet and a hoist of at least 10,000-pound capacity are required for hoisting.

If it is necessary to hoist the airplane with one or both engines removed, use a sling under the tail. The tail mooring hole can be used to hold the sling in position.

The 99-590029-1 hoisting sling (Ref. SPECIAL TOOLS, Chapter 12-20-00, 201), or its equivalent, is required to hoist the airplane. The hoisting sling is constructed of two broad-banded-nylon straps supported by a structural steel crossbar with wire ropes for attachment to an overhead crane or hoist. The sling is designed to support approximately 33,000 pounds to ensure a high safety factor during hoisting. The straps of the sling encircle the nose and midsection of the fuselage when the airplane is being hoisted as follows:

(1) Place one nylon band at Fuselage Station 107.0 in the nose area and the other nylon band at Fuselage Station 266.5 in the midsection portion of the fuselage.

(2) Attach the sling to the cable or chain on the overhead hoisting crane at a position corresponding to Fuselage Station 180.0.

(3) Guy ropes between the wing tip and lift point or a man stationed at each wing tip may be required to off set any unbalanced weight in the wings.

(4) Remove all loose equipment from the airplane and hoist the airplane smoothly. Take up slack in the hoist and begin the lift gently, watching carefully for any unbalance.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 08 - LEVELING AND WEIGHING

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GENERAL INFORMATION

08-00-00

Description and Operation 1
General 1
Leveling 1

LEVELING AND WEIGHING
GENERAL INFORMATION
DESCRIPTION AND OPERATION

1. GENERAL

A. Leveling

The airplane may be leveled by inflating or deflating the main and/or nose gear struts. Use a bubble level on the aft baggage compartment floorboard and deflate the tire or strut on the high side of the airplane to level the aircraft laterally. To level the airplane longitudinally, attach a cord and plumb bob to the upper Phillips head screw just forward of the cabin door. Inflate or deflate the nose gear shock strut as required to pass the cord over the center line of a second Phillips head screw directly below at the bottom of the door. Suspending the plumb bob in a can of light engine oil will dampen its movement.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 09 - TOWING AND TAXIING

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09-10-00

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General 1
Towing 1

TOWING AND TAXIING
GENERAL INFORMATION
DESCRIPTION AND OPERATION

1. GENERAL

A. Towing

With the tow bar connected to the towing lugs on the upper torque knee fitting of the nose strut, the airplane can be steered with the nose wheel when moving it by hand or with a tug. Although steering is automatic when the airplane is being towed by the nose strut, someone should ride in the pilot's seat to operate the brakes in the event of an emergency. Do not tow airplane with rudder locks installed, as severe damage to the steering linkage can result. When using a tug, observe turn limits marked on the nose gear strut to prevent damage to nose gear. When spotting the airplane, do not push on the propeller or control surfaces.

NOTE: The nose gear steering stop block is designed to withstand loads imposed through steering from the cockpit and not to prevent turn limitations being exceeded during ground handling. It is possible to overcome the stop during ground handling and this will result in damage to the steering linkage and the nose strut.

CAUTION: *Never tow or taxi with a deflated strut. Even brief towing or taxiing with a deflated strut may cause severe damage.*

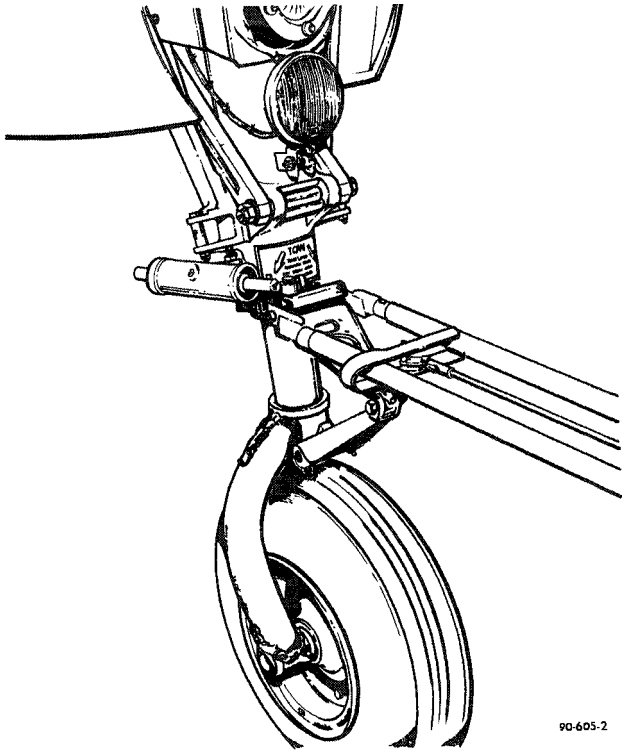

Figure 1
Towing the Airplane

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MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 10 - PARKING AND MOORING

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10-10-01

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ANCHORING AND MOORING

10-20-00

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PARKING AND MOORING
GENERAL INFORMATION
DESCRIPTION AND OPERATION

1. GENERAL

A. Parking

The brakes can be set for parking by depressing the pilot's brake pedals and pulling out the parking brake control. Do not attempt to lock the parking brake by applying force to the parking brake handle; it controls a valve only, and cannot apply pressure to the brake system. To release the brakes, depress the brake pedals and push the parking brake control in.

NOTE: Do not set the parking brakes during low temperatures when an accumulation of moisture may cause the brakes to freeze, or when they are hot from severe use.

B. Control Locks

The control lock consists of a U-shaped clamp and three pins connected by a chain. The pins lock the primary flight controls and the U-shaped clamp fits around the engine power control levers and serves to warn the pilot not to start the engines with the control locks installed. It is important that the locks be installed or removed together due to the possibility of an attempt to taxi or fly the airplane with the power levers released and the pins still installed in the flight controls.

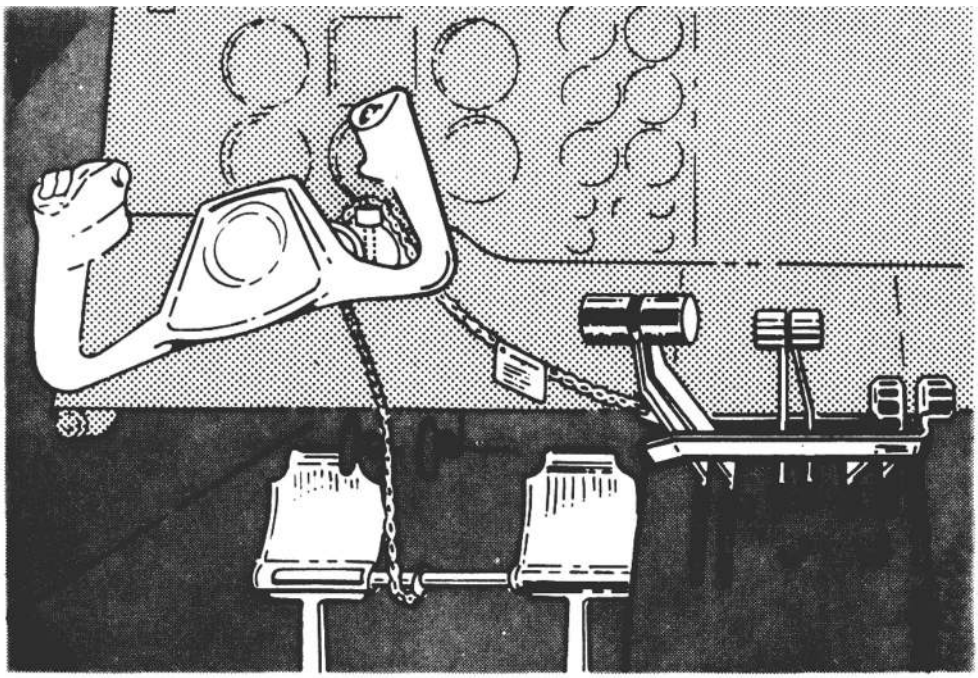

Figure 1
Control Lock

PARKING AND MOORING
PARKING
MAINTENANCE PRACTICES

1. CONTROL LOCKS

A. Installation

The control lock consists of a U-shaped clamp and three pins connected by a chain. The pins lock the primary flight controls and the U-shaped clamp fits around the engine power control levers and serves to warn the pilot not to start the engines with the control locks installed. It is important that the locks be installed or removed together due to the possibility of an attempt to taxi or fly the airplane with the power levers released and the pins still installed in the flight controls. Install the control locks in the following sequence:

(1) Position the U-clamp around the engine power controls.

(2) Insert the small pin in the elevator-aileron pilot's control from the upper side of the pilot's control column.

NOTE: On airplane serials U-165 and after, and airplanes U-1 thru U-164 that have incorporated Service Instructions No 0887-155, the control wheel must be fully forward and rotated to the left 15 degrees before the control pin can be inserted.

(3) Insert the largest pin in the pilot's right rudder pedal by pushing forward on the left rudder pedal and inserting the pin into the hole, located on the inside of the pilot's right rudder pedal

(4) Neutralize the rudder pedals and slide the pin into the hole in the left rudder pedal.

B. Removal

(1) Remove the pins from the pilot's left and right rudder pedals.

(2) Remove the elevator-aileron pin from the upper side of the pilot's control column.

(3) Remove the U-shaped clamp from the engine power controls.

NOTE: The control locks should be removed before towing, as serious damage could result in the steering linkage if towed by a tug with the rudder locks installed.

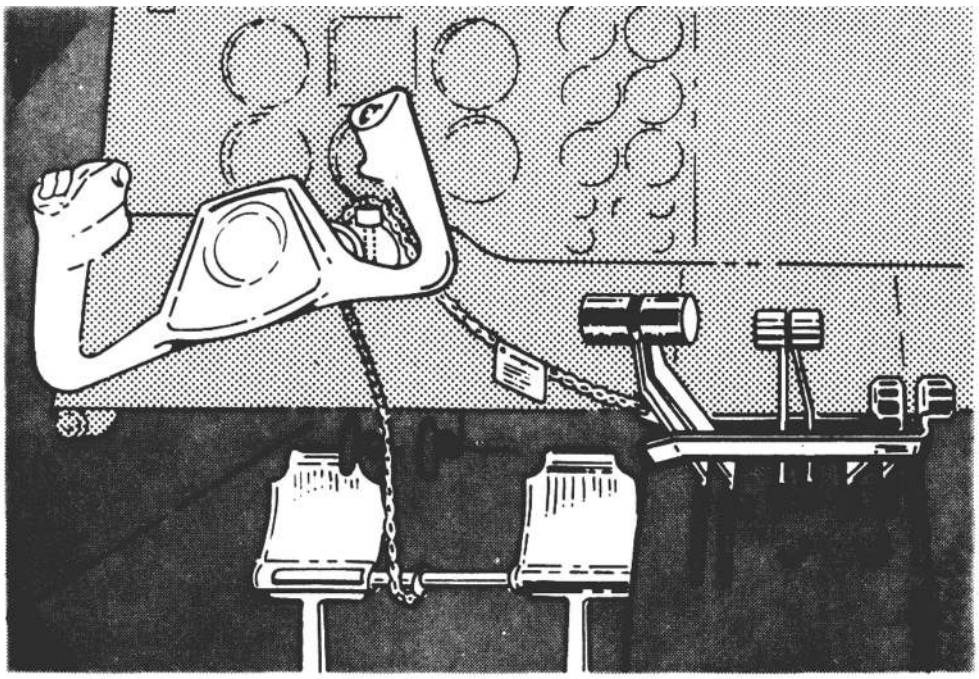

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Figure 201
Control Lock

PARKING, MOORING, STORAGE AND RETURN TO SERVICE
STORAGE
MAINTENANCE PRACTICES

1. GENERAL

A. Engine

CAUTION: *Under no circumstances should preservative oil be sprayed into the compressor or turbine end of the engine. Dirt particles deposited on the blades and vanes during engine operation will alter the airfoil shape and will adversely affect compressor efficiency.*

(1) **0 to 7 Days** - The engines may be left in an inactive state, with no preservation protection requirements, provided: the engine is sheltered; the humidity is not excessively high; the engine is not subject to extreme temperature changes, which would produce condensation.

CAUTION: *The propeller must be restrained to prevent windmilling (Ref. PROTECTIVE COVERS, 10-20-00).*

(2) **8 to 28 Days** - Engines that are inactive up to 28 days require no preservation provided all engine openings are sealed off and the relative humidity in the engine is maintained at less than 40%. Humidity control is maintained by placing desiccant bags (98, Table 6, 91-00-00) in the engine exhaust duct.

NOTE: Ensure that the desiccant bags are kept off of all engine parts by placing the bags on racks. Provide suitable windows in the exhaust closures for observation of the humidity indicators.

(3) **29 to 90 Days** - Engines inactive for a period exceeding 28 days, but less than 90 days, need only have the fuel system preserved as noted in the following paragraphs.

CAUTION: *Extreme care must be taken to prevent foreign material from being drawn into the engine fuel system. Equipment must be fitted with suitable filters, or strainers. no coarser than 10-micron ratings.*

(a) Close the normal fuel supply shutoff valve. Disconnect the fuel inlet line to the fuel pump or to the oil- to-fuel heater and connect a suitable oil supply line to the fuel pump or the oil-to-fuel heater. Supply lubricating oil (1, Table 6, 91-00-00) at a pressure of 5 to 25 psi and a temperature of at least 16°C (60°F).

CAUTION: *Under no circumstances permit preservative oil to enter engine where it may come in contact with the thermocouple probe assembly. Oil contamination of harness may cause complete failure of the thermocouple system.*

(b) To prevent lubricating oil from entering the fuel manifold, disconnect No. 1 and No. 2 manifolds and bypass and dump lines at the starting flow control. Cap off No. 1 manifold and the bypass and dump outlets. Allow No. 2 manifold to drain into a suitable container.

CAUTION: *Do not exceed the starter motor operating time of 40 seconds ON, 60 seconds OFF, 40 seconds ON, 60 seconds OFF, 40 seconds ON, then 30 minutes OFF.*

(c) With the ignition system OFF, the power lever at TAKEOFF position and the condition lever at HlGH IDLE, motor the engine while moving the condition lever from HIGH IDLE to CUTOFF and back to HlGH IDLE.

(d) After motoring the engine, check to see if oil is draining from the No. 2 manifold outlet If required, repeat the motoring cycle until oil drains from the outlet.

(e) Following the motoring cycle, return the power lever to IDLE and the CONDITION LEVER to CUTOFF. Reconnect all lines to the starting fuel control and reconnect the fuel supply line to the oil-to-fuel heater.

(f) Ensure that all shipping plugs, caps and covers are installed, that the engine intake and exhaust openings are covered, and that desiccant and humidity indicators are installed.

(4) **91 Days and Over** - Engines inactive over 90 days, or removed for long term storage must, in addition to the 29 to 90 day procedures, have the engine oil drained and have unused accessory drive-ends sprayed as noted in the following sub-paragraphs:

CAUTION: *Do not exceed the starter motor operating time of 40 seconds ON, 60 seconds OFF, 40 seconds ON, 60 seconds OFF, 40 seconds ON, then 30 minutes OFF.*

(a) Close the fuel shutoff valve and motor the engine until oil pressure and compressor speed (N_1) are indicated.

(b) Remove the drain plugs from the tank, accessory gearbox and propeller reduction gearbox.

(c) With the drains open, motor the engine to permit scavenger pumps to clear the engine as indicated by the cessation of a steady stream of oil from the drains. To prevent excessive operation with limited lubrication, limit rotation to the shortest possible time to accomplish the complete draining.

(d) Remove the oil filter element and allow it to drain to a slow drip (approximately one-half hour), then reinstall the filter and close the drains.

(e) Install plugs, caps and covers over all openings to prevent entry of foreign materials and accumulation of moisture.

NOTE: If the engine is to remain in the airplane, place desiccant bags in the exhaust and inlet ducts.

(f) Tag the oil filler cap with the date of preservation.

(g) Install humidity indicators in the air inlet end and exhaust end of the engine compartment. Cover with suitable airtight moisture barrier and provide inspection windows at each end for observation of the humidity indicators.

NOTE: Inspection of a preserved unit should be performed every two weeks if the airplane is stored outside, or every 30 days if the airplane is stored inside. If the relative humidity indicator is less then 40%, no further action is required. If humidity indicated is in excess of 40%, the desiccant bags must be replaced with freshly activated bags.

B. Airplane

(1) Disconnect and remove the nickel-cadmium battery, then place in a suitable storage area.

CAUTION: *It is imperative that the storage areas of nickel- cadmium batteries be separated (in different buildings if possible) from lead acid batteries so that no contamination, even from fumes, is possible.*

Clean the battery compartment, quick disconnect plug, cables, and vent hoses with a solution of 5 ounces of boric acid dissolved in one gallon of water, then rinse with dean water and allow to dry. Seal the battery vent tubes and cover the quick disconnect plug with barrier material.

(2) **Fuel Cells** - If the airplane is to be in storage 90 days or less, the fuel cells need only to be filled to capacity to minimize fuel vapor and protect the cell inner liners. Over 90 days, drain the cells, then flush, spray or rub a thin coating of light engine oil (99, Table 6, 91-00-00) on the inner liners of all fuel cells.

(3) **Propellers** - Remove dirt, oil, and bug accumulation from propellers with water and a soft brush. Clean the propellers with solvent (38, Table 6, 91-00-00), then coat the blades with corrosion preventive compound (88, Table 6, 91-00-00). Wrap the propeller blades with barrier material and secure with tape. Touch up the propeller spinners with paint if necessary, then secure the blades to prevent rotation.

(4) **Avionics** - Clean and cover any equipment sensitive to dust or moisture and take any additional precautions recommended by the manufacturer of such equipment.

(5) **Instruments** - If the airplane is to be in storage for more than 90 days, cover the instrument panel with barrier material and secure with tape.

(6) **Seats** - Clean and install protective covers.

(7) **Stall Warning Unit** - If the airplane is to be in storage for 90 days or more, remove the stall warning unit, then tape connections, and store in a room of low humidity.

(8) **Landing, Strobe, and Taxi Lights** - For 90 days or longer, cover with barrier material and tape.

(9) **Flight Control Surfaces** - Up to 90 days, lock with internal locks. Over 90 days, lubricate all hinge pins, bearings, bell cranks, chains, control rods, and quadrants in accordance with the Lubrication Schedule, (Figure 203, 12-20-00), and coat lightly with corrosion preventative compound (88, Table 6, 91-00-00). Lock with internal locks.

(10) **Wing Flap Tracks and Rollers** - If over 90 days, coat the flap tracks and rollers with corrosion preventative compound (88, Table 6, 91-00-00). Place flaps in retracted position.

(11) **Landing Gear** - Clean the brakes and apply a coating of primer (100, Table 6, 91-00-00) to the brake discs. Touch up all spots where paint has been chipped from the wheels. After mooring, cover or wrap the wheels with barrier material and secure with tape. Check air pressure periodically and inflate tires as necessary. It is recommended that unserviceable tires be used for prolonged storage. Coat the exposed surfaces of the shock strut pistons and nose gear shimmy dampener piston with preservative hydraulic fluid (101, Table 6, 91-00-00) and protect with barrier material. Fabricate hardwood collars for the strut pistons to prevent bottoming the struts when deflated. Install the collars over the barrier material. Slowly deflate struts until they rest on the wooden collars.

CAUTION: *Do not apply corrosion preventative to the exposed surfaces of the landing gear strut piston or to the extended polished surfaces of the hydraulic cylinders.*

(12) **Hydraulic System** - Fill the hydraulic reservoir to operational level and inspect the system for leaks, then repair as necessary prior to storage.

(13) **Loose Tools and Equipment** - Remove and store in a room of low humidity for storage over 90 days.

(14) **Airframe** - If over 90 days, install covers over pitot head and cover static ports with barrier material secured with tape. Clean all exposed antennas and connections. Remove windshield wipers and blades, then wrap, tag, and store in cockpit. Cover windows and windshield with either: (1) strippable coating, (2) alkali removable coating, or (3) barrier material secured with tape.

(15) **Mooring** - If the airplane cannot be placed in a hanger, tie it down securely using the three points provided. Do not use hemp or manila rope. It is recommended that a tail support be used to compress the nose strut and reduce the angle of attack of the wings. Attach a line to the nose gear.

(16) **Grounding** - Static ground the airplane securely and effectively.

C. Removal from Storage

(1) Remove all covers, tape, control locks, tie downs and tags from the airplane.

(2) Fill the oil tank to operating level with specified oil (1, Table 6, 91-00-00).

(3) Provide a source of specified fuel, as outlined in Chapter 12, and connect to the fuel inlet port on the fuel control assembly.

(4) Disconnect the fuel line between the fuel control assembly and fuel shutoff valve. Connect a line to the fuel control assembly to drain overboard.

(5) Motor the engine, without ignition, until clear fuel flows from the overboard drain.

(6) Stop motoring the engine and disconnect the overboard drain line. Reconnect the fuel line between the fuel shutoff valve and the fuel control assembly.

(7) Start engine, observing normal start procedure, and operate for approximately three minutes.

(8) Stop engine. When engine has completely stopped, shut off airplane fuel supply to the engine.

(9) Inspect all lubrication lines and fuel lines and their connection points for evidence of leakage.

(10) Remove the fuel filter strainer element and inspect, clean and reinstall. Refer to ENGINE FUEL FILTERS AND SCREENS, Chapter 12-10-00.

(11) Remove the oil filter element and inspect, clean and reinstall. Refer to OIL SYSTEM, Chapter 12-10-00.

(12) Give the airplane a thorough cleaning, visual inspection and test flight.

PARKING AND MOORING
ANCHORING AND MOORING
DESCRIPTION AND OPERATION

1. MOORING

Three mooring eyes are provided, one on each wing and one on the tail. To moor the airplane, chock the wheels fore and aft, install the control lock and tie the airplane down at all three points. Avoid overtightening the rear line and pulling the nose of the airplane up so far that wind will create lift on the wings. If extreme weather is anticipated, it is advisable to nose the airplane into the wind. Install engine inlet and exhaust covers and pitot mast covers when mooring the aircraft.

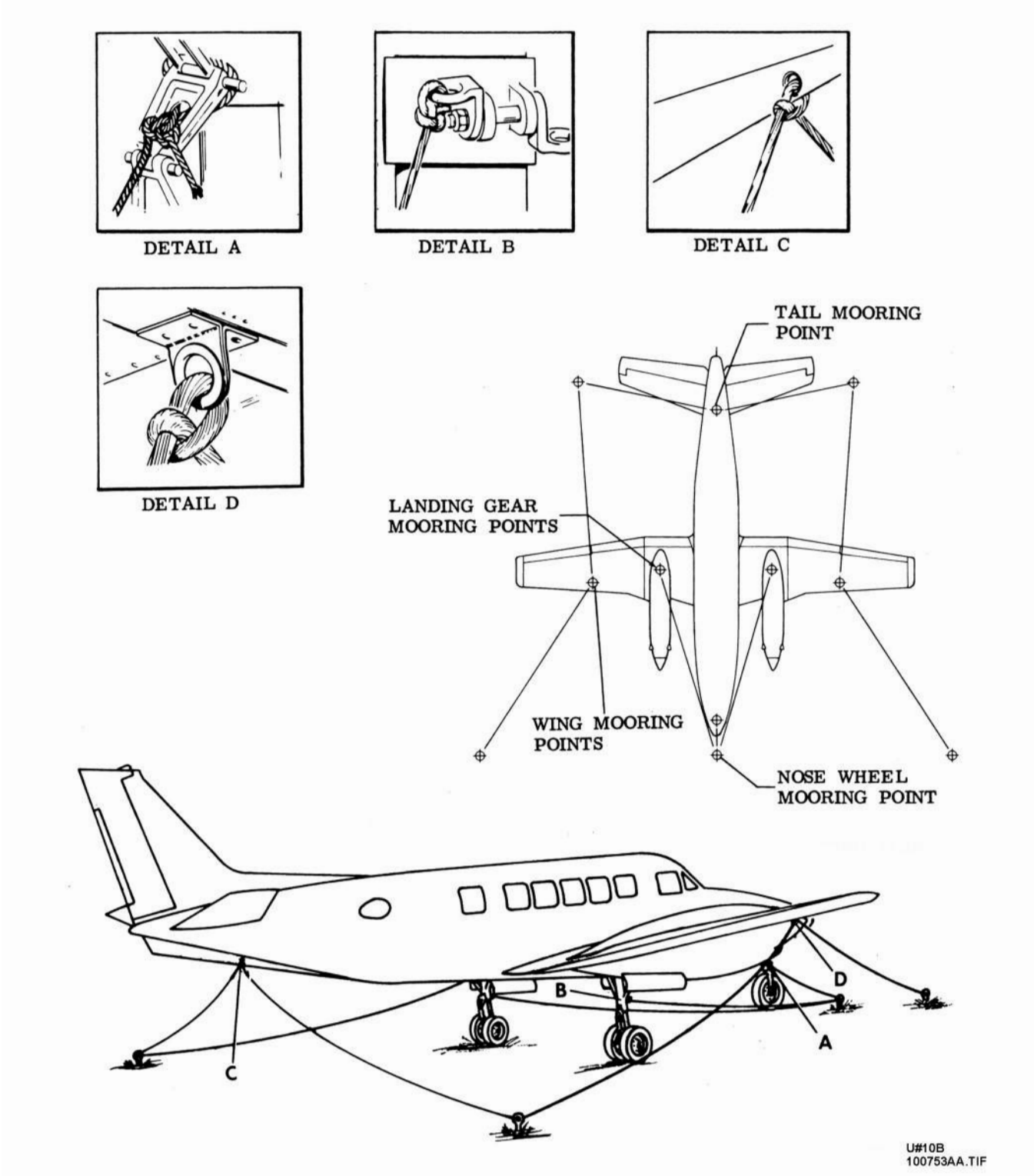

Figure 1
Anchoring and Mooring

PROTECTIVE COVERS FOR THE PROPELLERS

To prevent the propellers from windmilling and causing damage to the engine reduction gearing due to inadequate oil supply, damage to the propellers as a result of debris striking them, and as a safety precaution for personnel, it is recommended that the propellers be restrained by protective covers during prolonged storage of the aircraft outside of a hangar.

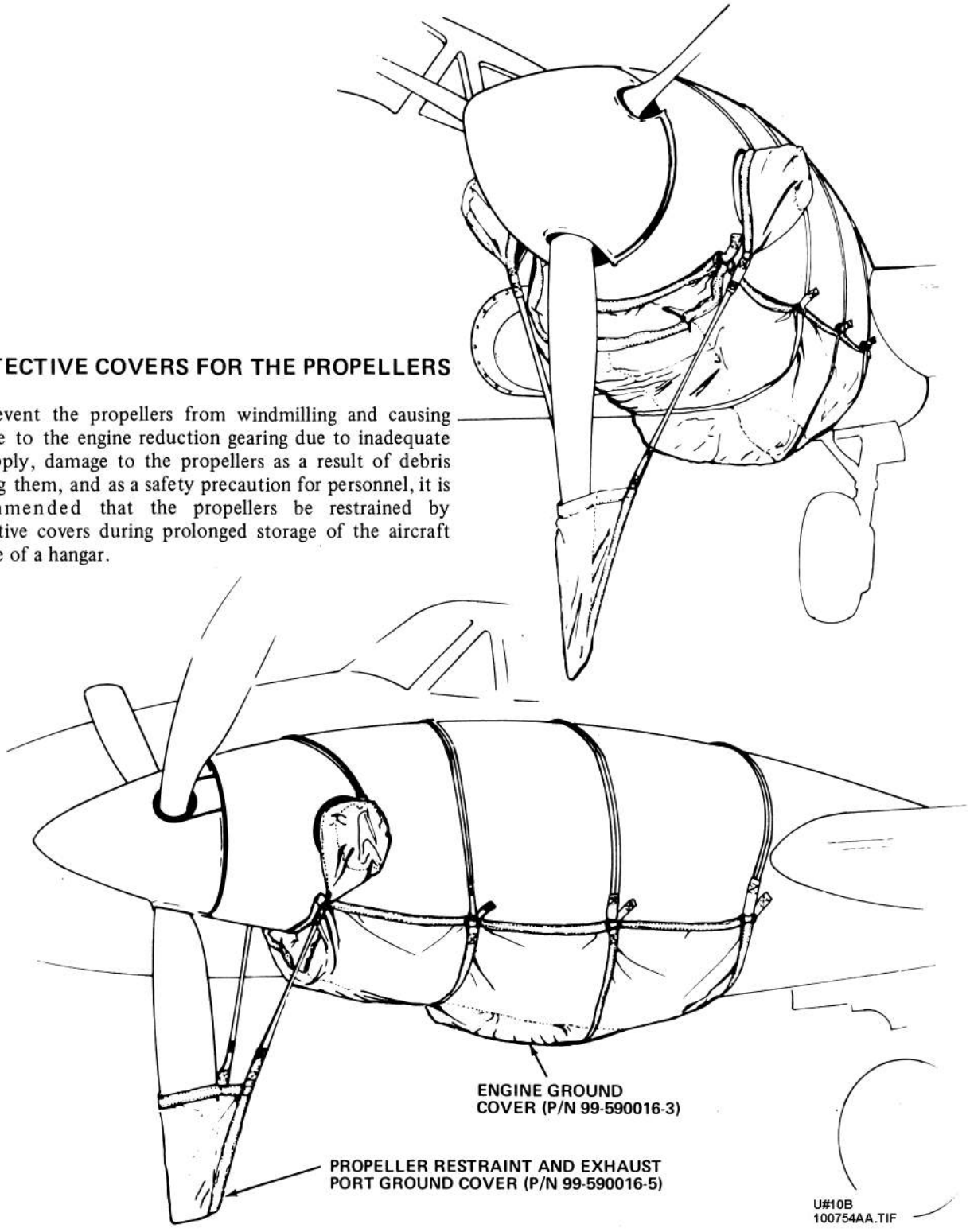

Figure 2
Protective Covers

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MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 11 - PLACARDS AND MARKINGS

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Model Designation Placard 1

EXTERIOR PLACARDS

11-20-00

Description and Operation 201
General 201

PLACARDS AND MARKINGS
GENERAL INFORMATION
DESCRIPTION AND OPERATION

1. GENERAL

A. Model Designation Placard

The model designation placard is located on the aft frame of the cabin entrance door. The door must be open to observe the placard. The placard identifies the airplane by its model number and serial number. Should a question arise concerning the care of the airplane, it is important to include the airplane serial number in any correspondence to Hawker Beechcraft Corporation.

All interior placards and markings are listed in the Limitations Section of the appropriate 99 Airliner Pilot's Operating Handbook.

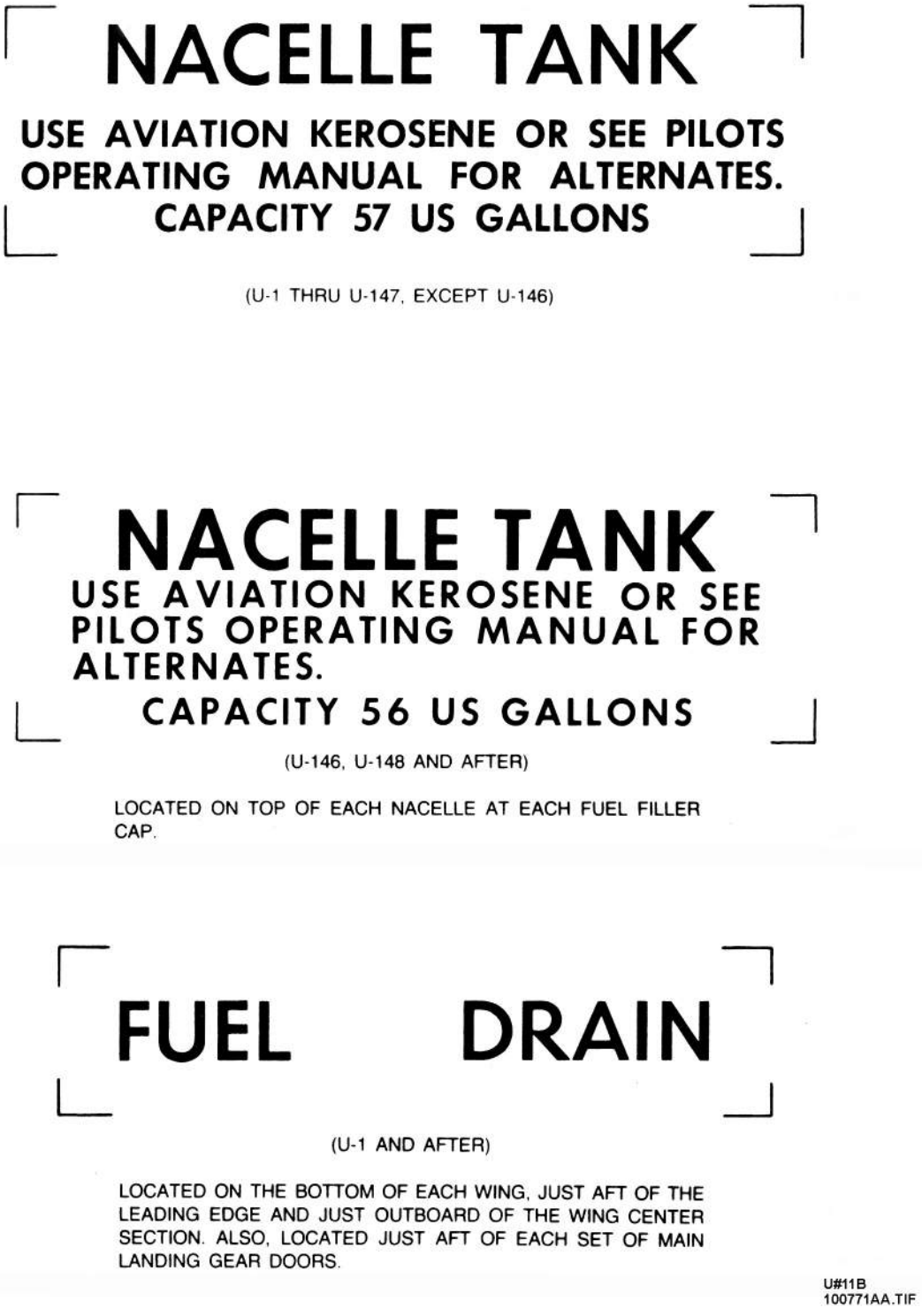
Exterior maintenance placard locations are shown in Chapter 11-20-00.

NOTE: Any time an airplane is repainted or touched up, inspect all placards to ensure that they are not covered with paint, are legible, and are securely attached.

PLACARDS AND MARKINGS
EXTERIOR PLACARDS
DESCRIPTION AND OPERATION

1. GENERAL

Shown in this chapter are exterior placards and markings which are essential for good maintenance practices, followed by location and description of each.



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Exterior Placards

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Exterior Placards

Figure 201 (Sheet 3 of 16)
Exterior Placards

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Exterior Placards

Figure 201 (Sheet 5 of 16)
Exterior Placards

(U-50, U-165 AND AFTER)

LOCATED ON THE BOTTOM OF THE LEFT WING BETWEEN THE FUSELAGE AND THE ENGINE NACELLE BENEATH THE FILL PLACARD ON THE TOP OF THE WING.

U-1 THRU U-164, EXCEPT U-50

LOCATED BETWEEN AND SLIGHTLY BELOW THE NOSE AVIONICS AND NOSE BAGGAGE COMPARTMENTS ON EACH SIDE OF THE FUSELAGE.

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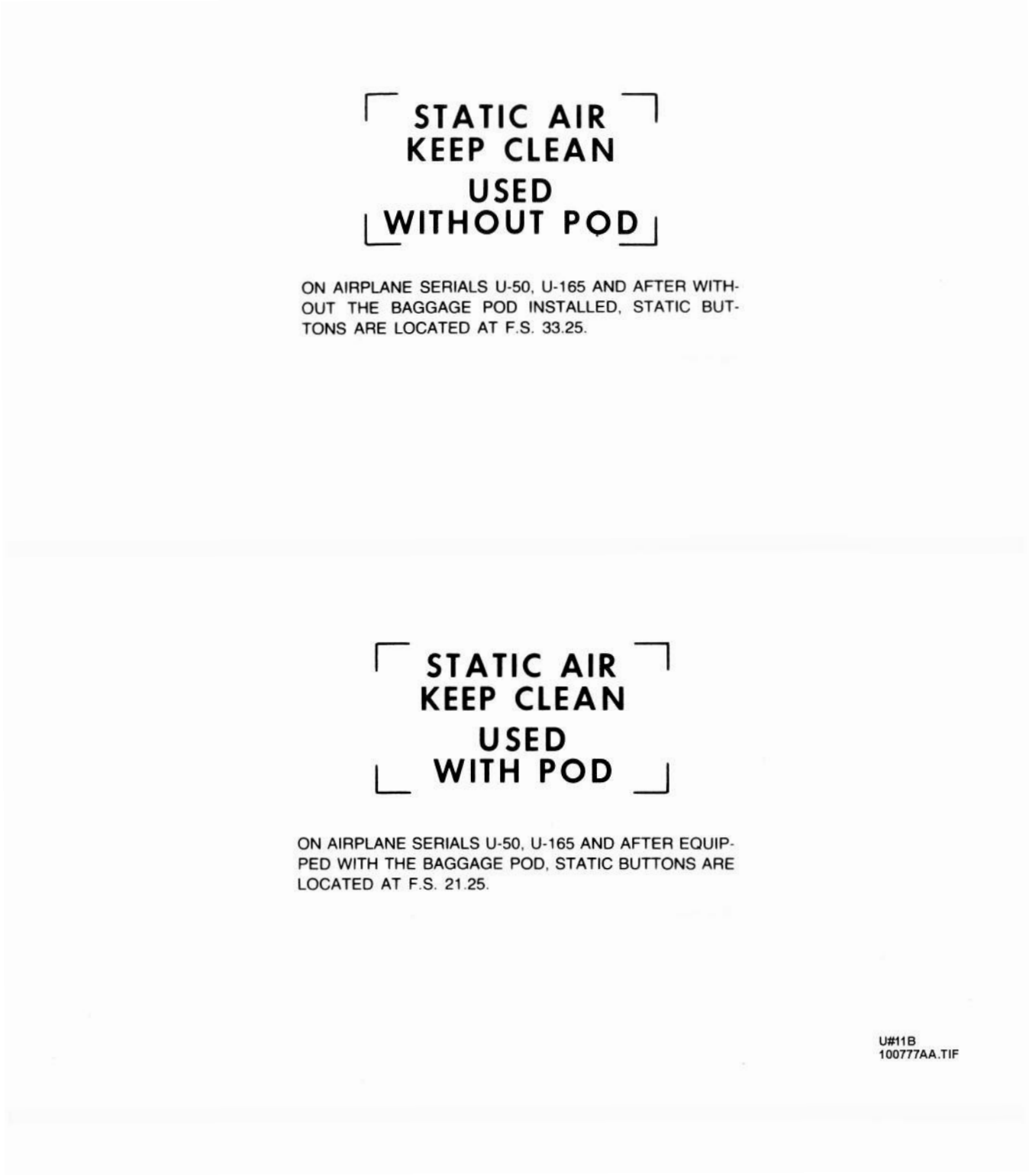
Figure 201 (Sheet 6 of 16)
Exterior Placards



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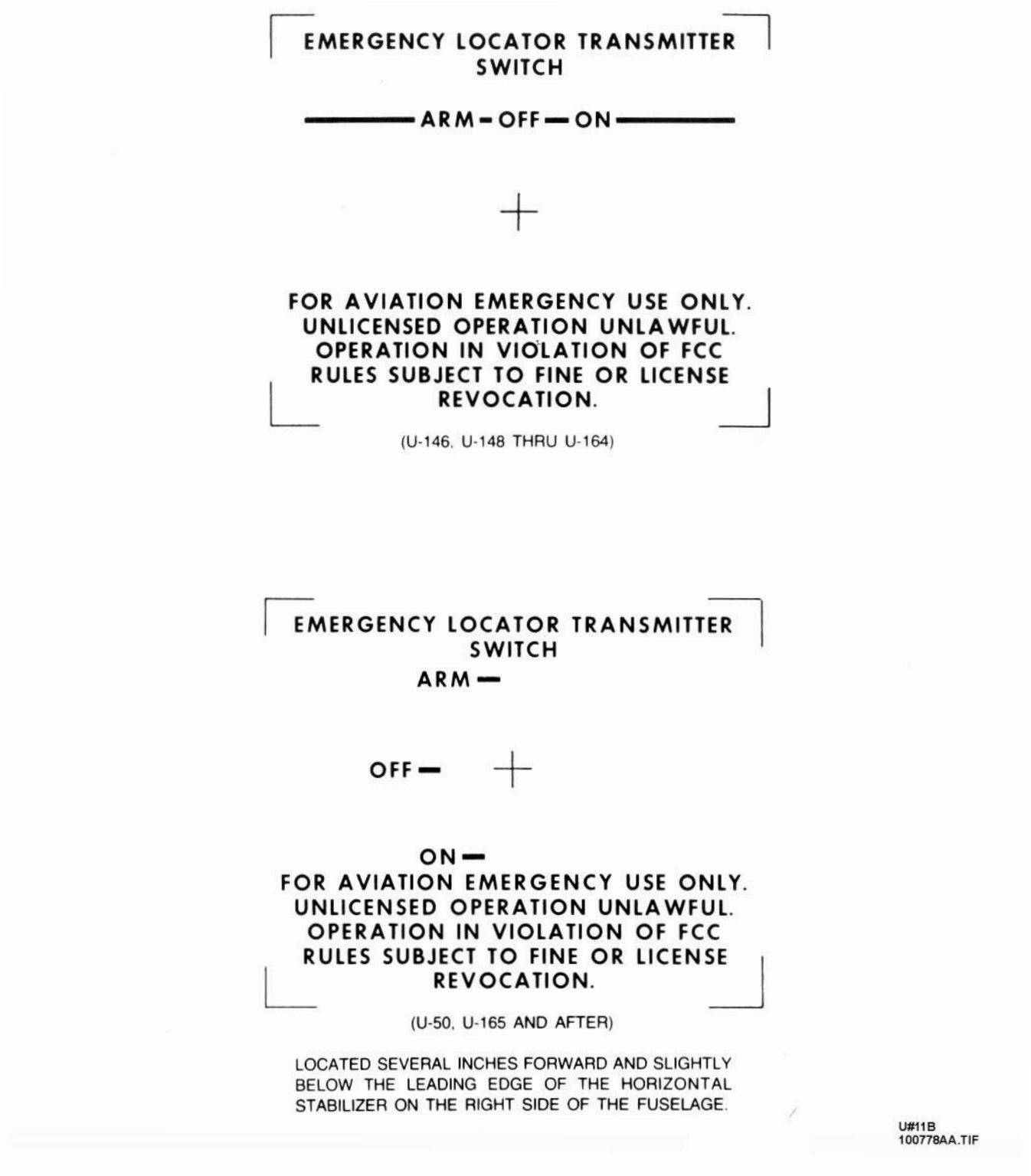
Exterior Placards



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Exterior Placards

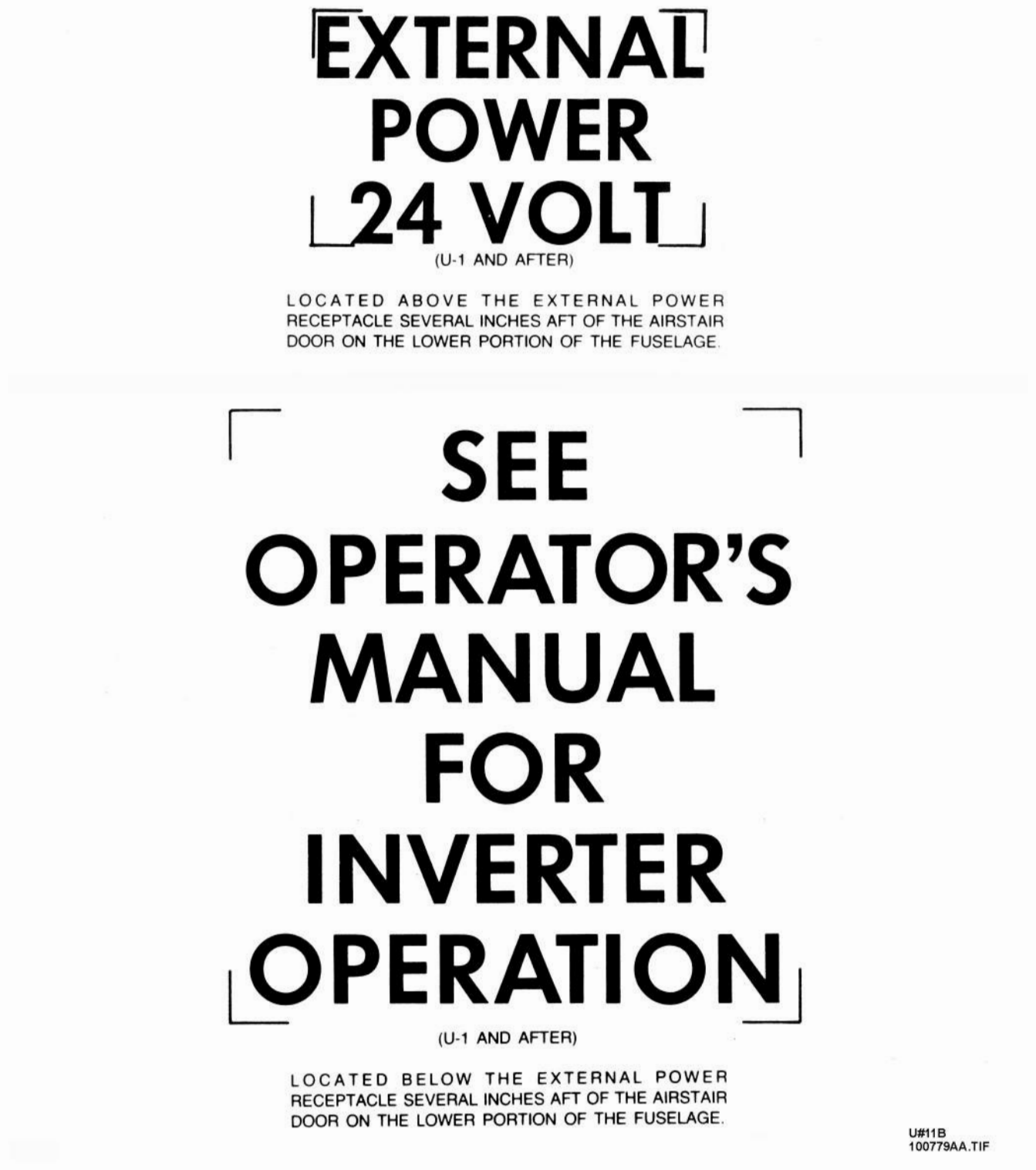
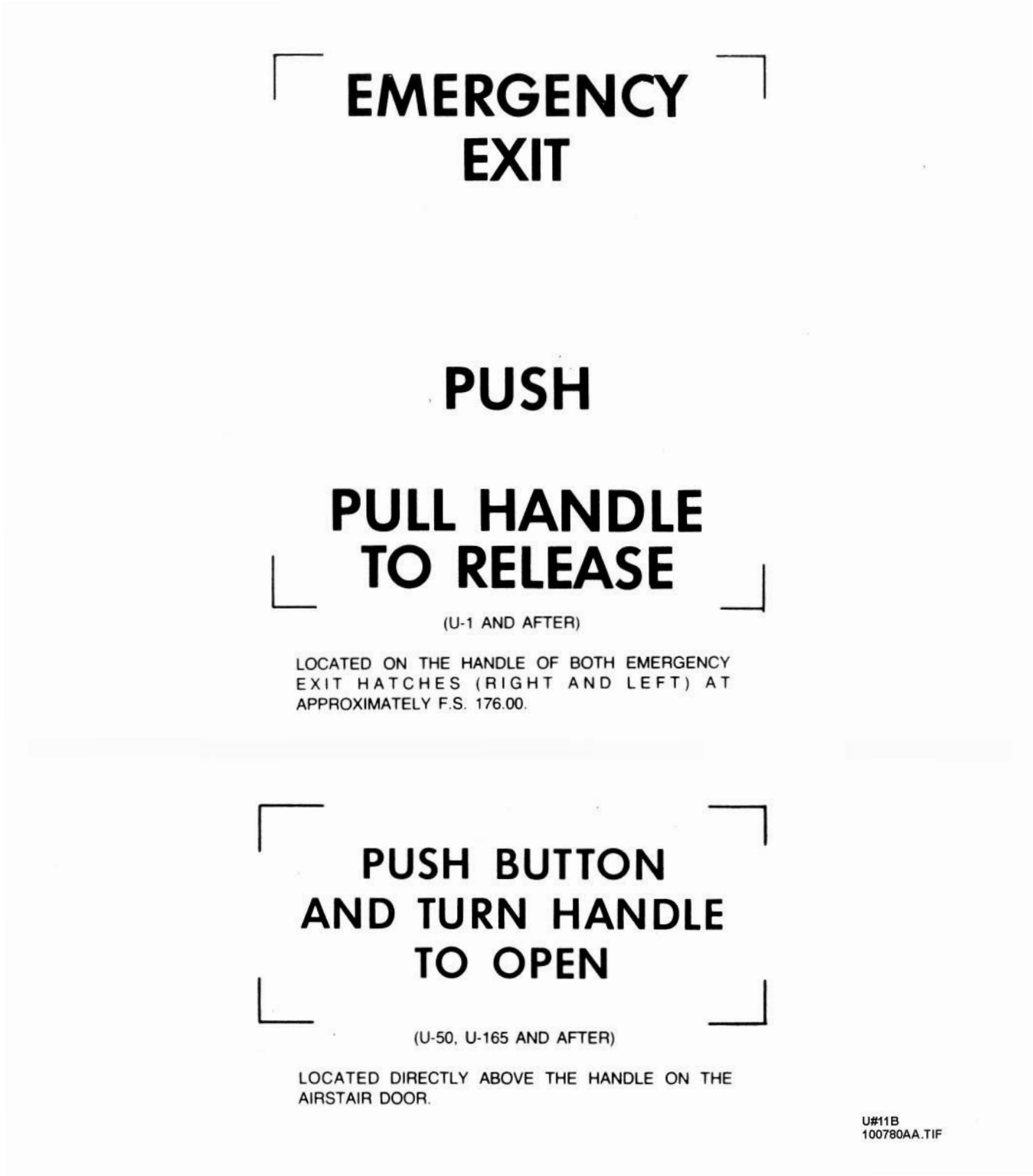


Figure 201 (Sheet 9 of 16)
Exterior Placards



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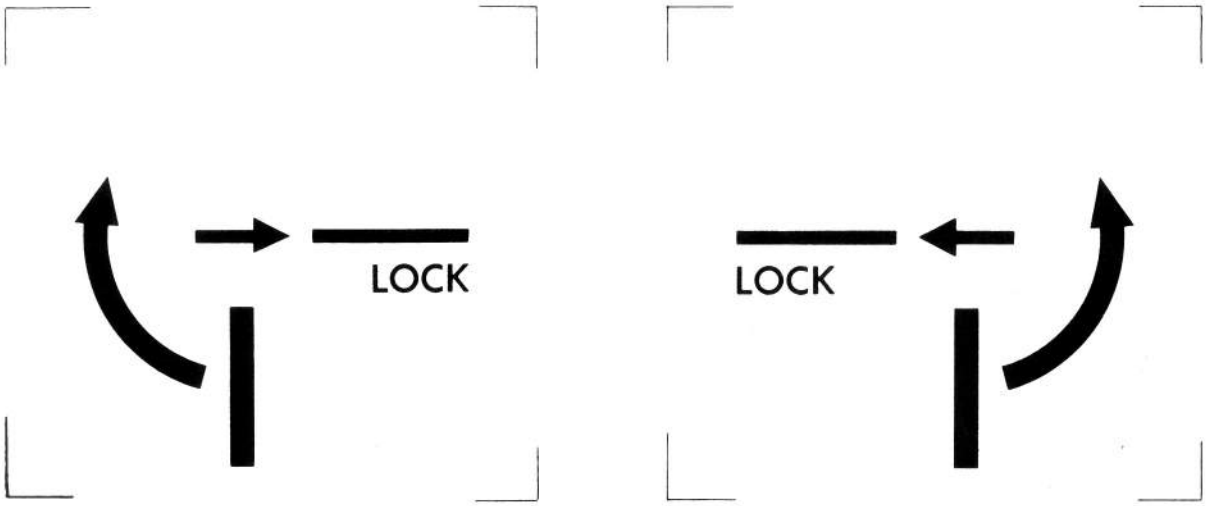
Exterior Placards



(U-1 AND AFTER)

THERE ARE A TOTAL OF FOUR PLACARDS FOR EACH ENGINE, TWO ON EACH SIDE OF THE ENGINE NACELLE AT THE POINT THE UPPER PORTION OF THE COWLING ATTACHES TO THE LOWER PORTION.

(U-1 AND AFTER)

LOCATED JUST FORWARD OF THE LEADING EDGE OF THE HORIZONTAL STABILIZER, ON THE FUSELAGE.

JACK PAD

(U-1 AND AFTER)

LOCATED ON THE UNDERSIDE OF EACH WING BETWEEN THE FUSELAGE AND THE NACELLE AND JUST AFT OF THE NOSE LANDING GEAR DOORS.

TIE DOWN

(U-1 AND AFTER)

LOCATED ON THE UNDERSIDE OF EACH WING AT APPROXIMATELY W.S. 184.00, AT THE TIE DOWN ON THE CENTERLINE OF THE MAIN WING SPAR. ALSO LOCATED ON EACH SIDE OF THE VENTRAL FIN AT THE FORWARD END OF THE FIN.

JACK

(U-1 AND AFTER)

LOCATED JUST AFT OF THE NOSE LANDING GEAR DOORS AT APPROXIMATELY F.S. 84.00.

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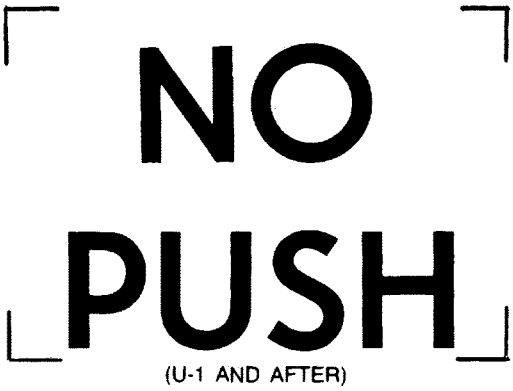
Figure 201 (Sheet 11 of 16)
Exterior Placards

(U-1 AND AFTER)

LOCATED AT F.S. 298.25 ON AIRPLANES WITHOUT THE CARGO DOOR INSTALLED. FOR AIRPLANES EQUIPPED WITH THE CARGO DOOR, THE PLACARD IS LOCATED AT F.S. 271.50. THERE ARE TWO PLACARDS, ONE DIRECTLY ABOVE THE OTHER, 31 INCHES APART.



(U-1 AND AFTER)

LOCATED ON THE NOSE OF THE AIRPLANE AND AT THE AFT END OF THE VENTRAL FIN ON EACH SIDE OF THE FIN.

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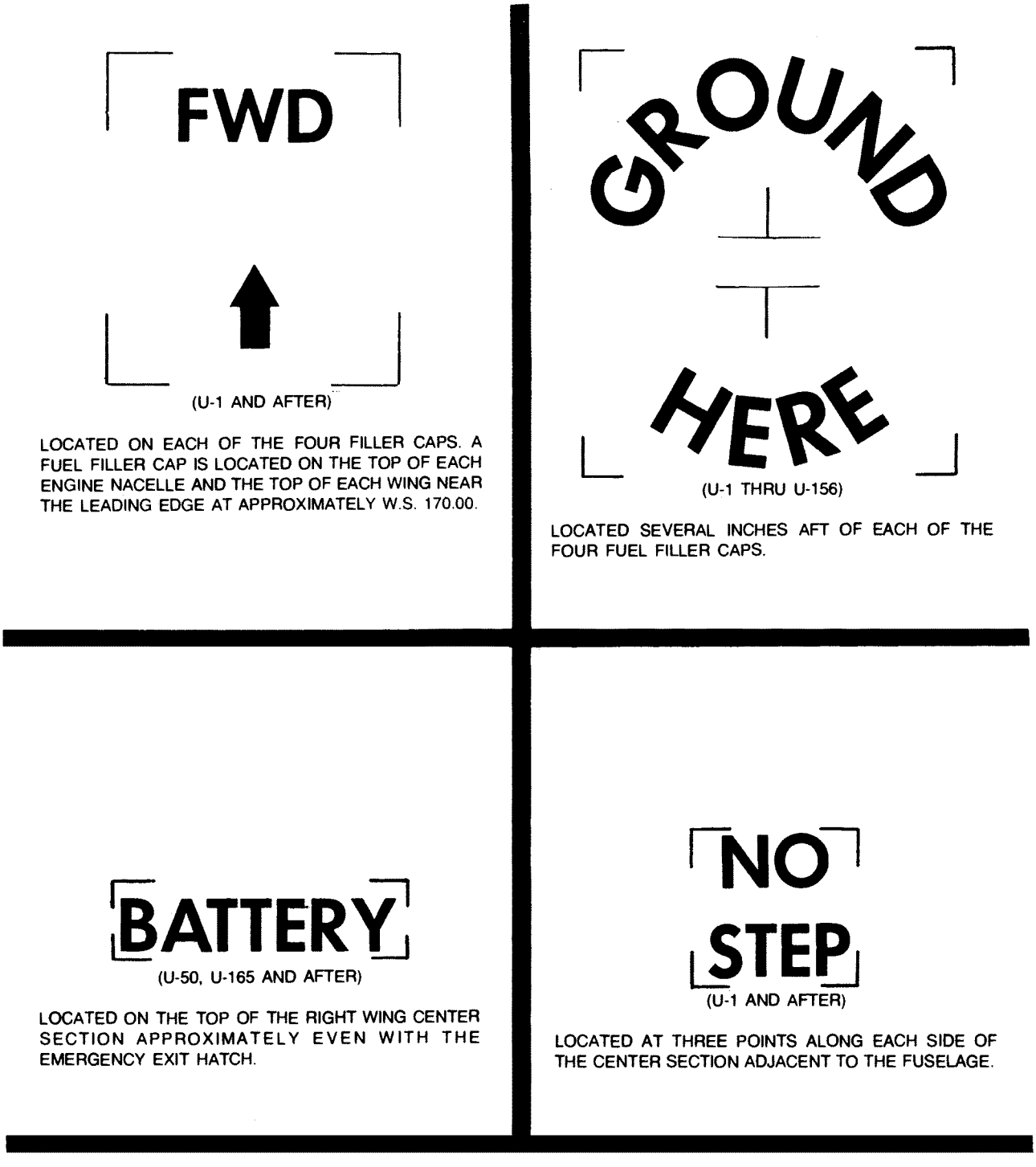
Exterior Placards



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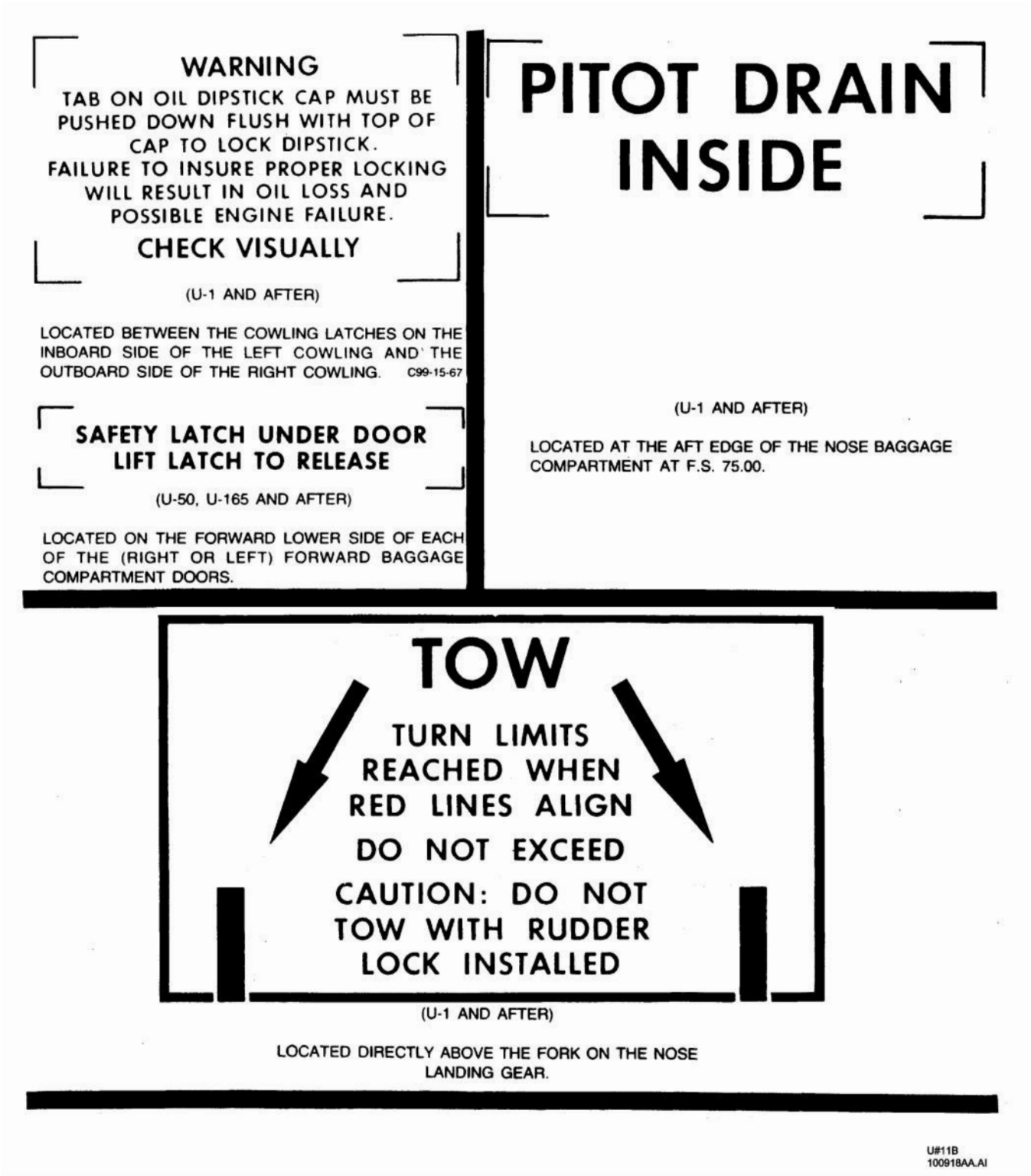
Exterior Placards



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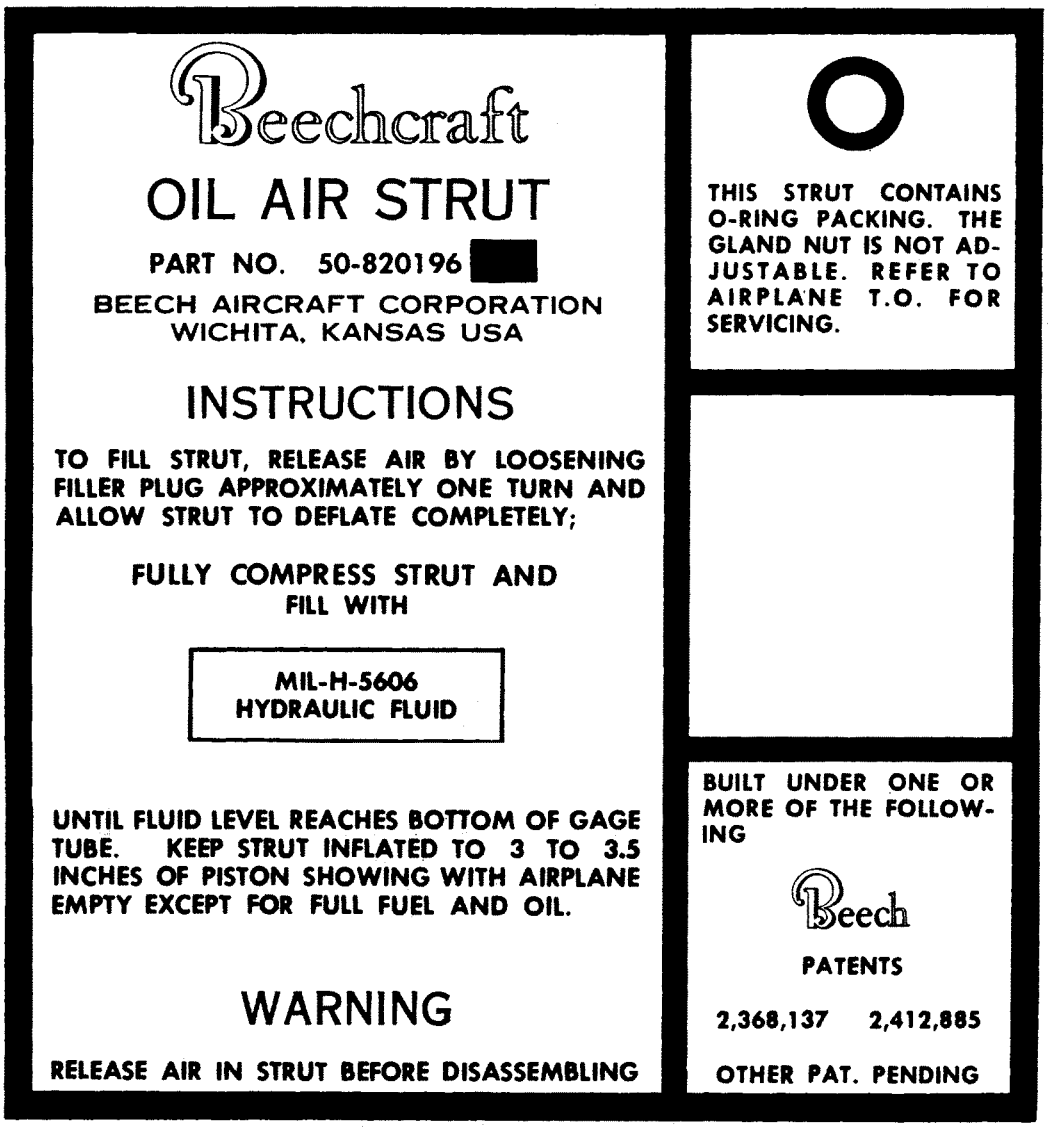
Exterior Placards



(U-1 AND AFTER)

LOCATED ON THE AFT SIDE OF THE NOSE LANDING GEAR STRUT.

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Figure 201 (Sheet 15 of 16)
Exterior Placards

Beechcraft®
OIL AIR STRUT
PART NO. 50-810317 [redacted]
SERIAL NO. [redacted]
BEECH AIRCRAFT CORPORATION
WICHITA, KANSAS, U.S.A.

INSTRUCTIONS
TO CHECK FLUID AND FILL
REMOVE VALVE CAP, DEPRESS VALVE CORE AND ALLOW STRUT TO FULLY COMPRESS. REMOVE VALVE BODY ASSEMBLY AND FILL WITH HYDRAULIC OIL CONFORMING TO INSTRUCTION MANUAL SPECIFICATIONS. SLOWLY CYCLE STRUT. REPEAT UNTIL ADDITIONAL OIL CAN NOT BE ADDED. REPLACE VALVE BODY.
WITH AIRPLANE EMPTY EXCEPT FOR FULL FUEL AND OIL KEEP STRUT INFLATED TO 4.25 TO 4.50 INCHES OF PISTON SHOWING.

WARNING
RELEASE AIR IN STRUT BEFORE DISASSEMBLING

(U-1 AND AFTER)

LOCATED ON THE AFT SIDE OF THE MAIN LANDING GEAR STRUT ON AIRPLANES WITH STANDARD MAIN LANDING GEAR.

Beechcraft®
OIL AIR STRUT
PART NO. 99-810027 [redacted]
SERIAL NO. [redacted]
BEECH AIRCRAFT CORPORATION
WICHITA, KANSAS, U.S.A.

INSTRUCTIONS
TO CHECK FLUID AND FILL
REMOVE VALVE CAP, DEPRESS VALVE CORE AND ALLOW STRUT TO FULLY COMPRESS. REMOVE VALVE BODY ASSEMBLY AND FILL WITH HYDRAULIC OIL CONFORMING TO INSTRUCTION MANUAL SPECIFICATIONS. SLOWLY CYCLE STRUT. REPEAT UNTIL ADDITIONAL OIL CAN NOT BE ADDED. REPLACE VALVE BODY.
WITH AIRPLANE EMPTY EXCEPT FOR FULL FUEL AND OIL KEEP STRUT INFLATED TO 4.50 TO 4.75 INCHES OF PISTON SHOWING.

WARNING
RELEASE AIR IN STRUT BEFORE DISASSEMBLING

(U-1 AND AFTER)

LOCATED ON THE AFT SIDE OF THE MAIN LANDING GEAR STRUT ON AIRPLANES WITH FLOTATION MAIN LANDING GEAR.

U#11B
100920AA.AI

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Exterior Placards

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 12 - SERVICING

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SERVICING
GENERAL INFORMATION
DESCRIPTION AND OPERATION

1. GENERAL

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gears are down and locked prior to removing the airplane from the jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

The information in the following chapters pertains to general maintenance procedures.

Detailed information on the engine or its systems will be found in the engine maintenance manual or in the chapter in this manual devoted to the system being serviced.

SERVICING
REPLENISHING
MAINTENANCE PRACTICES

1. FUEL SYSTEM

A. Fuel Handling Practices

All hydrocarbon fuels contain some dissolved and some suspended water. The quantity of water contained in the fuel depends on temperature and the type of fuel. Kerosene, with its higher aromatic content, tends to absorb and suspend more water than aviation gasoline. Along with the water, it will suspend rust, lint and other foreign materials longer. Given sufficient time, these suspended contaminants will settle to the bottom of the tank. However, the settling time for kerosene is five times that of aviation gasoline. Due to this fact, jet fuels require good fuel handling practices to assure that the Model 99 is serviced with clean fuel. If recommended ground procedures are carefully followed, solid contaminants will settle and free water can be reduced to 30 parts per million (ppm), a value that is currently accepted by the major airlines.

Since most suspended matter can be removed from the fuel by sufficient settling time and proper filtration, they are not a major problem. Dissolved water has been found to be the major fuel contamination problem. Its effects are multiplied in airplanes operating primarily in humid regions and warm climates.

Dissolved water cannot be filtered from the fuel by micronic-type filters, but can be released by lowering the fuel temperature, such as will occur in flight. For example, a kerosene fuel may contain 65 ppm (8 ounces per 1,000 gallons) of dissolved water at 80°F. When the fuel temperature is lowered to 15°F, only about 25 ppm will remain in solution. The difference of 40 ppm will have been released as supercooled water droplets which need only a minute shock to convert them to ice crystals. Tests indicate that these water droplets will not settle during flight and are pumped freely through the system. If they become ice crystals in the tank, they will not settle since the specific gravity of ice is approximately equal to that of kerosene. The 40 ppm of suspended water seems like a very small quantity but, when added to suspended water in the fuel at the time of delivery, is sufficient to ice a filter. While the critical fuel temperature range is from 0 to -20°F, which produces severe system icing, water droplets can freeze at any temperatures below 32°F.

Water in jet fuel also creates an environment favorable to the growth of a microbiological sludge in the settlement areas of the fuel cells. This sludge, plus other contaminants in the fuel, can cause corrosion of metal parts in the fuel system as well as clogging the fuel filters. Although the Beech 99 uses bladder-type fuel cells and all metal parts, except the main boost pumps and transfer pumps mounted above the settlement areas, the possibility of filter clogging and corrosion attacks on fuel pumps exist if contaminated fuels are consistently used.

Since fuel temperature and settling time affect total water content and foreign matter suspension, contamination can be minimized by keeping equipment clean, using adequate procedures, storing the fuel in the coolest areas possible, and allowing adequate settling time. Underground storage is recommended for fuel. Filtering the fuel each time it is transferred will minimize the quantity of suspended contaminants carried by the fuel.

The primary means of fuel contamination control by the owner/operator is Good Housekeeping. This applies not only to fuel supply, but to keeping the airplane system clean. The following is a list of Steps that may be taken to prevent contamination problems.

(1) Know your supplier. It is impractical to assume that fuel free from contaminants will always be available, but it is feasible to exercise caution and be watchful for the signs of fuel contamination.

(2) Assure, as much as possible, that the fuel obtained has been properly stored, filtered as it is pumped to the truck, and again as it is pumped from the truck to the airplane.

(3) Perform filter inspections to determine if sludge is present.

(4) Maintain good housekeeping by periodically flushing the fuel tankage system. The frequency of flushing will be determined by the climate and the presence of sludge.

(5) Use only clean fuel servicing equipment.

(6) After refueling, allow a three-hour settling period whenever possible, then drain a small amount of fuel from each drain.

CAUTION: *Remove spilled fuel immediately to prevent the surface thus contaminated from causing tire damage.*

B. Fuel Grades and Types

Jet A, Jet A-1, Jet B, JP-4 and JP-8 fuels may be mixed in any ratio. Aviation Gasoline, grades 90 (80/87),100LL (Blue) or 100 Green 100/130 and 115 (115/145) are alternate fuels and may be mixed in any ratio with the normal fuels when necessary. However, use of the lowest octane rating available is suggested due to its lower lead content. The use of aviation gasoline shall be limited to 150 hours of operation during each Time Between Overhaul (TBO) period.

Table 202 gives fuel refiner's brand name, along with the corresponding designations established by the American Petroleum Institute (API) and the American Society of Testing Material (ASTM). The brand names are listed for ready reference and are not specifically recommended by Hawker Beechcraft Corporation. Any product conforming to the recommended specification may be used.

2. JET FUEL ADDITIVES

Condensation of water in the fuel tanks increases the possibility of microbiological contamination of the fuel which can damage skins, coatings and sealants in the fuel tanks. Fuel additives, such as the anti-icing additives conforming to MIL-DTL-85470(B) (70, Table 6, 91-00-00) and a specific fuel biocide manufactured under the trade name Biobor JF (125, Table 6, 91-00-00), are compounds which contain antifungal and biocidal agents to reduce the possibility of contamination of the fuel, as well as clogging of fuel filters and lines. Either of these agents may be used separately or together in the fuel system with no detrimental effect on the fuel system components.

The following is a description of the two available fuel additive compounds which are effective biocidal and antifungal agents.

A. MIL-DTL-85470(B) Jet Fuel Anti-Icing Inhibitor

MIL-DTL-85470(B) Jet Fuel Anti-Icing Inhibitor (70, Table 6, 91-00-00) is primarily an anti-icing agent, but also has excellent microbiological-sludge-deterrent characteristics. It should be noted that anti-icing inhibitor does not alter the freeze point of the fuel; however, when dissolved water separates from the fuel during a drop in temperature, the additive quickly separates from the fuel to preferentially dissolve in the water, thereby depressing the freezing point of the water to prevent the formation of ice in the fuel. The additive must be precisely blended into the fuel by a metering device that permits injection of the agent into a flowing stream of fuel to ensure even dispersal. Fuel distributors may tank or batch blend, or may prefer to blend at the airplane when fueling. If the tanker truck is not equipped with the proper metering device, it may be necessary to carry the anti-icing agent and the blending device (HI FLOW Prist Blender Model PHF-204) or equivalent in the airplane. When blending the anti-icing agent with

fuel, the concentration of additive should not be less than 20 fluid ounces per 156 gallons of fuel or more than 20 fluid ounces per 104 gallons of fuel (at least 0.10% by volume but no more than 0.15 by volume). This additive should be used on a continuous basis.

B. Biobor JF

Biobor JF (125, Table 6, 91-00-00) is not an anti-icing agent, but is intended to be used specifically as a fuel biocide. The compound is an extremely efficient biocidal agent and is very soluble in fuel as well as water. Biobor JF disperses throughout the entire fuel system to even the most remote areas soon after introduction into the system. The compound is used as a periodic treatment in concentrations of 135 ppm when the airplane has been operated in an environment conducive to fungal or microbial contamination, or when such contamination is evidenced by dirty sump drains, clogged filters, odor or visual evidence in the tanks, etc.

Biobor JF may also be used in concentrations of 270 ppm as a single-dose shock treatment to clean out and sterilize a very contaminated system. Any system that is contaminated should be treated at the 270 ppm level. When sterility is achieved, the 135 ppm level may be used. Parked airplanes require only one treatment until fuel is burned off or replaced. Biobor JF is not volatile and may remain in the tank until the fuel is used.

The preferred method of introducing Biobor JF into the fuel is by injection through a metering device. If no metering device is available, blending may be accomplished by batch blending or by over-the-wing blending while filling the tanks. When half of the required quantity of fuel has been added, introduce the compound gradually, directly into the stream of fuel while adding the other half of the fuel. Complete mixing is necessary for fungicidal activity. Tank surface, gages, filters and linings should be inspected or replaced as necessary depending upon the severity of the contamination.

Biobor JF must be used at a high enough concentration to kill, not just control the infestation, and when used only periodically, it must be used often enough to prevent new growth. The mixture must be able to contact the entire surface of the fuel tank interiors and remain in contact up to 72 hours to effect the kill. Treated airplanes which are flown should be retreated at consecutive fills so that a 72 hour contact time is accomplished. Refer to Table 201 for fuel ratios for specified concentrations as an aid in blending. Be sure to account for residual fuel in the tanks so that the proper dosage is maintained.

C. Filling the Fuel Tanks

When filling the airplane fuel tanks, always observe the following:

(1) Make sure the airplane is statically grounded to the servicing unit and to the ramp.

CAUTION: *When inserting or removing the fuel nozzle, use extreme care to prevent the fuel hose from rubbing against the deicer boot. Also, do not allow fuel to contact the deicer boot.*

(2) Service the nacelle fuel tanks before filling the outboard fuel tanks. On the A99A, the nacelle tanks filler caps have been removed, thereby allowing single point refueling for each wing at the outboard edge.

(3) Allow a three-hour settling period whenever possible, then drain a small amount of fuel from each drain point.

Table 201
BIOBOR JF STERILIZATION AND MAINTENANCE TREATMENT LEVELS

TURBINE FUEL		BIOBOR JF AT 270 PPM			BIOBOR JF AT 135 PPM		
LBS.	GALS.	LBS.	GALS.	FL. OZS.	LBS.	GALS.	FL OZS.
670	100	0.18	0.02	2.63	0.09	0.01	1.32
1,340	200	0.36	0.04	5.26	0.18	0.02	2.63
2,010	300	0.54	0.06	7.89	0.27	0.03	3.95
2,680	400	0.72	0.08	10.53	0.36	0.04	5.26
3,350	500	0.90	0.10	13.16	0.45	0.05	6.58
6,700	1,000	1.18	0.21	26.46	0.90	0.10	13.16
13,400	2,000	3.62	0.41	52.92	1.81	0.21	26.46
16,750	2,500	4.52	0.52	66.08	2.26	0.26	33.04
33,500	5,000	9.01	1.03	132.16	4.52	0.52	66.08
67,000	10,000	18.09	2.07	264.47	9.05	1.03	132.31

To estimate the fluid ounces of Biobor JF required to give a concentration of 270 ppm multiply pounds of fuel by 0.004 and for 135 ppm by 0.002.

D. Draining the Fuel System (U-1 thru U-164, Except U-50)

The boost pumps may be utilized to drain the fuel system. To accomplish this, remove the cover from the underside of the LH center section adjacent to the inboard side of the nacelle to gain access to the tee connection used for draining. Connect a line to the tee and pump the fuel into a suitable tank or container. The crossfeed valve must be in the open position if the LH fuel system is to be drained.

A faster, alternate method of draining the fuel system with the boost pumps is to disconnect the flexible fuel heater line (under the engine cowling at the engine accessory case), connect a drain line to the flexible line and then pump the fuel overboard into a large, suitable container. By using this method and leaving the crossfeed valve closed, the fuel may be evacuated from only one wing, thereby not requiring complete draining of the entire system.

E. Draining the Fuel System (U-50, U-165 and After)

A defueling adapter, installed in the nacelle fuel cell and located aft of the standby pump, contains a check valve to prevent fuel drainage when the plug is removed. Each wing fuel system may be drained as follows:

(1) Remove the cover on the bottom of the nacelle to gain access to the adapter plug.

(2) Remove the plug and screw an AN832-1 union into the adapter. The fuel will begin draining as the check valve is unseated by the union.

(3) The fuel may be gravity drained or pumped out with the aid of a fuel truck.

NOTE: Remove the filler cap during draining to prevent damage or collapse of fuel cells.

(4) The quickest way to drain the wing center section is to turn the standby electric fuel pump on and use the airplane transfer system to move the fuel from the center section tank into the nacelle tank. The center section tank can also be drained by using the sump drain gravity flow into a suitable container. The wing center section tank need not be defueled in order to replace the standby boost pump.

F. Engine Fuel Filters and Screens

NOTE: All inline fuel filters and engine-driven pump filters should be inspected and cleaned every 100 hours as noted in the titled paragraphs following. Additional inspection and cleaning should be accomplished for the following conditions:

(1) At any time a suspected contaminated fuel is used.

(2) At any time the submerged fuel boost pumps are replaced. The sump tank area should also be properly cleaned at this time.

If abnormal quantities of contaminants are found, notify the local FAA General Aviation District Office so that corrective action may be taken regarding any other airplanes which may have received the same fuel.

3. CLEANING FILTERS

A. Firewall Filter

Clean the firewall filter every 100 hours as follows:

(1) Open access door on LH side of aft lower cowling to gain access to the firewall filter.

(2) Remove the drain hose from the firewall filter.

(3) Cut lockwire securing filter housing retaining nut and remove nut.

(4) Remove the filter housing from the filter body.

(5) Remove locknut and filter retaining nut and remove filter pack assembly (the packs need not be removed from the center tube).

(6) Inspect the filter pack for foreign material and microbiological sludge.

(7) Plug the open ends of the center tube and wash the unit in solvent (38, Table 6, 91-00-00) or its equivalent.

(8) Install the filter pack assembly; torque the retaining nut 10 to 15 inch-pounds.

(9) Install the locknut and torque the locknut 40 to 60 inch-pounds while holding the retaining nut in place.

(10) Install the filter housing and torque the retaining nut 80 to 120 inch-pounds and safety wire.

(11) Install the drain hose to the firewall filter.

B. Transfer Line Filter

Clean the transfer line filter every 100 hours as follows:

NOTE: The jet transfer system and filter are installed on the A99A. The jet transfer filter is not installed on the C99.

(1) Shut off the valve adjacent to the transfer line filter on the inboard side of each main landing gear wheel well.

(2) Disconnect drain line from transfer line filter.

(3) Cut the lockwire securing the filter housing retaining bolt and remove the bolt, filter housing, retaining spring and filter pack assembly.

(4) Inspect the filter pack assembly for foreign material and microbiological sludge. Do not remove the packs from the center tube.

(5) Plug the open ends of the center tube and wash the unit in solvent (38, Table 6, 91-00-00) or its equivalent.

(6) Install the filter pack assembly, retaining spring, housing and retaining bolt. Torque the retaining bolt 80 to 100 inch-pounds and safety wire.

(7) Connect the drain line to the transfer line filter. Open the valve in the transfer line after servicing the filter.

C. Fuel Pump Screen (Vickers Fuel Pump)

Clean as follows every 100 hours as follows:

(1) Cut the lockwire securing the fuel screen cap-nut to the fuel pump.

(2) Unscrew and remove the cap-nut and screen together, lifting the assembly vertically to avoid damaging the screen body.

(3) Disconnect the fuel inlet line at the pump end to drain off the residue of fuel.

(4) Blow all foreign matter from the screen with clean, dry compressed air.

(5) Install a new packing to the groove on the screen cap-nut.

(6) Install the screen and cap-nut assembly and tighten the cap-nut to a torque of 75 to 100 inch-pounds, then safety the cap-nut with lockwire.

(7) Connect the fuel inlet line to the pump, torque the coupling nut in accordance with the engine maintenance manual, and safety the nut with lockwire.

(8) Perform an engine run-up, then check the pump body for leaks after engine shutdown.

D. Fuel Grades and Types

NOTE: Jet A - aviation kerosene-type fuel with -40°F (-40°C) freeze point.

Jet A-1 - aviation kerosene-type fuel with -58°F (-50°C) freeze point.

Jet B - a low grade, kerosene-type fuel with a freeze point of -60°F (-51°C), similar to MIL-T-5624 grade JP-4, which has a freeze point of -76°F (-60°C).

Table 202
Fuel Brands and Type Designations

PRODUCT NAME	DESIG	PRODUCT NAME	DESIG
AMERICAN OIL		PURE OIL COMPANY	
American Jet Fuel Type A	Jet A	Purejet Turbine Fuel Type A	Jet A
American Jet Fuel Type A-1	Jet A-1	Purejet Turbine Fuel Type A-1	Jet A-1
ATLANTIC REFINING COMPANY		RICHFIELD PETROLEUM COMPANY	
Arcojet-A	Jet A	Richfield Turbine Fuel A	Jet A
Arcojet-A-1	Jet A-1	Richfield Turbine Fuel A-1	Jet A-1
Arcojet-B	Jet B		
BP TRADING COMPANY		SHELL OIL COMPANY	
BP A.T.K.	Jet A-1	Aeroshell Turbine Fuel 640	Jet A
BP A.T.G.	Jet B	Aeroshell Turbine Fuel 650	Jet A-1
		Aeroshell Turbine Fuel JP-4	Jet B
CALIFORNIA TEXAS COMPANY		SINCLAIR OIL COMPANY	
Caltex Jet A-1	Jet A-1	Sinclair Superjet Fuel	Jet A
Caltex Jet B	Jet B	Sinclair Superjet Fuel	Jet A-1
CITIES SERVICE COMPANY		STANDARD OIL OF CALIFORNIA	
Turbine Type A	Jet A	Chevron TF-1	Jet A-1
		Chevron JP-4	Jet B
CONTINENTAL OIL COMPANY		STANDARD OIL OF KENTUCKY	
Conoco Jet-40	Jet A	Standard JF A	Jet A
Conoco Jet-50	Jet A	Standard JF A-1	Jet A-1
Conoco Jet-60	Jet A-1	Standard JF B	Jet B
Conoco Jet JP-4	Jet B		
GULF OIL COMPANY		STANDARD OIL OF OHIO	
Gulf Jet A	Jet A	Jet A Kerosene	Jet A
Gulf Jet A-1	Jet A-1	Jet A-1 Kerosene	Jet A-1
Gulf Jet B	Jet B		
EXXON OIL COMPANY		TEXACO	
Exxon Turbo Fuel A	Jet A	Texaco Avjet K-40	Jet A
Exxon Turbo Fuel 1-A	Jet A-1	Texaco Avjet K-58	Jet A-1
Exxon Turbo Fuel 4	Jet B	Texaco Avjet JP-4	Jet B
MOBIL OIL COMPANY		UNION OIL COMPANY	
Mobil Jet A	Jet A	76 Turbine Fuel	Jet A-1
Mobil Jet A-1	Jet A-1	Union JP-4	Jet B
Mobil Jet B	Jet B		

Table 202
Fuel Brands and Type Designations (Continued)

PRODUCT NAME	DESIG	PRODUCT NAME	DESIG
PHILLIPS PETROLEUM COMPANY			
Philjet A-50	Jet A		
Philjet JP-4	Jet B		

E. Cleaning Pesco Fuel Pump Filter (Every 100 Hours)

(1) Cut the lockwire and remove the attaching bolts (1) and washers (2) from the filter cover (3) on the bottom of the fuel pump (Ref. Figure 201).

(2) Insert a No. 10 (0.190) 32UNF3B screw in the filter removal hole in the cover subassembly, and pull firmly on the screw to remove the filter (4) from the pump housing.

CAUTION: *Never pry the cover off with a sharp instrument for it may damage the pump housing or cover.*

(3) Separate the spring (5) and filter (4) from the cover (3) by removing the retaining bolt (6) and washer (7).

(4) Remove the packings (8 and 9) from the cover.

(5) Clean the filter with solvent (38, Table 6, 91-00-00) and blow dry with filtered, compressed air.

(6) Install the packings (8 and 9) on the filter cover (3).

(7) Secure the filter (4) and spring (5) to the filter cover (3) with the retaining bolt (6) and washer (7). Torque the retaining bolt 25 to 30 inch-pounds.

(8) Install the assembled filter and cover in the pump housing and torque the attaching bolts (1) to 40 to 46 inch-pounds. Safety the cover bolts with lockwire.

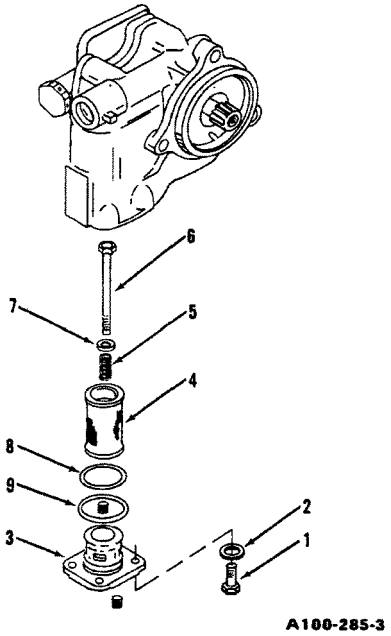


Figure 201
Pesco Fuel Pump

F. Cleaning Pesco Fuel Pump 10-Micron Filter

Clean as follows at intervals of 100 operating hours:

(1) Unscrew and remove filter housing from fuel pump (Ref. Figure 202).

(2) Remove filter element and discard.

(3) Before installing new filter element, unscrew inlet screen cover and withdraw assembly.

(4) Remove long bolt, spring, washers and separate screen from cover. Wash the screen with solvent and replace in reverse order of disassembly.

(5) Install two new packings on screen cover and assemble screen and cover assembly to pump body.

(6) Install the packings (8 and 9) on the filter cover (3).

(7) Tighten cover to compress the packing and obtain metal-to-metal contact and then lockwire.

(8) After cleaning and reinstalling screen, install a new fuel pump outlet filter. (Check for pressure or preformed packing in internal diameter. The element must be installed so that the preformed packing slides over the spigot in the pump body).

(9) Replace preformed packing on filter housing.

(10) Install filter housing and tighten to compress preformed packing to obtain metal-to-metal contact.

(11) Lockwire square section of housing to hole on nameplate boss. Check for leaks after engine ground checks.

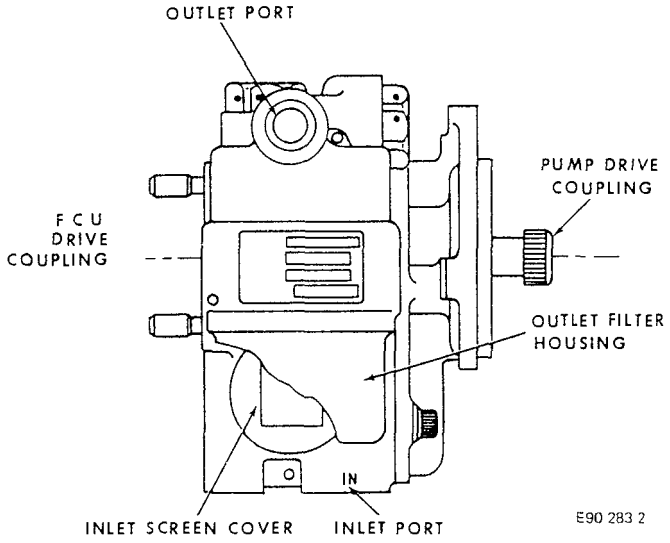


Figure 202
Pesco Fuel Pump 10 Micron Filter

G. Servicing Filter of the Fuel Drain Collector Pump

Clean the filter of the fuel pump as follows at 100 hour intervals:

(1) Remove the fuel pump from the airplane (Ref. FUEL DRAIN COLLECTOR PUMP REMOVAL, Chapter 71-70-00).

CAUTION: *Do not tamper with the seal in the center of the mounting bracket attached to the bottom of the pump. This seal retains the dry gas that surrounds the electrical components in the upper part of the pump.*

(2) Release the bottom cover from the bayonet fittings of the pump with a 5/8 inch wrench, then twist the cover off the pump by hand.

(3) Carefully remove the filter from the open end of the pump.

(4) Wash the filter in cleaning solvent (38, Table 6, 91-00-00) and blow out dirt and cleaning solvent with air pressure.

(5) Replace the cover gasket if deteriorated. Clean the cover and blow the magnet in the center of the cover clean.

(6) Place the cover gasket and magnet in the bottom of the cover, then place the filter screen around the magnet in the bottom of the cover. The screen must fit snugly at both ends.

(7) Taking care to avoid pinching or distorting the filter screen, insert the filter into the pump and twist the cover by hand to hold it in position on the pump housing. Tighten the cover securely in place with a 5/8 inch wrench.

(8) Install the fuel pump in the nacelle (Ref. FUEL DRAIN COLLECTOR PUMP INSTALLATION, Chapter 71-70-00).

4. SPARK IGNITERS

WARNING: If a spark igniter is to be removed, ground the lead on the engine for at least six minutes to dissipate the dangerous charge that has accumulated in the capacitor of the ignition exciter.

To check the spark igniters for proper operation, disconnect one lead at the ignition exciter and listen for the discharge of the spark igniter that is still connected, then repeat the procedure for the other spark igniter. The igniters will spark at a rate of 0.8 to 1.0 spark per second with 9 volts applied up to a rate of 1.4 to 4.0 sparks per second with 30 volts applied. Additional information as to the erosion limits of the spark igniters will be incorporated at a later date.

5. FUEL CELL PRESERVATION

CAUTION: *Never store fuel cells in the vicinity of electrical equipment, such as generators and motors. The movement of brushes across the commutators of these units results in minute sparks that cause the formation of ozone gas. Ozone has a highly destructive effect on fuel cells. Although it leaves no visible indication, ozone makes the material of which the cells are formed brittle so that the cells will disintegrate upon the application of stress.*

NOTE: Preservation is not necessary for in service fuel cells when the cell is to be empty for an indefinite period. The fuel cell should not be open to the atmosphere except for the normal vent lines in the airplane.

If AVGAS was in the fuel cell prior to emptying, purge the cell with aviation kerosene prior to leaving empty for the indefinite period.

For information pertaining to the storage of Uniroyal fuel cells, refer to Report FC 1473-73 entitled Recommended Handling and Storage Procedures for Bladder Type Fuel and Oil Cells, which is published by Uniroyal Inc., Engineered Systems Department, Mishawaka, Indiana.

Goodyear fuel cells are constructed of polyethylene; they need not be treated for storage. The only stipulation on storage of Goodyear cells is that they be carefully wrapped for protection against dust, then be stored at room temperatures in a location shielded from sunlight.

6. OIL SYSTEM

Servicing the engine oil system primarily involves maintaining the engine oil at the proper level, inspecting the oil filter at 100 hour intervals, and cleaning or changing the filter as outlined in the applicable Pratt & Whitney Aircraft of Canada LTD Maintenance Manual. The engine oil should be changed every 400 hours or 9 months (whichever occurs first) on airplanes operated 50 hours a month or less.

On high utility type of operation, change the engine oil every 800 hours. The time for engine oil change may be extended to 1200 hours if 5-centistoke oils are used, see the List of Consumable Materials (1, Table 6, 91-00-00). It is recommended that 5-centistoke oils be used in airplanes normally operated in climates with ambient temperatures at or below 0°F (-18°C).

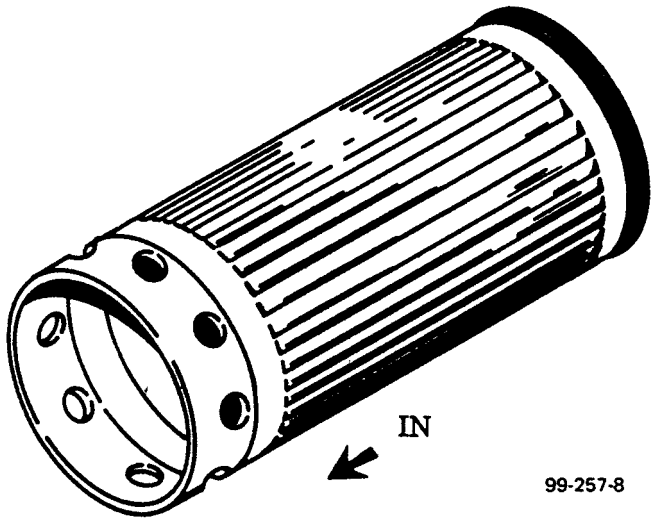


Figure 203
Installing Engine Oil Filter

NOTE: There is a difference of one quart between a hot engine and a cold engine. On the dipstick, the 1 mark for a hot engine is the full mark for a cold engine. The oil should be checked daily within 10 minutes after engine shutdown.

CAUTION: *Do not mix different brands, viscosities or types of oil since their different chemical structures may make them incompatible. If different types of oil become mixed, drain and flush the system and refill with new oil.*

The oil tank is provided with an oil filler neck and quantity dipstick cap which protrude through the accessory gearcase at the eleven o'clock position. The dipstick is marked in U.S. quarts and indicates the amount of oil required to fill the tank. Access to the dipstick cap is gained by opening the aft engine cowl. Service the oil system with oil as specified in the Consumable Materials Chart (1, Table 6, 91-00-00). Do not mix the oil brands or viscosities. The total oil tank capacity is 2.3 U.S. Gallons (9.2 quarts) of which only 1.5 gallons (6 quarts) is usable. An additional amount of approximately 5 quarts of oil is required to fill the lines and cooler, giving a total system capacity of 14 quarts; however, because of the residual oil trapped in the system, no more than 12 quarts should be added during an oil change.

CAUTION: *Remove spilled oil immediately to prevent the surface thus contaminated from causing tire damage.*

The engine oil filter is located under the square cover plate at the three o'clock position of the compressor inlet case and just behind the aft fire seal.

(1) Remove the four self-locking nuts and plain washers securing the filter cover to the compressor inlet case and remove the cover.

(2) Remove the element from the filter housing. Inspect the filter element for the presence of foreign particles.

(3) Insert a new filter element in the filter housing.

(4) Coat a new packing seal with engine oil and install the seal and cover on the engine.

(5) Secure the filter cover with four plain washers and self-locking nuts. Torque the nuts to between 20 and 30 inch-pounds above the torque necessary to turn the nuts before seating.

A. Oil Filter Cleaning

Some filter elements may be cleaned and reused. Some of these filters may be reused as long as the filter is found to be serviceable upon inspection. Refer to the applicable Pratt & Whitney Aircraft of Canada Maintenance Manual or Service Bulletin for the cleaning procedures and limitations of these filters.

B. Changing the Engine Oil

CAUTION: *When changing to a different brand of oil, completely drain the airplane oil system as indicated in the following procedure. Remove the oil filter and immerse it in the brand of oil to be used. Reinstall the oil filter and drain plugs. Fill the system to the proper level, and ground run the engines for 20 minutes to thoroughly circulate the new brand of oil throughout the system. Completely drain the airplane oil system and again remove the oil filter and immerse it in the new brand of oil. Refill the airplane oil system as indicated in the following procedure. This will thoroughly purge the system of the old oil to prevent chemical interaction between it and the new brand.*

(1) To gain access to the oil drain plug, remove the fiberglass duct from around the oil cooler and remove the metal bypass duct immediately aft of the oil cooler.

(2) Unsafety and remove the drain plug from the oil cooler and drain the oil into a container.

(3) Remove the safety wire from the oil drain cap.

(4) Position the oil drain funnel under the oil plug.

(5) Remove the oil drain cap. Allow all oil to drain from the engine.

(6) Remove the forward engine cowling and unsafety and remove the chip detector from the nose case. Refer to Chapter 71 for removal of the lower forward cowling.

(7) Remove the drain plug from the 6 O'clock position on the rear face of the accessory gearbox housing.

(8) With all the drain plugs removed, motor the engine over with the starter only (no ignition) to permit the scavenge pumps to clear the engine.

CAUTION: *Limit motoring to the time required to accomplish the above because of limited lubrication available to the engine during this operation and starter motor operating time limits. To prevent damage to the fuel control unit, leave the condition lever in IDLE CUTOFF while motoring the engine.*

(9) Install the cleaned or new oil filter element as described under the heading OIL FILTER CHANGING.

NOTE: Discard all old packings and replace with new packings.

(10) Install and safety wire the engine oil drain cap.

(11) Install and safety the chip detector. Torque 45 to 55 inch-pounds.

(12) Install the forward cowlings.

(13) Install and safety the oil cooler drain plug.

CAUTION: *Damage to the threads will result if the oil cooler drain plug is tightened to a torque exceeding 15 to 20 inch-pounds. Apply MIL-P-16232, Type A, Class 2, anti-seize compound to the drain prior to reinstallation.*

(14) Install the accessory gearbox drain plug. Torque 215 to 240 inch-pounds and safety.

(15) Fill the engine with the correct amount and type of oil as specified in (1, Table 6, 91-00-00).

(16) Motor the engine over, with the starter only, long enough to get an oil pressure reading.

CAUTION: *Do not exceed the starter motor operating time limits of 40 seconds ON, 60 seconds OFF, 40 seconds ON, 60 seconds OFF, 40 seconds ON, then 30 minutes OFF.*

(17) Check the engine for oil leaks.

(18) Refill the engine to the proper level.

(19) Install the metal bypass duct immediately aft of the oil cooler with the retaining screws and reinstall the fiberglass duct around the oil cooler on the lower cowl.

C. Magnetic Drain Plug Inspection

A magnetic drain plug (chip detector) is installed in each engine. The plug attracts ferrous particles which may be present in the sump area. Inspect the magnetic drain plug every 100 hours as follows:

(1) Remove the electrical connection.

(2) Remove the magnetic drain plug.

(3) Visually check the plug for metal particles.

(4) Check the plug for cracks or damaged threads.

(5) Check the electrical connector for a bent, broken or burned pin.

(6) Check the resistance from pin to ground. The magnetic plug shall not indicate continuity to ground.

(7) Install new packing on magnetic plug.

(8) Install the magnetic plug and torque 45 to 55 inch-pounds.

(9) Install the electrical connection finger tight.

(10) Remove the scavenge oil filter screen from the lower case of the reduction gearbox. Some oil will be lost from the gearbox.

(11) Fabricate the tool shown in Figure 204.

(12) Apply electrical power to the airplane.

(13) Insert the curved end of the fabricated tool through the scavenge oil filter screen hole and position the lip in the gap of the chip detector.

(14) Check to see if the chip detector light in the cockpit is illuminated. If not, repeat the above procedures.

(15) Remove the fabricated tool and check to ensure the light does not stay illuminated.

(16) Install the scavenge oil screen as instructed in the engine maintenance manual.

(17) Safety wire the chip detector.

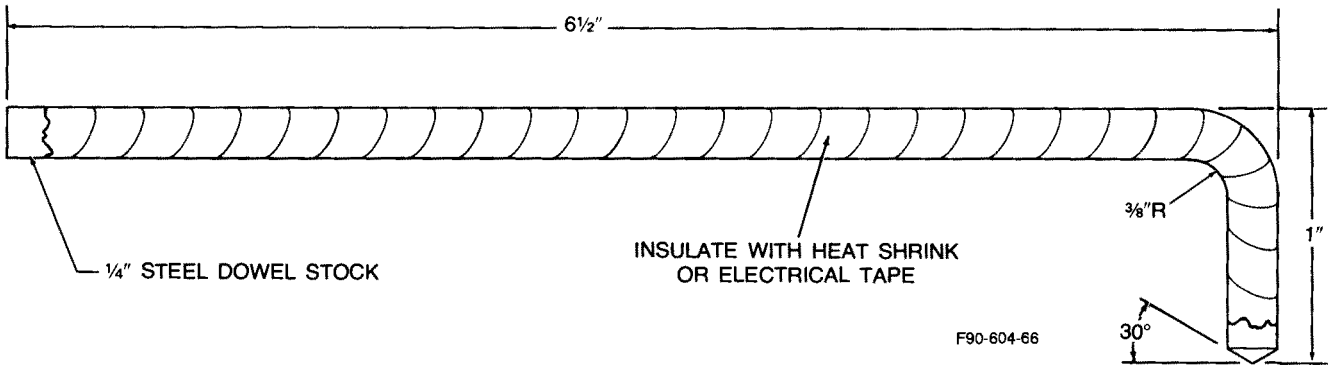


Figure 204
Magnetic Chip Detector Check Tool

7. BRAKE SYSTEM

Brake system servicing is limited primarily to maintaining the hydraulic fluid level in the reservoir mounted in the upper LH comer on the aft bulkhead of the nose baggage compartment. A dip stick is provided for measuring the fluid level. When the reservoir is low on fluid, add a sufficient quantity of hydraulic fluid (9, Table 6, 91-00-00) to fill the reservoir to the full mark on the dipstick. The only other requirements related to servicing involves the wheel brakes themselves. Although each brake is equipped with an automatic adjuster that eliminates the need for periodic adjustment of brake clearance, the brake must be checked periodically for indication of excessive wear. For detailed information relating to the proper inspection and repair procedures, refer to the Overhaul Manual, P/N 99-590015-3 for airplane serials U-1 thru U-164, except U-50. For airplane serials U-50, U-165 and After, refer to the C99 Component Maintenance Manual, P/N 99-590015-5.

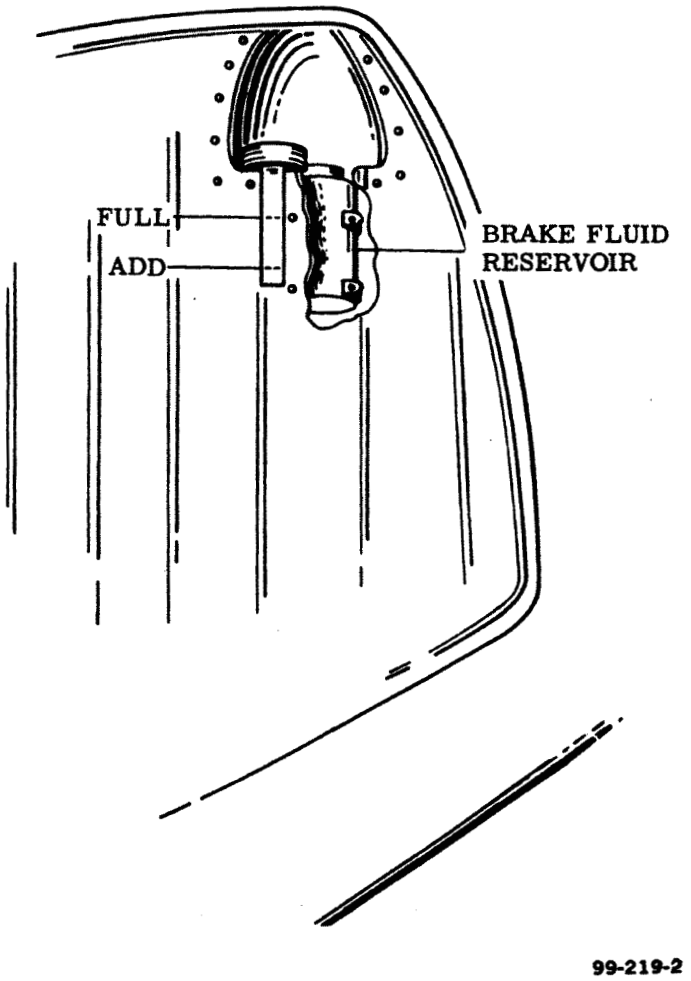


Figure 205
Brake Fluid Reservoir

8. CABIN ENTRANCE DOOR DAMPER (HYDRAULIC SNUBBER)

Brake system servicing is limited primarily to maintaining the hydraulic fluid level in the reservoir mounted in the upper LH comer on the aft bulkhead of the nose baggage compartment. A dip stick is provided for measuring the fluid level. When the reservoir is low on fluid, add a sufficient quantity of hydraulic fluid (9, Table 6, 91-00-00) to fill the reservoir to the full mark on the dipstick. The only other requirements related to servicing involves the wheel brakes themselves. Although each brake is equipped with an automatic adjuster that eliminates the need for periodic adjustment of brake clearance, the brake must be checked periodically for indication of excessive wear. For detailed information relating to the proper inspection and repair procedures, refer to the Overhaul Manual, P/N 99-590015-3 for airplane serials U-1 thru U-164, except U-50. For airplane serials U-50, U-165 and After, refer to the C99 Component Maintenance Manual, P/N 99-590015-5.

Observe the following precautions when installing or servicing the door snubber:

(1) Attach the barrel rod end to the door frame supports and the movable piston rod end to the cabin door supports when installing the hydraulic snubber. Use AN960 washers as shims to eliminate side play between the rod ends and their supports.

(2) To ensure proper travel of the snubber piston, adjust the rod ends until 2 threads are visible at each rod end.

(3) Turn the Allen head screw near the end of the barrel clockwise to slow down the action of the snubber or counterclockwise to speed up that action.

(4) Service the snubber with hydraulic fluid (9, Table 6, 91-00-00) as required. Fill the snubber barrel through the hole for the Allen head screw on the end of the barrel nearest the movable piston. Count the number of turns required to remove the screw so that the snubber can be adjusted for the same rate of action upon reinstallation of the screw after servicing.

NOTE: When installing the end cap on the barrel of the snubber, tighten the four retaining screws down until flush with the outside of the snubber barrel. If tightened beyond this point, the screws may bind on the piston rod, rendering it immovable.

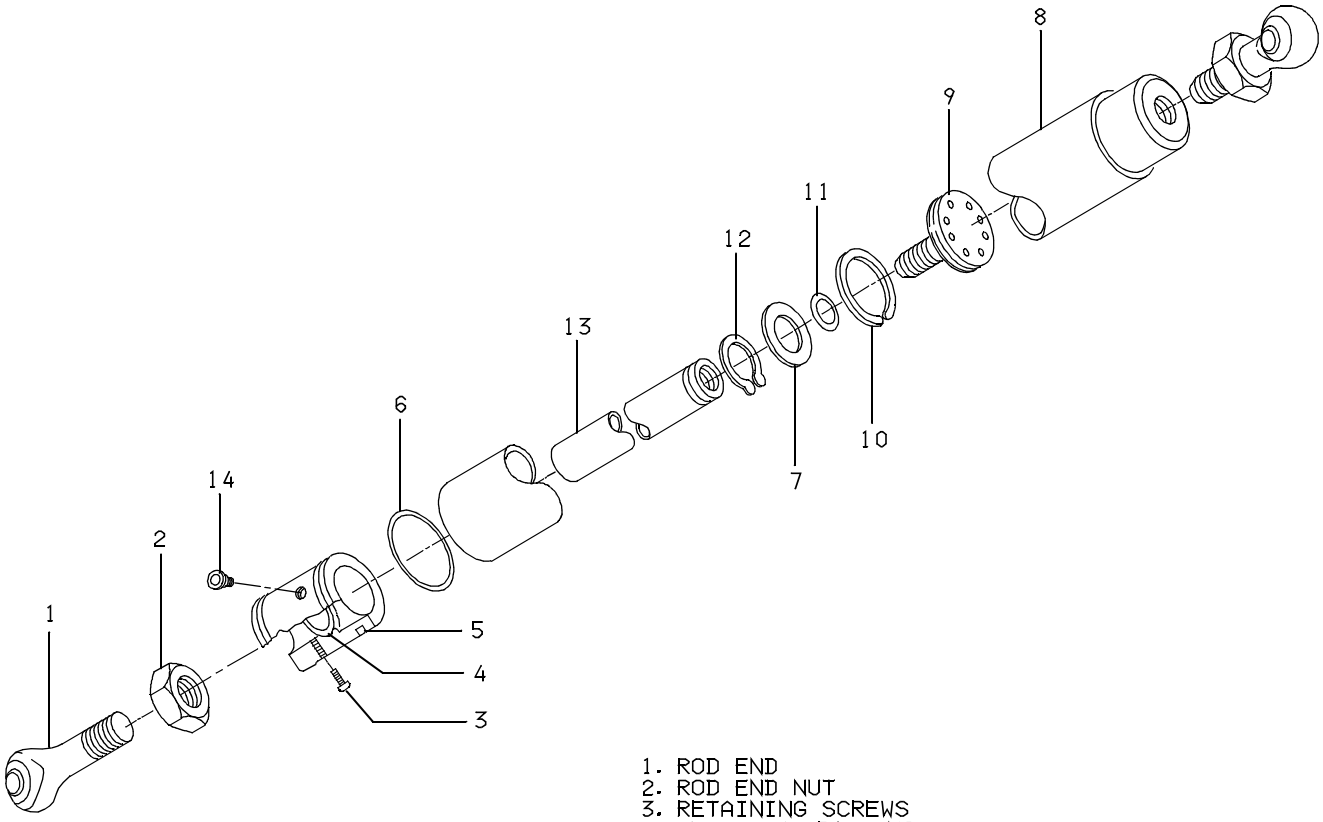
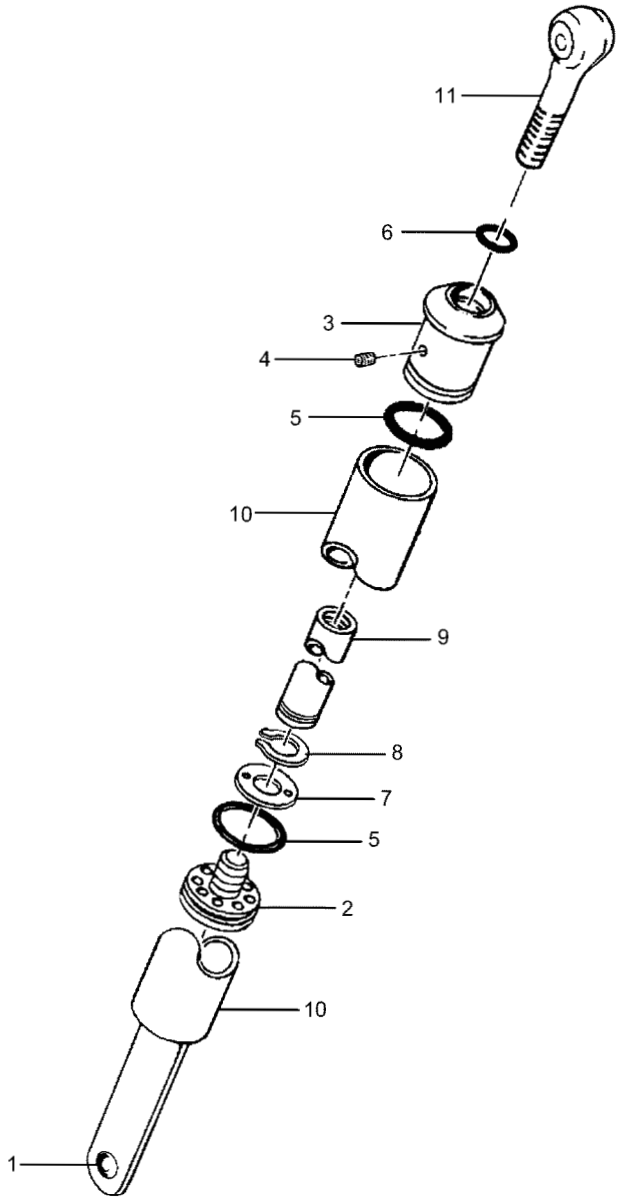


Figure 206
Cabin Entrance Door Damper

9. CABIN ENTRANCE DOOR DAMPER (HYDRAULIC SNUBBER) - HAWKER BEECHCRAFT MODEL

(1) Attach the barrel lower eye to the cabin door supports and the movable piston rod to the supports on the door frame when installing the hydraulic snubber. Shim as necessary with AN960-416 washers to eliminate side play between the rod ends and their supports.

(2) Service the snubber with hydraulic fluid (9, Table 6, 91-00-00). To service, loosen the cap on the end of the cylinder barrel with the movable rod by removing the three set screws securing the cap to the barrel. Push down on the rod end until the piston is bottomed in the cylinder barrel. Slide the cap up on the rod end and fill the cylinder barrel with hydraulic fluid up to where the cap is to be installed. Secure the cap to the cylinder barrel with the three set screws.



1. LOWER EYE
2. PISTON HEAD
3. CAP END
4. ALLEN-HEAD SCREW
5. PACKING
6. SEAL
7. ORIFICE
8. PISTON HEAD RETAINER
9. PISTON ROD
10. BARREL
11. UPPER ROD END

U#11B
100921AA.AI

Figure 207
Cabin Entrance Door Damper (Hawker Beechcraft Model)

10. SERVICING THE AIR CONDITIONING SYSTEM

Servicing the air conditioner system consists of checking and maintaining the correct refrigerant level, compressor oil level, belt tension and condition, system leak detection and replacement of the evaporator air filters.

A. Precautionary Service Measures

Before any service is attempted which requires opening of refrigeration pipes or units, the person doing the work should be thoroughly familiar with the instructions on servicing the system. He should follow very carefully these instructions when performing the tasks that will maintain this system in a proper functioning order.

The major reasons for these measures are for safety and to prevent dirt and moisture from entering into the system. Dirt contaminant may cause leaky valves or wear in the compressor. Moisture may not only freeze into ice at expansion valves, but can also cause the formation of hydrochloric or hydrofluoric acids in the system.

All subassemblies are shipped, sealed and dehydrated. They are to remain sealed until just prior to making connections. They should be at room temperature before uncapping. (This prevents condensation of moisture from the air that enters the system).

If for any reason, caps are removed but the connections are not made, then the tubes and other parts should not remain unsealed for more than 15 minutes. Reseal connections if period is to be longer.

Compressors are shipped with 10 to 11 ounces of oil (6, Table 6, 91-00-00) and charged with a mixture of Refrigerant-12 and dry nitrogen to provide an internal pressure at slightly above atmospheric pressure.

All precautions should be taken to prevent damage to fittings or connections. Even minute damage to a connection could cause it to leak. Any fittings getting grease or dirt on them should be wiped clean with a cloth dampened with alcohol. Do not use chlorinated solvents such as trichloroethylene for cleaning agent, for they are contaminants. If dirt, grease or moisture gets inside pipes and cannot be removed, the pipe will have to be replaced. Use a small amount of clean refrigeration oil (6, Table 6, 91-00-00) on all tube joints and dip the packing in this oil before assembling the joint. This will help in making a leak-proof joint.

WARNING: Due to air quality-control regulations being enacted in the United States, R-12 refrigerant cannot be vented into the atmosphere. When performing maintenance on the vapor cycle system where refrigerant R-12 can escape from the system, evacuate the system with a recovery or recycle servicing unit that will salvage the refrigerant.

Do not smoke when servicing the system with R-12, for refrigerant converts to a highly toxic gas when exposed to an open flame.

A face shield should be worn when servicing the lines; refrigerant coming in contact with the eyes can cause the loss of sight. Also wear gloves and avoid breathing any vapors.

CAUTION: *Use only Refrigerant-12 (R-12) in this system; other refrigerants, particularly those containing methyl chloride, will cause rapid deterioration of the aluminum components.*

Insufficient torque, when tightening tubing connections, can result in loose joints and excessive torque can result in deformed joint parts. Either condition can result in refrigerant leakage. When connecting aluminum fittings in the refrigerant system, torque all 5/8 inch fittings 18 to 21 foot-pounds and all 1/2 inch fittings 11 to 13 foot-pounds.

NOTE: This is a high pressure system. Before disconnecting a refrigerant line, depressurize the system with a recycle servicing unit, then purge the entire system with a vacuum pump operating at the 125 micron level.

The receiver-dryer is the last assembly to be connected. This is necessary to ensure maximum moisture protection of the refrigeration system.

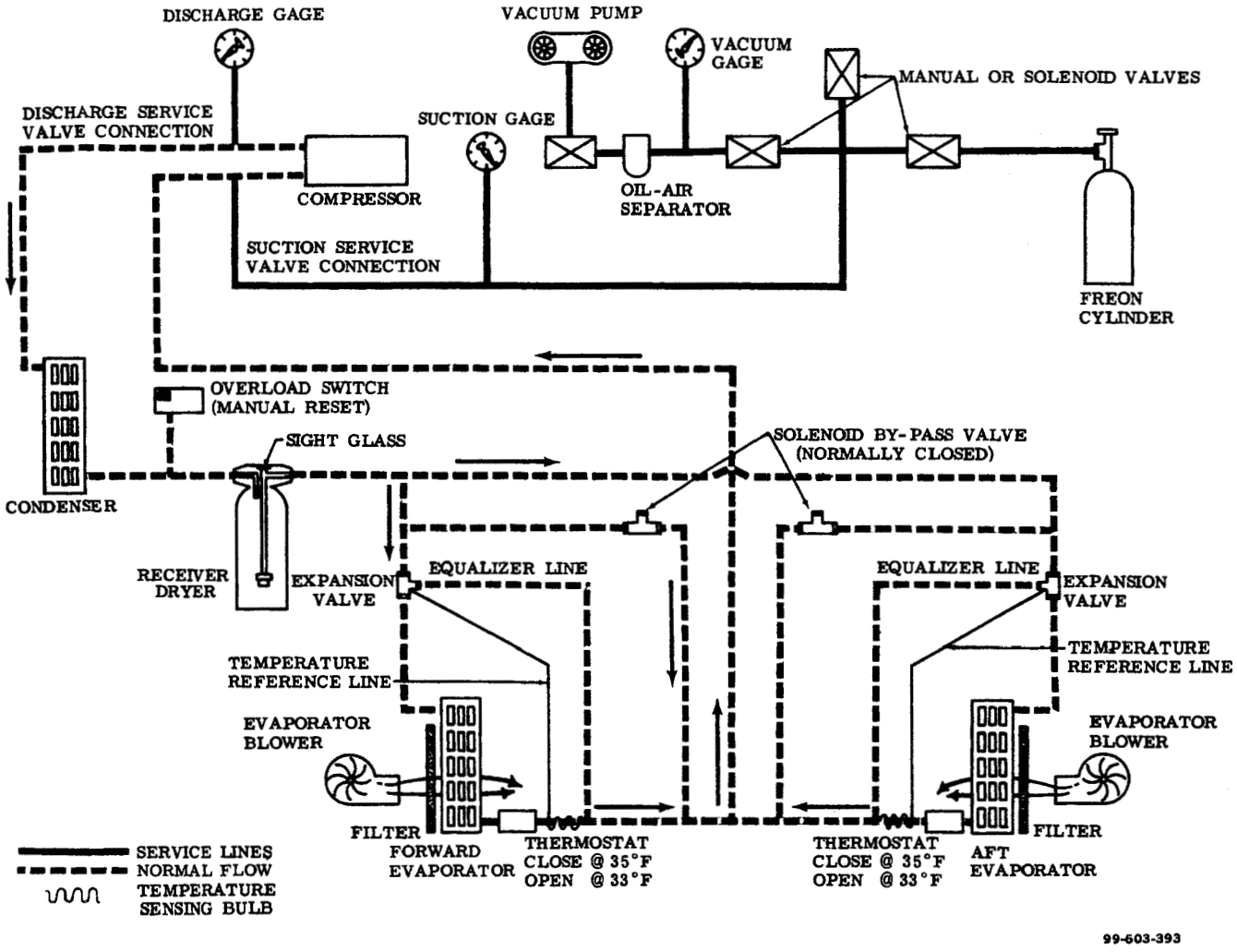


Figure 208
Cooling System Schematic (Prior to U-165, except U-50)

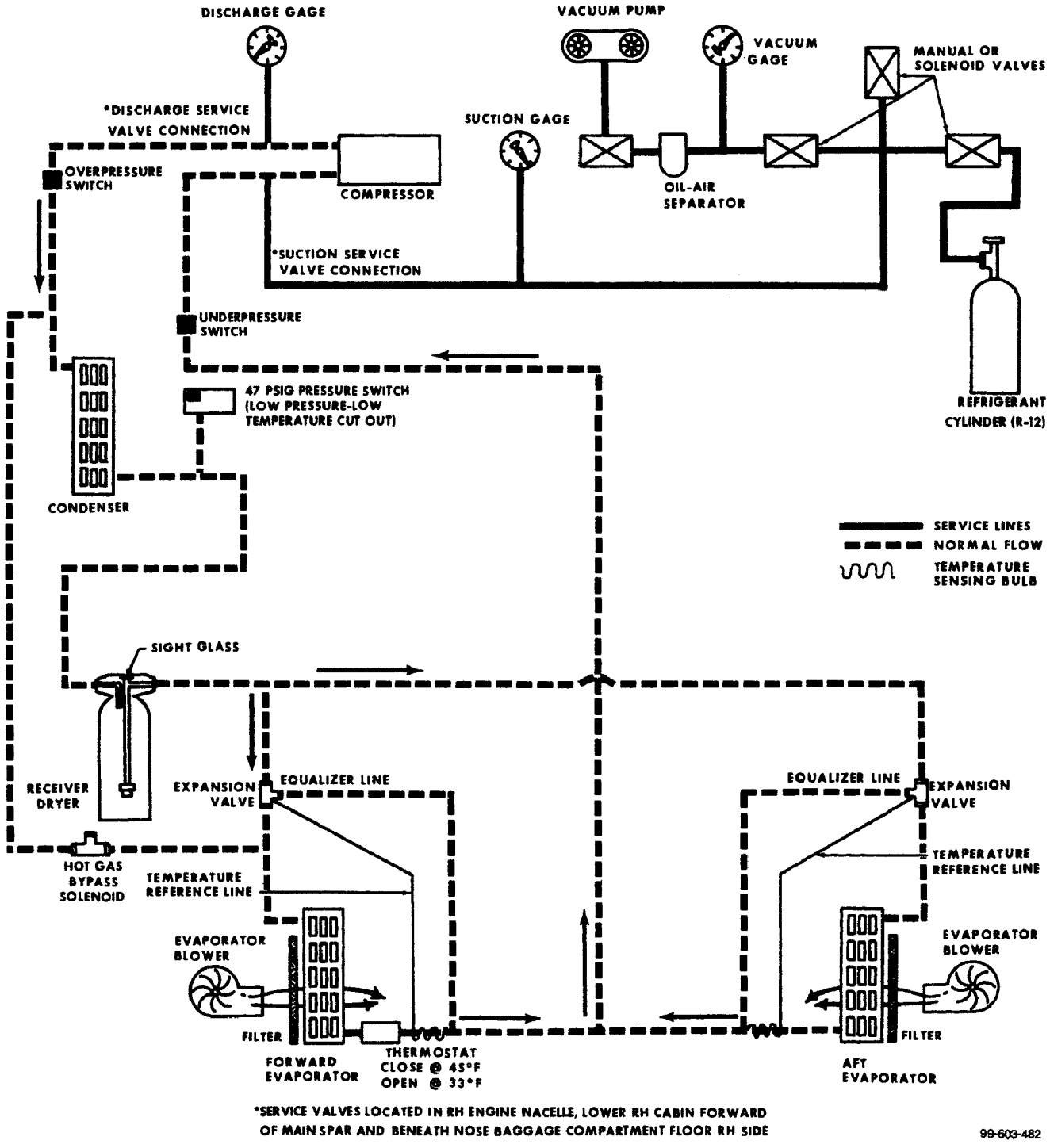


Figure 209
Cooling System Schematic (U-50, U-165 and After)

B. Checking System Oil Quantity

If a refrigerant leak has occurred, or a possibility exists that the system oil quantity is low, the quantity may be checked as follows: Operate the system for at least five minutes. Shut down for one to five minutes and, while the compressor is still hot, crack the compressor sump fitting. A flow of oil indicates adequate oil in the system, nearly dry gas indicates that the system is low on oil and the compressor must be removed from the system in order to check the oil content. Remove the drain plug and drain all the oil into a beaker while tipping the compressor back and forth and turning the compressor shaft. Add between 4 and 6 ounces of clean refrigerant oil (6, Table 6, 91-00-00) to the compressor and reinstall the drain plug.

Replacement compressors are shipped with approximately 10 ounces of oil in the compressor. Completely drain the oil from the new compressor and replace it with the same volume of oil that was drained from the old compressor if that volume is between 4 and 6 ounces. If the volume of oil drained from the old compressor is less than 4 ounces, then add 6 ounces of oil to the completely drained compressor.

C. System Leak Detection

NOTE: It is permissible to add 8 ounces of Trace, a red dye, to the suction side of the refrigerant system to detect leaks. A red film will appear in the areas of leakage.

A reduction of system cooling ability or the presence of bubbles in the refrigerant may indicate a partial loss of refrigerant. Open the right hand forward baggage compartment door and check for bubbles in the sight glass under the plexiglass cover in the floor of the baggage compartment at FS 57. The sight glass should be checked during operation at maximum available ambient and cabin temperatures. Streams of bubbles past the glass or foam in the glass indicates an inadequate refrigerant quantity. If a loss of refrigerant is suspected, an inspection of the system plumbing should be carried out to locate the source of the leak. Large leaks may be located by the appearance of oily spots where oil has been carried out by escaping refrigerant. Smaller leaks, which are much more difficult to locate, may be detected by detergent bubbles, or an electronic detector. In order to use the above detectors, the system must contain a partial charge and be operating. The detergent test is accomplished by covering a suspected area with soap suds and watching for bubbles. An electronic detector has a probe which is moved along the plumbing to pick up escaping refrigerant. R-12 is heavier than air so the probe should be held below the lines When the electronic probe picks up the presence of escaping refrigerant, the detector emits a flashing light or a high pitched squeal. If a leak is detected at a fitting the use of soft copper flare gaskets in the flare fitting is permissible if the connection does not seal by the normal method.

D. Torque Procedure for Wiggins Coupling

Correct torquing procedures for Wiggins couplings throughout the air conditioning system are as follows: Lubricate packings and internal coupling surfaces with the air conditioning system fluid. Tighten the coupling nuts hand tight and safety wire.

E. Depressurizing

With a service unit attached, slowly open the high pressure valve to allow a slow escape of refrigerant from the system. If oil accompanies the refrigerant, the refrigerant is escaping too rapidly. When hissing ceases, close the valve.

F. Evacuating the System

Using a service unit with a vacuum pump capable of obtaining an absolute pressure of 125 microns or less and a pressure gage capable of indicating an absolute pressure of 125 microns or less, evacuate the system as follows:

(1) Attach the LO service hose to the suction port.

(2) Attach the HI service hose to the discharge port.

(3) Start the vacuum pump, then throttle the pump down by adjusting the vacuum valve to control the turbulence of the oil in the pump.

(4) As the pump-down progresses and the turbulence reduces, gradually open the vacuum and HI and LO valves until they are fully open for maximum evacuation efficiency.

(5) Crack the ballast valve slightly until the pressure is down to about 200 microns, then close the valve to permit continuing the pump-down.

(6) The system must be evacuated to 125 microns or lower.

G. Evaporator Filter Replacement

The evaporator filter should be replaced every 300 hours or more frequently if operating in heavy dust or smoke conditions. To gain access to the filters, refer to the applicable evaporator access opening in Chapter 12. Replacement filters are installed with the lattice backing against the evaporator coils and taped to the evaporator case.

11. HEATING SYSTEM (U-1 THRU U-164, EXCEPT U-50)

Servicing the heating system consists of cleaning the fuel control valve filter and fuel filter every 100 hours.

(1) Close the shutoff valve adjacent to the filter. Cut the lockwire from the filter bowl and body mounted on the inboard side of the LH main wheel well.

(2) Remove the filter bowl and element.

(3) Clean the element with solvent (38, Table 6, 91-00-00) or its equivalent and blow the element dry with compressed air.

(4) Install the element and bowl. Safety the bowl with lockwire.

(5) Remove the large square access door from the underside of the left wing center section just inboard of the nacelle and forward of the main spar.

(6) Remove the fitting from the inlet port of the heater fuel control valve and remove the filter.

(7) Clean the filter with solvent (38, Table 6, 91-00-00) or its equivalent and blow the filter dry with compressed air.

(8) Install the filter and fitting at the inlet port of the fuel control valve.

12. ENGINE FIRE EXTINGUISHER

Access to the fire extinguisher cylinder is through the wheel wells of the main landing gear. On U-1 thru U-164, except U-50, each cylinder is charged with 2 1/2 pounds of Bromotrifluoromethane ($CBrF_3$) and pressurized to 450 psi at 70°F by dry nitrogen. On U-50, U-165 and After, each cylinder is filled with 2 1/2 pounds of Bromotrifluoromethane ($CBrF_3$) and charged to 450 + 25 - 0 psi at 70°F with dry nitrogen.

CAUTION: *The engine should be washed internally and externally as indicated under the heading ENGINE WASHING PROCEDURES in this chapter any time the engine fire extinguisher has been discharged. This will protect the engine against the corrosive effects of the fire extinguisher agent. This precaution is particularly critical on a hot engine.*

Fully discharged cylinders or any cylinder weighing less than the specified amount should be returned to a recharging station certified by the supplier for a proper charge of $CBrF_3$ extinguishing agent. Such stations are located in most major cities in this country. All warranty claims must be directed to the supplier Pacific Scientific Company (HTL/KIN Tech Division), 1800 Highland Avenue, Duarte, California 91010.

Table 203
FIRE EXTINGUISHER CYLINDER PRESSURE LIMITS

TEMP IN F°	**-40**	**-20**	**0**	**20**	**40**	**60**	**80**	**100**	**120**
INDICATED	190	220	250	290	340	390	455	525	605
PRESSURE	to	to	to	to	to	to	to	to	to
IN PSI	240	275	315	365	420	480	550	635	730

List of Certified Charging Stations

*Aviation Comprogas Services
10 Railroad Avenue
East Northport, L.I., N.Y. 11731
Phone: 516/368-2203

Bacon Equipment Co.
2709 N. Beckley Avenue
Dallas, Texas 75208
Phone: 214/742-5871

Protection C.F.H., Inc.
103 Gun Ave.
Pointe Claire, Que.
H9R 3X2, CANADA
Phone: 514/694-3980

*CB Enterprises
10704 Vanowen Street
North Hollywood, Calif. 91605
Phone: 213/980-2332

*Korean Air Lines
C.P.O. Box 864
Kimpo International Airport
Seoul, KOREA

*Japan Air Lines
Hanada International Airport
Tokyo, JAPAN

John Cameron Aviation
Hanger 16
Bankstown Airport
N.S.W. 2200
AUSTRALIA
Phone: 7055812

Air Support International, Inc.,
6753 E. 47th Avenue Dr.
Denver, Colorado 80216
Phone: 303/333-5441

Tym's
414 West Arbor Vitae
Inglewood, Calif. 90301
Phone: 213/673-3330 or 678-3330

City Sales

U. S. 2 and Lake Antoine Rd.
P. 0. Box 663
Iron Mountain, Michigan 49801
Phone: 906/774-3555

*Delta Air Lines, Inc.
Jet Maintenance Base
Hartsfield-Atlanta International Airport
Atlanta, Georgia 30320
Phone: 404/346-6011

Aero Electric, Inc.
113 S. Laura
Wichita, Kansas 67211
Phone: 316/263-0197

Safety Supply Company
172 Osborne Street, South
Winnipeg, Manitoba, R3J 1Z1
CANADA
Phone: 204/453-7838

*Deutsche Lufthansa
Weg Beim Jaeger
D2000 Hamburg 63
W. GERMANY

Eastern Aero Marine
3850 N.W. 25th Street
Miami, Florida 33142
Phone: 305/871-4050

Electronique Aerospatiale (EAS)
Boite Postale No. 4
Le Bourget - 93
FRANCE

Fire Fighter Sales & Service Co.
1721 Main Street
Pittsburgh, Pa. 15215
Phone: 412/782-2800

*KLM - Royal Dutch Airlines
Schiphol Airport
P.O. Box 770
Amsterdam, THE NETHERLANDS

List of Certified Charging Stations

*United Air Lines, Inc.
San Francisco Int'l Airport
San Francisco, Calif. 94128

Air Asia Co., Ltd.
Tainan Airfield
Tainan, Taiwan 700
Republic of CHINA
Phone: 234141

Forind-Avio S.R.I.
Via Bartolini, 1
20155 Milano, ITALY

General Fire Equipment Co.
E. 4004 Trent Ave.
Spokane, Washington 99202
Phone: 509/535-4255

Vancouver Welding
800 Harney St.
Vancouver, Washington 98660
Phone: 206/696-4605

City Sales

Aero Mercantil, Leaver & CIA
Guaymaral Airport
Bogota, COLUMBIA
Phone: 481761 or 488799

*Fas-Orient, (PTE) Ltd.
Jalan Kayu, P.O. Box 17
SINGAPORE 28

*Graviner, LTD
Poyle Road, CoInbrook
Slough, Buckinghamshire
SL3 OHB
ENGLAND

Houston Fire & Safety Equip. Co.
8555 Monroe Road
Houston, Texas 77061
Phone: 713/943-0920

*HTL Advanced Technology Division
1800 Highland Avenue
Duarte, California 91010
Phone: 213/359-9317 or 681-7487
TWX 910-585-3238

*Equipped for Hermetically-Sealed Containers

Each cylinder should be removed and weighed at one year intervals. The weight of the fully charged cylinder is marked on a decal attached to the cylinder.

To change the bulbs in the fire extinguisher switch and light combination, proceed as follows:

CAUTION: *To avoid inadvertently actuating the switch and discharging the supply cylinder, disconnect the battery or the electrical wiring at the supply cylinders when changing the bulbs.*

(1) Remove the light capsule from the panel front, which is held in place by spring clips, and replace the bulbs.

(2) After bulb replacement, slip the light capsule back into place.

13. OXYGEN SYSTEM

NOTE: Oxygen used for recharging must be aviators breathing oxygen (22, Table 6, 91-00-00).

WARNING: Do not use oxygen intended for medical purposes, or such industrial uses as welding. Such oxygen may contain excessive moisture that could freeze up the valves and lines of the oxygen system.

Access to the pressure indicator and filler valve of the oxygen system may be gained through an access door labeled OXYGEN RECHARGE located on the right side of the nose wheel well. To recharge the oxygen system, remove the protective filler cap from the filler valve and attach the hose from an oxygen recharging cart to the filler valve.

WARNING: Avoid making sparks and keep all burning cigarettes or fire away from the vicinity of the airplane. Make sure that the oxygen shutoff valve in the cockpit is in the closed position. Inspect the filler connection for cleanliness before attaching it to the filler valve. Make sure that your hands, tools, and clothing are clean, particularly of grease or oil, for these contaminants will ignite upon contact with pure oxygen under pressure. As a further precaution against fire, open and close all oxygen valves slowly.

Observe the following precautions when servicing the oxygen system:

(1) Always ground the system to be serviced and the servicing equipment before connecting the filler adapter.

(2) Close the oxygen bottle manual shutoff valve.

(3) Ensure that all airplane electrical power is OFF. Do not operate electrical switches, or connect or disconnect ground power generators during the oxygen charging operation.

(4) Do not service the oxygen system if fueling or other flammable fluid servicing is in process.

(5) Do not charge the system too fast. Rapid charging can create a dangerous overheating condition.

To prevent overheating, fill the oxygen system slowly by adjusting the recharging rate with the pressure regulating valve on the cart. Fill the cylinder until a pressure of 1800 $\pm$ 50 psig is attained at a temperature of 70°F. On U-50, U-165 and After, the standard 22 cubic foot bottle should be filled to 1800 psi. System pressure should be 1850 psi for optional oxygen installations requiring 49 and/or 66 cubic foot bottles. (This is a steady state condition, after the cylinder has cooled from the recharging heat build-up.) This pressure may be increased an additional 3.5 psi for each degree of increase in temperature; similarly, for each degree of drop in temperature, reduce the pressure for the cylinder by 3.5 psi. When the oxygen system is properly charged, disconnect the filler hose from the filler valve and replace the protective cap on the filler valve. If at any time, in the process of servicing and purging the system or replacing the oxygen cylinder, it becomes necessary to disconnect a fitting, the threads should be sealed with size 1 Teflon Tape (24, Table 6, 91-00-00). Apply to the first three threads of the male fitting. A swage-lock fitting should be tightened one and a quarter turns from the finger tight position.

NOTE: Refer to advisory circular 43.13-1A for additional servicing precautions recommended by the FAA for the oxygen system.

14. HYDRAULIC LANDING GEAR SYSTEM

Servicing the hydraulic landing gear consists primarily of maintaining the correct system fluid level, maintaining the correct accumulator precharge, changing the disposable system filter, cleaning the power pack finger screens, and checking all system plumbing for leaks, security, and attachment.

A. Accumulator Precharging

WARNING: TO PRECLUDE INJURY OR DAMAGE TO THE AIRPLANE, THE AIRPLANE MUST BE PLACED ON JACKS PRIOR TO SERVICING. If there is any movement of an S downlock hook or any actuator movement while the accumulator is being charged, the accumulator may have internal damage and should be replaced.

The accumulator is located adjacent to the power pack under the center aisle floorboards. Apply a source of dry compressed air or bottled nitrogen to the servicing valve on the accumulator and charge to 600 ± 30 psig (U-1 thru U-164, except U-50) or to 750 ± 38 psig (U-50, U-165 and After).

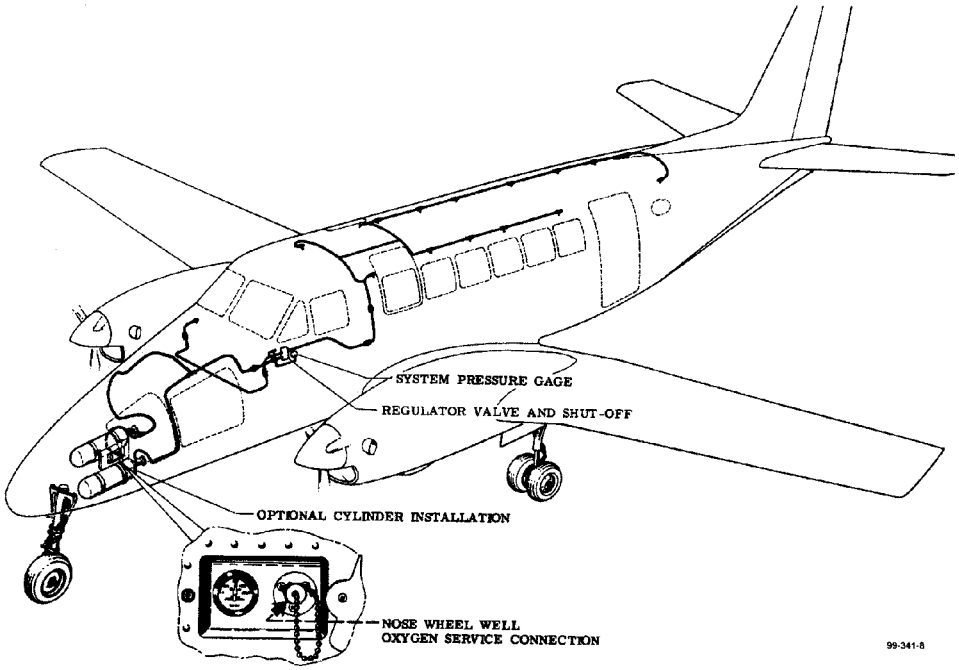


Figure 210
Oxygen System (U-1 thru U-164, except U-50)

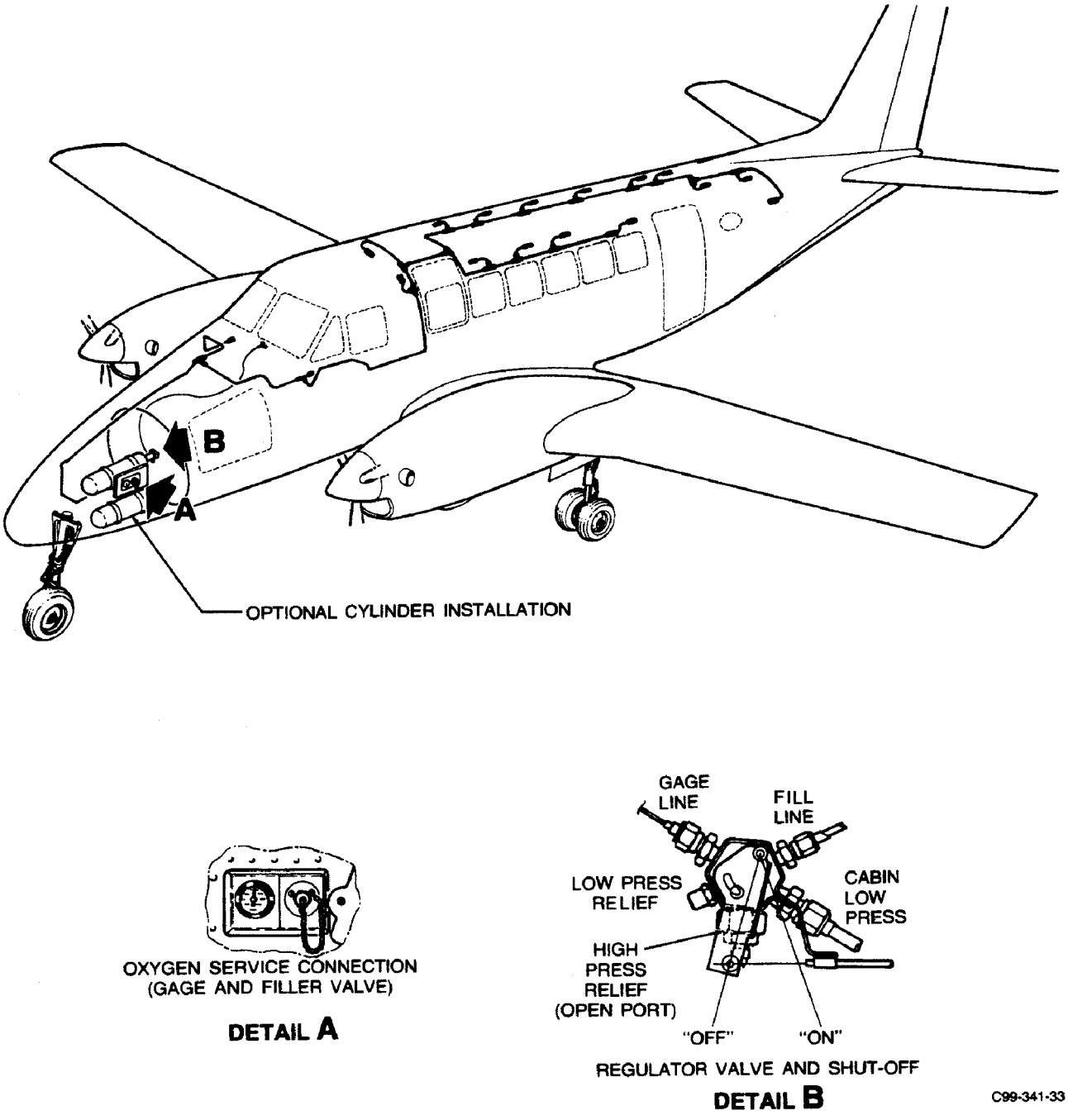


Figure 211
Oxygen System (U-50, U-165 and After)

B. Filling the System

A fill can, located just inboard of the LH nacelle and forward of the main spar, contains a cap and dipstick assembly, marked FULL and ADD, for convenience of maintaining system fluid level. The total hydraulic system capacity is 1 gallon (4 quarts), of which only 0.84 gallons (3.36 quarts) of hydraulic fluid is usable. Place the airplane on jacks and cycle the gears. After the gears have been cycled once, the fluid level may return to ADD. Refill the fill can and recycle the gears until the FULL level is maintained after two cycles. Use hydraulic fluid (9, Table 6, 91-00-00) when servicing the system.

C. System Filter Replacement

It is recommended that the system filter be replaced every 100 hours for the first 300 hours of service; then, as required. The system filter may be removed and replaced as follows:

(1) On airplanes prior to U-165, except U-50, remove the seats from the RH seat deck and the carpet and floorboard access openings just forward and just aft of the main spar. Remove the heat and vent duct that is installed over the filter.

(2) On airplanes U-50, U-165 and After, remove the center aisle floorboard just aft of the main spar.

(3) Cut the safety wire and unscrew the filter assembly. Remove and discard the filter cartridge but retain the cartridge spring.

NOTE: Shop towels will be needed to absorb the system fluid which will escape from the port when the filter assembly is removed. The port should be temporarily plugged to prevent excessive leakage until the new filter assembly is ready to install.

(4) Insert the spring and new filter cartridge into the filter housing.

(5) Lubricate the new filter cartridge packing and the filter housing packing with petrolatum (27, Table 6, 91-00-00). Install the packing on the filter housing.

(6) On airplanes prior to U-165, except U-50, install the heat and vent duct. Install the floorboard access panels, carpeting and seats previously removed.

(7) On airplanes U-50, U-165 and After, install the center aisle floorboard just aft of the main spar.

(8) Check the fluid level in the external fill can and add hydraulic fluid (9, Table 6, 91-00-00) to fill the system. Avoid overfilling the system as any excess hydraulic fluid will be dumped overboard.

D. Finger Screen Cleaning

It is recommended that the finger screens in the gear-up and gear-down ports of the power pack be cleaned every 100 hours for the first 300 hours of service, then as required. Remove and clean the screens as follows (Ref. Figure 212):

(1) On airplanes prior to U-165, except U-50, remove the seats from the RH seat deck and the carpet and floorboard access panels just forward and just aft of the main spar. Remove the heat and vent duct if necessary. Remove the center aisle floorboard just forward of the main spar.

(2) On airplanes U-50, U-165 and After, remove the center aisle floorboard just aft of the main spar.

(3) Remove the plumbing and unions from the gear-up and gear -down ports of the power pack.

(4) Cap the plumbing lines.

NOTE: Provision should be made to collect the fluid which will be lost when the plumbing is disconnected.

(5) Remove the filter screens with a magnet that can be inserted into the port, or with a small wire with a 45° bend at the tip.

(6) Clean the screens with solvent (38, Table 6, 91-00-00) and blow dry with shop air.

(7) Plug the ports after the screens have been removed.

(8) Insert the finger screens into the gear-up and gear-down ports.

(9) Install new packings, lubricated with petrolatum (27, Table 6, 91-00-00), and install the unions in the gear-up and gear-down ports.

(10) Connect the plumbing lines.

(11) Replace the fluid that was lost during the screen removal with hydraulic fluid (9, Table 6, 91-00-00) as required.

(12) Install any previously removed seats, carpets, access panels, and center aisle floorboards.

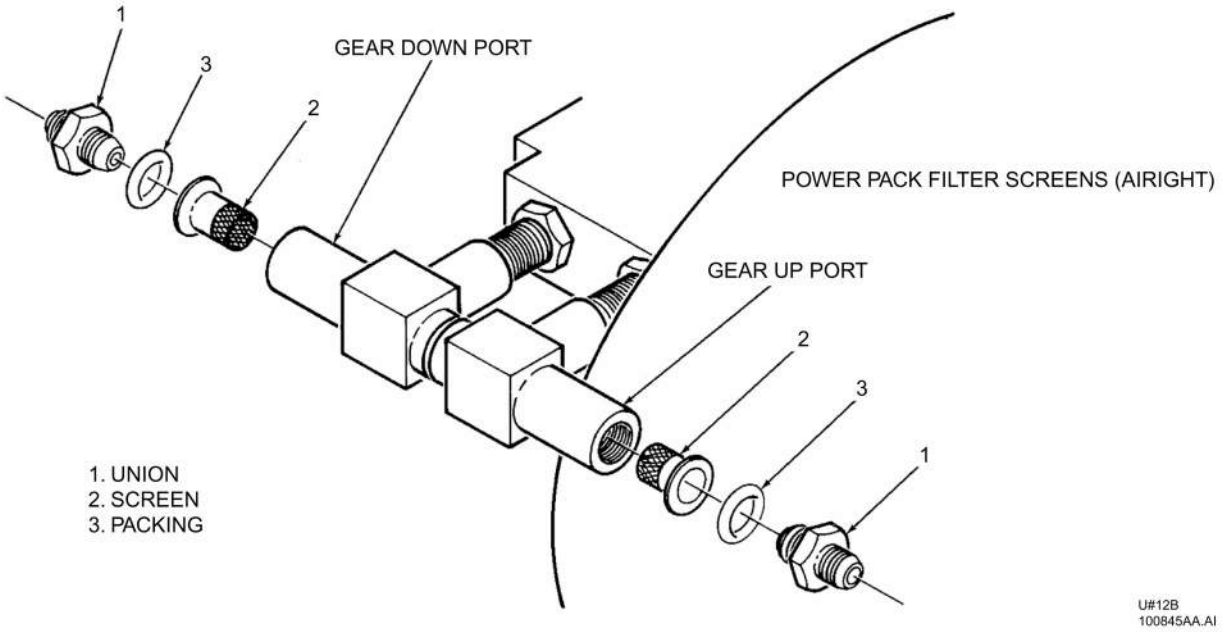


Figure 212
Power Pack Screen

15. WATER/METHANOL INJECTION

A. Filling the Water/Methanol System

WARNING: Methanol is a volatile, flammable, poisonous alcohol. Methanol/water mixtures must be handled in well ventilated areas. Inhalation of methanol vapors should be avoided. Normal precautions for handling and storage of flammable liquids must be complied with.

A small hinged door (placarded WATER/METHANOL FILL) located just aft of the wing center section leading edge and inboard of the nacelle provides access to the water/methanol fill ports for each engine. Two fill ports are installed in each location to provide both gravity-feed filling and pressurized filling of the water/methanol system. Filling the system under pressure can be accomplished much faster than by the gravity method; however, care must be exercised not to exceed 30 psi pressure during the filling operation. A standard 3 gallon size garden sprayer, equipped with the proper fitting to adapt to the pressure fill port on the airplane, may be used for pressure filling.

WARNING: Under no circumstances should 100% methanol ever be used for filling the water/methanol system.

A mixture (by volume) of 45% methanol and 55% distilled or demineralized water should be used for filling the system. Distilled or demineralized water should always be used to reduce the possibility of contamination or corrosion of the components. Each reservoir contains approximately 3 gallons of solution when full. Full indication of the reservoirs occurs when solution flows from the overboard vent line beneath the wing surface. The area around the fill cap should be cleaned before removing the cap to reduce the possibility of contamination.

B. Draining the Water/Methanol System

If the water/methanol system has not been used for 30 days and/or the airplane has not been moved for 7 days, the existing solution must be drained from the system and replaced with fresh mixture. At the end of the operational season, the system must be drained completely and blown out with an external air source to expedite evaporation of residual mixture. A drain valve is located in the plumbing line near the inboard end of each water/methanol reservoir. Access to the drain valve requires removal of the wing center section leading edge skin assembly.

C. Water/Methanol System Filter Cleaning and/or Replacement

An inline system filter is located just outboard of the water/methanol reservoir in each wing center section leading edge. Access to the filter requires removal of the leading edge skin assembly. It is recommended that this filter be removed once each year, preferably at the beginning of the hot weather season, and cleaned by backflushing with methanol or replaced as required.

SERVICING
SCHEDULED SERVICING
MAINTENANCE PRACTICES

1. AIRPLANE PRESERVATION

For preservation of the airplane and engine, refer to STORAGE in Chapter 10-10-00.

2. TIRES

CAUTION: *Tires that have picked up a fuel or oil film should be washed down as soon as possible with a detergent solution to prevent contamination of the rubber.*

Maintaining proper tire inflation will help to avoid damage from landing shock and contact with sharp stones and ruts, and will minimize tread wear. When inflating the tires, inspect them for cuts, cracks, breaks and tread wear. Inflate tire to pressures shown below (Ref. Tables 201, 202 or 203):

WARNING: Do not mix 8-ply rated tires with 10-ply rated tires on the same landing gear strut.

Table 201
Tire Pressure
(U-1 thru U-147, except U-50 and U-146 Without Kit No. 99-5014-1 or 99-5014-3 Installed)

LANDING GEAR	TIRE SIZE	AIR PRESSURE
Standard Main Gear	18 x 5.5 - 8 Ply	88 to 92 Pounds
Standard Main Gear	18 x 5.5 - 10 Ply	78 to 82 Pounds
Flotation Main Gear	10 x 6.50 - 6 Ply (tube type)	48 to 52 Pounds
Nose Gear	10 x 6.50 - 6 Ply	50 to 55 Pounds

Table 202
Tire Pressure
(U-146, U-148 thru U-164 and Prior Airplanes, except U-50, With Kit No. 99-5014-1 or 99-5014-3 Installed)

LANDING GEAR	TIRE SIZE	AIR PRESSURE
Standard Main Gear	18 x 5.5 - 8 Ply	92 to 96 Pounds
Standard Main Gear	18 x 5.5 - 10 Ply	82 to 86 Pounds
Flotation Main Gear	10 x 6.50 - 6 Ply (tube type)	51 to 55 Pounds
Nose Gear	10 x 6.50 - 6 Ply	53 to 57 Pounds

Table 203
Tire Pressure (U-50, U-165 and After)

LANDING GEAR	TIRE SIZE	AIR PRESSURE
Standard Main Gear	18 x 5.5 - 8 Ply	100 to 104 Pounds
Standard Main Gear	18 x 5.5 - 10 Ply	90 to 94 Pounds
Flotation Main Gear	10 x 6.50 - 6 Ply (tube type)	55 to 59 Pounds
Nose Gear	10 x 6.50 - 6 Ply	55 to 59 Pounds

NOTE: While Hawker Beechcraft Corporation cannot recommend the use of recapped tires, retreaded by an FAA approved repair station with a specialized service limited rating for TSO-C62c may be used. Tires recapped to any other specification may have a tendency to swell after use and may jeopardize proper function of the landing gear retract system, with the possibility of damage to the landing gear doors and retract mechanism.

A. Sealing Minor Leaks of Rim-inflated Tubeless Tires

Tire sealant (87, Table 6, 91-00-00) is recommended as an effective means of controlling the gradual loss of tire inflation pressure when the leakage rate does not exceed five percent over a 24-hour period. An M400 injector for applying the sealant and the tire sealant itself are available in either pint or gallon quantities from Hawker Beechcraft Aero or Aviation Centers and International Distributors and Dealers.

Tire sealant (87, Table 6, 91-00-00) may be applied as follows:

(1) Place the airplane on jacks and reduce the tire pressure to 35 psi or less.

(2) When using the M400 injector and gallon container:

(a) Screw the adapter (furnished with the injector) into the container.

(b) Place the M400 injector squarely on the adapter in a vertical position.

(c) Prime the injector with several slow strokes of the handle.

(d) Pull the handle full back to fill the injector with tire sealant (87, Table 6, 91-00-00).

(e) Place the tip of the injector squarely on the valve stem and inject the tire sealant (87, Table 6, 91-00-00) by pushing the handle completely in. The valve core does not need to be removed. One injection will be sufficient for each tire.

CAUTION: *Keep out of line with the injector handle and remove the injector from the valve stem before releasing the handle.*

(3) When using the pint container and the attached pump:

(a) Install the pump on the pint container and tighten firmly.

(b) Screw the adapter firmly onto the tire valve stem.

(c) Operate the pump until two ounces, as indicated on the container, has been pumped into the tire.

(d) Remove the adapter from the valve stem.

(4) Inflate the tire to the proper pressure to distribute the "Aero Seal", and remove the airplane from jacks.

3. EXTERNAL POWER

The airplane electrical system is protected against damage from reverse polarity by a relay and diodes in the external power circuit. Prior to U-90 the external power receptacle is located just outboard of the nacelle in the right center section. At serial U-90 and After the external power receptacle is located aft of the entrance door under the airplane. The receptacle is designed for a standard AN type plug. To supply power for ground checks, a ground power source capable of delivering a continuous load of 300 amperes at 24 to 30 volts is required. The ground power source must also be capable of delivering up to 1,000 amperes for 0.1 of a second during engine starts. Use of an inadequate ground power unit can cause a voltage drop below the dropout voltage of the starter relay, resulting in relay chatter and welded contacts. By the same token, a maximum continuous load in excess of 350 amperes will damage the external power relay and power cables of the airplane.

CAUTION: *The output setting must not exceed 1,000 amperes on external power sources with a higher current-carrying capability. Any current in excess of 1,000 amperes may overtorque the drive shaft of the starter-generator or produce heat sufficient to shorten the life of the unit.*

A. U-1 thru U-164, except U-50

Observe the following precaution when using an external power source:

(1) Use only an auxiliary power source that is negatively grounded. If the polarity of the power source is unknown, determine the polarity with a voltmeter before connecting the unit to the airplane.

(2) Before connecting the external power unit, turn OFF all radio equipment and generator switches, but leave the battery ON to protect transistorized equipment against transient spikes.

CAUTION: *When the battery switch is turned OFF for extended ground power operation, place an external battery in parallel with the output of the external power unit before operating any transistorized avionics equipment.*

(3) If the unit does not have a standard AN plug, check the polarity and connect the positive lead from the external power unit to the center post and the negative lead to the front post of the airplane's external power receptacle. The small pin of the receptacle must be supplied with +24 vdc to close the external power relay that provides protection against damage by reverse polarity.

B. U-50, U-165 and After

Observe the following precautions when using an external power source:

NOTE: The jet transfer system and filter are installed on the A99A. The jet transfer filter is not installed on the C99.

(1) Use only an auxiliary power source that is negative grounded. If the polarity of the power source is unknown, determine the polarity with a voltmeter before connecting the unit to the airplane.

(2) Before connecting an external power unit, turn off the Avionics Master switch or all radio equipment and the generator switches, but leave the battery switch ON to protect transistorized equipment against transient voltage spikes.

CAUTION: *Voltage is required to energize the Avionics Master Power relays to remove the power from the avionics equipment; therefore, never apply external power to the airplane without first applying battery voltage. For extended ground power operation, when it becomes necessary to turn off the battery switch to prevent depleting the battery or if the battery is removed from the airplane, place an external battery in parallel with the output of the external power unit before operating any avionics equipment.*

(3) After the external power plug is connected, turn on the battery switch before turning on any other equipment to protect transistorized equipment against transient spikes.

CAUTION: *The battery may be damaged if exposed to higher than 30 volts for extended periods of time.*

If the ground power unit does not have a standard AN plug, check the polarity of the plug. The positive lead from the ground power unit must connect to center post, the negative lead must connect to the front post, and a positive voltage of 24 to 28 vdc must be applied to the small polarizing pin of the airplane's external power receptacle.

4. STROBE LIGHT SYSTEM

Whenever one of the strobe lights fails to illuminate, check the condition of the flash tube and electrical connectors. If either of these appears defective, perform the system checkout described in Chapter 33 before replacing the inoperative light. To replace flash tubes in the strobe light system, refer to the illustrated procedure in the BULB REPLACEMENT GUIDE, Chapter 33.

On airplane serials U-1 thru U-99, except U-50, every 500 hours of airplane operation or once a year, whichever comes first, lightly lubricate the bearings and the cams of the motor in the timer module with Aeroshell No. 12. After 500 hours and 1,000 hours of airplane operation, check the condition of the relay contacts in each power supply module and perform a capacitance check on the condensers as described in Chapter 33.

WARNING: On airplanes equipped with Hoskins strobe lights, to avoid the chance of shock with the cable at the light fixture or with connector pins "A" and "B" at the power supply module, short out, these pins to pin "C" (ground) to dissipate any residual charge left in the condenser after the system has been turned off. Grimes strobe lights are equipped with a resistor that dissipates the high voltage in the circuit between the power supply and strobe light assemblies; however, wait 10 minutes after the system is turned OFF before handling the light assemblies to give the resistor time to dissipate voltage buildup.

5. BATTERY REPLACEMENT, EMERGENCY EXIT SIGN (U-50, U-165 AND AFTER)

Emergency exit signs are attached to the headliner above the airstair door and between the LH and RH cabin emergency exit doors. Each placarded EXIT sign contains two light bulbs, P/N GE354, and two batteries, P/N HS 150 Hercules (104, Table 6, 91-00-00) or equivalent. A switch placarded AUTOMATIC-MANUAL is located on the face of each EXIT sign. The signs are impact activated when the AUTOMATIC-MANUAL switch is placed in the AUTOMATIC position. The signs illuminate when the switch is placed in the MANUAL position. The batteries should be tested in this manner monthly and replaced at six-month intervals. The emergency exit sign switch should remain in the armed AUTOMATIC position at all other times.

The batteries should be replaced every six months as follows:

(1) Remove the screws attaching the lens to the EXIT sign assembly.

(2) Remove the batteries, two per sign assembly, four per airplane.

(3) Install new batteries (104, Table 6, 91-00-00) as placarded in the battery holder in the sign assembly.

(4) Position the lens on the sign assembly and install the attaching screws.

(5) Place the ATOMATIC-MANUAL switch in the MANUAL position to test the sign for proper operation and illumination. Return the switch to the AUTOMATIC position after testing.

CAUTION: *The emergency exit sign assemblies are NOT interchangeable.*

6. BATTERY MAINTENANCE

NOTE: Methods of servicing lead-acid batteries do not apply for the servicing of nickel-cadmium batteries.

The airplane is equipped with a 24-volt nickel-cadmium battery. To obtain optimum service from the nickel-cadmium battery, proper and regular maintenance of the battery MUST be performed.

A Systematic Battery Maintenance Program should be established and carefully followed:

CAUTION: *The electrolyte used in nickel-cadmium batteries is an alkali. It is important that the servicing areas for nickel-cadmium batteries be separated (in different buildings if possible) from lead-acid batteries so that no contamination, even from fumes, is possible. The tool and distilled water used for lead-acid batteries should be confined to areas used to service lead-acid batteries and MUST NOT be brought into the vicinity of nickel-cadmium batteries. Such tools MUST NOT BE USED on nickel-cadmium batteries. Even minute traces of lead-acid fumes can damage a nickel-cadmium battery.*

(1) The battery should be removed from the airplane for service.

(2) A log of the services performed on each battery should be maintained.

(3) The battery should be removed from the airplane and serviced after 100 flight hours or 30 days, whichever occurs first. If the ambient temperatures are above 90°F or the time between engine starts averages less than 30 minutes, the duty cycle should be reduced.

(4) The log of battery services performed should be evaluated to determine the need to service the battery at the above-recommended intervals or to extend intervals if justified. Accurate water consumption data is a valid barometer to use for adjustment of servicing intervals.

Since the proper battery servicing requires two days, an additional battery (or batteries) will be required where airplane utilization warrants. For additional information on battery maintenance, refer to the Operating and Service for General Electric Nickel-Cadmium Vented-Cell Batteries (BBD-OEM-234). If the airplane is equipped with a SAFT battery, refer to the Nickel-Cadmium Aircraft Battery Maintenance Manual (DC3-01-78-3176-5A) for additional maintenance information. FAA Advisory circular AC 00-33, is another good source of battery maintenance information.

7. ENGINE WASHING PROCEDURES

CAUTION: *Any time the engine compressor is to be washed, both the compressor bleed air and the P_3 air filter to the fuel control unit must be capped off.*

For detailed instructions of the washing procedure recommended by the engine manufacturer, refer to the applicable Engine Maintenance Manual as detailed in Chapter 72 and Aircraft Gas Turbine Operation Information Letter No. 7, Rev. 2 or subsequent, by Pratt and Whitney Aircraft of Canada Limited.

CAUTION: *Take precautions against contaminating the clutch portion of the air conditioner compressor with cleaning solvent during the engine wash procedure. Such contamination may dilute the grease in the compressor pulley bearing and result in premature failure.*

A. Engine External Washing Procedure

When the exterior surface of the engine is contaminated with salt, it should be washed clean with water prior to flight of the airplane. Demineralized water is not required for this purpose. At no time should an engine be left in a contaminated (salted) condition for any extended period of time (such as overnight).

CAUTION: *Never wash an engine while it is hot or running. After the engine has been shut down, allow it to cool for at least an hour prior to washing. Electrical components and plugs on the engine and in the engine compartment should be covered/protected during the engine wash. Use dry shop air to dry all components after washing. Do not allow water into the engine exhaust or air inlet. Do not direct high pressure water or solvent directly into electrical components or mechanical parts having air vent holes, e.g. fuel control units.*

8. LUBRICATION OF SPLINE DRIVES

When spline drives of any type are to be mated to the engine, they must be properly lubricated. Use lubricant (111, Table 6, 91-00-00) or equivalent on all tight press fit splines unless otherwise specified in the engine maintenance manual. Lubricating grease (12, Table 6, 91-00-00) or an equivalent compound should be used on all loose fit spline drives which are external to the engine and have no other means of lubrication. Take care to apply the compounds in an even thin coat only on the surfaces where they are required. Excessive lubricant may cause malfunctioning or even failure of the engine.

9. SURFACE DEICER BOOT CLEANING

The surfaces of the deicer boots should be checked for indications of engine oil after servicing and at the end of each flight. Any oil spots that are found should be removed with a non-detergent soap and water solution. Care should be exercised during cleaning to avoid scrubbing the surface of the boots, as this will tend to remove the special A56B conductive coating surface. The deicer boots are made of soft - flexible stock which may be damaged if gasoline hoses are dragged over the surface of the boots or if ladders and platforms are rested against them.

10. INSTRUMENT AIR FILTER (U-1 THRU U-151, EXCEPT U-150)

Servicing the air conditioner system consists of checking and maintaining the correct refrigerant level, compressor oil level, belt tension and condition, system leak detection and replacement of the evaporator air filters.

A filter is mounted in the instrument air lines under the floor just aft of the rear spar to provide protection for the instruments from dust, smoke and other foreign the particles in the air. The filter is a sealed disposable element which should be checked every 100 hours and changed each 400 hours or sooner, during operation in dusty or heavy smoke conditions. Replacement is as follows:

(1) Remove the floorboard on the left side of the passenger compartment just aft of the rear spar.

(2) Disconnect the line at the base of the filter housing and remove the filter housing.

(3) Discard the old filter housing and reinstall a new filter housing in place by connecting the lines at the base of the filter housing.

(4) Install the floorboard on the left side of the passenger compartment just aft of the rear spar.

11. INSTRUMENT AIR FILTER (U-50, U-152 AND AFTER)

On U-50, U-152 thru U-172, an instrument air filter is mounted on the upper right hand corner of the bulkhead in the nose compartment. On U-173 and After, the instrument air filter is recessed in the bulkhead in the nose compartment and is protected by a filter cover. The filter is incorporated in the lines to provide additional protection for the instruments from dust and other foreign particles in the air. The disposable filter unit should be checked every 100 hours and replaced every 400 hours of operation or sooner during operation in dusty conditions. Replacement is as follows:

(1) Open the right hand nose compartment door.

(2) On U-173 and After, remove the filter cover.

(3) Remove and discard the filter by rotating the filter in a counterclockwise motion.

(4) Install the new filter by rotating the filter in a clockwise motion.

(5) On U-173 and After, install the filter cover.

(6) Close the right hand nose compartment door.

12. CLEANING AIRPLANE FINISHES

Prior to cleaning, cover the wheels, making certain the brake discs are covered. Attach the pitot cover securely and plug or mask off all other openings. Be particularly careful to mask off the static air buttons before washing or waxing.

CAUTION: *High pressure washing can carry the lubricant away from the wheel bearings, uplock rollers and propeller hub bearings if directed on or into these components. Avionics equipment can also be damaged if the stream is directed into overboard cooling ducts. In power plant areas there are garlock type seals which could be unseated by a high pressure stream which, in turn, will allow contaminants to enter the sealed system. High pressure sprays should not be directed upon the wings, fuselage or tail assembly from the rear, where the structure is exposed to invasion by moisture and chemicals which can result in corrosion damage to structural members, as well as moving parts. Do not expose the aileron, rudder and elevator trim tab hinge lines or their push rod systems. After high pressure or hand washing, and at each periodic inspection, lubricate the hinge lines and push rod end fittings with a water displacement lubricant (108, 109 or 110, Table 6, 91-00-00). After washing the airplane, lubricate all lubrication points.*

Flush loose dirt away with clean water, then wash with a soft cloth or chamois, and a mild soap and water. Avoid harsh, abrasive or alkaline soaps and detergents which could cause corrosion or scratches.

Oil residue, exhaust soot and stubborn soils may be removed with solvent (33, Table 6, 91-00-00) or solvent (34, Table 6, 91-00-00). If solvent is used, care must be exercised to assure that prolonged soaking of the finish is not allowed; furthermore, even slight contact of solvent with acrylic windows or light lenses must be avoided.

Wax areas finished with epoxy paint. If a substance cannot be removed with either solvent, consult a Hawker Beechcraft Aero or Aviation Center or International Distributor or Dealer.

High pressure water hoses used in conjunction with strong cleaners, winter operation on salty runways, etc., will contribute to and aggravate corrosion problems of the landing gear braces. The landing gear (main and nose) should be washed with low-pressure water and a mild detergent as soon as practical following operation on salty or muddy runways. Blow the wash water off the landing gear with low-pressure air after washing and before flight or storage of the airplane.

On exposure to ultra violet source (sunlight) epoxies undergo a surface degradation referred to as chalking. Due to this chalking characteristic, epoxy finishes must be waxed on a regular basis. If chafing occurs, polishing and buffing will restore a good appearance. A good grade of wax should be used; however, a “cleaner" wax or polish may be used where a severe case of chalking has occurred. Even though urethane finishes should not be polished, they may be waxed, but it is not necessary for approximately two to five years after application. A build-up of wax on exterior paint finishes will yellow with age; therefore, wax should be removed periodically. Generally, solvent (33, Table 6, 91-00-00) is adequate and safe for this purpose.

CAUTION: *Do not apply wax to uncured paint. Waxes and polishes seal the paint from the air and prevent curing. Urethane and epoxy finishes cure in seven to ten days at 75°F. Wash uncured paint with cold or lukewarm water and MILD soap. Any rubbing of the paint should be done gently and held to a minimum, to avoid cracking the paint film.*

13. CLEANING PLASTIC WINDOWS

The plastic windows should be kept clean and waxed at all times. Any nonacid, nonabrasive material may be used for this purpose. Windex, Permatex, Whiz, and similar commercial window cleaners may be safely used. Automotive cleaners and waxes may be used on plastic windows only if they are nonacrylic. To prevent scratches and crazing, wash the windows carefully with plenty of soap and water, using the palm of the hand to feel and dislodge dirt and mud. A soft cloth, chamois or sponge may be used, but only to carry water to the surface. Rinse thoroughly, then dry with a clean, moist chamois. Rubbing the surface of the plastic with a dry cloth builds up an electrostatic charge which attracts dust particles in the air.

Remove oil and grease with a cloth moistened with kerosene. Never use gasoline, benzine, alcohol, acetone, carbon tetrachloride, fire extinguisher or anti-ice fluid, lacquer thinner or glass cleaner. These material will soften the plastic and may cause it to craze.

After removing dirt and grease, if the surface is not badly scratched, it should be waxed with a good grade of commercial wax. The wax will fill in minor scratches and help prevent further scratching. Apply a thin, even coat of wax and bring it to a high polish by rubbing lightly with a clean, dry, soft flannel cloth. Do not use a power buffer; the heat generated by the buffing pad may soften the plastic.

14. INTERIOR CLEANING

Frequently the seats, rug, upholstery panels and head lining should be vacuum-cleaned to remove as much surface dust as possible.

Experience has shown that commercial foam-type cleaners or “shampoos" can be used to condition the surface of rugs, carpets and upholstered materials. The upholstery is first vacuum-cleaned, stains removed, then a solution of the cleaner is prepared by mixing a small amount in a bucket of water, and beating the mixture until a heavy foam is formed.

Apply the foam uniformly over the surface to be cleaned with a brush, then remove the suds with a vacuum cleaner, or by wiping off with a brush or cloth. Since there is very little moisture in this foam, wetting of the fabric or retention of moisture in the warp does not occur. Unlacquered metal fittings and furnishings within the airplane can be cleaned with most commercial metal polishes. Use a soft, clean rag for application; then polish to a brilliant gloss with a dry cloth. Protect the finish with a good grade of wax.

A. Cleaning Upholstery

The most effective method of cleaning upholstery is directly dependent upon the type of upholstery involved. For instance, a fabric type upholstery that has been flame proofed should never be treated by the application of cleaners with a water base. The reason for this is that the natural capillary action of water in the fabric will cause the salts from the flame proofing to rise to the surface, resulting in unsightly faded spots. The most effective way to clean fabric upholstery is as follows:

NOTE: The manufacturers of wool and wool blended upholstery fabrics recommend that these materials be dry-cleaned.

(1) Brush the upholstery along the weave of the fabric with a stiff bristled brush, such as a nylon bristled fingernail brush.

(2) Vacuum the entire surface to remove any salt residue or dirt stains.

(3) Gently rub the upholstery along the weave of the fabric with a lint free cloth moistened with solvent (38, Table 6, 91-00-00). Apply the solvent sparingly and do not reinstall the upholstery until completely dry.

Leather upholstery should be kept waxed for maximum protection. For cleaning, a nonabrasive chemically neutral, nonreactive emulsion type cleaner of cream-like consistency is recommended. Dilute the cleaner with water, and apply it over the dirty surface with a sponge or soft cloth (use gentle wiping motion; do not scrub). The solution should not be allowed to stand, but should be wiped off before drying.

B. Cleaning Interior Cabin Trim

Wash the plastic used for interior trim with soap and water. Scrubbing with a brush and detergent soap will usually provide adequate cleansing; however, alcohol may also be used to remove contaminants that are soluble in alcohol.

CAUTION: *To prevent damage to the plastic used for interior trim, never use methyl propyl ketone, naphtha, mufti, solvent, gasoline, lacquer thinner, or other types of paint cleaner as a cleaning agent.*

15. VACUUM REGULATOR VALVE FILTER (U-50, U-152 AND AFTER)

The vacuum regulator valve filter is mounted on top of the vacuum regulator. On U-152 through U-164, the vacuum regulator filter is located on the aft side of the forward bulkhead at fuselage station 84.00 on the pilot's side. On U-50, U-165 and After, the vacuum regulator filter is located at fuselage station 94.00 near the left side of the fuselage. The valve is protected by a foam type filter that should be cleaned with solvent (38, Table 6, 91-00-00) at 100-hour intervals. If the vacuum in the system exceeds 5.9 inches Hg., clean the filter and recheck vacuum pressure before attempting to adjust the regulator valve.

16. PITOT SYSTEM SERVICING

Moisture in the lines is drained through a pair of drain pet cocks located immediately inside and aft of the nose baggage compartment door on each side. These drain pet cocks should be opened to release any trapped moisture following each wash down and each flight in an atmosphere containing precipitation or high humidity. After the pet cocks have emptied any moisture, position the pet cocks in the closed position.

17. ENGINE CONTROL CABLES

Engine control cables are prelubricated by the manufacturer with a special lubricant compounded for effective lubrication in the excessively low temperatures encountered during flight at high altitudes. For this reason, any oil, grease, solvent, etc., used to lubricate the sealed cables will only hinder operation; however, check cables may be periodically purged as follows with dry nitrogen to prevent moisture condensation and subsequent freezing.

NOTE: If the control cable has already been contaminated with a lubricant that congeals at low temperatures, the nitrogen purge will be ineffective, and the affected control cable must be replaced.

(1) Remove the rubber seal at each end of the control cable.

(2) Connect a dry nitrogen source, regulated at 18 psi, to the end of the control cable that is attached to the engine.

(3) Purge the control cable for 15 minutes while observing the end attached to the cockpit controls to ensure that all moisture has been purged. Continuously cycle the cable control lever throughout its full range of travel during the purging procedure.

18. ENGINE INERTIAL ANTI-ICING SYSTEM SERVICING

An inertial separator vane (ice vane) is installed in each lower forward cowling section to preclude entry of freezing rain and snow to the compressor inlet screen. A by-pass door in the lower aft cowling section provides an outlet for the ice particles and snow. Servicing will consist of lubricating the actuating linkage periodically with grease (16, Table 6, 91-00-00) to assure freedom of movement and to reduce wear.

19. CLEANING PROPELLER-REVERSING HARDWARE

As a rust preventive measure, propeller-reversing hardware may be rubbed clean with steel wool and coated with aerosol clear lacquer spray.

20. SERVICE SCHEDULE

A. Check

ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(1) Engine Oil Level	11 o'clock position of accessory gear case.	Refer to Engine Oil (1, Table 6, 91-00-00)	Preflight.
(2) Refrigerant Level	Sight gage under the baggage compartment floorboards.	Refrigerant 12 (52, Table 6, 91-00-00)	100 hours.

ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(3) Instrument Air Filter	U-1 thru U-151, except U-50 - Mounted in air pressure line under left floorboard area in passenger compartment aft of rear spar. U-50, U-152 thru U-172 - Mounted on the upper right hand corner of the bulkhead in the nose compartment. U-173 and After - Mounted in the nose compartment, recessed in the bulkhead and protected by a filter cover (U-173 and After).		100 hours.
(4) Hydraulic Landing Gear System Fluid Level	Dipstick in fill can located inboard of the LH nacelle and forward of the main spar.	Hydraulic Fluid (9, Table 6, 91-00-00)	100 hours.

B. Change

ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(1) Engine Oil	Remove forward cowlings to gain access to nose case drain. Remove fiber duct and oil cooler bypass duct to gain access to engine drain plug, oil cooler drain plug and chip detector plug. Refill at 11 o'clock position on accessory gear case.	Refer to Pratt and Whitney Service Bulletin No. 1001	Only required when oil is contaminated.

NOTE

DO NOT exceed torque of 15 to 20 inch-pounds when reinstalling oil cooler drain plug.

(2) Water/methanol Solution	Remove wing center section leading edge skin for access to drain valve in line at inboard end of reservoir. Filler cap under hinged door just aft of leading edge and inboard of nacelle.	Mixture of 45% methanol and 55% distilled or demineralized water	Drain and replace the water/ methanol solution any time system is not used for 30 days or airplane is not moved for 7 days.

C. Clean

ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(1) Engine	Access through cowling.	Cleaning agent (31, Table 6, 91-00-00)	As Required.
(2) Engine Driven Fuel Pump Screen	RH side of engine accessory section.	Dry compressed air	100 hours.

ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(3) Fuel Filters	For access to the filter on each engine, remove fiberglass duct and oil cooler bypass duct. Transfer pump filter mounted in each main wheel well. (Not installed on A99A or C99).	Clean with solvent (38, Table 6, 91-00-00) and blow dry with compressed air	100 hours.
(4) Fuel Drain Collector Pump Filter	U-1 thru U-164, except U-50 - Mounted on the RH side of the fire seal, just forward of engine accessory section.	Clean with solvent (38, Table 6, 91-00-00) and blow dry with compressed air	300 hours.
(5) Fuel Drain Collector Tank	U-1 thru U-164, except U-50 - Mounted on RH side of fireseal just forward of engine accessory section.	Drain and clean with solvent (38, Table 6, 91-00-00)	Every 12 months.
(6) Fuel Purge System Air Filter	U-50, U-165 and After - P_3 air line to purge tank.	Check for rust and corrosion and blow filter clean with shop air	Every 12 months.
(7) Fuel Purge Tank	U-50, U-165 and After - Mounted on truss mount just forward of engine accessory section.	Flush tank with solvent (38, Table 6, 91-00-00)	Every 12 months.
(8) Heater Fuel Filter	U-1 thru U-164, except U-50 - Mounted on inboard side of left hand main wheel well.	Clean with solvent (38, Table 6, 91-00-00)	100 hours.
(9) Heater Fuel Control Valve Filter	U-1 thru U-164, except U-50 - Remove large square access cover from underside of left center section just inboard of nacelle and forward of main spar.	Clean with solvent (38, Table 6, 91-00-00)	100 hours.
(10) Oil Filter (Non-Disposable)	3 o'clock position of the compressor inlet case.	Clean as outlined in the applicable Pratt and Whitney Aircraft of Canada LTD Maintenance Manual	100 hours.
(11) Hydraulic Power Pack Permanent Finger Screens	Located inside valve body attached to normal gear extend and gear retract ports.	Clean with solvent (38, Table 6, 91-00-00)	Every 100 hours for the first 300 hours and then as required.
(12) Water/methanol Filter	Located in wing center section leading edge just outboard of water/methanol reservoir.	Clean by back flushing with methanol	Every 12 months.

D. Service

	ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(1)	Air Conditioner Compressor	Access panel in inboard right nacelle area.	Refrigerant Oil (6, Table 6, 91-00-00)	Whenever the system is recharged with refrigerant.
(2)	Battery	Nose area or wing RH center section forward of the main spar.	Distilled water	Check and service every 100 flight hours or 30 days whichever occurs first. Refer to Chapter 24.
(3)	Oxygen Supply Cylinders	U-1 thru U-238 - Service through nose wheel well. U-239 and After - Service through baggage compartment.	Oxygen (22, Table 6, 91-00-00)	As required.
(4)	Brake Fluid Reservoir	Upper left hand corner of the bulkhead in forward baggage compartment.	Hydraulic fluid (9, Table 6, 91-00-00)	As required.
(5)	Nacelle Fuel Tanks	Access panel in top of nacelle aft of the firewall.	Refer to Table 202, 12-20-00	Preflight.
(6)	Wing Fuel Tanks	Access panel near leading edge of outboard wings	Refer to Table 202, 12-20-00	Preflight.
(7)	Main Landing Gear Struts		Hydraulic fluid (9, Table 6, 91-00-00)	100 hours.
(8)	Nose Landing Gear Strut	Filler plug at top of nose gear strut.	Hydraulic fluid (9, Table 6, 91-00-00)	100 hours.
(9)	Shimmy Damper	Mounted at upper knee of nose landing gear.	Hydraulic fluid (9, Table 6, 91-00-00)	100 hours.
(10)	Hydraulic Landing Gear System Accumulator	Under the center aisle floorboard adjacent to the power pack.	Charge accumulator to 600 ± 30 psig (U-1 thru U-164, except U-50) or 750 ± 38 psig (U-50, U-165 and After) using dry compressed air or bottled nitrogen	Every 100 hours for the first 200 hours and then as required.

E. Drain

	ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(1)	Fuel Strainer	Pull ring located on right side of firewall.		Preflight.
(2)	Wing Center Section Fuel Tank	Drain cock on underside of wing center section, adjacent to the fuselage.		Preflight.

ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(3) Wing Fuel Tank	U-1 thru U-164, except U-50 - Drain cock on under side of outboard wing just forward of main spar.		Preflight.
(4) Boost Pumps and Nacelle Fuel Tanks	Drain cocks on underside of nacelle just forward of wheel well.		Preflight.
(5) Gravity Feed Line	Forward drain valve in the manifold is located: U-1 thru U-79, except U-50, between boost pump drains. U-50, U-80 and After, aft portion of the wheel well.		Preflight.
(6) Transfer Pump Drain	Aft drain valve in the manifold is located: U-1 thru U-79, except U-50, between boost pump drains (99 only). U-50, U-80 and After, in aft portion of the wheel well (99, 99A, B99 and C99).		Preflight.
(7) Pitot Line Drain	In pitot lines, access is gained through forward baggage compartment, immediately aft of the compartment opening.		100 hours.
(8) Water/methanol System	Drain valve located in plumbing line at inboard end of water/methanol reservoir in wing center section leading edge.		At end of operating season.

F. Replace

ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
(1) Engine Oil Filter (Disposable)	3 o'clock position of the compressor inlet case.		200 hours.
(2) Forward Evaporator Filter	Access panel in forward baggage compartment, left aft corner floorboard.		300 hours, or more often if required by operating conditions.
(3) Aft Evaporator Filter	Right rear floorboard area in passenger compartment: U-1 thru U-164, except U-50. Center floorboard area: U-50, U-165 and After.		300 hours, or more often if required by operating conditions.
(4) Instrument Air Filter	U-1 thru U-151, except U-50 - Mounted in air pressure lines under floorboard area in passenger compartment on left side, aft of the rear spar.		400 hours or sooner if operated in heavy dust or smoke conditions.

ITEM	LOCATION	SERVICE WITH	INTERVAL HRS
	U-50, U-152 thru U-172 - Mounted in the upper right portion of the bulkhead in the nose compartment.		400 hours or sooner if operated in dusty conditions.
	U-173 and After - Mounted in the nose compartment, recessed in the bulkhead and protected by a filter cover. (U-173 and After).		400 hours or sooner if operated in dusty conditions.
(5) Hydraulic Landing System Filter (Disposable)	Valve body end of power pack.		Every 100 hours for the first 300 hours and then as required.

21. LUBRICATION SCHEDULE

A. Engine Lubrication

NOTE: Refer to Figure 201 and the Table shown below for information on lubricating the airplane.

Engine Lubrication

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	ENGINE		
(1)	Control Rod Ends	Grease (16, Table 6, 91-00-00)	100 Hours.
(2)	Cam Plate and Pins	Lubricating grease (13, Table 6, 91-00-00)	100 Hours.
(3)	Generator Drive Shaft Spline (U-1 thru U-164, except U-50)	Lubricant (111, Table 6, 91-00-00)	200 Hours and anytime the generator is removed (Some may require more frequent lube intervals).
(4)	Air Conditioner Quill	Lubricant (111, Table 6, 91-00-00)	600 Hours.

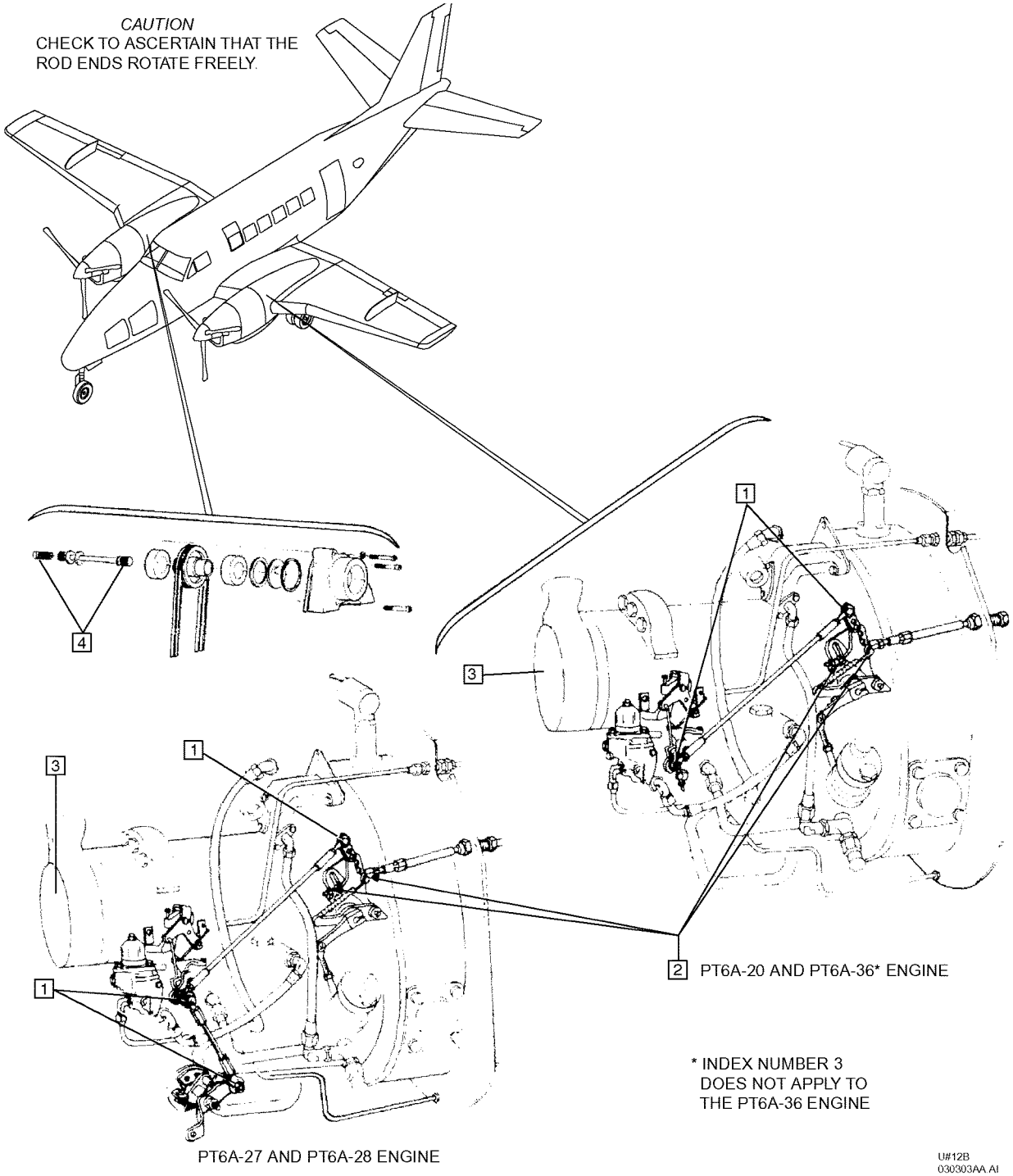


Figure 201
Engine Lubrication

B. Engine Controls and Propeller Lubrication

NOTE: Refer to Figure 202 and the Table shown below for information on lubricating the airplane.

Engine Controls and Propeller Lubrication

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	ENGINE CONTROLS		
(1)	Linkage (All moving parts)	Grease (18, Table 6, 91-00-00)	100 Hours.
	NOTE		
	Do not lubricate engine control cables under any circumstances. These cables are prelubricated by the manufacturer in accordance with their specifications. The manufacturer recommends that no further lubrication be applied.		
	PROPELLER		
(2)	Propeller Hub	Grease (16, Table 6, 91-00-00)	100 Hours.
	NOTE		
	When lubricating Hartzell hubs, remove one Zerk in each blade clamp of the hub (this allows excess pressure to vent before the seals rupture) before pumping grease into the other Zerk. Install the removed Zerks.		
(3)	Low Pitch Stop Rods	Marvel Mystery Oil (7, Table 6, 91-00-00)	100 Hours.

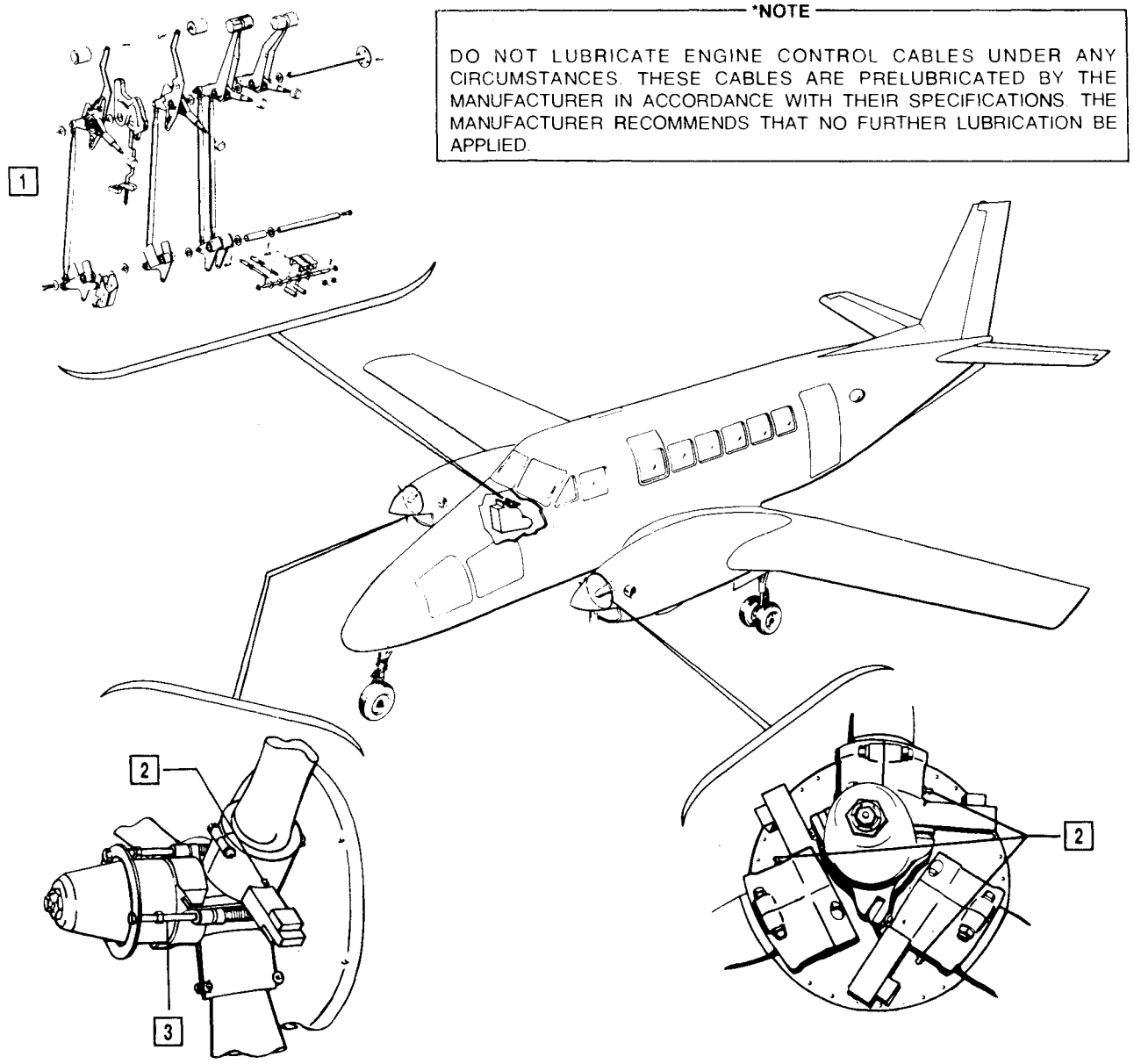


Figure 202
Engine Controls and Propeller Lubrication

C. Control Column, Rudder Pedals And Bellcranks, Rudder Tab System, Strobe Light System (Non-motorized) and Landing Gear Emergency Extension Lubrication

NOTE: Refer to Figure 203 and the Table shown below for information on lubricating the airplane.

Control Column, Rudder Pedals And Bell Cranks, Rudder Tab System, Strobe Light System (Non-motorized) and Landing Gear Emergency Extension Lubrication

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	CONTROL COLUMN		
(1)	Chain	Preclean chain with a cloth dampened with Stoddard Solvent ((Mineral Spirts). Lubricate chain with SAE 30W Mineral Oil. Wipe off excess oil.	500 Hours.
	RUDDER PEDALS AND BELL CRANKS		
(2)	Pedal and Bellcrank Linkage	Oil (3, Table 6, 91-00-00)	200 Hours.
	RUDDER TAB SYSTEM		
(3)	Rudder Trim Tab Tube	Oil (3, Table 6, 91-00-00)	100 Hours.
(4)	Rudder Trim Tab Actuator	Grease (16, Table 6, 91-00-00)	200 Hours.
(5)	Rudder Trim Hinge	Lubricating Oil (2, Table 6, 91-00-00)	100 Hours.
	STROBE LIGHT SYSTEM (NON-TRANSISTORIZED)		
(6)	Timer motor (cams and bearings) (U-1 thru U-99, except U-50)	Lubricating Oil (8, Table 6, 91-00-00)	500 Hours.
	LANDING GEAR EMERGENCY EXTENSION		
(7)	Extension Handle	Oil (3, Table 6, 91-00-00)	100 Hours.

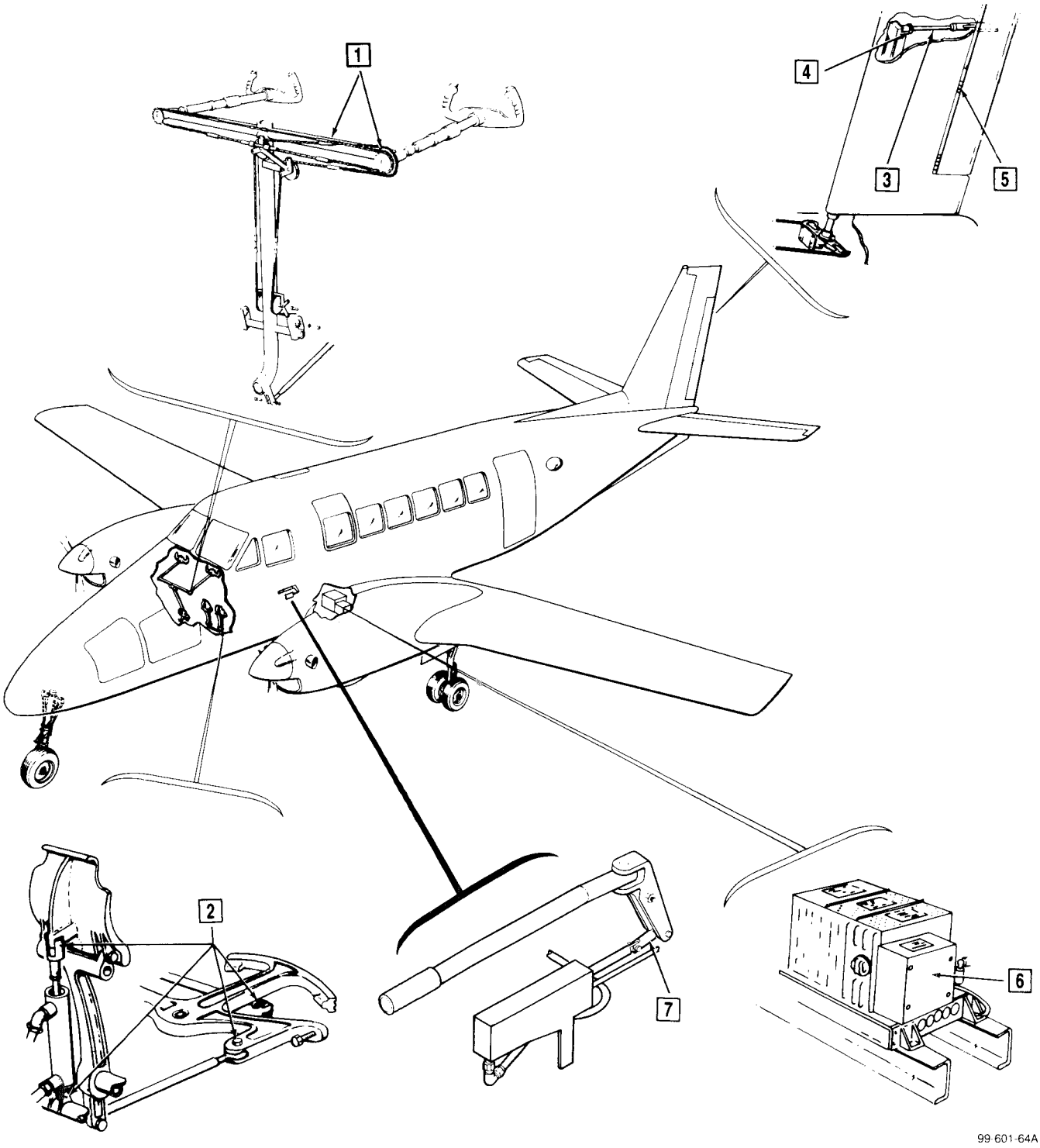


Figure 203
Control Column, Rudder Pedals And Bellcranks, Rudder Tab System, Strobe Light System (Non-motorized) and Landing Gear Emergency Extension Lubrication

D. Aileron Control System and Flap Control System Lubrication

NOTE: Refer to Figure 204 and the Table shown below for information on lubricating the airplane.

Aileron Control System and Flap Control System Lubrication

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	AILERON CONTROL COLUMN		
(1)	Aileron Quadrant	Oil (3, Table 6, 91-00-00)	200 Hours.
(2)	Control Cables	Corrosion Preventive Compound (88, Table 6, 91-00-00). Brush on and wipe off excess with a dry cloth.	Whenever control cables are to be reinstalled in the airplane.
(3)	Aileron Bellcranks	Oil (3, Table 6, 91-00-00)	200 Hours.
(4)	Trim Tab Actuator	Grease (16, Table 6, 91-00-00)	200 Hours.
(5)	Trim Tab Hinge	Lubricating Oil (2, Table 6, 91-00-00)	100 Hours.
(6)	Trim Tab Push Rods	Lubricating Oil (2, Table 6, 91-00-00)	100 Hours.
	FLAP CONTROL SYSTEM		
(7)	Flap Actuator Pistons	Lubricating Oil (5, Table 6, 91-00-00)	As required.
(8)	Flap Actuator 90° Drives	Lubricating grease (18, Table 6, 91-00-00)	1,000 Hours.
(9)	Flap Motor Gearbox	Grease (20, Table 6, 91-00-00)	1,000 Hours.

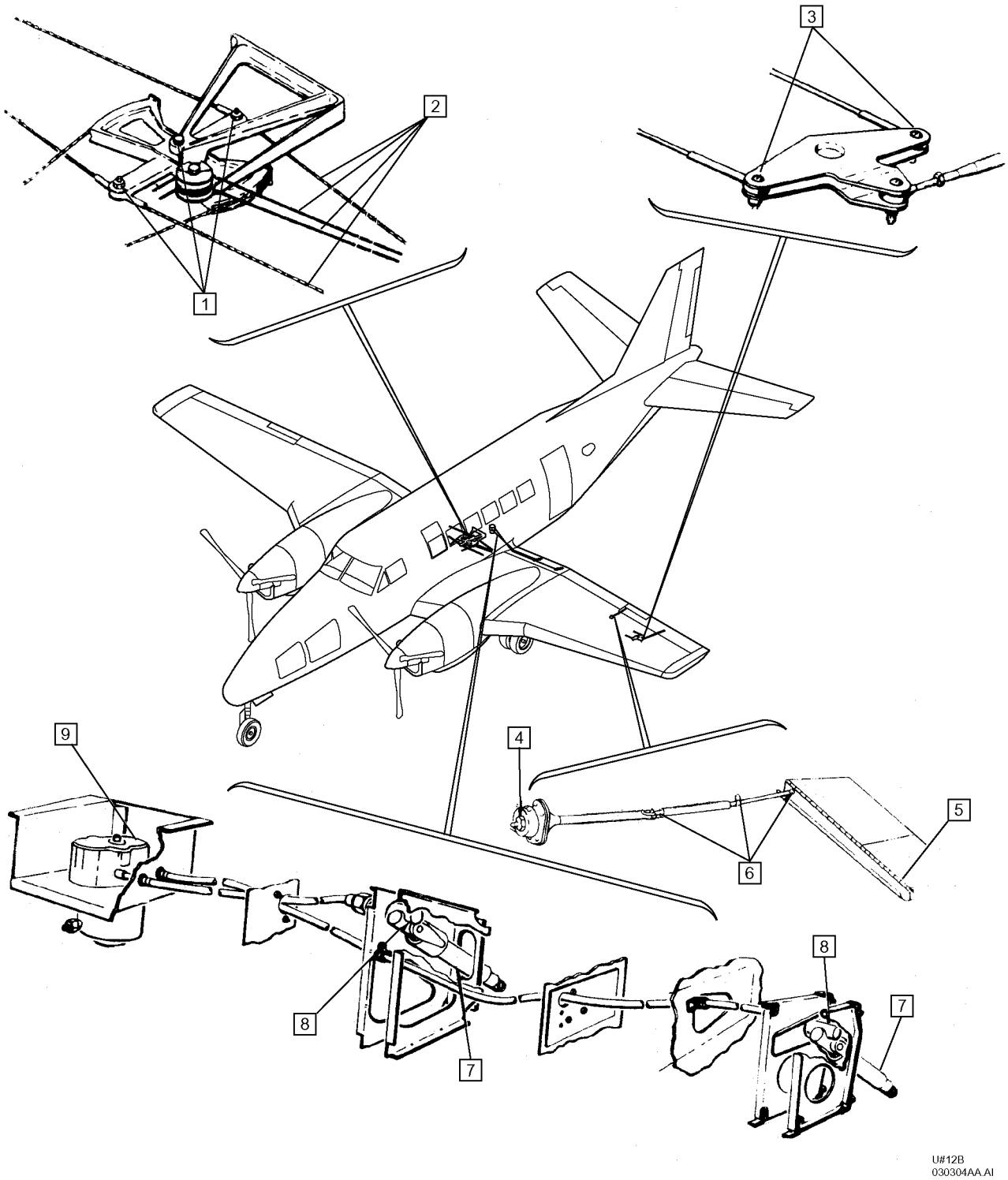


Figure 204
Aileron Control System and Flap Control System Lubrication

E. Landing Gear Retract Mechanical System (U-1 thru U-164, except U-50) Lubrication

NOTE: Refer to Figure 205 and the Table shown below for information on lubricating the airplane.

Landing Gear Retract Mechanical System Lubrication (U-1 thru U-164, except U-50)

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	LANDING GEAR RETRACT MECHANICAL SYSTEM (U-1 thru U-164, except U-50)		
(1)	Flotation Gear Door Eye Bolts and Bushings	Metal Protector (74, Table 6, 91-00-00)	100 Hours.
(2)	Retract Chains	Mineral Oil (83, Table 6), 91-00-00)	100 Hours.
(3)	Emergency Extension Mechanism	Lubricating Oil (3, Table 6, 91-00-00)	100 Hours.
(4)	Torque Tube Splines	Grease (16, Table 6, 91-00-00)	1,000 cycles.
(5)	Chain Idler Sprocket Needle Bearings	Grease (16, Table 6, 91-00-00)	1,000 cycles or 30 months, whichever occurs first or whenever the nose gear actuator is removed.

NOTE

After washing the airplane, lubricate all lubrication points.

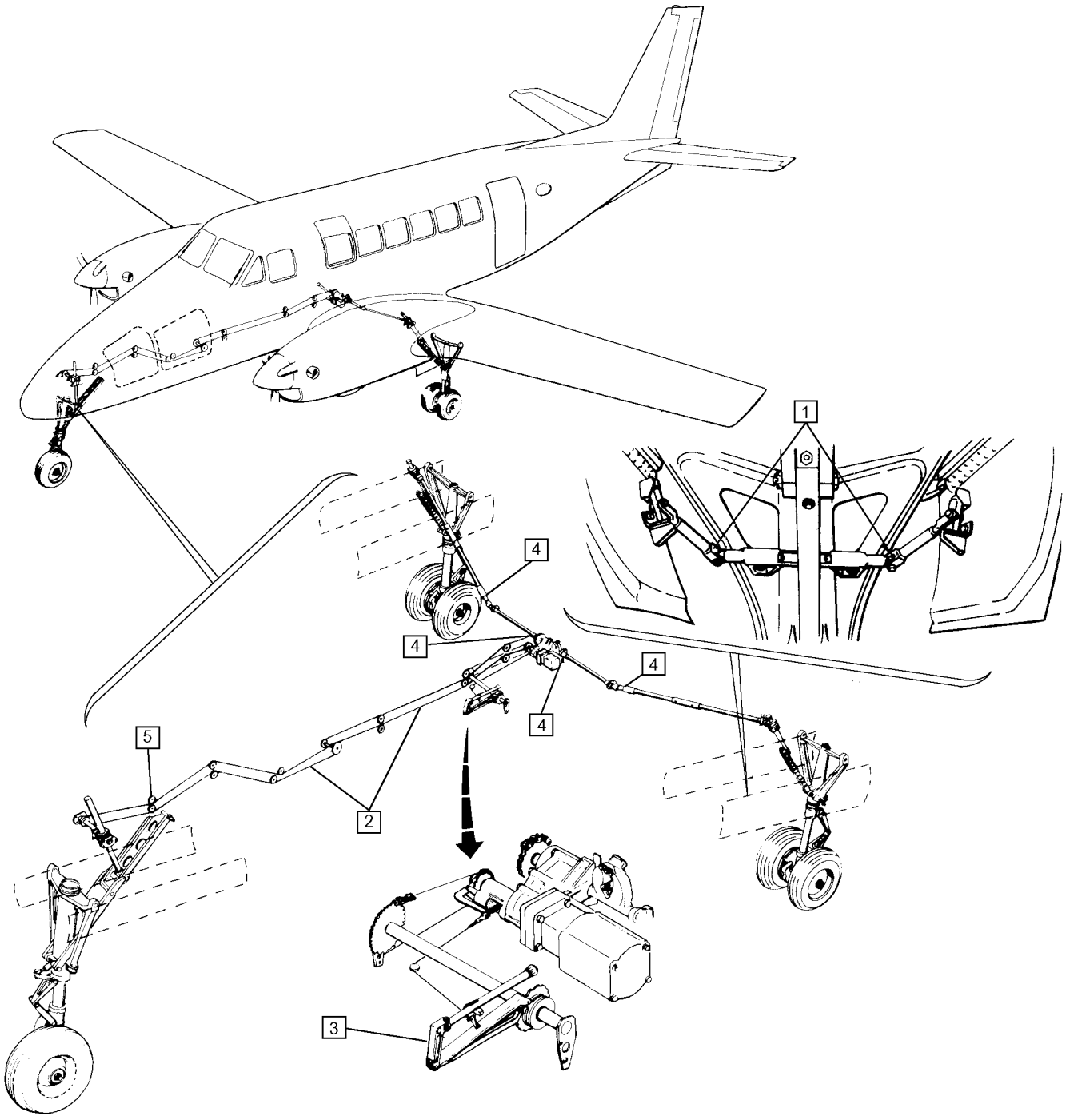


Figure 205
Landing Gear Retract Mechanical System Lubrication (U-1 thru U-164, except U-50)

F. Nose Landing Gear Lubrication

NOTE: Refer to Figure 206 and the Table shown below for information on lubricating the airplane.

Nose Landing Gear Lubrication

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	NOSE LANDING GEAR		
(1)	Door Hinges and Retract Linkage	Lubricating Oil (3, Table 6, 91-00-00)	100 Hours.
(2)	Grease Fittings	Grease (17, Table 6, 91-00-00)	50 Hours.
(3)	Wheel Bearings	Grease* (112, Table 6, 91-00-00)	100 Hours.
(4)	Nose Wheel Steering Mechanism	Grease (17, Table 6, 91-00-00)	50 Hours.
(5)	Retract Actuator Jackscrew (Mechanical Gear)	Grease (18, Table 6, 91-00-00)	1,000 cycles or 30 months, whichever occurs first.
(6)	Actuator Support Bearings	Grease (16, Table 6, 91-00-00)	1,000 cycles or 30 months, whichever occurs first.
(7)	Drag Leg Bolts and Bushings **	Metal Protector (74, Table 6, 91-00-00)	6 Months.
(8)	Grease Fittings ***	Grease (17, Table 6, 91-00-00)	50 Hours.
(9)	Hinge Bolts and Bushings **	Metal Protector (74, Table 6, 91-00-00)	6 Months.
(10)	Grease Fittings ***	Grease (17, Table 6, 91-00-00)	50 Hours.
(11)	Shimmy Damper Bolt	Metal Protector (74, Table 6, 91-00-00)	6 Months.
(12)	Actuator Yoke Grease Fitting	Grease (17, Table 6, 91-00-00)	50 Hours.
(13)	Actuator Yoke Plate Grease Fitting	Grease (17, Table 6, 91-00-00)	50 Hours.
(14)	Brace Upper and Lower Bearings	Aeroshell #17 Grease	50 Hours.

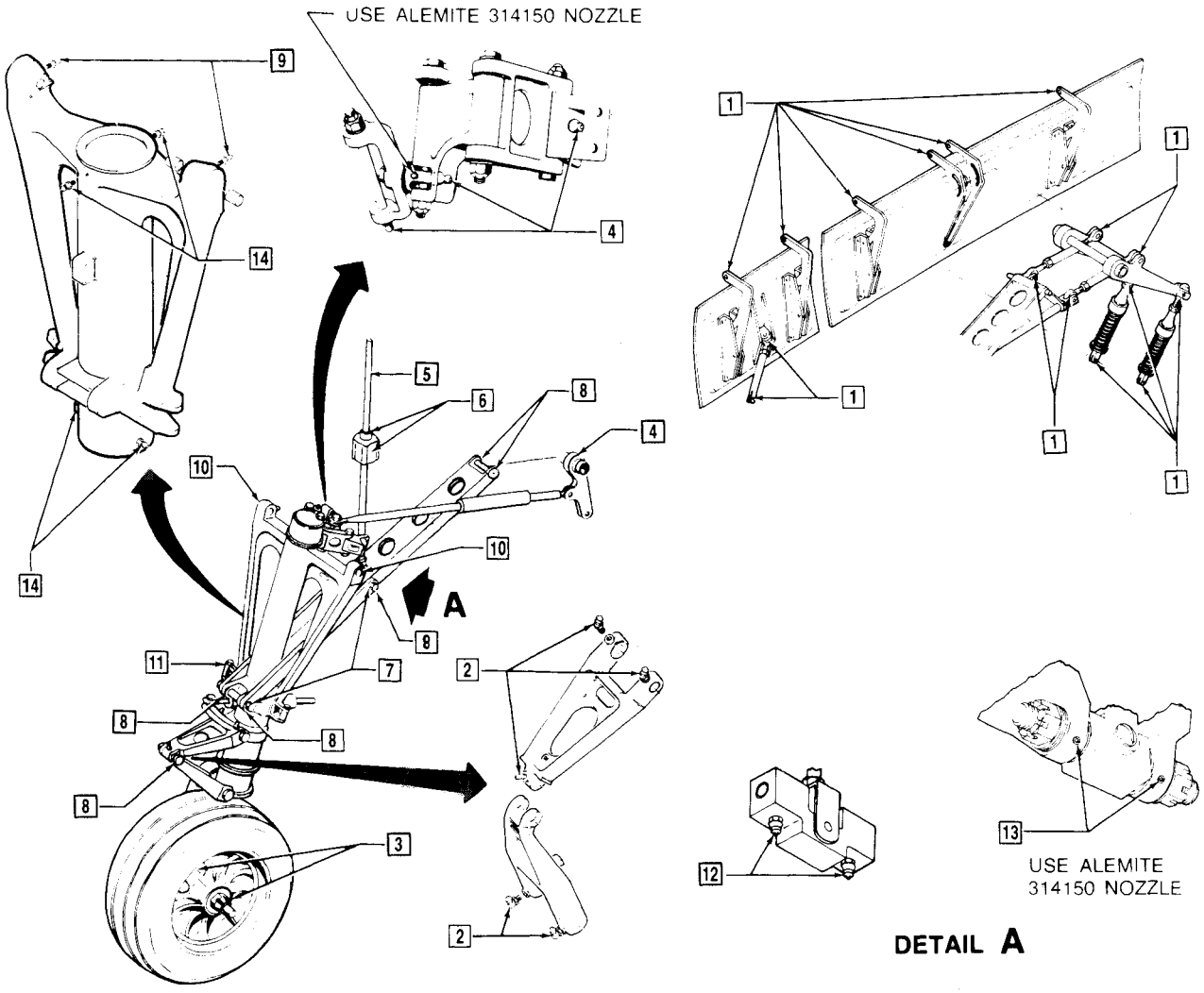
* Aeroshell Grease 5 preferred, but MIL-G-81322 may be used. Do not mix greases.

** Prior to U-152, except U-50 and U-149 that have not incorporated Service Instructions No. 0516-200.

*** U-50, U-149, U-152 and After and earlier serials that have incorporated Service Instructions No. 0516-200.

NOTE

After washing the airplane, lubricate all lubrication points



NOTE

WHEN LUBRICATING THE LOWER AND THE UPPER BEARINGS ON THE NOSE BRACE, PLACE THE NOSE WHEEL ON A SWIVEL PLATE OR JACK UP THE AIRPLANE AND PARTIALLY RETRACT THE LANDING GEAR AND TURN THE NOSE WHEEL TIRE FROM SIDE-TO-SIDE WHILE INJECTING THE GREASE. THIS WILL ASSURE COMPLETE LUBRICATION OF THE BEARINGS. NOTE THERE ARE TWO LUBRICATION FITTINGS FOR BOTH THE UPPER AND THE LOWER BEARINGS

NOTE

TO LUBRICATE THE CENTER ATTACHMENT BOLTS AND BUSHINGS OF THE NOSE GEAR DRAG LEG ON U-50, U-149, U-152 AND AFTER AND ON EARLIER AIRPLANES REWORKED TO BEECHCRAFT SERVICE INSTRUCTIONS NO. 0516-200, USE THE ALEMITE Z-737 NOZZLE, INCLUDED IN THE LOOSE TOOLS AND ACCESSORIES OF THE AIRPLANE, ON THE FLUSH TYPE FITTING.

99-604-44
001

Figure 206
Nose Landing Gear Lubrication

G. Main Landing Gear Lubrication

NOTE: Refer to Figure 207 and the Table shown below for information on lubricating the airplane.

Main Landing Gear Lubrication

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	MAIN LANDING GEAR		
(1)	Retract Actuator (Mechanical)	Grease (18, Table 6, 91-00-00)	1,000 cycles or 30 months, whichever occurs first.
(2)	Actuator Support Bearings	Grease (16, Table 6, 91-00-00)	1,000 cycles or 30 months, whichever occurs first.
(3)	“H” Link Attach Bolt	Grease (17, Table 6, 91-00-00)	50 Hours.
(4)	Door Hinges and Retract Linkage	Lubricating Oil (3, Table 6, 91-00-00)	100 Hours.
(5)	Drag Leg Bolts and Bushings	Metal Protector (74, Table 6, 91-00-00)	100 Hours.
(6)	Grease Fittings **	Grease (17, Table 6, 91-00-00)	50 Hours.
(7)	Hinge Bolts and Bushings **	Metal Protector (74, Table 6, 91-00-00)	100 Hours.
(8)	Grease Fittings ***	Grease (17, Table 6, 91-00-00)	50 Hours.
(9)	Torque Knee Bolt and Bushings ****	Metal Protector (74, Table 6, 91-00-00)	100 Hours.
(10)	Grease Fittings *****	Grease (17, Table 6, 91-00-00)	50 Hours.
(11)	Drag Leg (U-50, U-165 and After)	Metal Protector (74, Table 6, 91-00-00)	100 Hours.
(12)	Wheel Bearings	Grease* (112, Table 6, 91-00-00)	100 Hours.
(13)	Flotation Gear Door Eye Bolts and Bushings	Metal Protector (74, Table 6, 91-00-00)	100 Hours.
(14)	Flotation Gear Door Bracket	Grease (17, Table 6, 91-00-00)	50 Hours.

* Grease (112, Table 6, 91-00-00) preferred, but Grease (17, Table 6, 91-00-00) may be used. Do not mix greases.

** Prior to U-152, except U-50 and U-149 that have not incorporated Service Instructions No. 0516-200.

*** U-50, U-149, U-152 and After and earlier serials that have incorporated Service Instructions No. 0516-200.

**** Prior to U-149, except U-50, that have not incorporated Service Instructions No. 0516-200.

***** U-50, U-149 and After and earlier serials that have incorporated Service Instructions No. 0516-200.

NOTE

After washing the airplane, lubricate all lubrication points

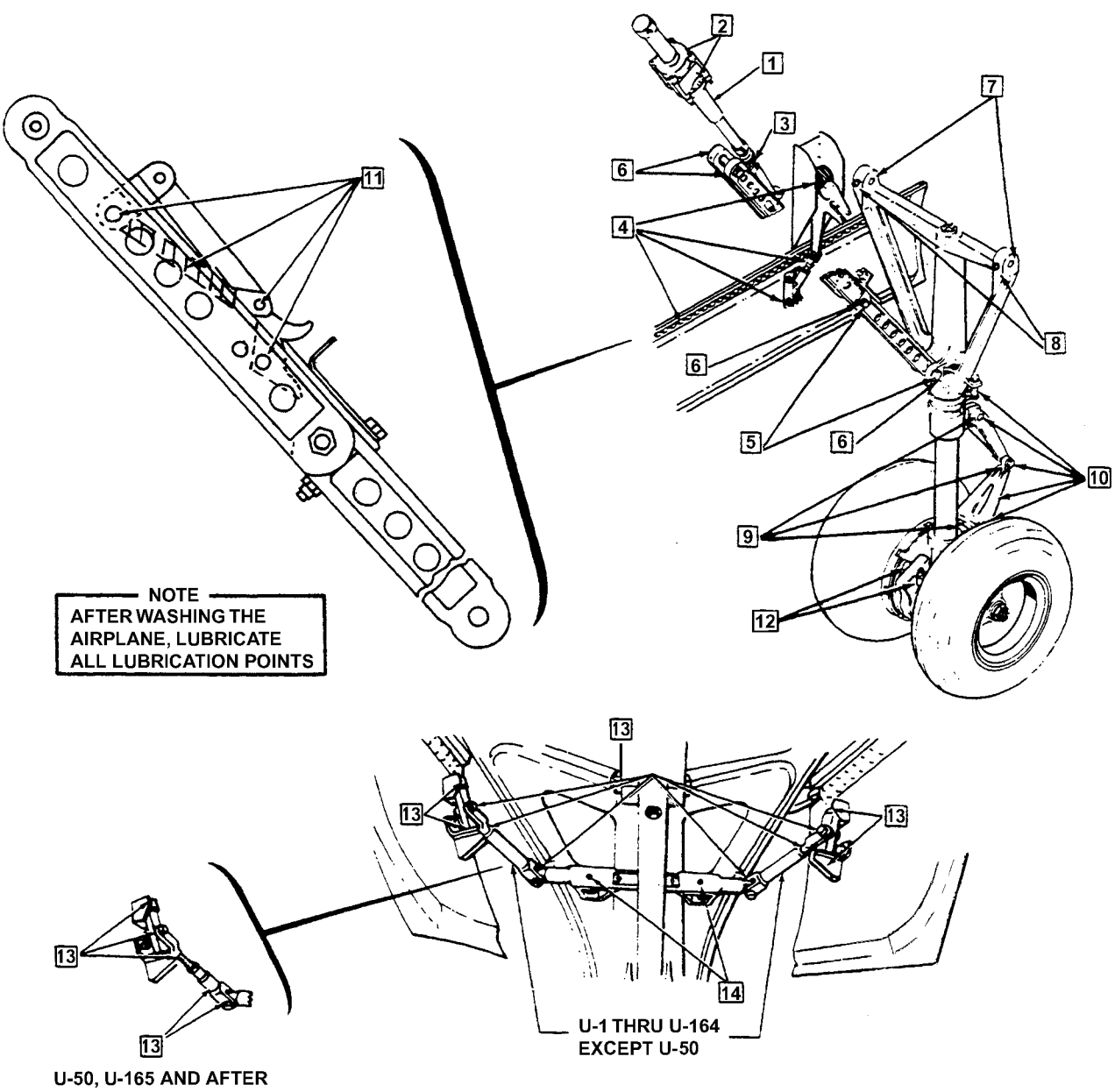


Figure 207
Main Landing Gear Lubrication

H. Emergency Exit Doors and Cabin Door Lubrication

NOTE: Refer to Figure 208 and the Table shown below for information on lubricating the airplane.

Emergency Exit Doors and Cabin Door Lubrication

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	EMERGENCY EXIT DOORS		
(1)	Tracks, Guides and Latches	Molykote 505 Paste Product	500 Hours.
	CABIN DOOR		
(2)	Latching Mechanism	Lubricating Oil (3, Table 6, 91-00-00)	100 Hours.

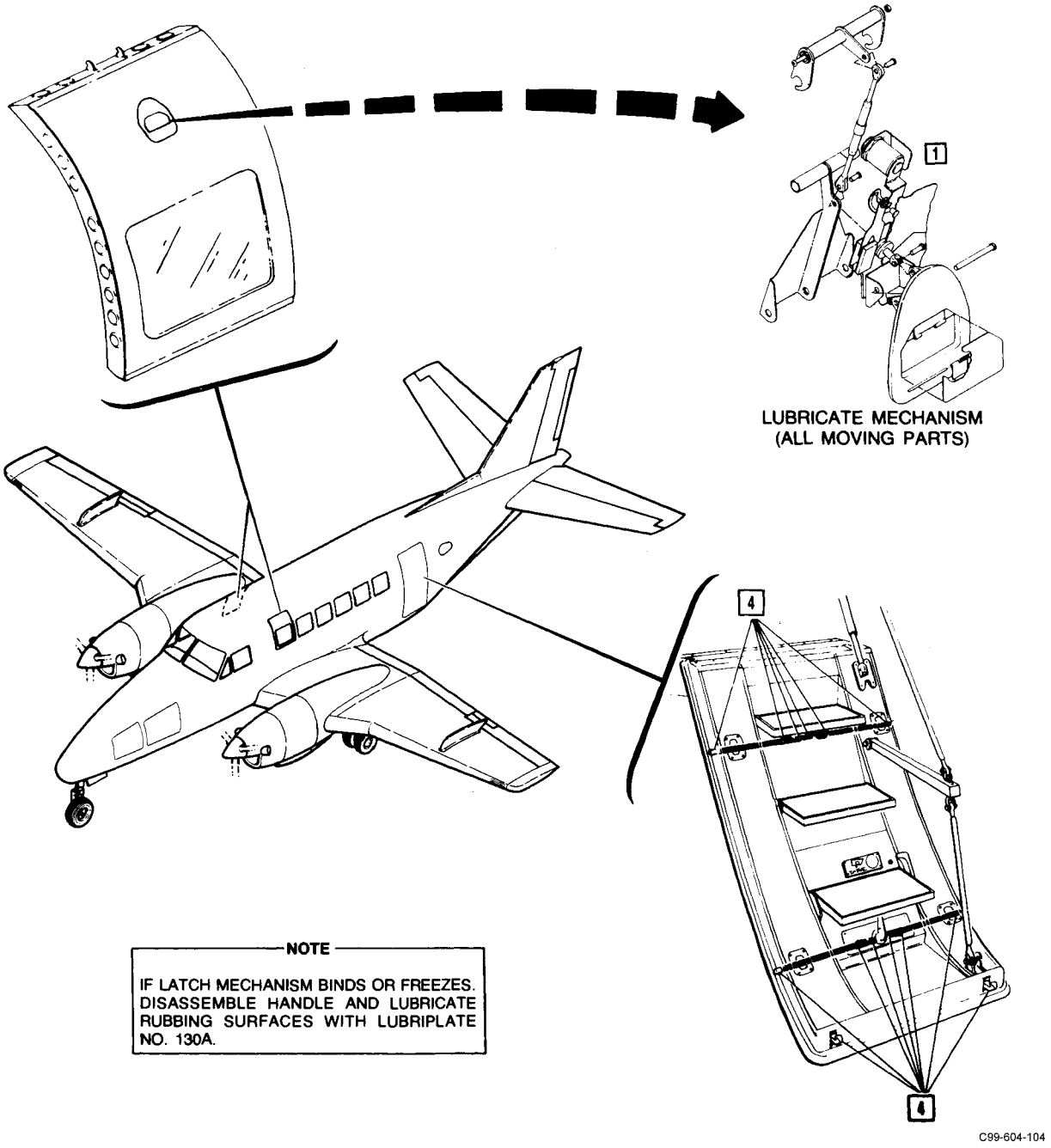


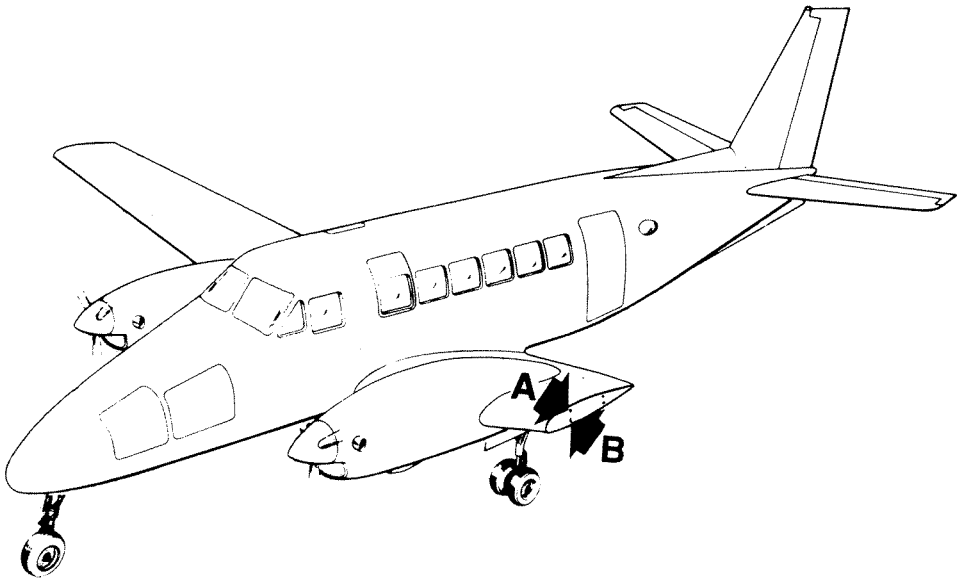
Figure 208
Emergency Exit Doors and Cabin Door Lubrication

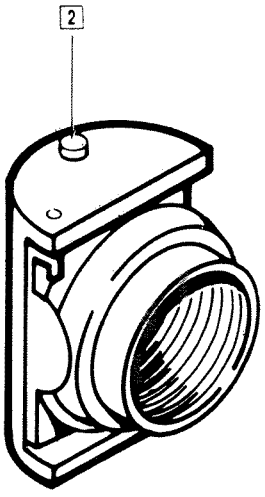
I. Wings Lubrication

NOTE: Refer to Figure 209 and the Table shown below for information on lubricating the airplane.

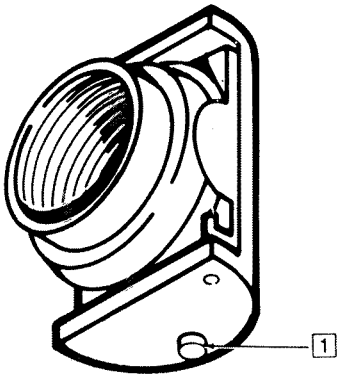
Wing Lubrication

INDEX NO.	ITEM	LUBRICATION	INTERVAL HRS.
	WINGS		
(1)	Lower Forward Wing Bolt Barrel Nut with Grease Fitting (U-165 and After) and prior airplanes with Kit No. 99-4023-1S installed	Corrosion Preventive Compound (88, Table 6, 91-00-00)	First scheduled inspection after wing installation and annually thereafter.
(2)	Upper Forward Wing Bolt Barrel Nut with Grease Fitting (U-165 and After)	Corrosion Preventive Compound (88, Table 6, 91-00-00)	First scheduled inspection after wing installation and annually thereafter.





DETAIL A



DETAIL B

U#12B
030094AA

Figure 209
Wing Lubrication

22. LUBRICATION OF THREADS

All plumbing fittings on external lines, including their points of attachment at the engine and other components, and the threads on the oil pressure and torque pressure transmitters shall be lubricated with the proper lubricant as specified in Table 4, 91-00-00. When applying thread lubricants, the following rules should be observed:

(1) Clean threads before applying lubricant.

(2) Use only thin coats of selected thread lubricant.

(3) Apply lubricant to male threads only (Ref. Figure 210).

(4) On straight thread fittings, lubricate the first 3 threads.

(5) On tapered thread fittings, do not lubricate first 2 threads. Apply lubricant to next 3 threads only.

(6) Never allow lubricant to enter fittings or flare areas.

(7) Engine fittings should be lubricated only with the fluid transported in the line, i.e. engine oil, fuel. Air lines are not lubricated.

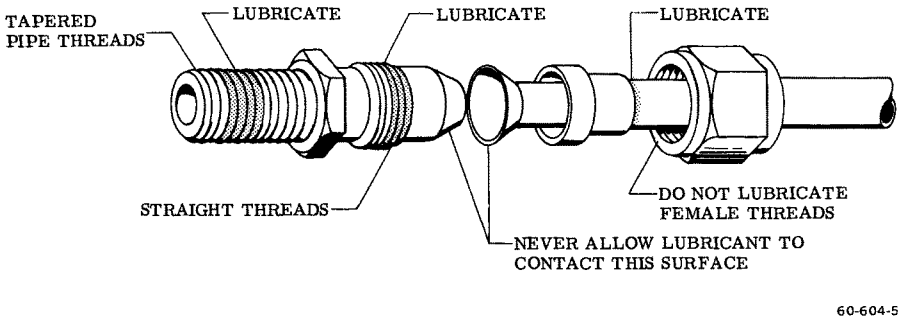


Figure 210
Lubrication of Threads

23. LUBRICATION OF THREADS

Gaskets and packings requiring lubrication are to be lubricated with the fluid type which they are sealing.

24. LANDING GEAR SHOCK ABSORBERS

Gaskets and packings requiring lubrication are to be lubricated with the fluid type which they are sealing.

WARNING: Release the air pressure entirely before removing the valve adapter.

To check the fluid level in the landing gear shock absorbers, deflate the strut by releasing the air through the valve, then remove the filler valve adapter. The fluid level should be at the bottom of the valve standpipe with the struts fully compressed. If the level is low, add hydraulic fluid (9, Table 6, 91-00-00) to reach the standpipe. Work the strut slightly to eliminate any trapped air, then add more fluid as necessary.

CAUTION: *Damage to the torque knees may result if the struts are inflated while the airplane is on jacks.*

With the airplane empty except for fuel and oil, inflate the nose strut until the piston is extended 3 to 3-1/2 inches and the main strut until the piston is extended 4-1/4 to 4-1/2 inches (4-1/2 to 4-3/4 inches with flotation landing gear).

25. NOSE LANDING GEAR ACTUATOR LUBRICATION

Refer to Chapter 32-30-00 for lubrication and inspection procedures.

26. MAIN LANDING GEAR ACTUATOR LUBRICATION

Refer to Chapter 32-30-00 for lubrication and inspection procedures.

27. LANDING GEAR SHIMMY DAMPER

To check, the fluid level in the shimmy damper, insert a wire, approximately 1/16 inch in diameter, through the hole in the washer at the aft end of the piston rod until the wire touches the bottom of the hole in the floating piston. Mark the wire, remove it and measure the depth of insertion. Inserting the wire in the hole of the floating piston, rather than letting it rest against the face of the piston will give a more accurate check. The wire can be inserted approximately 5-1/4 inches when the reservoir is empty.

NOTE: To determine if the wire is inserted in the hole of the floating piston, insert the wire several times, noting each insertion depth. When the wire is in the hole, the depth will be approximately 1/2 inch greater.

When the shimmy damper is full, the distance from the end of the piston rod to the bottom of the hole in the aft floating piston is 2-7/8 inches. When the distance is 0.50 inch from full (i.e. 3-1/8 to 3-3/8 inches) the shimmy damper should be serviced with hydraulic fluid (9, Table 6, 91-00-00) as follows:

(1) Remove the shimmy damper.

(2) Remove the piston rod end fitting and secure the shimmy damper in a fixed position with the fitting end of the piston rod down.

WARNING: The spring in the piston rod is under compression. Remove the snap ring and washer carefully.

(3) Remove the snap ring, disc and spring from the aft end of the piston rod.

(4) Remove the aft floating piston from the piston rod by screwing a long 10-32 bolt into the piston and pulling it out.

(5) Fill the piston rod with fluid.

(6) Insert the 10-32 bolt through the fitting end of the piston rod and engage the forward floating piston. Pull the piston toward the fitting end of the piston; at the same time, insert the aft floating piston and spring and install the disc and snap ring.

(7) Remove the bolt from the forward floating piston and install the piston rod end fitting.

28. SPECIAL TOOLS

Figure 211 provides information on the various special tools used on this airplane.

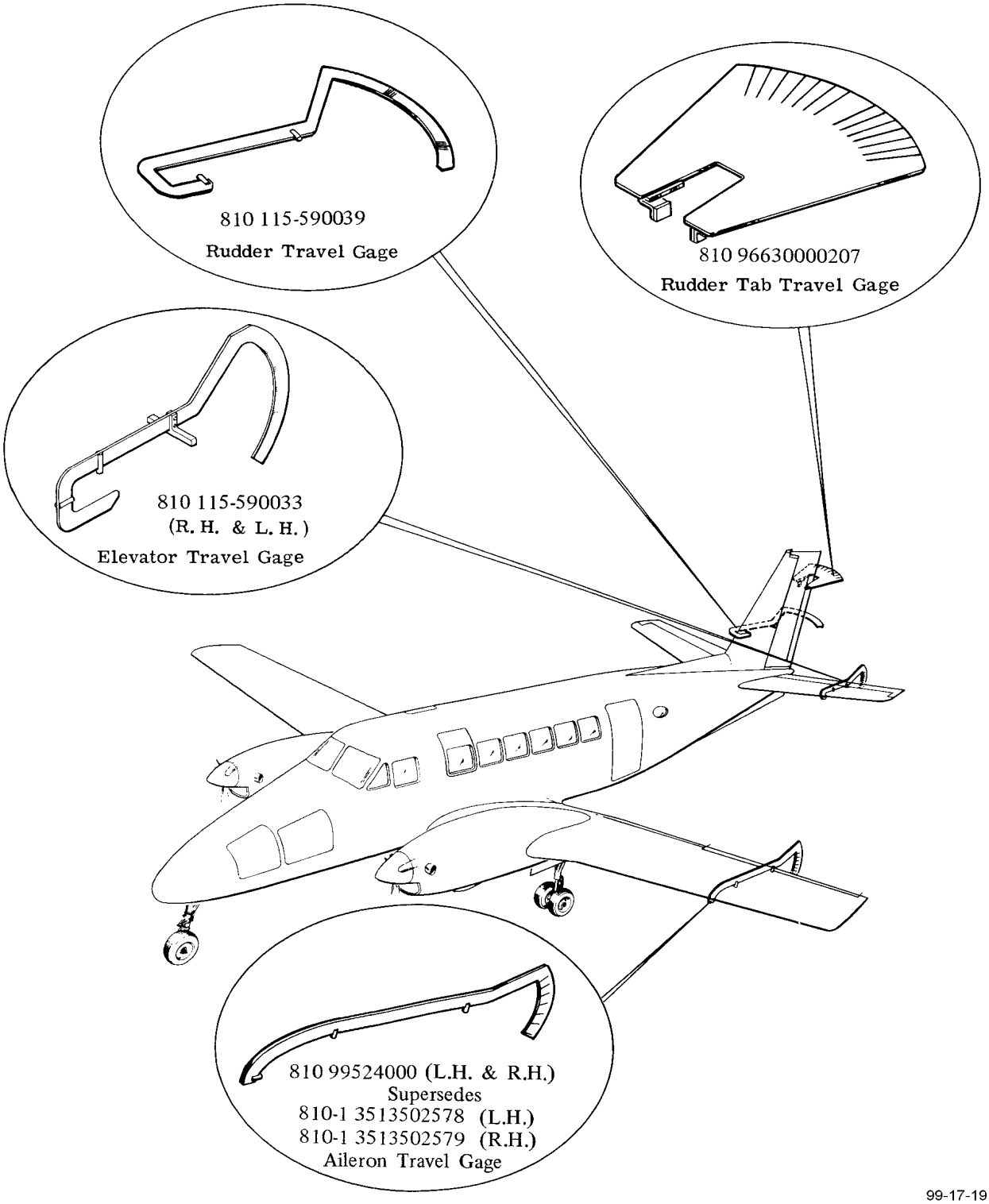


Figure 211 (Sheet 1 of 6)
Special Tools

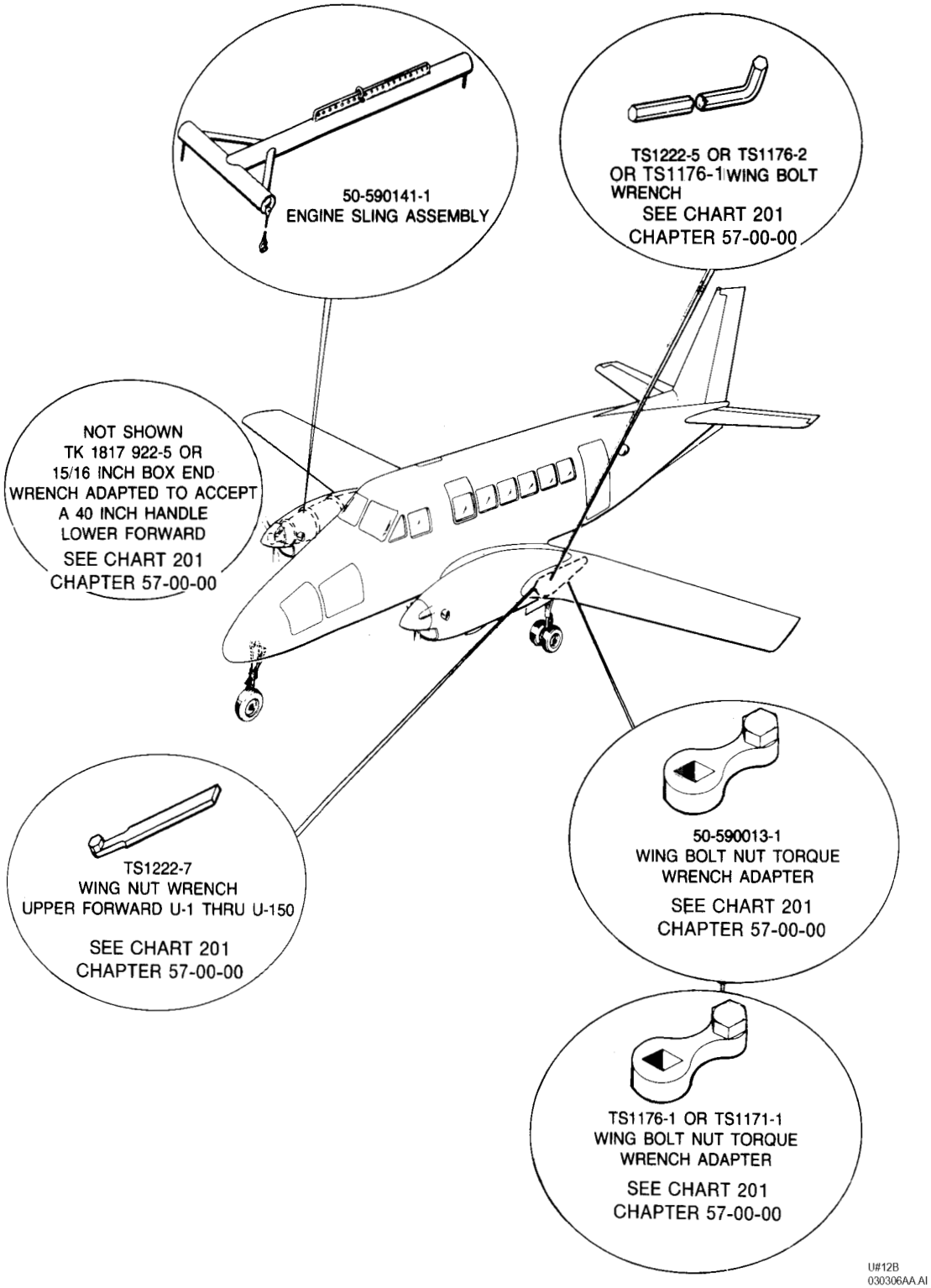


Figure 211 (Sheet 2 of 6)
Special Tools

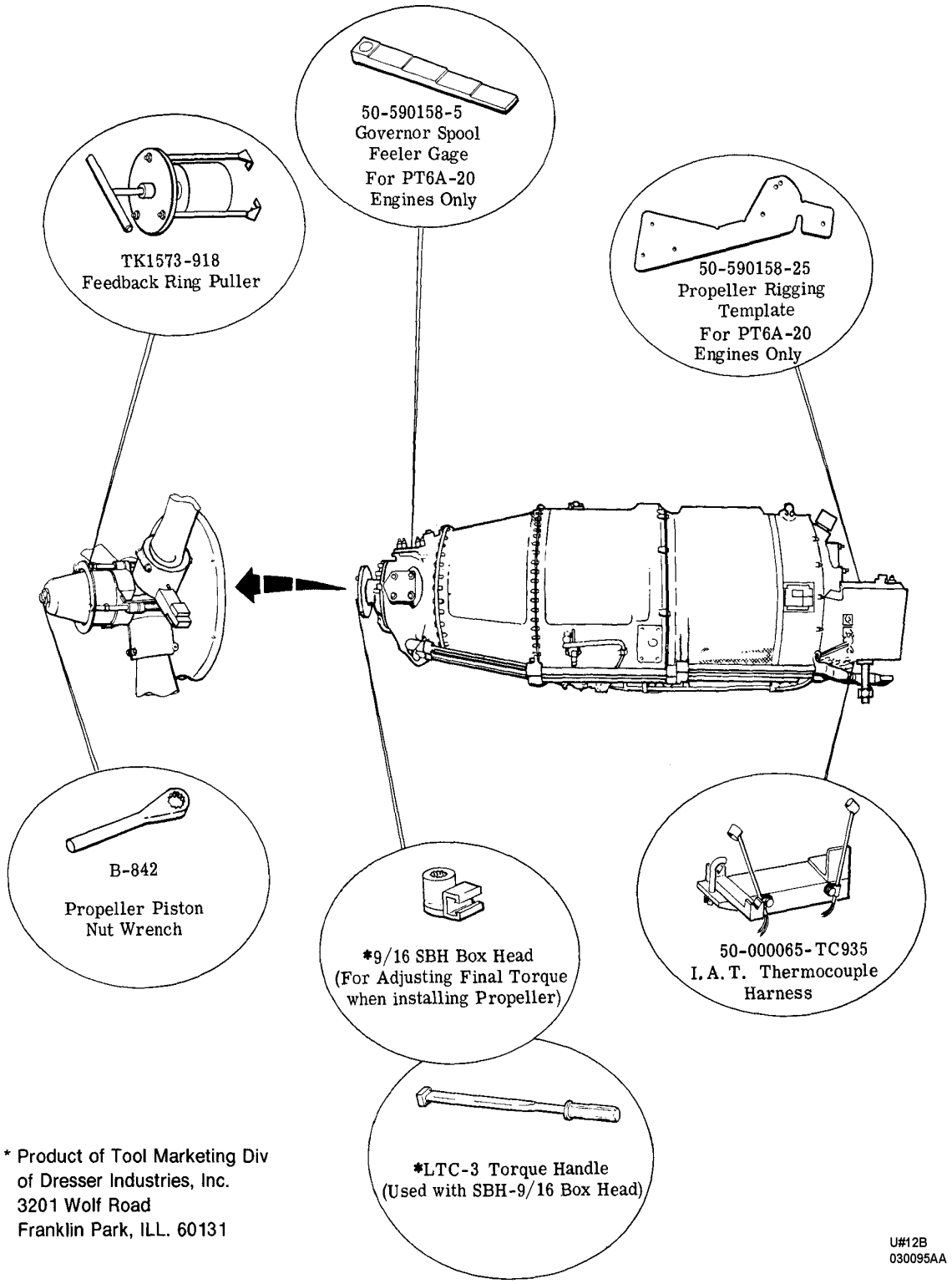


Figure 211 (Sheet 3 of 6)
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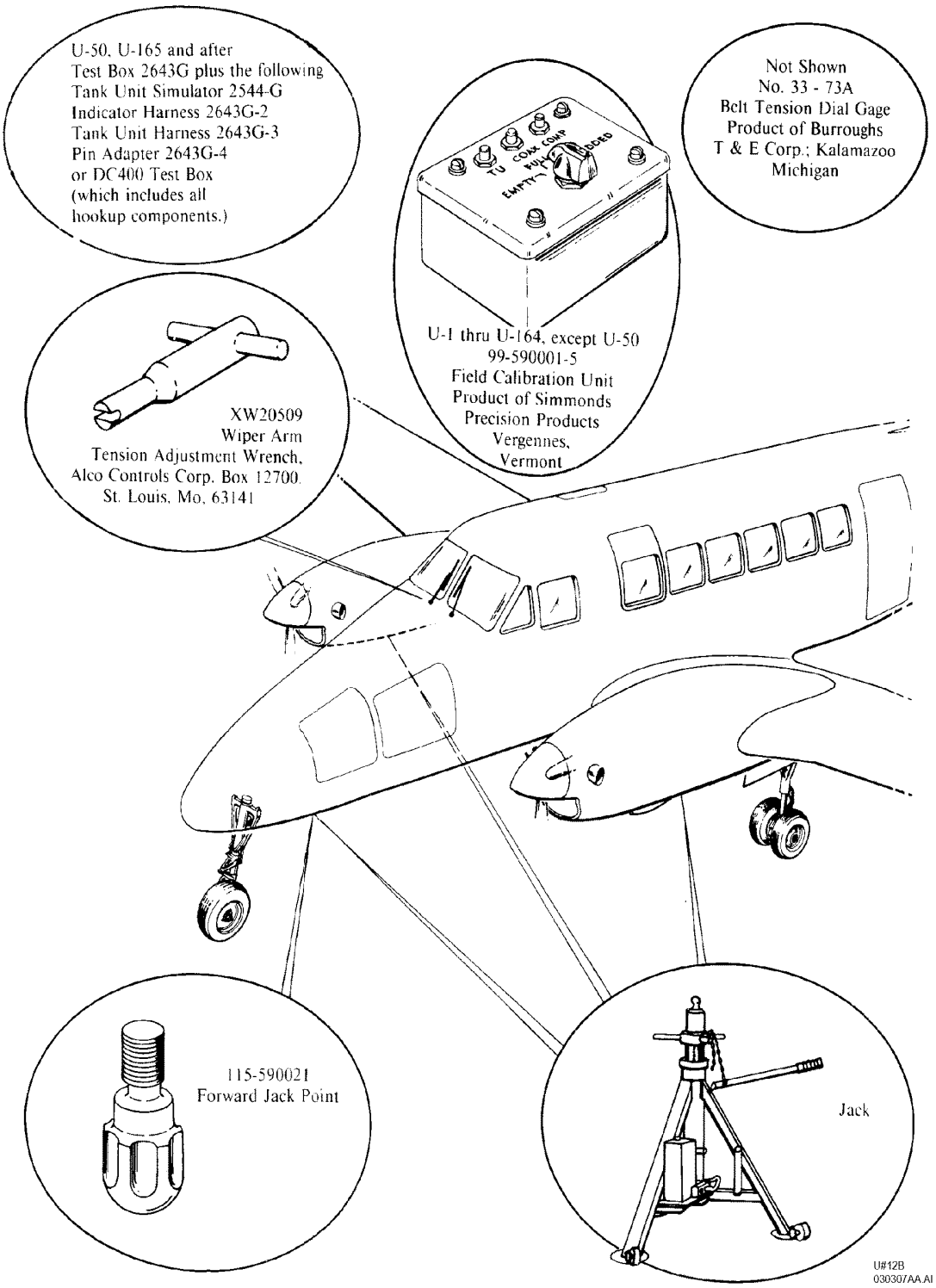

Figure 211 (Sheet 4 of 6)
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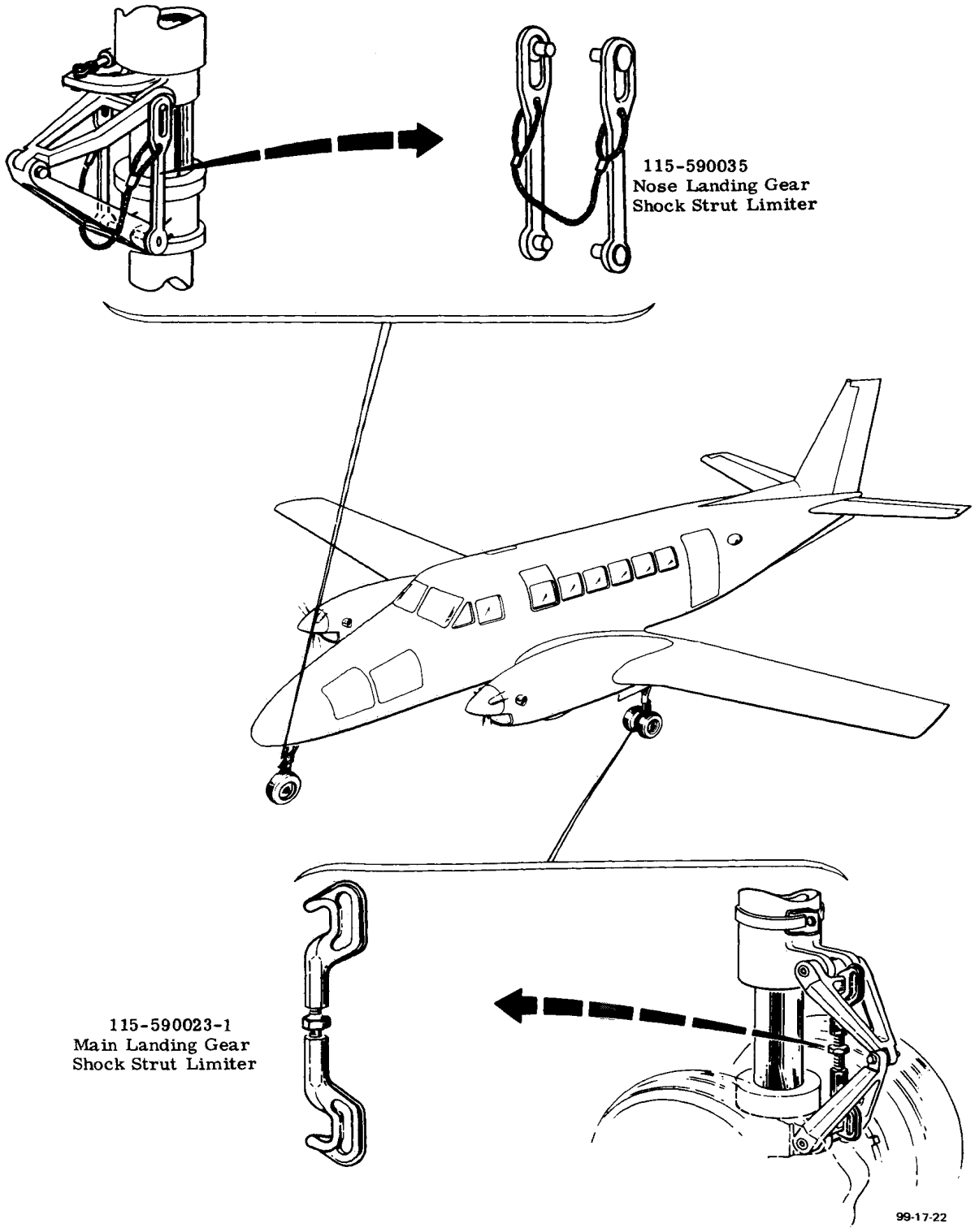

Figure 211 (Sheet 5 of 6)
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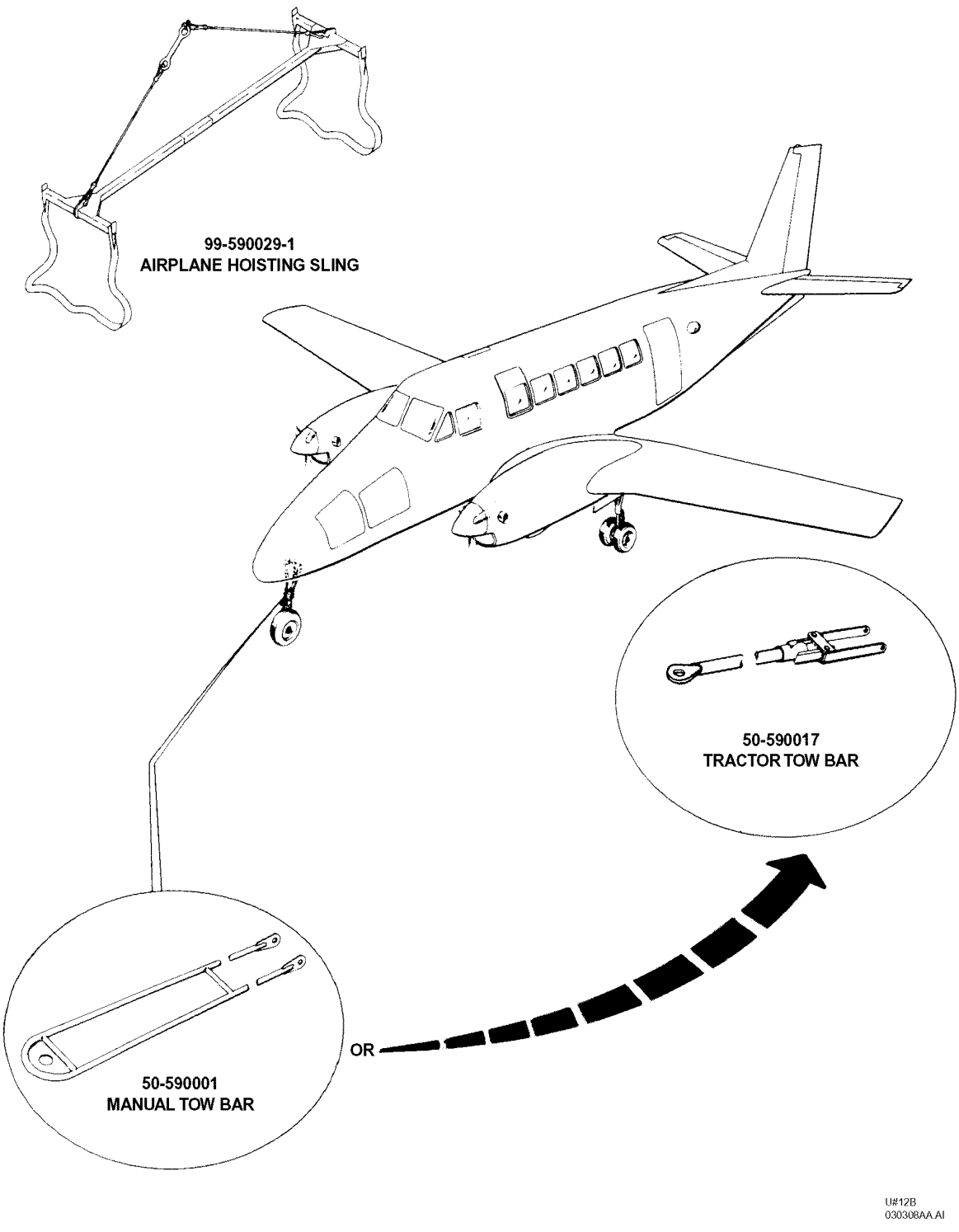

Figure 211 (Sheet 6 of 6)
Special Tools

SERVICING
UNSCHEDULED SERVICING
MAINTENANCE PRACTICES

1. DEICING AND ANTI-ICING OF AIRPLANES ON THE GROUND

Deicing is the removal of ice, frost and snow from the airplane’s exterior after it has formed. Anti-icing is a means of keeping the surface clear of subsequent accumulations of ice, snow and frost.

Snow and ice on an airplane will seriously affect its performance. Even formation of a smooth covering of ice on the wing will change the contour of the wing, producing an increase in drag and a reduction in effective lift coefficient. Frost or frozen snow may present an even greater hazard since the surface texture is rough and will seriously disrupt the smooth flow of air across the wing.

2. DEICING AND ANTI-ICING FLUIDS

Hawker Beechcraft Corporation has evaluated and approved the following deice/anti-ice fluids for use on this model:

A. SAE Type I Anti-icing Fluids (Unthickened-Type Fluids)

Type I fluids (SAE AMS 1424 and ISO 11075) mainly provide protection against refreezing when there is no precipitation.

B. SAE Type II Anti-icing Fluids (Unthickened-Type Fluids)

Type II fluids (SAE AMS 1428 and ISO 11078) provide protection against refreezing when precipitation occurs.

C. SAE Type III Anti-icing Fluids (Thickened-Type Fluids)

Type III fluids (SAE AMS 1428) provide protection against refreezing when precipitation occurs.

D. SAE Type IV Anti-icing Fluids (Thickened-Type Fluids)

Type IV fluids (SAE AMS 1428) provide protection against refreezing when precipitation occurs.

All approved Type I, Type II, Type III, and Type IV fluids may be used for either deicing or anti-icing at any dilution as allowed by the fluid manufacturer's recommendations.

Hawker Beechcraft Corporation cannot accept responsibility for damage to the airplane finish, windows, rubber seals, etc. resulting from the use of deicing fluids not conforming to the specified specifications.

These fluids were chosen according to the following specifics:

(1) Noncorrosive.

(2) Do not deteriorate rubber, painted surfaces, or plastics.

(3) Have a high flash point.

(4) Nontoxic.

(5) Good self-wetting and antifoaming characteristics.

3. REMOVE FROST, SNOW OR ICE

WARNING: Remove all snow, ice and frost before flight.

Type I, Type II, Type III and Type IV; Glycols are listed in the 'harmful' category. You must wear protective goggles, protective gloves and clothing when you handle this material. Keep the material away from skin and eyes.

Inhalation of glycol mists, aerosols, or high concentration of heated vapors poses a hazard to humans. Apply deicing fluid only in well-ventilated areas. Avoid inhaling vapors or mists. If adequate ventilation, designed to keep mists or vapors below harmful levels, is not available, maintenance personnel must wear approved respiratory protective devices.

Deicing fluid Type I, has a limited period of effectiveness (referred to as 'Holdover Times (HOT)'). The use of Type I fluid should only be considered where Type II, Type III or Type IV fluids are not available.

CAUTION: *Make sure that the correct fluid application equipment and correct procedures are used by qualified personnel so that the fluid will perform to the specifications.*

Fluid failure is complex and dependent on:

- *Outside air temperature (OAT)*
- *Percent mix*
- *Type and rate of precipitation*
- *Product Holdover Time (HOT), and other variables*

Each product is unique and reference must be made to the manufacturers recommendations and the FAA guidelines which are published each year.

Though these fluids pass the DTD crazing test for transparent panels, do not apply hot spray directly onto window panels or seals as damage may occur.

The fluids identified in this procedure are for ground deicing/anti-icing only and are not intended for and do not provide protection during flight.

Residue from thickened deicing/anti-icing fluids (Type ll, Type III or Type IV) can remain in aerodynamically quiet areas and accumulate over time. This residue can re-hydrate and expand into a gel-like material that could freeze during flight and cause restrictions in the flight control systems. The accumulation of residual fluid is more prevalent when using a one-step deicing/ anti-icing process which is commonly used in Europe.

For operators using the one-step application process, visually inspect flight control systems for the presence of dry or re-hydrated fluid at least twice during the winter operations season.

For operators using the two-step application process, visually inspect flight control systems for the presence of dry or re-hydrated fluid once at the end of the winter operations season, as a recommended minimum procedure.

Each operator must determine their own frequency of inspection based on the operational tempo of their airplane or fleet. For additional information concerning this guidance please contact the HBC field service representative or the Technical Support telephone line.

A. Guidelines to Holdover Times (Hot)

(1) Visit FAA web site (www.faa.gov) and in the search field type [Holdover Time Tables] and press go/search. A listing will be called up that will include (but is not limited to) documents that read “FAA - Approved deicing Program Updates” for the current year. When the Web Site cannot be accessed or questions arise, contact FAA Flight Standards, Washington DC at 202.267.8166 or Hawker Beechcraft Corporation 1.800.429.5372 or 316.676.3140.

(2) Before take-off the pilot in command must be satisfied the airplane is clear of frost, snow or ice, within the limitations stated in the applicable Aircraft Flight Manual.

(3) If the holdover time is exceeded and visual/tactile investigation of flight surfaces is not possible, then the airplane should be returned for further treatment with deicing/anti-icing fluid prior to takeoff.

(4) Freezing point of SAE/ISO Type I fluid mixture used must be at a minimum of 10°C (18°F) below OAT.

B. General Application

WARNING: Make sure the airplane is grounded before any work is started.

While these procedures are done, the surfaces will be very slippery due to snow, ice, or deicing/anti-icing fluid; use a servicing stand whenever possible. If a stand is not available, use mainplane mats, safety belts, and take other similar precautions. Always stand upwind of the airplane.

Deicing/anti-icing fluids are toxic. Avoid contact with skin and eyes, use goggles and protective clothing.

Functional check of flight controls - If an airplane has been extremely iced or snow covered, a flight control check should be done. This check should be repeated after deicing.

CAUTION: *Wing skin temperatures may differ and in some instances be lower than the OAT. A stronger mix (more glycol) can be used under these conditions.*

The deicing/anti-icing fluid used must be to a concentration and at a temperature in accordance with the fluid manufacturer's instructions, provided the following limitations are not exceeded:

- For heated fluids, a temperature of 60° C (140° F) at the nozzle is desirable. The upper temperature limit shall not exceed fluid and airplane manufacturer’s recommendations.
- Before you apply fluid, ground the airplane and make sure the covers and blanks are installed to the equipment that follows:
  - Static vents
  - Pitot heads
  - Stall detector vanes
- To prevent possible ingestion of deicing fluid, do not operate the engines or APU (if applicable) while deicing is in progress.
- After snow is cleared from the surfaces, make sure the areas listed below are free from snow or ice:

- All air intakes
- Control surface hinges
- Gaps between wing trailing edge shroud and flaps, air brakes and ailerons
- Gaps between horizontal stabilizer and elevators
- Gaps between rudder and vertical stabilizer
- Wheel brakes
- Landing gear bays
- All doors and external drains
- Cold air unit ram air exhaust outlet and duct

The directions given in the fluid manufacturer's instructions with reference to the possibility of mixing UCAR ULTRA with the residues of other fluids must be strictly adhered to. Minor contamination with Type I fluids can significantly degrade the anti-icing performance.

C. Remove Frost

(1) Spray with deicing/anti-icing fluid to fluid manufacturer's instructions.

(2) Under severe frost forming conditions, after defrosting, give a further light application of the concentrated fluid to make sure the maximum holdover period.

D. Remove Sleet and Freezing Rain

CAUTION: *After spraying, examine surface thoroughly, as ice formed by freezing rain can be difficult to see under the deicing fluid and may require touch for confirmation.*

- Spray with undiluted deicing/anti-icing fluid.

4. REMOVE SNOW

A. Dry Snow

(1) Remove by brushing, or by the application of a cold air blast, taking care not to trap the snow in the control surface gaps and hinges.

(2) If the ambient temperature is above freezing, the snow can be removed by the application of a hot air blast, and the cleared surface sprayed with diluted deicing/anti-icing fluid.

B. Wet Snow

(1) Remove with rubber squeegees, taking care not to trap the snow in the control surface gaps and hinges.

(2) If ice has formed under the snow, clear by spraying with deicing/anti-icing fluid.

C. Frozen Snow and Ice Films

(1) Clear off any loose snow, then apply a heavy spray of deicing/anti-icing fluid to the manufacturer's instructions.

- Brush the snow as the fluid is being applied; this will assist in breaking up the deposit and help to retain the fluid on the deposit. When all frozen deposits have been removed, give a final light spray.

(2) If the ambient temperature is above 0°C (32°F), ducted hot air blasts may be used to disperse the ice.

- Do not use hot air blasts near windows, and take extreme care to prevent damage by overheating to painted surfaces, rubber, glass, acrylic or glass fibre, hydraulic pipelines, grease or oiled surfaces.
- Brush off or mop up water resulting from melted ice, as soon as possible.

5. DEICE THE WINDSHIELD

- Lightly spray windshield with fluid, windshield washing (DTD.900AA/4939A). If smearing occurs, wipe clear using a warm damp rag.

6. DEICE THE LANDING GEAR

CAUTION: *Deicing/anti-icing fluid must not come into contact with landing gear electrical plugs and harnesses.*

Do not let deicing/anti-icing fluid contact brake units.

(1) Brush-off loose accumulations of snow with soft brush.

(2) Remove stubborn deposits with a rag soaked in deicing/anti-icing fluid.

7. ANTI-ICING

- If meteorological reports predict the onset of freezing rain or heavy frost deposits, an anti-icing spray of cold (or hot if cold not available) concentrated ground deicing/anti-icing fluid Type II, Type III or Type IV, will give limited holdover protection times, guidance for which is contained in Web Site mentioned in paragraph GUIDELINES TO HOLDOVER TIMES (HOT). If the precipitation is extremely heavy, deposits will form on top of the coating, but these will have little, if any, adhesion to the surfaces and can be easily removed with a light spray of hot diluted fluid.

8. FLUID SPILLS

Glycol-based deicing fluids are biodegradable in water. Only gross contamination of slow moving or restricted bodies of water would be likely to cause any serious environmental impact. Typical field-use concentrations of deicing fluids, particularly when diluted by snow, ice or water, causes little or no injury to most broad leaf plants, grasses perennial ground cover, and woody plants.

Minor leaks or spills of deicing fluid in storage areas must be soaked up with an absorbent material, such as sawdust, vermiculite, an all-purpose commercial oil absorbent, or sand. Carefully shovel the absorbent/ deicing fluid into an appropriate container for disposal.

Spilled, leaked, or contaminated deicing fluid must be disposed of in strict compliance with all applicable federal, state, and local regulations and ordinances.

9. REMOVE SALT OR CHEMICAL AGENTS

CAUTION: *Do not use high pressure water; damage may result to electrical equipment and lubricated components.*

(1) Where contamination has occurred on the structure, due to the airplane landing on airfields where the snow or ice has been dispersed with salt or chemical melting agents:

(a) Wash down the affected areas with clean water as soon as possible. A wetting agent, such as detergent cleaner (Teepol TS610) or Comprex A, may be added in small quantities.

(b) When the time or conditions prevent removal of the contamination at outstations, this must be noted in the technical log, so that appropriate action may be taken to remove it at main base.

10. RESIDUE FROM DEICE/ANTI-ICE FLUIDS

CAUTION: *Do not use high pressure water; damage may result to electrical equipment and lubricated components.*

A. Inspection

(1) Gain access to areas, potentially affected by residue, where flight controls and other systems components are located.

(2) Visually inspect for the presence of dry or rehydrated residue anywhere in these areas. The residue may be difficult to see, especially if dry. Dry residue tends to be in the form of a thin film that may be partially covered with grease or dirt. Rehydrated residue often appears as a gel-like substance which is thicker and therefore more visible.

B. Cleaning

(1) After identifying any residue it should be removed by using warm water with rags and/or soft bristle brushes to clean the gel-like material away.

CAUTION: *Make sure the water or compressed air does not cause any residue to enter areas that are not accessible. Do not allow runoff from the cleaning process to enter other areas of the airplane.*

Avoid spraying cleaning fluids onto bearings, fittings, control cables and electrical connectors.

Do not spray controls with water if the ambient temperature is below freezing unless the airplane is located in a heated hangar.

This cleaning process can, potentially, remove grease from control system bearings and fittings, and remove corrosion inhibitors from control cables.

(2) The use of a low-pressure stream of water or compressed air to rinse away any residue may prove helpful. Using Type I deicing fluid, or a mixture of water and Type I fluid, is also a good cleaning agent for removal of residue.

C. Relubrication (If Required)

- If residue has been found and removed by cleaning, it is recommended that all bearings, fittings, and control cables in the affected area be relubricated (Ref. Chapter 12-20-00, 201, SCHEDULED SERVICING).

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 20 - STANDARD PRACTICES - AIRFRAME

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AIRPLANE FINISH CARE

20-08-00

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STANDARD PRACTICES - AIRFRAME
GENERAL INFORMATION
DESCRIPTION AND OPERATION

1. GENERAL

This chapter describes the standard maintenance practices used in maintaining the airplane in an airworthy condition. The information in this chapter is applicable during maintenance in several chapters of this maintenance manual. Maintenance procedures in a particular chapter supersedes the general information in this chapter.

WARNING: Any maintenance requiring the disconnection and connection of flight control cables, plumbing, electrical connectors or wiring requires identification of each side of the component being disconnected to facilitate correct reassembly. At or prior to disassembly, components should be color coded, tagged or properly identified in a way that it will be obvious how to correctly reconnect the components.

After connection of any component, remove all identification tags. Check all associated systems for correct function prior to returning the airplane to service.

2. TORQUE WRENCHES

When a torque wrench and adapter is used, compensation must be made for the extra leverage gained (Ref. Figure 1). New indicator readings must be calculated before the wrench is used. To figure the desired lower readings which will actually give the torque specified, use the following formula:

Example:D = Desired reading
L = Length of torque wrench
A = Adapter length
T = Torque

D =?
L = 33 in.
A = 11 in.
T = 5,000 in.-lbs

$$\frac{\text{Original wrench length x specified torque}}{\text{length of wrench + adapter}} = \text{Desired reading}$$

$$\frac{33 \times 5{,}000}{33 + 11} = \frac{165{,}000}{44} = 3{,}750 \text{ in.-lbs}$$

Refer to Tables 1 and 2 for nut-bolt torque values.

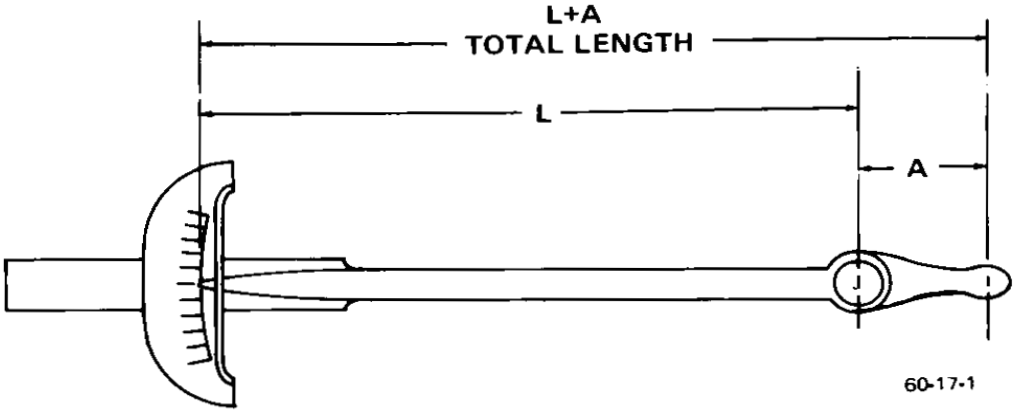


Figure 1
Torque Wrench and Adapter

An acceptable method of checking the torque, if a torque wrench is not available, is to attach a spring scale to a conventional flex or "T" handle inserted in an adapter (Ref. Figure 2). Force should be applied in a direction perpendicular to an imaginary line extending from the center of the bolt through the spring scale attaching point.

To calculate the force in pounds (scale reading) required to obtain the specified torque, divide the torque in inch-pounds by the distance in inches between the center of the bolt and the scale attaching point. For example, if the specified torque is 5,000 inch-pounds and the distance is 25 inch, a pull of 200 pounds must be applied. Except for the lubricated wing bolts, all bolts must be clean and free of all lubricant, otherwise loss of normal friction allowed for establishing the torque values may result in over torquing the bolt.

When a torque wrench adapter is used, the length of the adapter must be added to the length of the flex or "T" handle wrench and a value calculated for that particular combination. The following is a typical example in finding a desired value.

Effective length of flex or “T” handle wrench:..... 12 in.
Length of adapter..... 3 in.
Total length..... 15 in.
Desired torque on bolt..... 2,000 in.-lbs

$$\frac{\text{2,000 in.-lbs}}{\text{15 in.}} = \text{133.3 lbs (scale reading)}$$

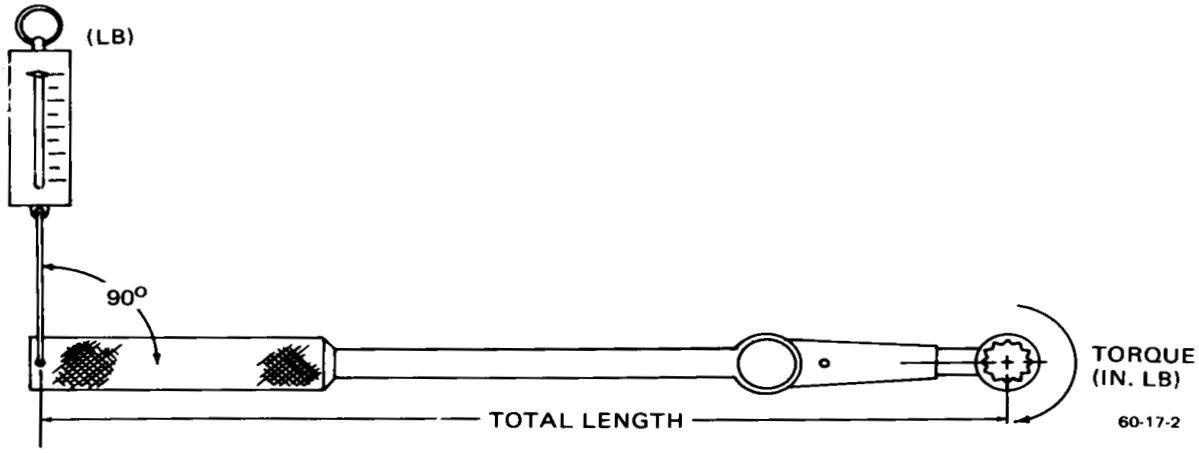


Figure 2
Computing Torque with Spring Scale

Table 1
RECOMMENDED TORQUE VALUES FOR NUT-BOLT COMBINATIONS - (WITHOUT LUBRICATION).
FINE THREAD SERIES.

BOLTS **Steel Tension**	**BOLTS** **Steel Tension**	**BOLTS** **Aluminum**
AN 3 thru AN 20 AN 42 thru AN 49 AN 73 thru AN 81 AN 173 thru AN 186 AN 509 NK9 AN 525 NK525 MS 20033 thru MS 20046 MS 20073 MS 20074 MS 24694 MS 27030	MS 20004 thru MS 20024 NAS 144 thru NAS 158 NAS 172 NAS 174 NAS 333 thru NAS 340 NAS 517 NAS 583 thru NAS 590 NAS 624 thru NAS 644 NAS 1303 thru NAS 1320	AN 3DD thru AN 20DD AN 173DD thru AN 186DD AN 509DD AN 525D MS 24694DD MS 27039D

Steel Shear Bolt
NAS 464

NUTS		**NUTS**		**NUTS**	
Steel Tension	**Steel Shear**	**Steel Tension**	**Steel Shear**	**Aluminum Tension**	**Aluminum Shear**
AN 310 AN 315 AN 363 AN 365 MS 17825 MS 20365 MS 20500 MS 21045 NAS 679 NAS 1021	AN 320 AN 364 MS 17826 MS 20364 NAS 1022	AN 310 AN 315 AN 363 AN 365 MS 17825 MS 20365 MS 21045 NAS 679 NAS 1021 NAS 1291	AN 320 AN 364 MS 17826 MS 20364 NAS 1022	AN 310D AN 365D NAS 1021D	AN 320D AN 364D NAS 1022D

Nut-Bolt Size	**Torque Limits (in.-lbs)**											
	Min.	**Max.**	**Min.**	**Max.**	**Min.**	**Max.**	**Min.**	**Max.**	**Min.**	**Max.**	**Min.**	**Max.**
8-36	12	15	7	9	-	-	-	-	5	10	3	6
10-32	20	25	12	15	25	30	15	20	10	15	5	10
1/4-28	50	70	30	40	80	100	50	60	30	45	15	30
3/16-24	100	140	60	85	120	145	70	90	40	65	25	40
3/8-24	160	190	95	110	200	250	120	150	75	110	45	70
7/16-20	450	500	270	300	520	630	300	400	180	280	110	170
1/2-20	480	690	290	410	770	950	450	550	280	410	160	260
9/16-18	800	1000	480	600	1100	1300	650	800	380	580	230	360
5/8-18	1100	1300	660	780	1250	1550	750	950	550	670	270	420
3/4-16	2300	2500	1300	1500	2650	3200	1600	1900	950	1250	560	880
7/8-14	2500	3000	1500	1800	3550	4350	2100	2600	1250	1900	750	1200
1 -14	3700	4500	2200	3300	4500	5500	2700	3300	1600	2400	950	1500
1 1/8-12	5000	7000	3000	4200	6000	7300	3600	4400	2100	3200	1250	2000
1 1/4-12	9000	11000	5400	6600	11000	13400	6600	8000	3900	5600	2300	3650

Table 2
RECOMMENDED TORQUE VALUES FOR NUT-BOLT COMBINATIONS - (WITHOUT LUBRICATION).
COARSE THREAD SERIES

	BOLTS **Steel Tension**		**BOLTS** **Steel Tension**		**BOLTS** **Aluminum**	
	AN 3 thru AN 20 AN 42 thru AN 49 AN 73 thru AN 81 AN 173 thru AN 186 AN 509 NK9 AN 525 NK525 MS 20033 thru MS 20046 MS 20073 MS 20074 MS 24694 MS 27030		MS 20004 thru MS 20024 NAS 144 thru NAS 158 NAS 172 NAS 174 NAS 333 thru NAS 340 NAS 517 NAS 583 thru NAS 590 NAS 624 thru NAS 644 NAS 1303 thru NAS 1320	**Steel Shear bolt** NAS 464	AN 3DD thru AN 20DD AN 173DD thru AN 186DD AN 509DD AN 525D MS 24694DD MS 27039D	
	NUTS		**NUTS**		**NUTS**	
	Steel Tension	**Steel Shear**	**Steel Tension**	**Steel Shear**	**Aluminum Tension**	**Aluminum Shear**
	AN 310 AN 315 AN 363 AN 365 MS 17825 MS 20365 MS 20500 MS 21045 NAS 679 NAS 1021	AN 320 AN 364 MS 17826 MS 20364 NAS 1022	AN 310 AN 315 AN 363 AN 365 MS 17825 MS 20365 MS 21045 NAS 679 NAS 1021 NAS 1291	AN 320 AN 364 MS 17826 MS 20364 NAS 1022	AN 310D AN 365D NAS 1021D	AN 320D AN 364D NAS 1022D

Nut-Bolt Size	**Torque Limits (in.-lbs)**											
	Min.	**Max.**	**Min.**	**Max.**	**Min.**	**Max.**	**Min.**	**Max.**	**Min.**	**Max.**	**Min.**	**Max.**
8-32	12	15	7	9	-	-	-	-	-	-	-	-
10-24	20	25	12	15	-	-	-	-	-	-	-	-
1/4-20	40	50	25	30	-	-	-	-	-	-	-	-
3/16-18	80	90	48	55	-	-	-	-	-	-	-	-
3/8-16	160	185	95	110	-	-	-	-	-	-	-	-
7/16-14	235	255	140	155	-	-	-	-	-	-	-	-
1/2-13	400	480	240	290	-	-	-	-	-	-	-	-
9/16-12	500	700	300	420	-	-	-	-	-	-	-	-
5/8-11	700	900	420	540	-	-	-	-	-	-	-	-
3/4-10	1150	1600	700	950	-	-	-	-	-	-	-	-
7/8- 9	2200	3000	1300	1800	-	-	-	-	-	-	-	-
1 - 8	3700	5000	2200	3000	-	-	-	-	-	-	-	-
1 1/8- 8	5500	6500	3300	4000	-	-	-	-	-	-	-	-
1 1/4- 8	6500	8000	4000	5000	-	-	-	-	-	-	-	-

STANDARD PRACTICES - AIRFRAME
TUBING, HOSE AND FITTINGS
MAINTENANCE PRACTICES

1. HYDRAULIC SYSTEM PLUMBING

A. Repair

Effective repair of damaged hydraulic plumbing lines used on the Model 99 Airliner landing gear retraction and extension system may be accomplished using "CryoFit” couplings, manufactured by Raychem Corp., 300 Constitution Drive, Menlo Park, CA 94025. Should such repair be required, contact the Commercial Service Department, Hawker Beechcraft Corporation, Wichita, KS 67201. You will be advised of procedures for obtaining repair parts and personnel to train and guide your service staff in the correct technique for installation of the “CryoFit”' couplings.

Procedures for repairing hydraulic plumbing lines with "CryoFit" couplings are detailed in Chapter 29-10-00 of the King Air Series Components Maintenance Manual. These procedures, however, should only be used by properly trained and qualified personnel.

STANDARD PRACTICES - AIRFRAME
LOCKING DEVICES
MAINTENANCE PRACTICES

1. PROCEDURES FOR INSTALLATION OF SELF-LOCKING NUTS, LOCKWIRE, COTTER PINS AND LOCK WASHERS

Except where specific instructions may be required to satisfy certain applications, the following procedures are standard methods for installing the various locking devices used on bolts, screws and nuts.

A. Self Locking Nuts

Where self-locking nuts are used, the following procedure applies:

(1) For each item, note the torque necessary to turn the nut on the bolt before seating the nut.

(2) Add the above torque to the value detailed in the assembly instruction for the application. Use this new value as the total applied torque.

NOTE: Before reinstallation of self- locking nuts, nuts should be checked for effectivity of the self-locking feature. Reject suspect nuts as necessary.

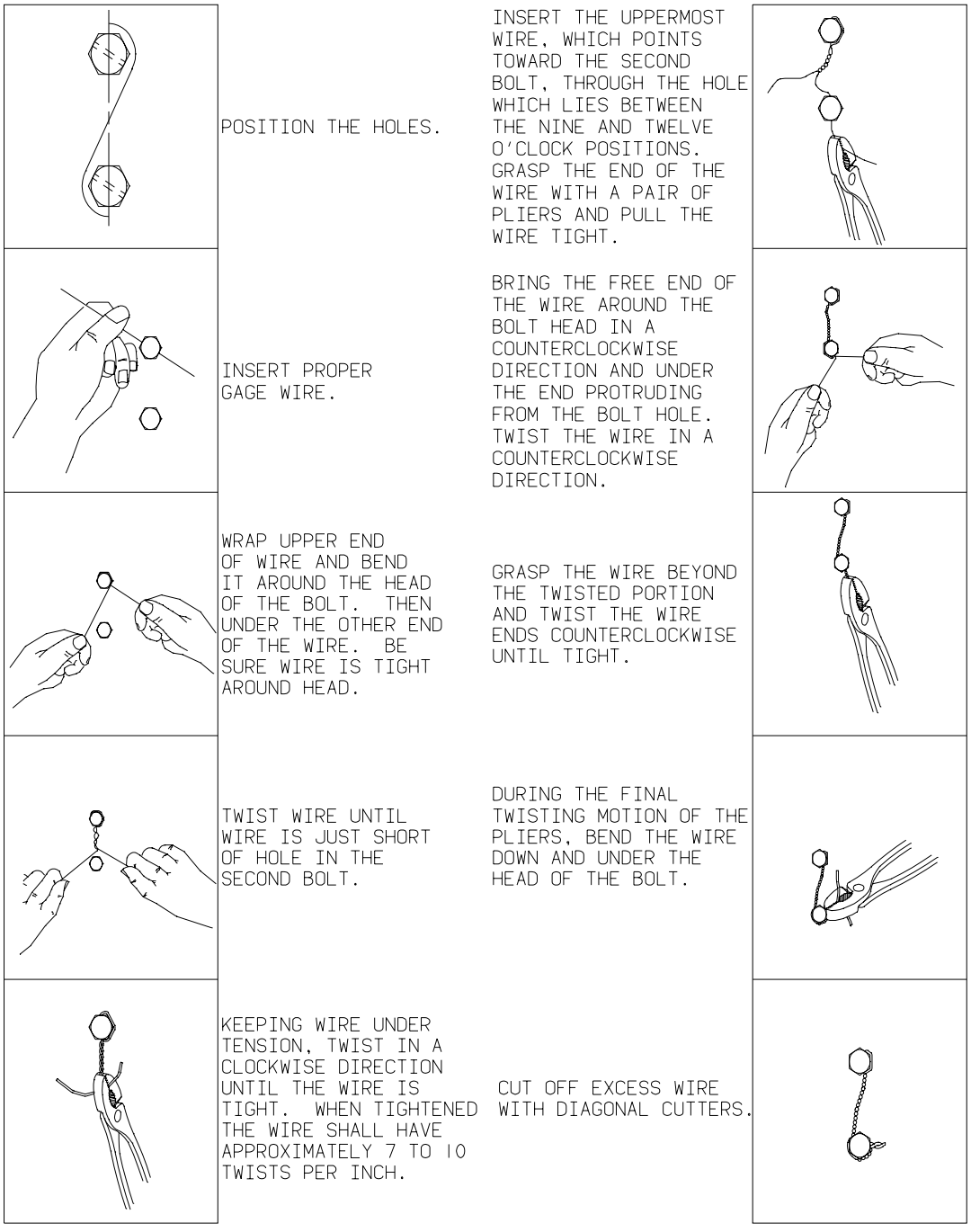
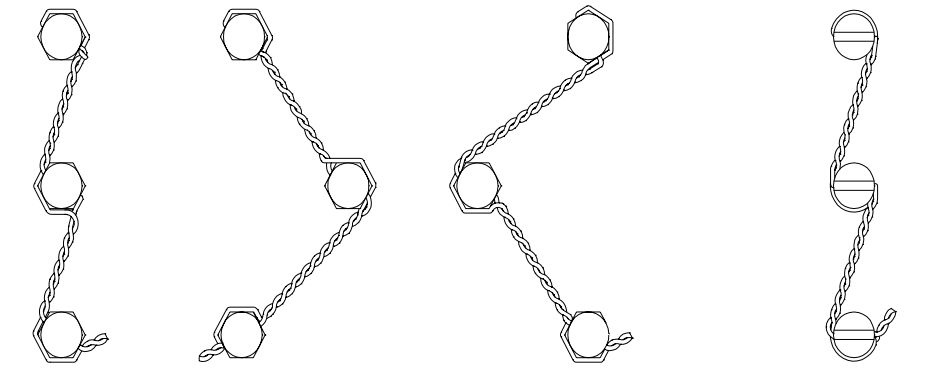
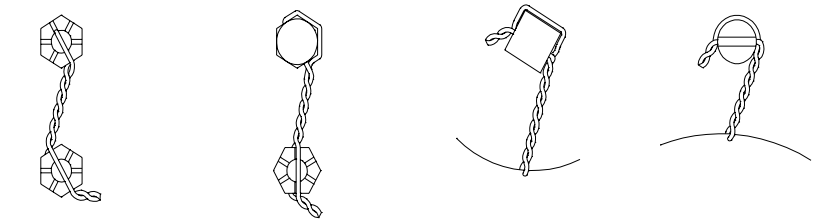


Figure 201
Lockwiring Procedure



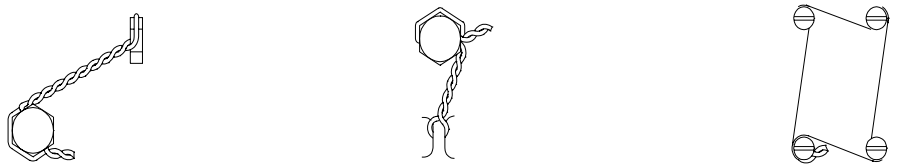
EXAMPLE 1 EXAMPLE 2 EXAMPLE 3 EXAMPLE 4

BOLTS, FILLISTER HEAD SCREWS, SQUARE HEAD PLUGS, AND OTHER SIMILAR PARTS ARE WIRED SO THAT THE LOOSENING TENDENCY OF EITHER PART IS COUNTERACTED BY THE TIGHTENING OF THE OTHER PART. THE DIRECTION OF TWIST FROM THE SECOND TO THE THIRD UNIT IS COUNTERCLOCKWISE TO KEEP THE LOOP IN POSITION AGAINST THE HEAD OF THE BOLT. THE WIRE ENTERING THE HOLE IN THE THIRD UNIT WILL BE THE LOWER WIRE AND BY MAKING A COUNTERCLOCKWISE TWIST AFTER IT LEAVES THE HOLE, THE LOOP WILL BE SECURED IN PLACE AROUND THE HEAD OF THAT BOLT.



EXAMPLE 5 EXAMPLE 6 EXAMPLE 7 EXAMPLE 8

METHODS FOR WIRING VARIOUS STANDARD ITEMS. WIRE MAY BE WRAPPED OVER THE UNIT RATHER THAN AROUND IT WHEN WIRING CASTELLATED NUTS OR ON OTHER ITEMS WHEN THERE IS A CLEARANCE PROBLEM.



EXAMPLE 9

THE METHOD FOR WIRING BOLTS IN DIFFERENT PLANES. NOTE THAT WIRE SHOULD ALWAYS BE APPLIED SO THAT TENSION IS IN THE TIGHTENING DIRECTION.

EXAMPLE 10

HOLLOW HEAD PLUGS WIRED WITH THE TAB BENT INSIDE THE HOLE TO AVOID SNAGS AND POSSIBLE INJURY TO PERSONNEL.

EXAMPLE 11

CORRECT APPLICATION OF SINGLE WIRE TO CLOSELY SPACED MULTIPLE GROUP.

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Figure 202
Lockwiring Procedure

B. Lockwire and Cotter Pin Requirements

When tightening a castellated nut, alignment of the slot must be obtained without exceeding the maximum torque. If this is not possible, replace the nut with another one. After tightening the nut to the recommended torque, the nut must not be loosened to permit insertion of lockwire or a cotter pin. If the slot in the nut or lockwire hole in the bolt or screw is not correctly aligned at the minimum torque value given, the nut, screw or bolt should be further tightened to the next alignment position, but the maximum torque value given must not be exceeded. Should alignment still be impossible without exceeding the maximum torque, back off the nut, screw or bolt one-half turn and retorque.

C. Slotted, Steel Locknuts (Prevailing Torque Type)

Effective locking of slotted, steel locknuts on bolts or studs requires full engagement of all locknut threads. The chamfered section of the locknut ID does not exert force on the bolt or stud; therefore, it is not necessary that the bolt or stud be flush with, or protrude from the outer face of the locknut.

D. Standard Stepped Nuts

When the torque required to drive a stud to the correct protrusion does not reach the minimum value given, or exceeds the maximum value given, a new stud must be selected.

E. Hose, Tubing and Threaded Couplings

If leakage occurs at a coupling, do not attempt to correct it by overtorquing. Disassemble the fitting and check for nicks, burrs and/or foreign matter. Use new parts to rectify.

2. LOCKING DEVICES

A. General

Lockwire, lock washers, tab locks, tab washers, key washers, cup washers and cotter pins must never be reused. All lockwire and cotter pins must fit snugly into drilled holes in the bolts and studs for locking purposes. Bushings and plugs must be lockwired to boss or casing. Do not lockwire bushing to plug. Install cotter pins so that the head fits into the slot of the castellated nut and, unless otherwise specified, bend one end of the, pin back over the stud or bolt and the other end flat against the flat on the nut.

3. LOCKWIRE

Use the same type and diameter of lockwire as that employed during the initial assembly. Except where otherwise specified, the wire used on the airplane power plant is heat and corrosion resistant steel wire of 0.025 inch diameter.

A. Basic Rules

(1) Lockwire must be tight after installation to prevent failure due to rubbing or vibration.

(2) Lockwire must be installed in a manner that tends to tighten and keep a part locked in place thus counteracting the natural tendency of the part to loosen.

(3) Lockwire must never be over-stressed. It will break under vibrations if twisted too tightly. The lockwire shall be pulled taut when being twisted, but shall have minimum tension, if any, when secured.

(4) Lockwire ends must be bent toward the engine, or structure, to avoid sharp or projecting ends which might present a safety hazard or vibrate in the air stream.

(5) Internal wiring must cross over or obstruct a flow passage when an alternate method can be used.

B. Lockwire Hole Alignment

(1) Check the units to be lockwired to make sure that they have been correctly torqued and that the wiring holes are properly positioned in relation to each other. When there are two or more units, it is desirable that the holes in the units be in the same relationship to each other. Never overtorque or loosen units to obtain proper alignment of the holes. It should be possible to align the wiring holes when the units are torqued within the specified limits. However, if it is impossible to obtain a proper alignment of the holes without either over or under torquing, select another unit which will permit proper alignment within the specified torque limits.

C. Lockwire Twisting

(1) To prevent mutilation of the twisted section of the wire when using pliers, grasp the wire at the ends or at a point that will not be twisted. Lockwire must not be nicked, kinked or mutilated. Never twist the wire ends off with the pliers and, when cutting off ends, leave at least three complete turns after the loop, exercising extreme care to prevent the wire ends from falling into areas where they might create a hazard or damage. The strength of the lockwire holes is marginal; never twist the wire off with pliers. Cut the lockwire close to the hole, exercising extreme care.

D. Lockwire Illustration

(1) Figure 201 illustrates a typical lockwiring procedure. Although there are numerous lockwiring operations performed on the airplane, practically all are derived from the basic examples shown in Figure 202.

4. KEY WASHERS (TAB AND CUP TYPES)

The terms key washer, tab washer and cup washer are interchangeable.

NOTE: These types of washers must be used once only. Use new washers at each assembly.

When bending or setting the tabs on these washers, do not use sharp pointed tools. Use of such tools can lead to subsequent failure of the locking tabs which, on becoming detached, can cause extensive damage.

A. Retaining Rings (Spirolox, Etc)

Retaining rings must be installed using approved retaining ring pliers. Internal rings must not be compressed beyond the point where ends of the ring meet. External type rings must be expanded only enough to allow installation without becoming bent. After installation, ensure each retaining ring is completely seated in its groove, without looseness or distortion.

STANDARD PRACTICES - AIRFRAME
AIRPLANE FINISH CARE
MAINTENANCE PRACTICES

1. PROCEDURES

A. Paint Care Procedure

Epoxy and urethane finishes. being two component chemically cured materials, do not require an extended cure period (ultimate cure occurs in seven to ten days at 75°F.). On exposure to ultra violet source (sunlight) epoxies undergo a surface degradation referred to as chalking. Due to this chalking characteristic, epoxy finishes must be waxed on a regular basis. If chalking occurs, polishing and buffing will restore a good appearance. A good grade of wax should be used; however. a cleaner wax or polish may be used where a severe case of chalking has occurred. Even though urethane finishes should not be polished, they may be waxed, but it is not necessary for approximately two to five years after application. A build-up of wax on exterior paint finishes will yellow with age: therefore wax should be removed periodically. generally naphtha is adequate and safe for this purpose.

Oil residue, exhaust soot and stubborn soils may be removed with solvent (33, Table 6, 91-00-00). If solvent (34, Table 6, 91-00-00) is used, care must be exercised to assure that prolonged soaking of the finish is not allowed; furthermore, even slight contact of MPK with acrylic windows or light lenses must be avoided.

2. CLEANING

A. Exterior Painted Surfaces

Prior to cleaning, cover the wheels, making certain the brake discs are covered. Attach the pitot cover securely and plug or mask off all other openings. Be particularly careful to mask off the static air buttons before washing or waxing.

CAUTION: *Do not apply wax to uncured paint. waxes and polishes seal the paint from the air and prevent curing. Urethane and epoxy finishes cure in seven to ten days at 75°F. Wash uncured paint with cold or lukewarm water and a mild soap. Any rubbing of the paint should be done gently and held to a minimum to avoid cracking the paint film.*

Flush loose dirt away with clean water, then wash with a soft cloth or chamois, and a mild soap and water. Avoid harsh, abrasive or alkaline soaps and detergents which could cause corrosion or scratches. To remove oil residue or exhaust soot, use a cloth dampened with solvent (33, Table 6, 91-00-00) or solvent (34, Table 6. 91-00-00), then wax areas finished with epoxy paint. If a substance cannot be removed with either solvent, consult a Hawker Beechcraft Aero or Aviation Center or International Distributor or Dealer.

CAUTION: *When washing the airplane with soap and water, use special care to avoid washing grease away from any lubricated area. After washing the wheel well areas with solvent, lubricate all lubrication points. Premature wear of the lubricated surfaces may result if these precautions are not observed.*

A build-up of wax on the paint will yellow with age; therefore, wax should be removed periodically. generally naphtha is adequate for this purpose.

3. AIRPLANE FINISHES

Since both epoxy and urethane paint finishes are available for the Model 99 Airliner Series, this chapter will deal with the care of each.

A. Epoxy Paint

Because it is impervious to synthetic oil and most solvents and has excellent abrasion resistance, epoxy paint is offered for use on the Model 99 Airliner exterior. Epoxy paint is necessary because the turbine oil used in the PT-6 engines will damage enamel and lacquer finishes.

Besides forming a tougher film than enamel or lacquer, epoxy has it very lustrous sparkle; however, the painted surface oxidizes a little faster than enamel or lacquer, and must be polished more frequently to retain the sheen. Oxidation is accelerated by exposure to the sun, hence, in hot weather oxidation will occur faster than in cold weather. A good coat of wax will protect the airplane surface from the suns rays and keep the surface from oxidizing as fast. Any good automotive polish or wax can be used.

B. Paint Repair Procedures

NOTE: Any time an airplane is repainted, inspect all placards to assure that they are not covered with paint, are easily readable, and are securely attached.

At temperatures below 70°F, the paint cure time required for any of the paints used in this procedure will extend beyond the time normally required. Under no circumstances should the paint be applied at temperatures below 60°F, since the paint will not cure.

The Model 99 Airliner Series is finished with pretreatment (wash) primer, epoxy primer, and a topcoat of epoxy enamel. The following procedures including cleaning, paint stripping, prepaint preparation, priming, epoxy painting, and an alternate method for small repairs that does not involve paint stripping. Careful observance of these procedures should result in a smooth, high lustre finish with firm adhesion for maximum life.

Besides forming a tougher film than enamel or lacquer, epoxy has it very lustrous sparkle; however, the painted surface oxidizes a little faster than enamel or lacquer, and must be polished more frequently to retain the sheen. Oxidation is accelerated by exposure to the sun, hence, in hot weather oxidation will occur faster than in cold weather. A good coat of wax will protect the airplane surface from the sun's rays and keep the surface from oxidizing as fast. Any good automotive polish or wax can be used.

NOTE: Do not intermix paint colors or paint and paint catalyst from different vendors.

Precut stripe and number patterns are available through Medagraphics of Kansas, 1720 S. 151 W., Route 1, Goddard, Kansas 67052.

C. Paint Stripping And Cleaning

CAUTION: *Never use aluminum foil to mask electrothermal windshields during painting. The metal brighteners used on the aluminum and the stannous oxide used as an anti-static coating on electrothermal windshield, may combine to form lead oxide, producing a mirror-like glaze on the windshield surface. If metal brighteners are to be used, cover the windshield with pasteboard or paper masking material.*

Epoxy paints and primer are difficult to strip because of their resistance to chemicals and solvents; therefore, a paint stripper made specifically for epoxy paints should be used.

(1) Mask around the edge of the skin or skins containing the damaged area. Use a double thickness of heavy paper to prevent accidental splashes of paint stripper from penetrating the masking.

(2) Apply Epoxy Stripper as indicated by the manufacturer's directions. try to stay approximately 1/8 inch away from the masking tape. This will necessitate a little more clean up upon finishing, but will prevent any damage to the finish on the next skin. The stripper will not attack aluminum during the stripping process and can be neutralized by rinsing the affected area with water.

CAUTION: *Epoxy Strippers usually contain acids that will irritate or burn the skin. Wear rubber gloves and eye protection when using the stripper.*

(3) Rinse the area with water and dry.

NOTE: If washing in the vicinity of lubricated fittings, relubricate the fittings. This will prevent unnecessary wear.

(4) Wash the stripped area carefully with a solvent such as acetone, methyl propyl ketone, or lacquer thinner. This will prevent tiny particles of loose paint from adhering to the stripped area.

(5) Using a nylon scratch pad or aluminum wool dipped in clean water, clean the surface with a cleanser such as bon ami, ajax, comet cleaner, etc. A good scouring will leave the surface completely clean.

(6) Rinse thoroughly with clean water and dry the affected area carefully. If the stripped area includes several joints or skin laps, let the airplane sit until all moisture has dried. This may be accelerated by blowing the skin laps with compressed air. Wet masking should be replaced.

4. PRETREATMENT (WASH) PRIMER

An acid etching primer that conforms to specification MIL-C-8514 should be applied to improve adhesion of the finishing coats. EX2016G base and EX2016 catalyst (products of Pratt and Lambert Industrial Coatings Div., P.O. Box 2153, Wichita, Kansas 67201) are used in equal parts as a pretreatment wash primer at the factory.

(1) Mix the primer in accordance with the manufacturer’s instructions.

(2) Apply a thin wet coat of primer. It should dry for at least one hour, but not over 6 hours, before the next coat of Epoxy primer.

A. Epoxy Primer

(1) Mix 54-23985 Epoxy Primer (product of Pratt and Lambert Industrial Coatings Div., P.O. Box 2153, Wichita, Kansas 67201) in accordance with the manufacturer's instructions.

NOTE: For the best results, these directions must be followed carefully, since some manufacturers require that the primers be allowed to set for 1/2 hour after the catalyst and base have been mixed while others allow immediate use after mixing.

(2) Apply a thin coat of Epoxy primer with a spray gun using approximately 35 to 55 psi of air pressure. A dappled appearance indicates that the coat is too thin.

(3) If the initial coat is allowed to cure for more than 24 hours before topcoating, sand the primer slightly to roughen the surface to assure adhesion. Wipe off the surface with a cloth dampened with solvent (such as lacquer thinner), then apply the topcoat.

B. Special Procedures

The following special procedures should be used when repainting the specific areas indicated.

C. Urinal Surface Area

Give additional corrosion protection to surfaces and areas affected by spray discharged from the urinal venture tube in the following manner:

(1) Give the unpainted surface areas of the fuselage belly a coat of MIL-L-6806 finish, beginning 4 inches aft of any urinal venturi tube outlet and extending to the tip of the tailcone.

(2) Seal all the belly skin laps aft of the venturi tube outlet at fuselage station 346.5 with EC-750 sealer or equivalent.

5. NOSE RADOME

CAUTION: *Radar operation can be adversely affected by abrupt variations in the thickness of the material in the nose radome, which could be caused by excessive and/or uneven buildup of repair material during repair of damage. Therefore, it is recommended that a damaged nose radome be replaced rather than repaired.*

Repainting the radome will affect the transmissivity requirements of the radome; therefore, a maximum of three thin coats of poly-urethane paint only should not be exceeded. Use of any paint other than poly-urethane is not recommended. When preparing the radome for repainting, use a fine grit sandpaper, and do not sand into the black anti-static coating beneath the finish paint coat.

New radomes purchased from Hawker Beechcraft Corporation are finished complete with primer and anti-static coating, ready for final painting; however, the new radome must be fit, trimmed, drilled and counter-sunk to the individual airplane. To assure electrical bond to the fuselage, a small metal tab or strap should be attached to the inner aft edge of the radome with one rivet after fitting. After installation of the tab, the radome should be painted with one coat of 528-104 flat black anti- static paint with 910-119 catalyst or equivalent. (Sterling anti-static black 22-3359 is a recommended equivalent optional coating).

6. BATTERY BOX AND LID

Apply a minimum of three coats in a vinyl coating procedure or equivalent to the interior surfaces of the battery box and lid as follows:

(1) One coat of EX2016G wash primer.

(2) One coat of intermediate vinyl.

(3) One coat of finish vinyl.

To ensure complete coverage, each coat must be of a different color, and must completely hide the preceding coat. The final coat shall be gray in color.

7. RUBBER SEALS

Apply one coat of a thoroughly dissolved solution of one part Oakite No. 6 and two parts water to all rubber surfaces that are come in contact with either metal or other rubber surfaces.

8. WHEELS AND BRAKES

To ensure protection of surface susceptible to mud and spray, prime and apply two coats of white epoxy paint to the following:

(1) Main landing gear wheel wells.

(2) Nose landing gear wheel well.

(3) Interior surfaces of the landing gear doors.

(4) Main and nose landing gear assemblies.

CAUTION: *Excessive paint can cause adverse affects on operation of wheel and brake. Internal areas of the brake (brake disc, linings, brake pistons) and wheel assembly (wheel bearings) must be masked off and kept free of paint.*

9. EPOXY TOPCOAT

When interior or exterior paint is required, refer to the airplane log book for the part number of the paint used on the airplane as it was delivered new. For older airplanes that do not have the paint part number listed in the airplane log book, contact the Commercial Service Department, Wichita, Kansas 67201, to determine the paint used on the airplane as it was delivered.

NOTE: All orders for Epoxy paint MUST include No. 5400 catalyst.

(1) Mix the paint and catalyst as directed by the manufacturer.

(2) Apply one topcoat with a spray gun using approximately 35 to 55 psi of air pressure. Two coats are normally required to fully conceal the primer and buildup the topcoat film to a thickness necessary for adequate service life and beauty. The Epoxy finish will normally cure to approximately 85% of its full hardness in 24 hours at temperatures of 80°F or higher.

10. TOUCH-UP REPAIR

(1) Mask around the skin containing the damaged area.

(2) Remove any loose edges of paint by using a high tack adhesive tape around the edge of the damaged area.

(3) Using coarse sandpaper, fair the edge of the damaged area with the metal.

(4) When the edge of the paint begins to fair into a smooth joint, use a fine grade of sandpaper to eliminate the sand scratches left by the coarse paper so that the finish will be perfectly smooth. Take care to avoid removing any more metal than is absolutely necessary.

(5) Wash the sanded area with a solvent, such as lacquer thinner or methyl propyl ketone. Change the washcloths used for this purpose frequently so that all of the sanding dirt will be picked up. After the area to be touched up has been cleaned with solvent until all trace of discoloration is gone, apply a thin coat of pretreatment primer to the damaged area.

NOTE: If a metal conversion coating such as iridite or alodine is used, the wash primer coating can be dispensed with. If the metal has not been treated with a metal conversion coating and no wash primer is available, carefully clean the surface to be touched up and apply Epoxy primer to the bare metal. This will produce a satisfactory undercoat for the repair area.

(6) Spray two or three coats of Epoxy for a heavier than normal buildup of primer.

(7) After the Epoxy primer has cured for 24 hours, sand the area being repaired with a medium fine sandpaper. Sand the edge of the repair area until the indentation where the metal and the old paint meet is gone. If necessary, apply additional Epoxy primer until the conjuction of the old paint with metal is no longer visible.

(8) Spray on two topcoats.

11. URETHANE PAINTS

The need for an extremely hard finish for protection against sandblast during take-off and landings led to the development of urethane coatings for airplanes. Urethane paint dries to a high gloss and retains color much better than standard finishes. It is unaffected by the chemicals in hydraulic fluids, deicers, and fuels and requires far less care and maintenance than other finishes.

A. Urethane Paint Repair Procedures

NOTE: Any time an airplane is repainted, inspect all placards to assure that they are not covered with paint, are easily readable, and are securely attached.

The time normally required for urethane paint to cure must be extended at temperatures below 70°F. The paint will not cure at temperatures below 60°F.

The surface on the Model 99 Airliner series is finished with pretreatment (wash) primer, urethane primer, and a topcoat of urethane. The following procedures include cleaning, paint stripping, pre-paint preparation, priming, applying a urethane topcoat, and an alternate method for small repairs not requiring paint stripping. Careful observance of these procedures should result in a smooth, hard, glossy finish with firm adhesion for maximum life.

NOTE: Do not intermix paint colors or paint and paint catalyst from different vendors.

Precut stripe and number patterns are available through Medagraphics of Kansas, 1720 S. 151 W., Route 1, Goddard, Kansas 67052.

B. Paint Stripping and Cleaning Urethane Paint

CAUTION: *Never use aluminum foil to mask electrothermal windshields during painting. The metal brighteners used on the aluminum and the stannous oxide used as an anti-static coating on electrothermal windshields may combine to form lead oxide, producing a mirror like glaze on windshield surface. If metal brighteners are to be used, cover the windshield with pasteboard or paper masking material.*

Because of their resistance to chemicals and solvents, urethane paints and primers require a special paint stripper.

(1) Mask around the edge of the skin or skins containing the damaged area. Use a double thickness of heavy paper to prevent accidental splashes of paint stripper from penetrating the masking.

(2) Apply urethane stripper as indicated by the manufacturer's directions. Try to stay approximately 1/8 inch away from the masking tape. This will necessitate a little more cleanup upon finishing, but will prevent damage to the finish on the next skin. The stripper will not attack aluminum during the stripping process and can be neutralized afterwards by rinsing the affected area with water.

CAUTION: *Urethane strippers usually contain acids that irritate or burn the skin. Wear rubber gloves and eye protection when using the stripper.*

(3) Rinse the area with water and dry.

(4) Wash the stripped area carefully with solvent such as acetone, methyl propyl ketone, or lacquer thinner. This will prevent tiny particles of loose paint from adhering to the stripped area.

(5) Using a nylon scratch pad or aluminum wool dipped in clean water, clean the surface with a cleanser such as Bon Ami, Ajax, Comet cleaner, etc. A good scouring will leave the surface completely clean.

(6) Thoroughly rinse with clean water and carefully dry the affected area. If the stripped area includes several joints; or skin laps, let the airplane sit until all moisture has dried. This may be accelerated by blowing the skin laps and reams with compressed air. Wet masking should be replaced.

C. Pretreatment (Wash) Primer for Urethane Paint

An acid etching primer that conforms to Specification MIL-C-8514 should be applied to improve adhesion of the finishing coats. EX2016G base and T6070 catalyst (products of Pratt and Lambert Industrial Coatings Division, P.O. Box 2153, Wichita, Kansas 67201) are used in equal parts as a pretreatment wash primer at the factory.

(1) Mix the primer in accordance with the manufacturer’s instructions.

(2) Apply a thin coat of primer. It should be permitted to dry for at least one hour, but not over six hours, before the next coat of urethane primer is applied.

D. Urethane Primer

(1) Mix the urethane catalyst and base in accordance with the manufacturer's instructions when preparing the primer.

NOTE: For the best results, these directions must be followed carefully, for some manufacturers require that the primer be allowed to set for 1/2 hour after the catalyst and base have been mixed while others recommend immediate use after mixing.

(2) Apply a coat of urethane primer with a spray gun using approximately 35 to 55 psi of air pressure. A dapple appearance only indicates that the coat is thin.

NOTE: The minimum drying time for urethane primer is approximately two hours under conditions of low humidity at temperatures of 85 to 90° F. When the primer coat cannot be scratched by fingernail or when sandpapering does not cause the primer to ball up, the urethane topcoat may be safely applied.

(3) If the initial coat is allowed to cure for more than 24 hours before the topcoat is applied, sand the primer slightly to roughen the surface and assure adhesion. Wipe off the surface with a cloth dampened with a solvent, (such as lacquer thinner), then apply the topcoat.

E. Urethane Topcoats

When interior or exterior paint is required, refer to the airplane log book for the part number of the paint used on the airplane as it was delivered new. For older airplanes that do not have the part number listed in the airplane log book, contact the Commercial Service Department, Hawker Beechcraft Corporation, Wichita, Kansas 67201 to determine the paint used on the airplane as delivered.

(1) Mix the paint and catalyst as directed by the manufacturer.

(2) Apply the topcoat with a spray gun under approximately 35 to 55 psi of air pressure. Two coats are required to fully conceal the primer and build up the topcoat film necessary for adequate service life and beauty. The urethane finish will normally cure to approximately 85% of its full hardness in 24 hours at temperatures of 80°F or higher.

F. Urethane Touch-Up Repair

(1) Mask around the skin containing the damaged area.

(2) Remove all loose edges of paint by using a high tack adhesive tape around the edge of the damaged skin.

(3) Using a coarse sandpaper, fair the edge of the damaged area with the metal.

(4) When the edge of the paint begins to fair into a smooth joint, use a fine grade sandpaper to eliminate the scratches left by the coarse paper. Take care to avoid removing any more metal than is absolutely necessary.

(5) Wash the sanded area with a solvent, such as lacquer thinner or toluol. Change the wash cloths often so that all the sanding dirt will be picked up.

(6) After the area to be touched up has been cleaned with solvent until all traces of discoloration are gone, apply a thin coat of pretreatment primer to the damaged area.

NOTE: If a metal conversion coating such as iridite or alodine is used, the wash primer coating can be dispensed with. If the metal has not been treated with a metal conversion coating and no wash primer is available, carefully clean the surface to be touched up and apply urethane primer to the bare metal. This should produce a satisfactory undercoat for the repair area.

(7) After the urethane primer has cured for 24 hours, sand the area under repair with medium fine sandpaper. Sand the edge of the repair area until the indentation where the metal and old paint meet is gone. If necessary, apply additional urethane primer until the juncture of old paint with metal is not longer visible.

(8) Spray on two topcoats.

12. PAINTING MAGNESIUM

A. Paint Removal from Magnesium Surfaces

(1) Mask around the edge of the damaged area with a double thickness of heavy paper to prevent accidental splashes of paint stripper from penetrating the masking.

(2) Apply Turco No. 4260 (product of Turco Products Inc., Los Angeles, California) to the skin under repair with a brush of non-atomizing gun.

CAUTION: *Stripping should be accomplished in a well ventilated area since prolonged exposure to high concentrates of Turco No. 4260 vapor may irritate the eyes and lungs.*

(3) Allow the paint stripper to work for 20 to 30 minutes, then work the remaining paint loose with a bristle brush.

CAUTION: *Never use a wire brush for it will damage the magnesium surface.*

(4) Remove the masking paper and wash the affected area thoroughly with water under high pressure. Remove all remnants of paint with lacquer thinner.

(5) Sand the repaired area lightly, then prepare and apply Dow Treatment Number 19, as follows:

(a) Place 3/4 gallon of 70 to 90°F water in a one gallon stainless steel, aluminum, or vinyl polyethylene container.

(b) Add 1-1/3 oz. of chromic acid (Cr03), then 1 oz. of calcium sulfate (CaS04) (obtain chemicals locally).

(c) Add 1/4 gallon (70 to 90°F) of water and stir vigorously for at least 15 minutes.

(d) Brush the solution on the bare area, keeping the area wet for at least 1 to 3 minutes until a brown film appears.

CAUTION: *DO NOT exceed 3 minutes.*

(6) Rinse the treated area in cold (not hot) running water and dry in an oven, or dry by exposure to hot air blast.

NOTE: If cold running water is not available, rinsing may be eliminated and the area dried as stated above.

(7) Brush two coats (minimum of 30 minutes apart) of epoxy primer (MIL-P-23377) on and around the reworked area. Assure adequate penetration of primer into the treated area.

B. Painting Magnesium Surfaces

(1) Prepare the surface to be repainted as indicated under PAINT REMOVAL FROM MAGNESIUM SURFACES. Clean the affected area thoroughly with lacquer thinner or an equivalent solvent.

(2) Prime the affected area and apply the topcoat as indicated under the applicable (for urethane or epoxy) paint procedure in this Chapter.

NOTE: *Do not apply wash primer to magnesium surfaces. Allow a minimum of four hours drying time between application of the primer and topcoat.*

When interior or exterior paint is required, refer to the airplane log book for the part number of the paint used on the airplane as it was delivered new. For older airplanes that do not have the paint part number listed in the airplane log book, contact the Commercial Service Department, Hawker Beechcraft Corporation, Wichita, Kansas 67201 to determine the paint used on the airplane as delivered.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

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AIR CONDITIONING
HEATING
DESCRIPTION AND OPERATION

1. HEATING SYSTEM

A. Prior to U-165, Except U-50

The combustion heater is a 100,000 BTU unit designed to operate on any fuel that the engine will bum. Fuel for the heater is tapped off the inboard side of the left nacelle fuel cell. The fuel to the heater flows through a filter on the inboard side of the wheel well to the fuel pump and fuel regulator in the left wing center section. The fuel regulator is set at 95 to 105 psi and returns all fuel in excess of this setting to the cross vent line of the fuel system. The regulated fuel then flows to a heater fuel control valve located adjacent to the fuel pump in the left wing center section An additional fuel filter is incorporated in the inlet port of the heater fuel control valve. The fuel is then routed to the heater compartment in the nose section, where it passes through a fuel line heater located adjacent to the airplane heater When the heater is operating, the igniter, mounted just aft of the heater, is on continuously.

Air is supplied to the heater by a ventilation blower located under the forward baggage compartment to the left of the heater. Vent blower operation is sensed by a differential pressure switch mounted under the inboard end of the blower. Should the blower malfunction, the pressure switch closes and grounds the heater control circuitry to blow the heater fuse. The heater fuse is located on the relay panel beneath the left cabin floor forward of the main spar. The blower is controlled through the CABIN TEMP MODE switch or the manual VENT BLOWER switch. The manual switch has HIGH, LOW, and OFF positions. With the manual VENT BLOWER switch in the OFF position, the blower will continue to operate at low speed unless the CABIN TEMP MODE switch is also in the OFF position. An additional vent blower is available as an optional installation to increase cabin air circulation. This aft blower is installed under the RH floorboard at fuselage station 256.75. Both blowers provide ventilation air for the evaporators on airplanes equipped with air conditioning.

Heater combustion air is provided by a blower located under the baggage compartment floor between the heater and the well for the nose landing gear tire If air flow through the heater is too low to support combustion, a differential pressure switch, located directly under the heater, closes and grounds the heater control circuitry to blow the heater fuse. A combustion air solenoid valve, attached to the duct between the combustion air blower and heater, closes when the air conditioner is operating to prevent the loss of conditioned air through the combustion air duct.

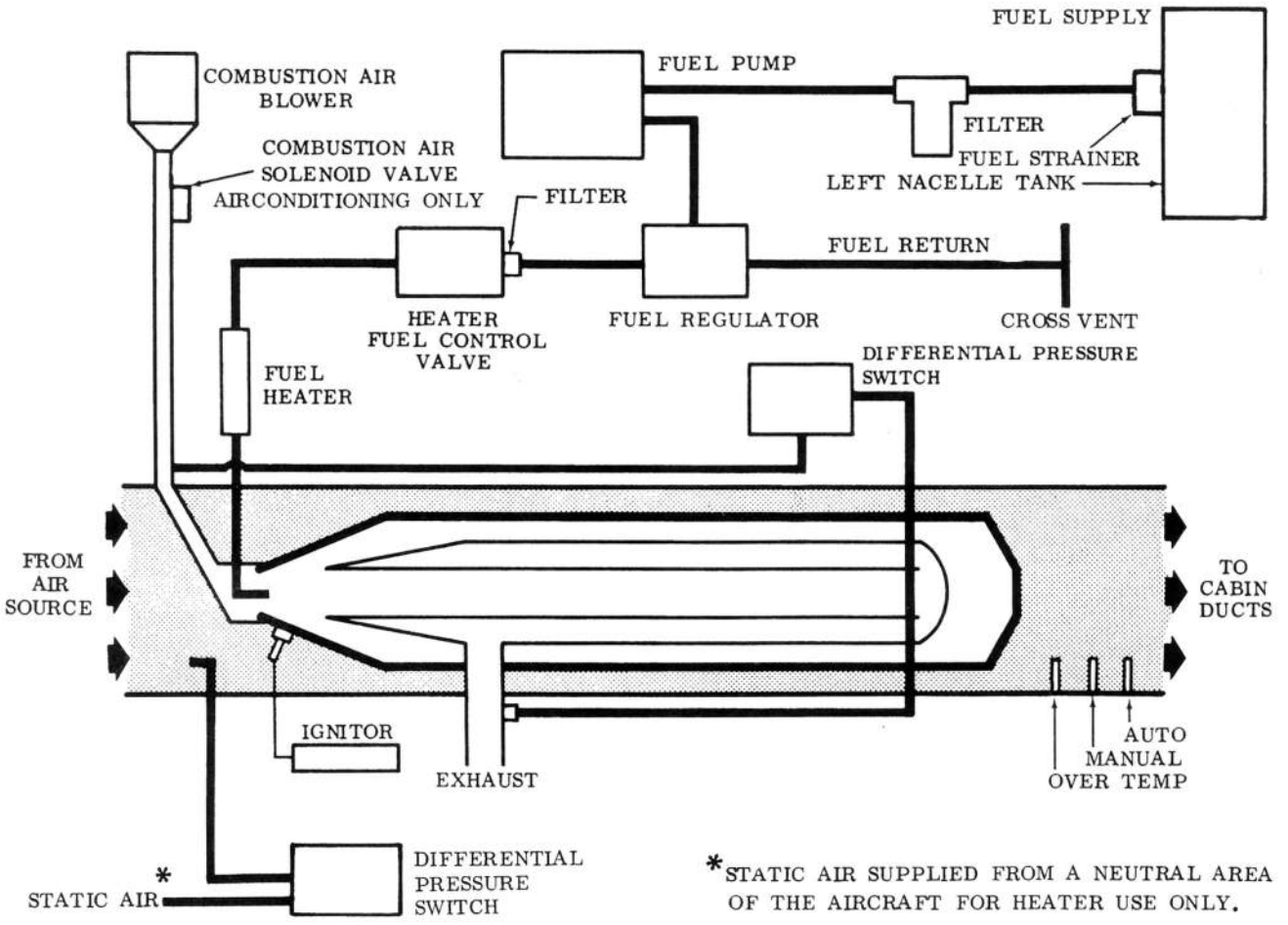

Figure 1
Heating System Schematic

1. Cabin Air Exhausts
2. Cabin Air Exhaust Plenum
3. Ram Air Intake
4. Outside Air Sense Element
5. Ram Air Door
6. Ventilation Blower
7. Ventilation Air Plenum
8. Recirculating Air Return
9. Combustion Air Blower
10. Combustion Air Heater
11. Ignitor
12. Heater Discharge Sensing Element
13. Copilot's Outlet Shutoff
14. Windshield Defroster Outlet
15. Defroster Shutoff Valve
16. Pilot's Outlet Shutoff
17. Heater Discharge Duct Shutoff
18. Cockpit Fresh Air Outlets
19. Passenger Fresh Air Outlets
20. Air Outlets
21. Temperature Control Box
22. Optional Aft Vent Blower
23. Cabin Sense Element
24. Heater Fuel Shutoff Valve
25. Fuel Pressure Regulator
26. Heater Fuel Pump
27. Heater Fuel Filter
28. Ram Air Control
29. Heater Cycling Switch
30. Over Temperature Safety Switch

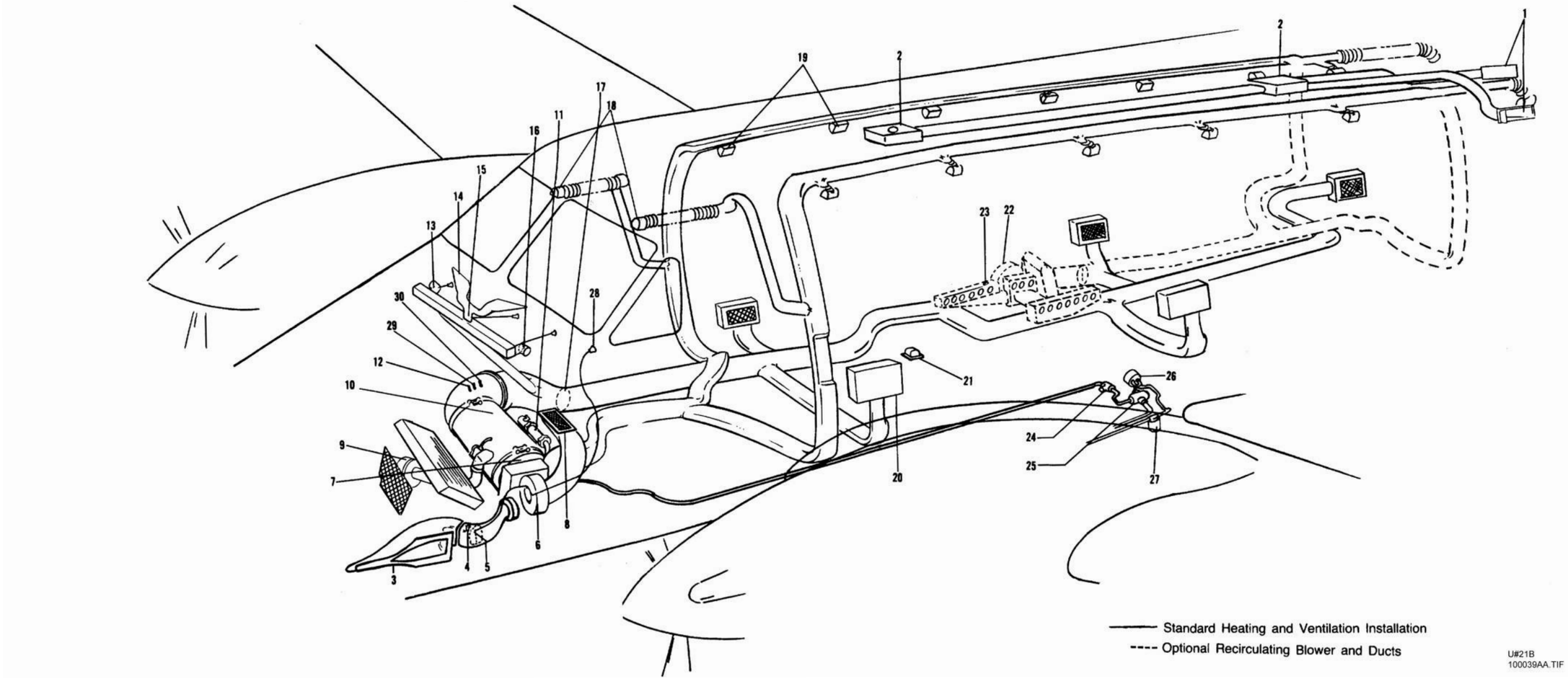

Figure 1
Standard Heating System Installation
(Prior to U-165, Except U-50)

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The heater is controlled by the CABIN TEMP MODE switch located on the RH subpanel. When the switch is in the MANUAL HEAT position, a 225°F thermal cycling switch controls heater operation. If the cycling thermostat fails to open, a heater over-temperature safety switch will close when the temperature in the heater discharge duct reaches 300°F. This will cause a shorting or overload condition that blows the fuse in the circuit supplying power to the heater fuel pump, ignition unit, and fuel line heater. The thermal cycling switch is mounted just aft of the heater discharge air sensor and forward of the over-temperature safety switch on the heater discharge duct at the RH end of the heater installation. When the CABIN TEMP MODE switch is in the AUTO position, heater cycling is automatically controlled by a temperature control box mounted under the center aisle floor immediately aft of the main spar. The temperature control box regulates heater cycling in response to the information supplied from a balanced wheatstone bridge circuit made up of a cabin air sensor, outside air temperature sensor, heater discharge temperature sensor, and a cabin control rheostat. The outside air temperature sensor is attached to the ram air intake duct under the nose baggage compartment floor adjacent to the well for the nose landing gear tire. The heater discharge temperature sensor is mounted directly in front of the heater cycling and over-temperature safety switches on the discharge duct at the RH end of the heater installation. The location of the cabin air temperature sensor depends upon the type of heater installation with which the airplane is equipped. On the standard heater installation (without the aft vent blower), the sensor is installed in the cabin air exhaust plenum at the top center of the fuselage parallel to the second cabin window. On airplane equipped with the aft vent blower, the sensor is installed on the duct under the floor on the left side of the fuselage immediately forward of the vent blower and even with the fourth cabin window. A thermostat switches between the heating and coding modes automatically to maintain an even temperature in the cabin. A 4°F dead zone between the heating and cooling modes prevents the system from constantly fluctuating back and forth between the two modes. Interior temperature during the automatic mode is controlled through a rheostat in the left subpanel.

The pilot and copilot have separate controls to operate individual air outlets under the instrument panel. The DEFROST AIR control is located next to the copilot's outlet control. The defroster valve, which regulates air flow to the windshield defrost ducts, works in conjunction with the heater discharge duct shutoff valve. As the defroster valve starts to open, the shut-off valve begins to close the heater discharge duct. If the DEFROST AIR control is pulled completely out, all heat will be routed through the defroster duct. If the defroster control is only partially pulled, air will divert to the defroster ducts and still allow heater air to the cabin outlets. When the RAM AIR control is pulled, a door in the ram air scoop opens to allow outside air to the ventilation blower. The outside air then displaces the cabin air out the overhead exhaust ducts.

B. U-50, U-165 and After

Bleed air from the engine compressors, stage P_3, is the primary source of cabin heat for the Model C99 Airliner Series airplanes. Normally, either engine is capable of providing adequate bleed air to heat the cabin and cockpit areas of the airplane. If, however, bleed air heating is interrupted or inadequate to maintain cabin heat at the desired level, an optional electrical resistance heater can be used to augment the bleed air heating. The electrical resistance heater also makes it possible to heat the airplane interior during ground operation while the airplane is connected to an auxiliary power source.

Bleed air from the engines is controlled through shutoff valve/regulators mounted in the engine nacelle area just aft of the respective firewalls. A check valve/tee junction between the two bleed air sources supplies bleed air to the cabin through a modulating valve, which modulates the temperature of the air entering the distributing ductwork. A venturi, downstream of the modulating valve, provides pressure differential for determining bleed air flow.

A forward ventilation blower. located under the forward baggage compartment, supplies air for both cooling and heating operations. The operation of the vent blower is sensed by a differential pressure switch, which will remove power from the electric heater and close the bleed air modulating valve in the event of a ventilation blower failure.

2. BLEED AIR HEAT

The LH and RH shutoff valve/regulators are controlled through two switches on the right inboard subpanel. When the switches are in the ON position, the valve/regulators are energized, opening the valves and regulating the bleed air flow into the cabin. When in the OFF position, they are de-energized and the power is shunted to a pair of lights on the right and left advisory annunciator panels, alerting the pilot that the environmental bleed air is off.

The bleed air modulating valve is primarily a motor- actuated butterfly valve which responds to a continually varying degree of demand. The commands for the valve to open or close is transmitted by the cabin temperature controller.

The bleed air venturi high and low pressure outlets are used to measure bleed air flow according to the pressure differentials. The outlets are connected by tubing to a test port fixture, mounted under the center cabin floorboards forward of the main spar.

Check valves in the tee junction between the two bleed air sources, prevent the reverse flow of bleed air in the event of failure of either of the bleed air sources.

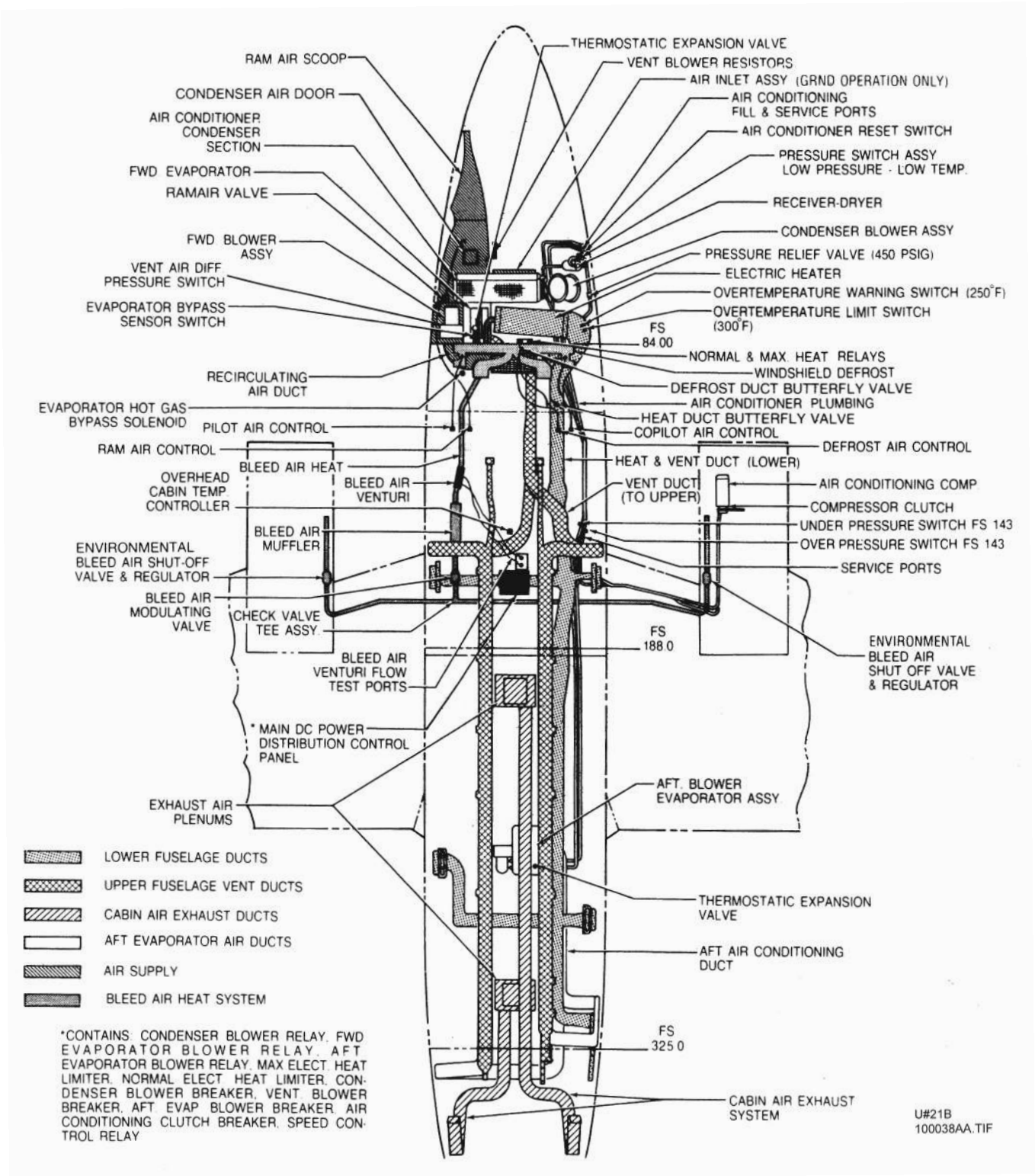


Figure 3
Heating and Cooling System Schematic (U-50, U-165 and After)

3. ELECTRICAL RESISTANCE HEAT

The optional electrical resistance heater (maximum output 23,700 BTU at 8,000 watts) is installed in place of a section of ductwork to the right of the blower assembly. Bilevel heating is accomplished by selecting only part of the heating elements for normal heating, or all of the heating elements for maximum heating. Maximum heat can only be obtained while on the ground and an external power source is connected to the external power receptacle. Energizing current for the maximum heating relay is taken from the small pin of the external power plug. Normal or maximum heating is controlled through a magnetic-hold electric heat switch, located in the right inboard subpanel, which will keep the switch contacts closed in the maximum heating position only if the airplane is connected to an external power supply. If the electric heater is to be used as supplemental heat during flight operations, the electric heater switch must be placed in the normal electric heat position. When the electric heater is being used, either in flight or on the ground, it will not be cycled out unless the over temperature switch, located in the ductwork, interrupts power to the heater or in the event of a blower failure. Any time the electric heater is operating, the forward ventilation blower will be switched to high speed regardless of the position of the blower switch.

When the air conditioning relay is closed, the electric heater is switched off.

4. FORWARD VENTILATION BLOWER

The forward ventilation blower supplies the necessary air volume for both heating and cooling modes of the environmental system. The blower is controlled through a three position switch on the right inboard subpanel. The switch positions are HI, LO and AUTO. When the switch is in the AUTO position, the blower will be controlled by the cabin temperature mode switch, which will switch the blower to low speed at any time the cabin temperature mode switch is positioned to any on mode.

A differential pressure switch, mounted in the blower assembly, serves as a safety switch for the heating system should the airflow from the blower be interrupted for any reason. The pressure switch is open during blower operation and is closed if airflow is interrupted. When the pressure switch closes, a relay is energized and power to the electric heater is stopped. At the same time, the relay transmits a simulated cool command to the modulating valve, causing it to close and shut off the bleed air flow to the cabin.

5. DUCT OVERTEMPERATURE SWITCHES

Two overtemperature switches, which close on rising temperature, are installed in the ductwork. One closes at 250°F and the other closes at 300°F. When closed, the 250°F switch transmits a signal to the warning light on the annunciator panel, advising the pilot of an impending duct overtemperature condition. If remedial action is not taken and the duct temperature continues to rise, the second switch removes the power from the electric heater and transmits a simulated cool command to the modulator valve, causing it to close and stop cabin bleed air flow.

AIR CONDITIONING
HEATING
TROUBLESHOOTING

1. HEATING SYSTEM

Table 101
Troubleshooting - Heater System
(Prior to U-165, Except U-50)

PROBLEM	POSSIBLE CAUSE	CORRECTIVE ACTION
1. Blower runs but heater will not start.	a. Fuse blown due to overheat condition causing overtemperature switch to close.	a. Replace fuse, check entire circuit for cause of overheat condition.
	b. Defective temperature limit switch.	b. Check switch and lead.
	c. Faulty ignition unit, vibrator.	c. Switch to reverse vibrator contacts. If this corrects trouble, replace vibrator at 1000 hours.
	d. Fuel control valve inoperative.	d. Disconnect fuel line and check for fuel flow.
	e. Faulty ignition unit coil.	e. Remove shielded lead from spark plug and check for spark.
	f. Fouled or defective spark plug.	f. If test (d) above produces spark, remove and clean or replace igniter plug.
	g. Heater control rheostat defective.	g. Connect voltmeter across rheosat leads and turn rheostat through its range. Voltage should decrease as knob is turned toward high temperature. If not, replace rheostat.
	h. Shorted lead between cabin thermostat and heater control box.	
	i. Defective heater control box.	i. Switch to MANUAL heat. If heater starts, replace heater control box.
	j. Faulty shielded ignition lead.	
	k. Worn ground electrodes.	
	l. Fuel regulator faulty.	
	m. Faulty combustion air blower.	
	n. Fuel filter clogged.	
	o. Spray nozzle clogged.	

Table 101
Troubleshooting - Heater System
(Prior to U-165, Except U-50)

PROBLEM	POSSIBLE CAUSE	CORRECTIVE ACTION
1. Blower runs but heater will not start. (Continued)	p. Leaks or obstruction in combustion or supply line.	
	q. Faulty or out-of-adjustment differential pressure switch.	q. Readjust or replace.
2. Heater will not shut off automatically.	a. One or both heater controls defective.	a. Check as in 1., item f.
	b. Open circuit from rheostat to cabin thermostat.	b. Check circuit from rheostat to cabin thermostat.
	c. Defective ground at cabin thermostat.	
	d. Defective cabin thermostat.	d. Ground lead from thermostat to heater control box; if heater turns off, replace the thermostat.
3. Heater backfires intermittently.	a. Loose connection in control circuit, loose high-tension lead to spark plug.	a. Check connections, check wiring breaks.
	b. Mixture too rich.	b. Make checks listed in item 4.
4. Fuel mixture too rich, pulsating combustion or smoky exhaust.	a. Restriction in combustion air duct.	a. Check air ducts for obstructions.
	b. Restriction in exhaust duct.	b. Check exhaust outlet for obstructions.
	c. Loose core in fuel nozzle.	c. Clean nozzle. Make sure core is seated securely in shell.
	d. Fuel regulator may be faulty.	d. See 1., item I.
	e. Fouled spark plug.	

AIR CONDITIONING
HEATING
MAINTENANCE PRACTICES

1. COMBUSTION HEATER (Prior to U-165, Except U-50)

A. Removal

(1) Remove the aft flooring in the nose baggage compartment to gain access to the heater.

(2) Remove the differential pressure sense line (1) at the heater (Ref. Figure 201).

(3) Remove the shroud covering the fuel line and then disconnect the fuel line.

(4) Loosen the clamps securing the combustion air blower duct (2), exhaust shroud (3), and fuel drain shroud (4).

(5) Disconnect the igniter spark plug lead (5). Loosen the clamps on the igniter enough to slide it clear of the heater.

(6) Remove the clamps (6) at each end of the heater.

(7) Remove the eight bolts (7) at the heater discharge duct and disconnect the heater sensing elements in the discharge duct.

(8) Lift the heater straight up and out of the compartment.

NOTE: Replace both the front and rear clamp seals and the gasket at the heater discharge duct any time the heater is removed.

B. Installation

(1) Install the heater through the access under the aft flooring of the nose baggage compartments (Ref. Figure 201).

(2) Connect the heater sensing elements.

(3) Install the eight bolts (7) at the heater discharge duct.

(4) Install the clamps (6) at each end of the heater.

(5) Install the igniter spark plug lead (5) and tighten clamps.

(6) Secure the drain shroud (4), exhaust shroud (3) and blower duct by tightening clamps.

(7) Connect fuel line and then install fuel line covering shroud.

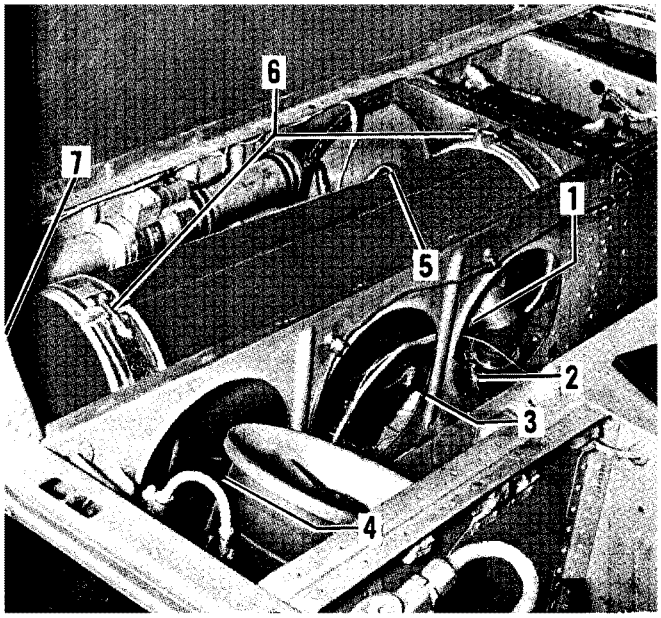


Figure 201
Heater Removal and Installation

2. HEATER FUEL PUMP

A. Removal

(1) Remove the large access door located on the lower side of the left wing center section.

(2) Remove strobe light power supply (if installed).

(3) Disconnect the inlet, outlet, and drain lines from the pump.

(4) Disconnect the electrical plug from the pump.

(5) Remove the four bolts securing the pump mounting base to the airplane.

(6) Remove the four bolts securing the pump to the base.

B. Installation

(1) Position the pump on its mounting base and install the four retaining bolts. Safety the bolts with lockwire.

(2) Position the base on its mounting bracket and install the retaining bolts.

(3) Connect the inlet, outlet, and drain lines to the pump.

(4) Connect the electrical leads to the pump.

(5) Install the access door.

3. COMBUSTION AIR BLOWER

A. Removal

(1) Disconnect the aft nose gear doors to gain access to the aft portion of the nose wheel well.

(2) Remove the plate (or screen on airplane with air conditioning installed) at the aft uppermost portion of the wheel well to gain access to the combustion air blower.

(3) Remove the retaining clamps to the combustion air blower and the clamp attaching the blower duct to the hose and remove the blower.

B. Installation

(1) Install the combustion blower through the access at the aft uppermost portion of the nose wheel well.

(2) Attach the hose to the blower duct with the attaching clamp.

(3) Install retaining clamps to the combustion air blower.

(4) Replace access plate (screen on airplane with air conditioning installed).

4. DIFFERENTIAL PRESSURE SWITCH

A. Adjustment

If the switch has not been set or if it is desired to change the set point, turn the slotted adjustment screw clockwise to increase and counterclockwise to decrease the set point. The proper setting for the switch is 1.5 inches water.

5. HEATER CONTROL BOX

A. Adjustment

The heater control box is located beneath the center aisle immediately aft of the main spar. If the desired temperature cannot be obtained by operating the cockpit rheostat, the heater control box should be checked for proper adjustment.

(1) Rotate the "INCR. FREQ." knob counterclockwise as far as possible and secure it in place with the locknut (Ref. Figure 202).

(2) Rotate the "INCR. TEMP." knob clockwise as far as possible and secure it in place with the locknut.

NOTE: The preceding adjustments provide optimum heater operation. Should the heater control system continue to operate improperly with these settings, have the system inspected at an approved heater repair facility instead of attempting further adjustments.

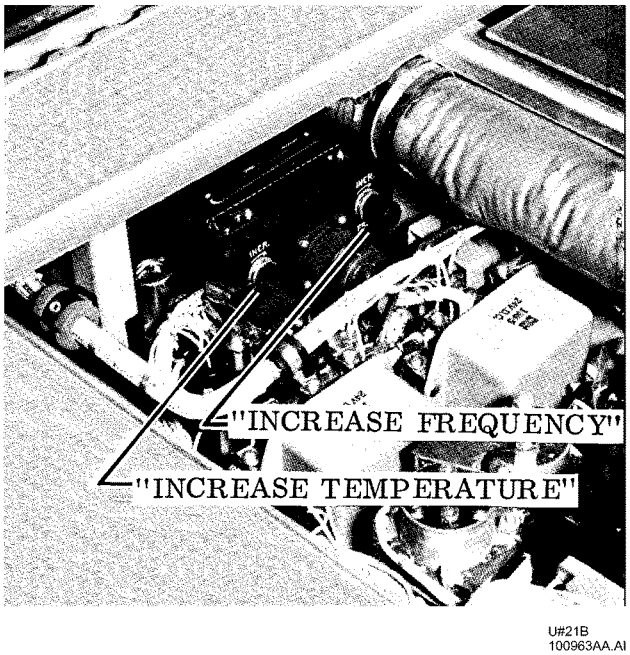


Figure 202
Heater Control Box Adjustment (Typical)

6. HEATER SPARK PLUG

A. Cleaning

Clean heater spark plugs as follows:

(1) Remove the spark plug located under the RH aft section of the nose baggage compartment floor. With a deep socket, remove the spark plug. Always place the wrench on the lower (larger) hex section.

(2) Before cleaning, examine the spark plug for evidence of cracked or broken porcelain, arcing, or carbon tracks inside the spark plug well. If cracks are found in the porcelain, the plug should be discarded without further examination. Arcing or carbon tracks may be caused by shorting of the plug or by dirt on the spring connector that seats in the well. In either case the fault should be corrected before installing the plug in service or using a new plug.

(3) Wipe out the inside of the well of the plug with a clean cloth dampened with solvent (38, Table 6, 91-00-00) to remove any grease or carbon deposits.

(4) Clean the spark plug by sandblasting.

NOTE: Close the well of the spark plug with a stopper or cap to prevent dirt or sand from lodging there during the cleaning. One or more adapters may be required to position the electrode where it can be cleaned effectively. The use of adapters may make it impossible to use the regular levers for depressing the sand and air valves. In the latter event, use apiece of flat heavy metal stock against the valve pins to control the valves of the sandblast machine in proper sequence. The use of goggles is recommended to protect the eyes.

Sandblast the spark plug only long enough to remove the carbon deposit from the electrode and ceramic area. If necessary, rotate the plug and move it up and down by means of additional adapters to finish cleaning the entire electrode and ceramic areas.

(5) Examine the ceramic area for porosity. Spark plugs with extremely porous surfaces have a limited service life and should be replaced.

(6) Inspect the base of the threaded portion of the spark plug for any carbon deposits that sandblasting may have failed to remove. Remove any such indications of carbon prior to reinstalling the spark plug. Blow out any sand that may be lodged in the spark plug.

(7) Inspect the electrode for concentricity. Any appreciable misalignment would make setting the proper spark plug gap extremely difficult. Check the spark plug for proper gapping.

B. Gap

Check that the gap between the spark plug electrode and ground electrode is between 0.187 (3/16) and 0.218 (7/32) inch. To adjust the spark plug gap, add or remove shims under the ground electrode (Ref. Figure 203).

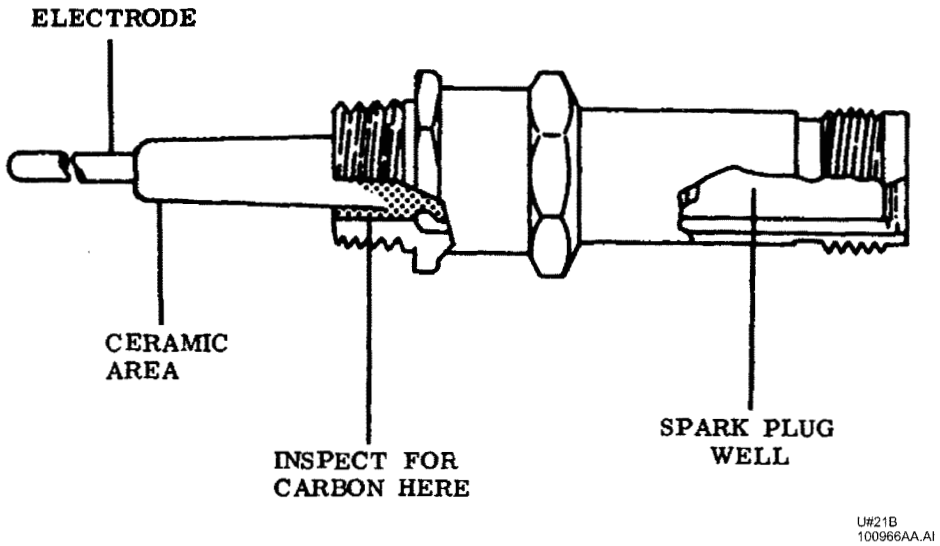


Figure 203
Heater Spark Plug

C. Torque Values

Failure to observe the torque values below may result in damage to igniter units or cause the upper and lower hex sections of the spark plug to separate, leaving the lower section still installed in the heater when the upper section is removed.

(1) Apply a torque of 28 foot-pounds when installing the spark plug in the heater.

NOTE: Always place the wrench on the lower (larger) hex section when removing or installing the heater spark plug.

(2) Any time that the spark plug has been disassembled for cleaning, apply a torque of 36 foot-pounds when installing the upper section in the lower hex section of the spark plug.

(3) Apply a torque of 20 foot-pounds when attaching the shielded lead assembly to the spark plug.

(4) Apply a torque of 25 inch-pounds when attaching the shielded lead assembly to the igniter unit.

7. IGNITER VIBRATOR

A. Replacement

(1) Remove the safety wire from the vibrator retaining ring (1) and remove the ring (Ref. Figure 204).

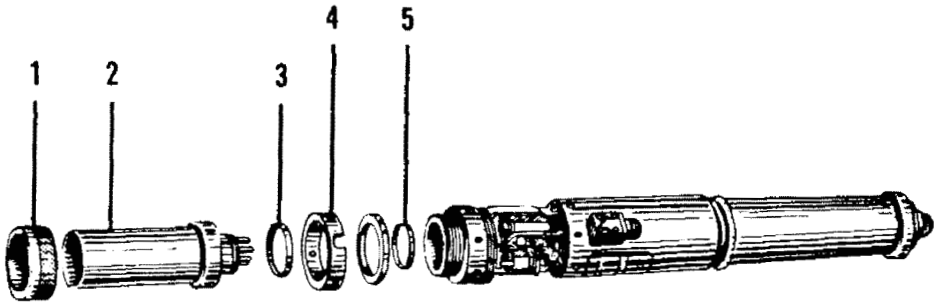
(2) Remove the vibrator (2) and radio noise shield (3).

NOTE: Make sure that the packing (5) remains in place.

(3) Install a new vibrator (2) with the radio noise shield (3).

(4) Install the vibrator retaining ring (1) and safety it with wire to one of the screws in the vibrator support collar (4).

CAUTION: *To protect the alternate set of vibrator points from damage install a jumper wire between terminals 7 and 8 in the nose gear wheel well after approximately 1000 hours of heater operation on airplanes serials U-1 through U- 59, U-61 through U-65, U-67 through U-87, U- 89 and U-90 that have not incorporated Kit No. 99-5002S. On the above noted airplanes that have incorporated Kit No. 99-50028 and airplanes serials U-60. U-66, U-88, U-99, and after, a selector switch is located under the right hand sub panel to change to an alternate set of points after approximately 1000 hours of heater operation. Replace the igniter vibrator on all airplanes after 2000 hours of heater operation.*



1. Retaining Ring
2. Vibrator
3. Radio Noise Shield
4. Support Collar
5. Packing

U#21B
100965AA.AI

Figure 204
Heater Igniter

8. HEATER FUEL SYSTEM

A. Checks

(1) Inspect the fuel nozzle for a buildup of carbon. The presence of such a restriction will result in an incorrect fuel-air ratio by preventing complete atomization of fuel in the heater. In this case, the nozzle must be removed for cleaning or replacement.

(2) Should smoke appear in the cabin during heater operation, bake the fiberglass ducts. This may be accomplished by operating the heater for approximately three hours in the manual heat mode. This will cure the heater ducts until they no longer smoke. Keep the cabin door and cockpit windows open during the curing period.

B. Regulator Adjustment

The fuel regulator, located in the center section of the left wing may be adjusted as follows:

(1) Attach a dead head pressure gage to the line on the inlet side of the heater fuel control valve in the left wing center section.

(2) Turn the locknut and cap at the aft end of the fuel regulator (IN to increase the setting or OUT to decrease it) until the gage registers between 95 and 105 psi.

9. HEATER FUEL NOZZLE

A. Aeration Gap

Run a wire type feeler gage around the nozzle inside the combustion head to check for a clearance of 0.010 to 0.035 inch between the nozzle and head. Increase this clearance with an A55D91 gasket if less than the prescribed limits. For better protection against carbon deposits on the face of the nozzle, it is recommended that this clearance be maintained at its maximum limit.

B. Disassembly, Cleaning, and Reassembly

NOTE: The heater must be removed to gain access to the heater fuel nozzles on aircraft equipped with air conditioning.

(1) Remove the floor panels over the heater and inlet plenum (Ref. Figure 205).

(2) Disconnect the air tube from the jacket fitting inside the heater jacket and remove the fuel fitting shroud.

(3) Remove the nut and washer that attaches the fuel fitting to the jacket.

(4) Remove the safety wire and screws attaching the fuel and air assembly to the heater combustion head and lift the assembly out.

(5) Remove the spray nozzle from the fuel and air assembly by unscrewing the nozzle body from the assembly.

(6) Unscrew the fuel strainer from the spray nozzle body. The strainer should only be finger tight.

(7) Unscrew the core from the body using a screwdriver.

(8) Clean the fuel strainer, core and body by immersing them in solvent (38, Table 6, 91-00-00).

(9) Remove the parts and check them for cleanliness. Use a small, soft non-metallic brush to aid in removing the more stubborn foreign material or particles.

NOTE: Fuel gum or varnish may be removed by soaking the parts in isopropyl alcohol (Specification TT-1-735 or its equivalent) (90, Table 6, 91-00-00) for two or three hours.

(10) If brushing fails to remove the foreign matter from the grooves in the body, core, and orifice, they may be cleaned using a small stick made by shaping a piece of soft wood (a toothpick is satisfactory).

CAUTION: *Do not use a metal tool to scrape these parts as this upsets the flow characteristics of the nozzle.*

(11) Rinse the parts carefully in clean solvent (38, Table 6, 91-00-00) and dry with filtered compressed air.

(12) When assembling the spray nozzle, screw the core into the body using a screwdriver.

(13) Screw the filter into the spray nozzle finger tight.

(14) Screw the node into the fuel and air assembly.

(15) Replace the fuel and air assembly on the combustion head and connect lines and safety wire.

(16) Reinstall the fuel fitting shroud, reconnect the air tube to the jacket fitting inside the heater jacket and reinstall the floor panels over the heater and inlet plenum.

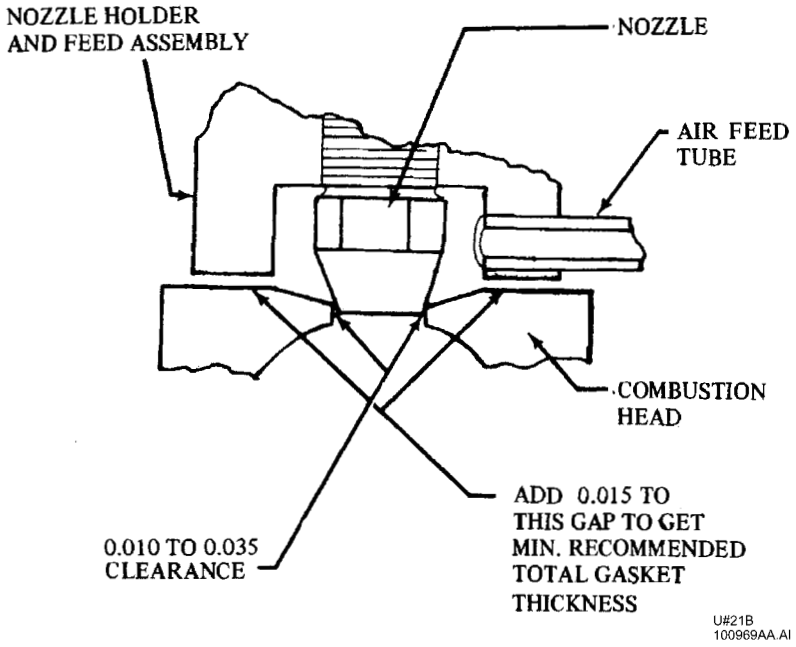


Figure 205
Heater Fuel Nozzle Aeration Gap

10. TESTS

A. Heater Pressure

The heater normally should be overhauled every 1500 hours of heater operation or 3000 hours of airplane operation. Two pressure tests should be conducted during this period (at intervals of 500 hours of heater operation or 1000 airplane hours) to assure that the heater is not leaking excessively. If excess leakage occurs, the heater should be overhauled at that time.

Periodic pressure tests may be conducted with the heater in the airplane as follows: Seal the fuel inlet, combustion air duct, and exhaust opening and connect an air source to the drain connection. The air line should have two valves in series to assure positive air shutoff and a 20-inch mercury manometer or an accurate pressure gage between the valves and the heater. Pressurize the heater to 7 psi (14.25 inches of mercury) and close both valves. The maximum allowable pressure drop is 2.5 psi 95.1 inches of mercury) in 15 seconds. Should a greater pressure drop be observed, remove the heater and repair the leak. If the pressure drop is still excessive, the heater should be overhauled.

B. Time Delay Relay

(1) Remove the relay and place an ohmmeter (RX1scale) across pins 3 and 8.

(2) Connect a 28 VDC power source to pins 1 and 6 of the time delay relay. After 5 seconds the ohmmeter should indicate a closed circuit at the relay points.

C. Temperature Sensors and Control Rheostats

NOTE: The airplane must be in a location, such as a hangar, where temperatures inside and outside of the airplane are equal and stable. When making the tests, slight variances may occur; however, a defective component will give a definite indication.

(1) Disconnect the electrical plug from the temperature control box.

(2) Using the graph in Figure 206, determine the correct resistance for the cabin air sensing element.

NOTE: On standard installations (without the aft vent blower), the sensor is located in the cabin air exhaust plenum at the top center of the fuselage parallel to the second cabin window.

On airplanes equipped with the aft vent blower, the sensor is located on the duct under the floor on the right side of the fuselage immediately forward of the vent blower and even with the fourth cabin window.

(3) Using the graphs in Figures 207 and 208, determine the correct resistance for the heater discharge sensing element and the outside air sensing element respectively, then add these resistance values.

NOTE: The heater discharge temperature sensor is attached to the discharge duct on the RH end of the heater installation. The outside air sensor is attached to the ram air intake duct under the nose baggage compartment floor adjacent to the well for the nose landing gear tire.

Measure the resistance between pin C and ground. This resistance should equal the added resistance of the heater discharge sensing element at pins C and D on the element. If the resistance does not equal the determined value, it is defective. If the resistance is correct, the outside air sensing element is defective.

(4) Rotate the cabin rheostat to the full increase position. Using the graph in Figure 208, determine the resistance for the heater discharge sensing element.

(5) Measure the resistance between pin E and D. If the resistance does not equal the determined value, the heater discharge element is defective.

(6) Rotate the cabin temperature control rheostat to the decrease position. The resistance on the ohmmeter should increase 250 ohms. If the resistance increase is high or low, the cabin temperature control rheostat is defective.

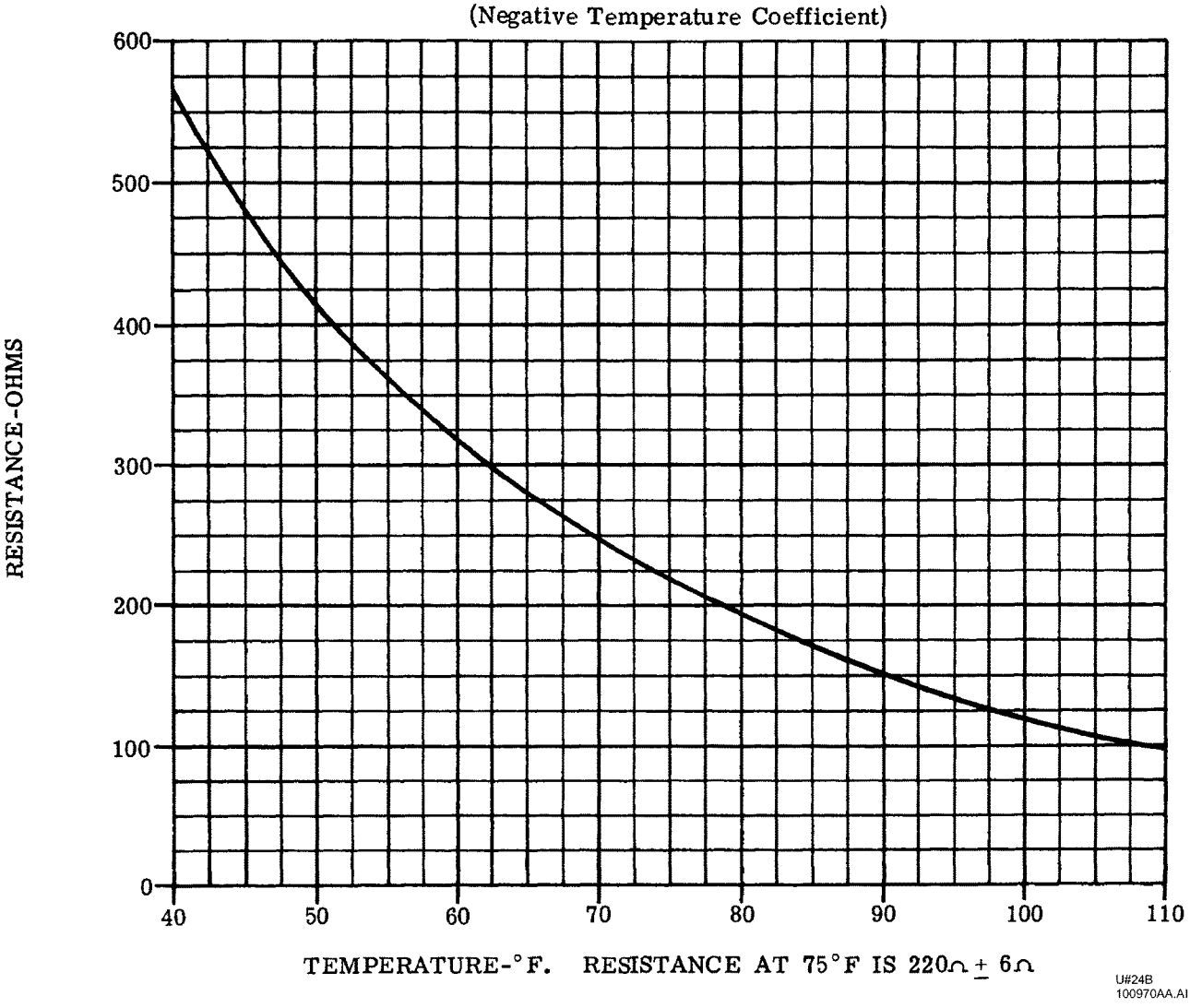


Figure 206
Temperature - Resistance Characteristic Curve for Cabin Air Sensing Element

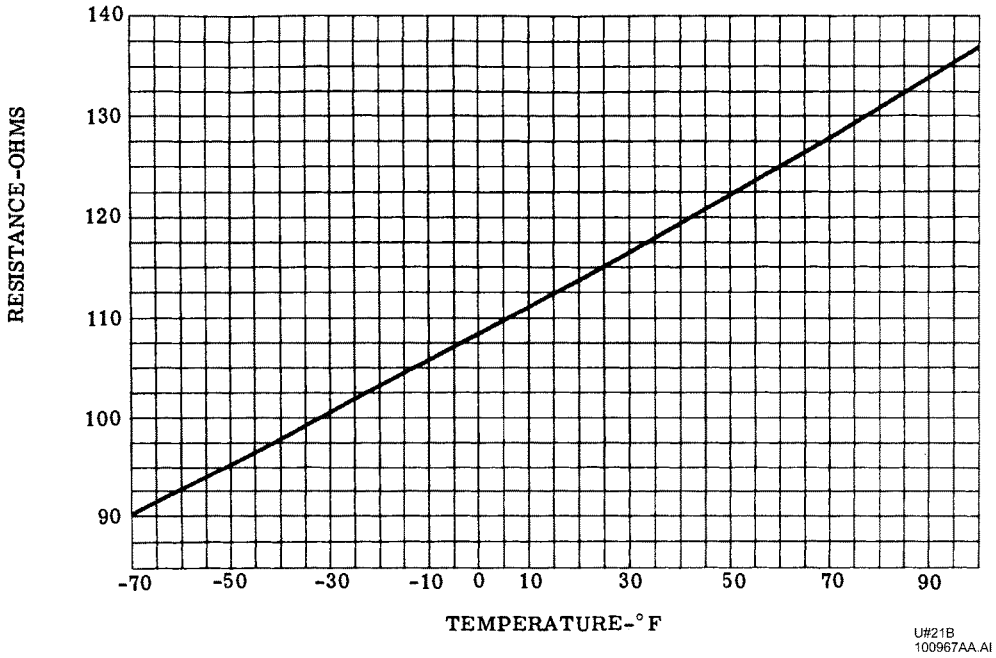


Figure 207
Temperature - Resistance Characteristic Curve for Outside Air Sensing Element

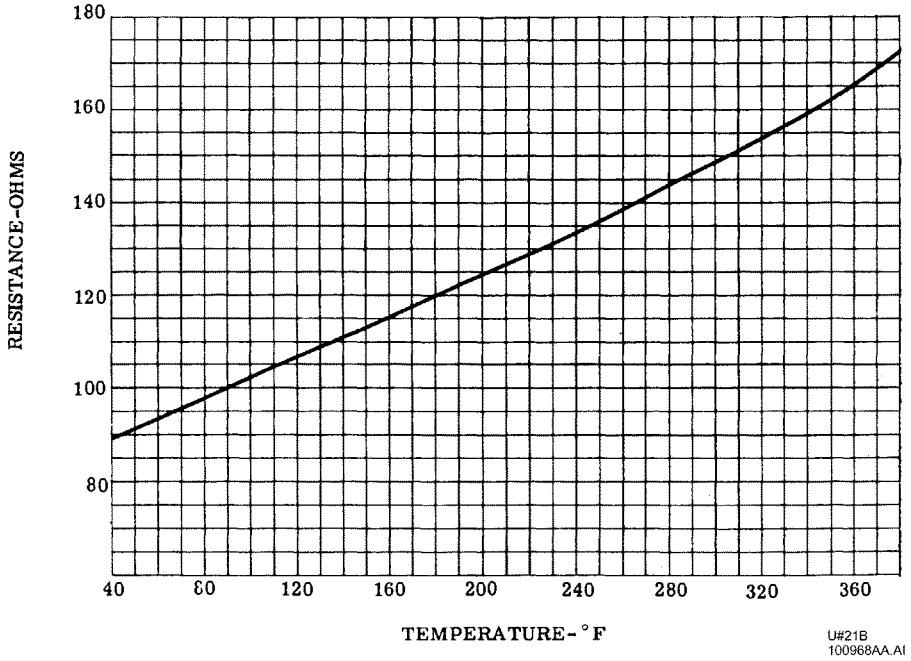


Figure 208
Temperature - Resistance Characteristic Curve for Heater Discharge Air Sensing Element

D. Igniter and Ignition Lead

When the igniter and ignition lead are tested, the following precautions must be taken:

(1) When testing the ignition lead, clamp a metal strip to the lead and spark plug in such a manner that the gap between the metal strip and the end of the spark plug electrode is 1/2 inch. This will provide a ground return to prevent the high voltages developed during the test from arcing.

(2) Never fire the igniter without the ignition lead attached, for the resultant voltage will arc across the igniter insulator and ruin the unit unless the lead is attached.

11. ELECTRIC HEATER (U-50, U-165 AND AFTER)

A. Removal

(1) Remove the aft flooring in the nose baggage compartment to gain access to the heater.

(2) Disconnect the heater electrical wiring from the heater and tag or otherwise identify the wiring to facilitate reinstallation.

(3) Disconnect the clamps at the end of the heater.

(4) Remove the heater from the airplane.

B. Installation

(1) Replace both the front and rear clamp seals anytime the heater is removed.

(2) Position the heater into its compartment.

(3) Install the heater clamps and tighten them.

(4) Connect the electrical wiring.

(5) Install the aft flooring in the nose baggage compartment.

12. HEATER ELEMENTS

The heater has eight electrical heating elements made into two sets with four elements in each set. The four elements in each set are connected in parallel. One set provides heat for NORMAL HEAT operation and both sets combine for MAX HEAT operation. The heating elements can be replaced individually.

A. Removal

The individual elements are placed by removing and dismantling the heater as follows:

(1) Remove the heater as instructed under the heading ELECTRIC HEATER REMOVAL.

NOTE: Each set of elements is removed from its respective end of the heater. One or both sets may be removed as required.

(2) Remove the aluminum pressure sensitive foil tape and the industrial oven insulation covering the heater.

(3) Remove the electrical terminals.

(4) Remove the screws attaching the inner and outer shells.

(5) Pull the inner element free from its outer shell.

(6) Disconnect the individual element from the electrical connection and from the screws attaching it to the inner shell.

B. Installation

(1) Place the heating element into the inner shell and attach it to the electrical connector and the attaching screws.

(2) Insert the inner shell into the outer shell.

(3) Install the attaching screws and tighten them.

(4) Replace the heater industrial oven insulation and apply aluminum pressure sensitive tape, leaving half of the tape width exposed on each lap.

(5) Install the heater into the airplane as instructed under the heading ELECTRIC HEATER INSTAUATION.

13. DEFROST AIR CONTROL

When the defrost air control is pulled open, a link cable, between the defrost duct butterfly and the heat duct butterfly (Ref. 21-40-00, Figure 3). simultaneously closes off the heat duct, thereby diverting air through the windshield defrost duct work. Due to the elliptical design of the heat duct butterfly, only 600 of travel will fully close off the heat duct. The defrost duct butterfly pivot arm has two control cable wire clamps; the clamp farthest from the butterfly valve shaft is for clamping the cable wire from the control handle, the clamp nearest to the butterfly valve shaft is for clamping the link cable wire from the heat duct butterfly.

A. Rigging

Rig the butterfly valves according to the following procedure:

(1) Locate the defrost duct butterfly valve.

(2) Loosen both cable clamps on the pivot arm so that the cable wires run freely through the clamps.

(3) Push the defrost air control handle fully in and hold in that position.

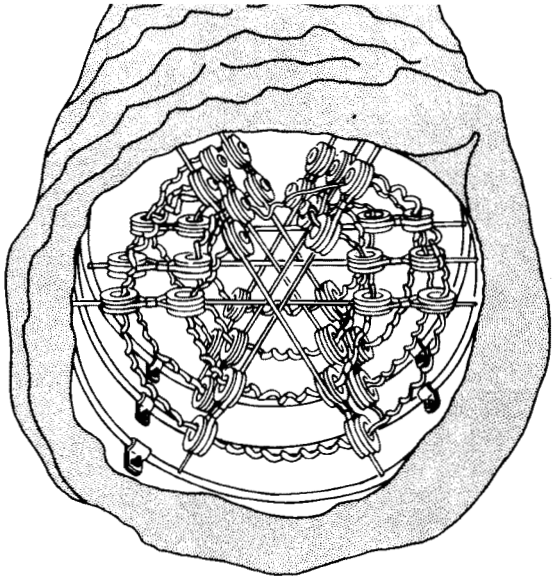
(4) Fully close the defrost duct butterfly valve; the valve pivot arm will be perpendicular to the duct in the fully closed position.

(5) Tighten the cable wire clamp farthest from the butterfly valve pivot shaft.

(6) Locate the heat duct butterfly valve.

(7) While maintaining the position of the defrost duct butterfly, set the heat duct butterfly fully open; the butterfly pivot arm should be parallel to the duct when this valve is fully open.

(8) Tighten the cable wire clamp on the defrost duct pivot arm nearest the butterfly valve pivot shaft.



F90-412-5

Figure 209
Heater and Elements

AIR CONDITIONING
BLEED AIR HEAT
DESCRIPTION AND OPERATION

1. GENERAL

A. U-50, U-165 and After

Bleed air from the P_3 stage of both engines is supplied to the cabin through two shutoff valve/regulators installed in the engine nacelle area on the aft side of the firewall. The valve/regulators are controlled through two switches on the right inboard subpanel in the cockpit.

The RH and LH bleed air lines join at a check valve/tee junction mounted under the center cabin floor. The check valves in the tee junction prevent bleed air from one engine from entering the other engine in the event of failure of either of the bleed air sources.

A bleed air venturi is installed downstream of the check valve/tee junction and provides high and low pressure outlets to measure bleed air flow during ground maintenance. The high and low pressure outlets are connected by tubing to a test port fixture, mounted under the center cabin floor forward of the main spar.

A bleed air modulating valve meters the flow of bleed air into the cabin ducting system at a rate necessary to modulate the temperature of the ambient air entering the distributing duct work. The modulating valve is primarily a motor-actuated butterfly valve which responds to a continually varying degree of demand. The commands for the valve to open or close are transmitted by the cabin temperature controller.

Troubleshooting information for the modulating valve is found in Chapter 24-60-00.

B. Auxiliary Bleed Air Heat installed with Kit 99-9003 Prior to U-165, Except U-50

Kit 99-9003 provides the parts and rework instructions required to install an auxiliary heating system utilizing hot engine bleed air tapped from the compressor stage of the left engine. Bleed air heat is used as a backup system rather than as a supplement to combustion heater operation. The kit consists of two separate systems; bleed air heat and bleed air failure warning line.

The bleed air heat system consists of a cabin temperature selector switch with an added BLEED AIR HEAT mode, an in-line 3-amp fuse adjacent to the switch, an air heat switching relay, bleed air line adapter and attaching hardware (adapters are different, depending on application to -20 or -27 and -28 and -36 engines), bleed air shutoff solenoid valve, manual bleed air firewall shutoff valve and control cable, and an in-line check valve. Auxiliary bleed air heat installations which have a bleed air failure warning system are provided with an annunciator warning light located in the annunciator panel.

When the cabin temperature selector switch is placed in the BLEED AIR HEAT mode, the air/heat switching relay, mounted on the forward bulkhead adjacent to the combustion air heaters, is energized. The relay removes power from the combustion heater overtemperature and cycling switches to supply power for the BLEED AIR HEAT ON light in the annunciator panel. Power through the relay simultaneously opens the bleed air shutoff solenoid valve located forward of the left engine firewall and allows engine bleed air to enter the duct. A manually operated firewall shutoff valve located directly aft of the firewall may be adjusted by a control knob under the copilot's subpanel to select the desired amount of heat delivered to the cabin. Pulling the knob out increases the temperature and pushing the knob all the way in will shut off the bleed air at the firewall. The bleed air line is routed through the nacelle, along the leading edge, into the cabin and forward under the left hand seat track floorboards and into the combustion air heater plenum. An in-line check valve, located under the pilot's cockpit floorboards, prevents the loss of combustion heater air through the bleed air duct.

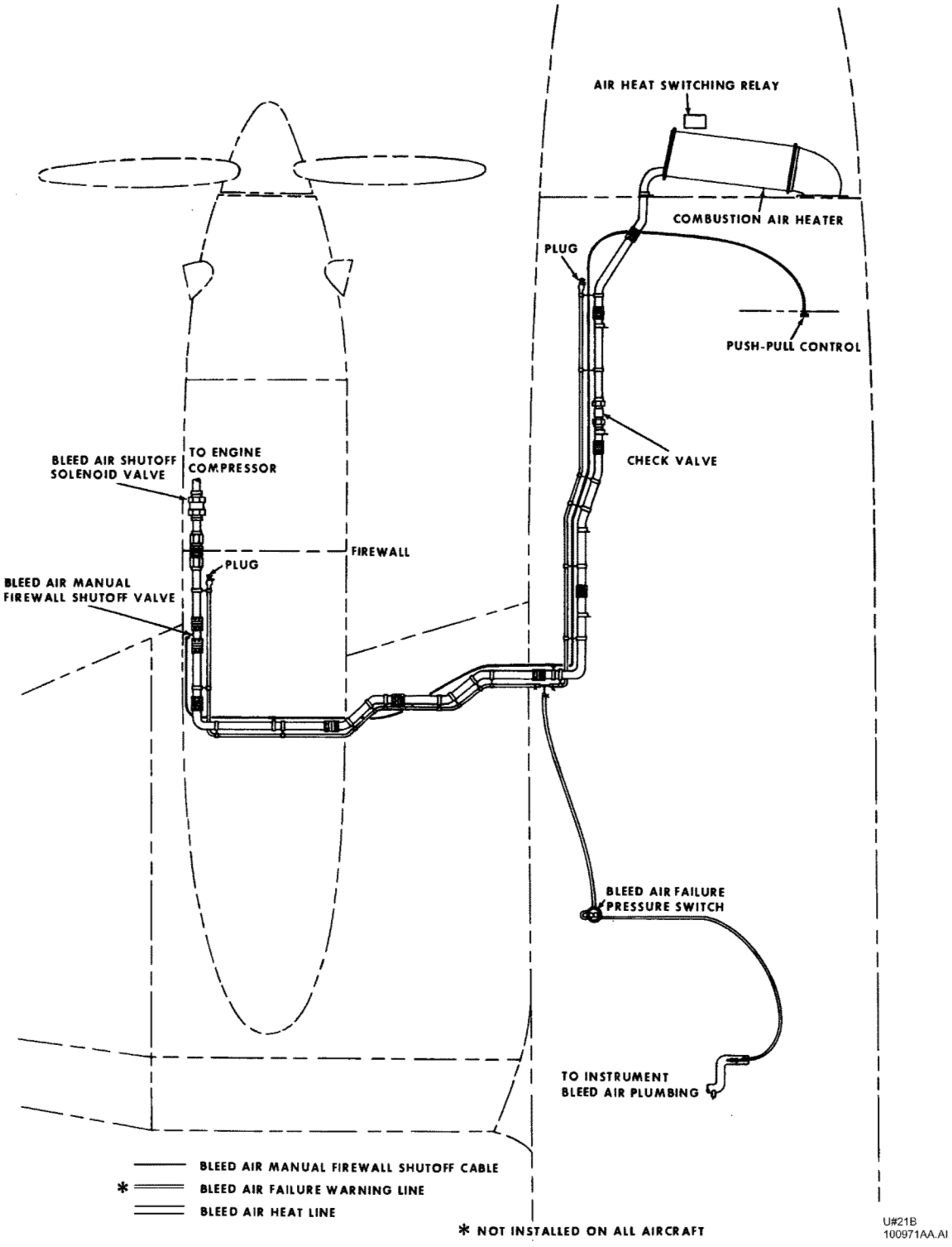


Figure 1
Auxiliary Bleed Air Heat (Prior to U-165, Except U-50)

AIR CONDITIONING
BLEED AIR HEATING
TROUBLESHOOTING

1. BLEED AIR FLOW

Single engine environmental bleed air flow for the C99 Airliner Series Airplanes should be between 5 pounds per minute and 7 pounds per minute, corrected to standard atmospheric pressure and 70°F. Two procedures for making the flow determination are discussed in the following text.

Two manometers are used to determine bleed air flow (Ref. Figure 101). One manometer must be mercury filled and will measure the bleed air pressure at the inlet to the bleed air venturi. The second manometer must be water filled and will measure the differential bleed air pressure between the inlet of the venturi and the throat of the venturi. A temperature measuring device capable of measuring in a range between 200°F and 600°F is used to determine the temperature of the bleed air at the inlet of the venturi.

The bleed air temperature and pressure measurements are applied to the following formula in order to correct for the nonstandard conditions:

$$\frac{T_1}{P_1} \times \frac{P_2}{T_2} = Sigma$$

Where:

T_1 is 530° Rankin (460 + 70°F).

P_1 is 29.92 inch Hg. (Standard Sea Level Pressure).

P_2 is absolute bleed air pressure (Standard Field Elevation Pressure (Ref. Figure 102) plus Measured Bleed Air Pressure at the venturi inlet).

T_2 is absolute bleed air temperature (460 plus bleed air temperature measured at the venturi inlet).

Delta P is the differential pressure measured between the venturi inlet and the venturi throat.

Sigma Delta P is the differential pressure reading corrected to a standard set of conditions. Sigma Delta P is plotted against bleed air flow rate on the Venturi Calibration Curve (Figure 103).

The bleed air flow determination graph (Figure 104) provides a more simplified approach to determining bleed air flow. All of the mathematics have been incorporated into the graph. The reading on the mercury manometer is entered at "A". A line from "A" on the graph is followed to the right until an intercept point is met on the altitude graph which most closely matches the field elevation of the service facility at "B". From "B", a line is followed up the page to point "C" which should correspond to the temperature measured at the venturi inlet. A line from this point is followed to the right to point "D" (Differential pressure measured between 'he venturi ports) and finally down the page to point "E", which represents the bleed air flow rate of the engine.

Pressure measurements are taken at the bleed air test port fixture located in the center aisle subfloor. Temperature measurements must be taken at the venturi located in the left cabin subfloor forward of the test port fixture (Ref. Chapter 21-40-00, Figure 3).

(1) Locate the bleed air venturi and peel back the insulation from the venturi inlet.

(2) Secure the probe of the temperature measuring device against the venturi inlet and cover with the insulation.

(3) Connect the manometers to the test ports as shown in Figure 101. Be sure the ON/OFF valve of the test setup is OFF.

(4) Place the cabin temperature mode switch in MAN HEAT and open the modulating valve fully by holding the manual temperature control switch in INCREASE for 60 seconds.

(5) Start one engine only and bring the generator on line. Engine speed should be adjusted to 65% N_1.

(6) Very slowly open the ON/OFF valve of the test setup to allow the high side bleed air to enter the manometers.

(7) Allow a few minutes for the bleed air temperature to peak and stabilize.

(8) Take a reading from the mercury manometer. Note the reading of mercury level in both sides of the manometer, the difference between the readings will become "A" (Ref. Figure 101).

(9) Read the water manometer in the same manner as the mercury manometer; this difference will become "C" (Ref. Figure 101).

(10) Record the temperature of the bleed air at the inlet to the venturi; this will become "B" (Ref. Figure 104).

(11) Repeat the procedure using the opposite engine and apply the readings obtained to the graph (Ref. Figure 104) to determine the bleed air flow rate in pounds per minute for each engine.

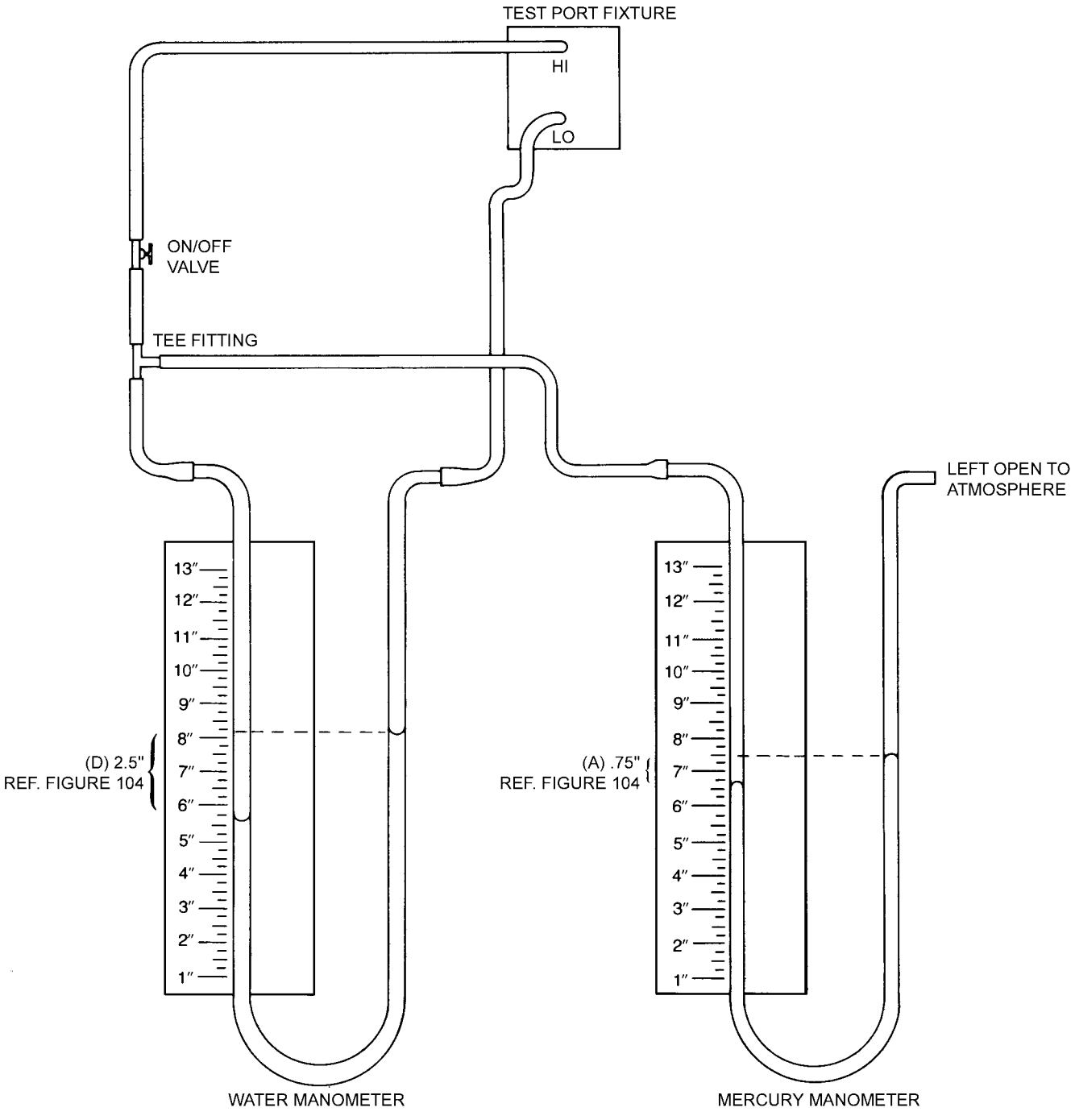


Figure 101
Bleed Air Flow Check

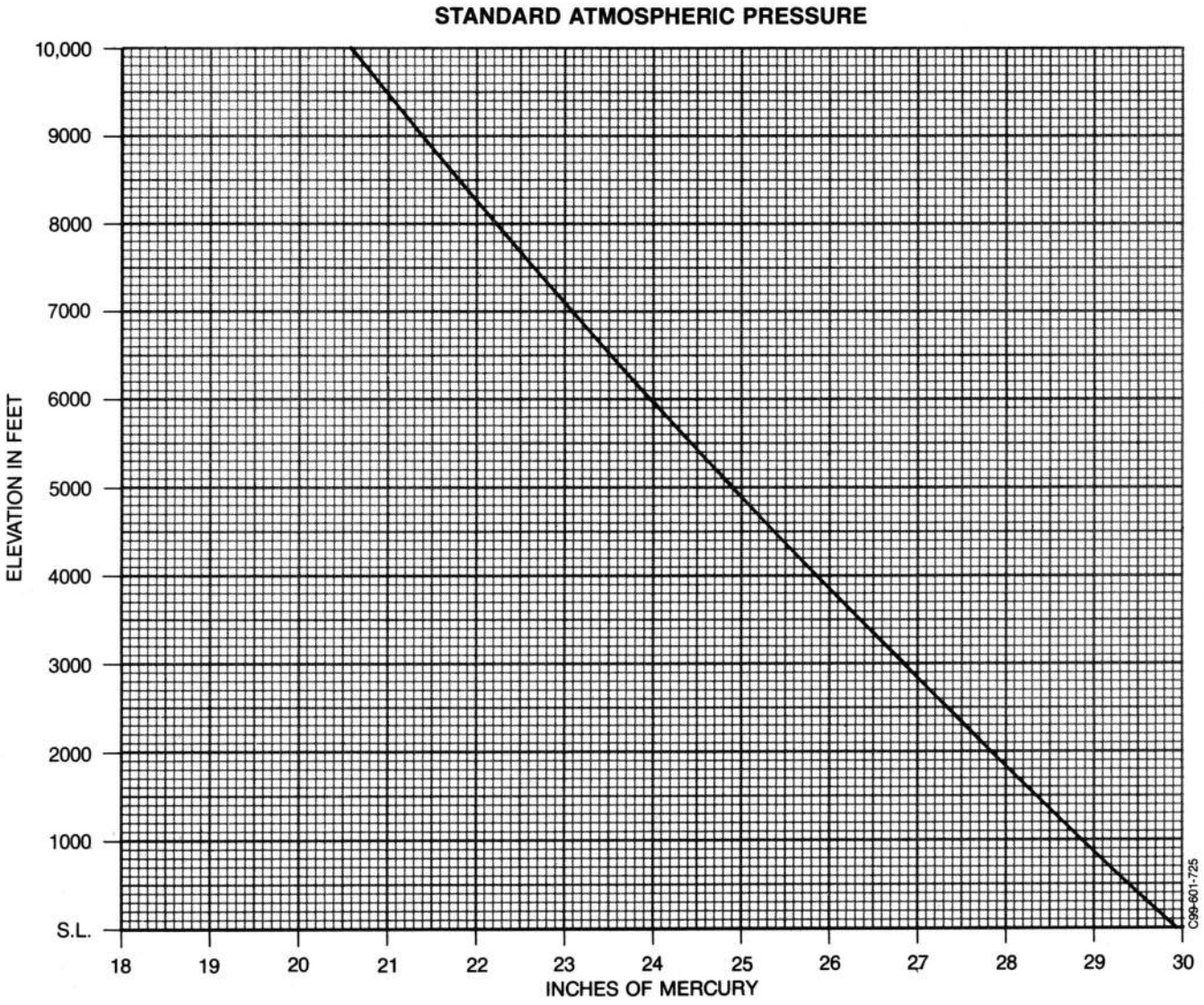


Figure 102
Atmospheric Pressure

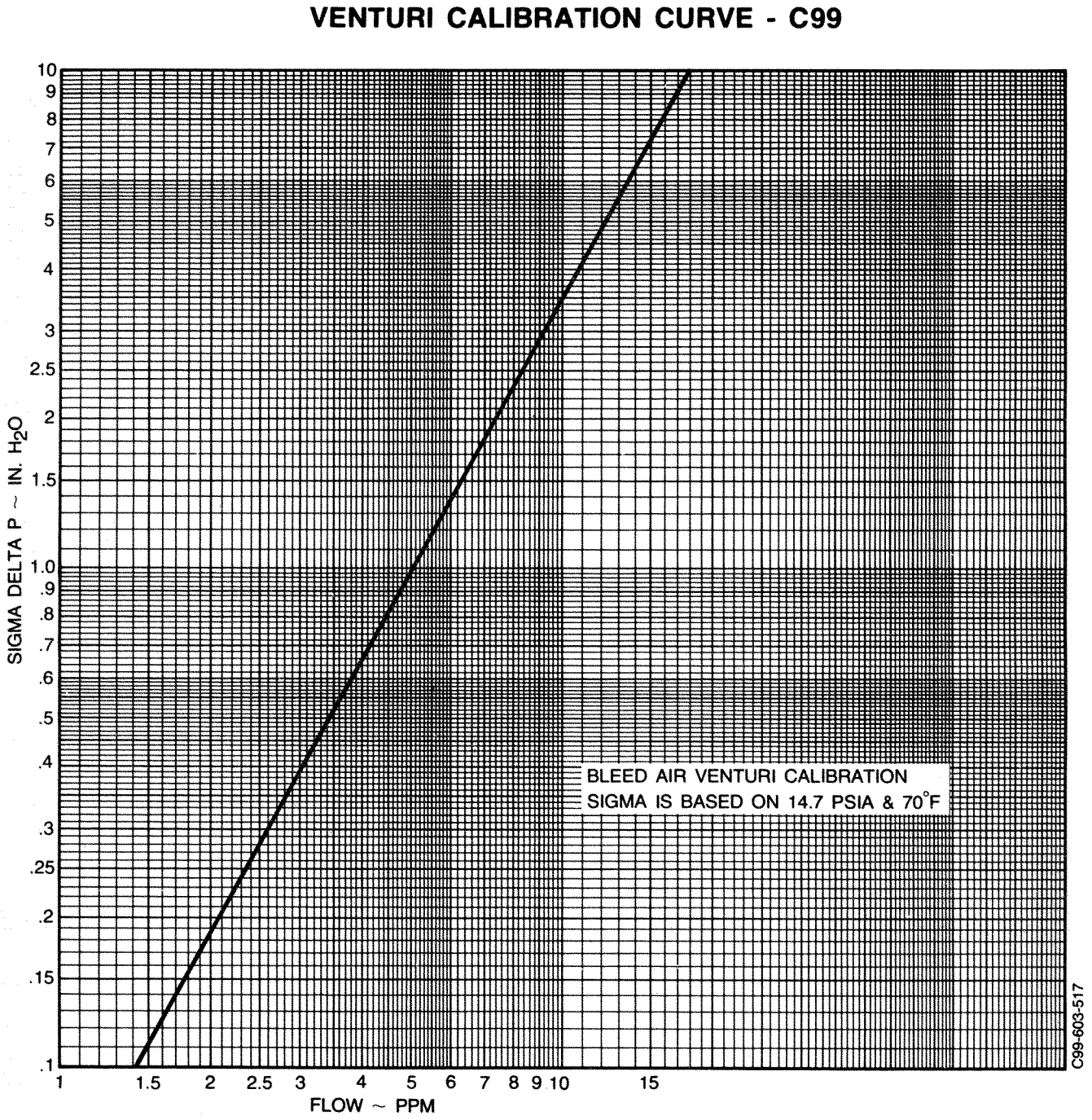


Figure 103
Bleed Air Flow (Pounds Per Minute)

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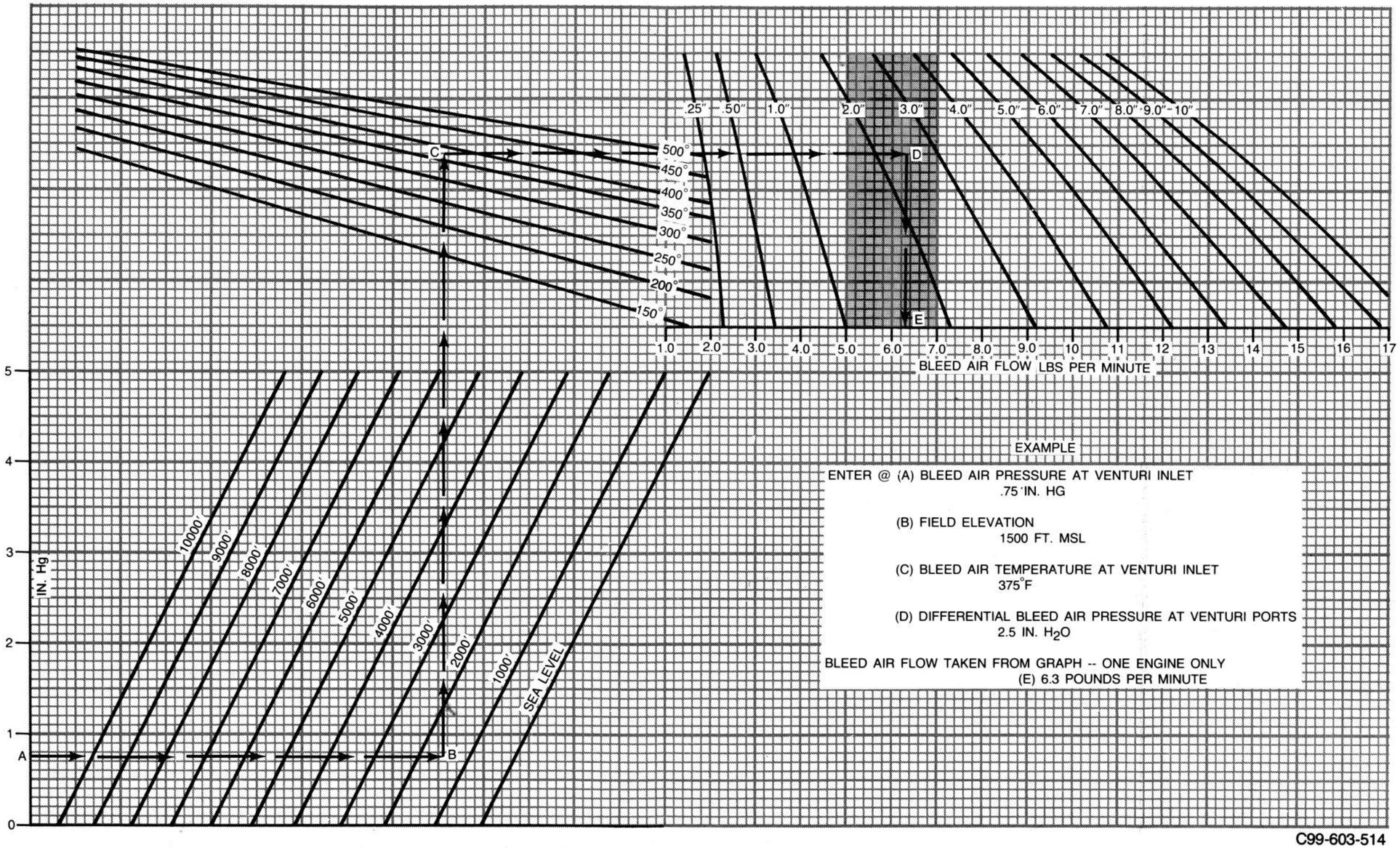
BLEED AIR FLOW DETERMINATION



C99-603-514

Figure 104
Bleed Air Flow Determination

AIR CONDITIONING
BLEED AIR HEATING
MAINTENANCE PRACTICES

1. BLEED AIR SHUTOFF SOLENOID VALVE

A. Removal

Earlier serials, prior to U-165 except U-50, use one bleed air shutoff valve installed on the left engine and located forward of the firewall on the outboard side of the engine nacelle. Later serials, U-50, U-165 and after, use a bleed air shutoff valve on each engine and are located just aft of the firewall on the inboard side of the engine nacelles. The valve may be removed as follows:

(1) Disconnect the electrical leads from the valve solenoid.

(2) Peel back the aluminum foil tape and insulation.

(3) Loosen the clamps and remove the valve.

B. Installation

(1) Position the valve and secure with the attaching clamps.

(2) Replace the 1-inch thick industrial oven insulation (69, Table 6, 91-00-00) leaving half the width of the tape exposed at each lap.

2. BLEED AIR SHUTOFF MANUAL VALVE (PRIOR TO U-165, EXCEPT U-50)

A. Removal

To gain access to the manual firewall shutoff valve, remove the access panel on the outboard side of the left nacelle, immediately forward of the wing leading edge. Remove the valve as follows:

(1) Remove the bolt securing the control cable clevis to the valve actuating arm.

NOTE: To retain proper operation of the control cable, do not disturb the adjustment at the clevis end.

(2) Peel back the foil tape and insulation from the valve and remove the valve from the bleed air line fittings.

B. Installation

Upon installation of the valve, always replace the silicone packings whenever the bleed air duct couplings are separated. Install the couplings finger-tight and safety wire. Install the valve as follows:

(1) Replace the 1-inch thick industrial oven insulation (69, Table 6, 91-00-00) leaving half the width of the tape exposed at each lap.

(2) Position the valve in the bleed air line fittings with the solenoid up and install the bolt securing the control cable clevis to the valve actuating arm.

3. BLEED AIR MODULATING VALVE (U-50, U-165 AND AFTER)

A. Removal

The bleed air modulating valve is installed under an access panel in the left hand cabin subfloor under the first passenger seat. Modulating valve removal is as follows:

(1) Remove the electrical connector plug from the valve.

(2) Remove both retaining clamps from the valve inlet and outlet flanges.

(3) Remove the valve through the access opening.

B. Installation

(1) Position the valve between the two bleed air line flanges.

(2) Install the two retaining clamps on the inlet and outlet flanges.

(3) Connect the electrical connector plug to the modulating valve.

NOTE: The flow direction arrow on the valve body should point forward, indicating proper flow direction through the valve.

4. BLEED AIR HEAT LINE

A. Repair

Peel back the foil tape and insulation from the damaged section of tubing. Cut the safety wire and disconnect the Wiggins coupling. Install the new section of bleed air tubing.

NOTE: Whenever the couplings are disconnected the silicone packings must be replaced.

Wrap the entire section of bleed air tubing with 1-inch thick industrial oven insulation and aluminum foil tape (69 and 67, Table 6, 91-00-00) leaving half the width of the tape exposed on each lap.

AIR CONDITIONING
COOLING
DESCRIPTION AND OPERATION

1. PRIOR TO U-165, EXCEPT U-50

The optional air conditioner is a vapor-cycle system containing two 12,000 BTU, refrigerative type evaporators. The refrigerant compressor is belt-driven off the right engine accessory case. One of the evaporators is installed in the nose under the baggage compartment floor and the other under the right seat track at fuselage station 266.50 (even with the aft side of the fifth cabin window). A valve controlled by two solenoids is located immediately aft of the aft evaporator. During the heating mode, one of the solenoids closes the valve to prevent evaporator air from entering the heater distribution ducts. During the cooling mode, the other solenoid opens the valve to facilitate air conditioning by enabling the flow of evaporator air to rapidly cool down the distribution ducts. Each evaporator has a two-speed ventilation blower controlled through a VENT BLOWER switch on the copilot's subpanel next to the air conditioning mode selector switch. A 33°F thermal switch at each evaporator actuates a refrigerative bypass valve to prevent the evaporators from icing up.

When the airplane is on the ground, a vane-axial blower (operating through the LH landing gear up-lock switch), draws air through a grill in the aft end of the nose wheel well to the condenser and exhausts air through louvers on the lower right hand side of the nose. During flight the condenser blower is inoperative and rain air is directed to the condenser from the air scoop along the left side of the nose. The air conditioning system operates either automatically or manually as selected by the mode switch. When the AUTO position is selected, the heating and air conditioning systems are automatically controlled through a balanced wheatstone bridge by the control box as described in the heater portion of this Chapter. When the cabin has cooled down to the selected temperature setting, the automatic temperature control opens the bypass solenoids that route the refrigerant around the evaporators. A 4°F dead zone between the healing and cooling modes prevents cycling between the two. When the selector switch is placed in the MANUAL COOL position, the automatic temperature control system is bypassed. Also incorporated within the system, is a time delay relay that prevents the air conditioner from starting when there is a high head pressure.

To protect the compressor from excessive line pressure, an overload switch (located under the right nose baggage compartment floor) will release the electric compressor clutch whenever the pressure exceeds 400 psi.

WARNING: The overload switch cannot be reset in flight. The switch must be reset manually, but only after a complete system checkout to determine the cause.

A thermal switch in the temperature control box prevents the air conditioning system from operating at temperatures below 50°F to protect the compressor and evaporators against icing.

2. U-50, U-165 AND AFTER

The optional refrigerative cabin air cooling system in the C99 is essentially the same installation as the earlier airplanes, but with some refinements in the control system. Cabin cooling is controlled through the cabin temperature mode switch and the cabin temperature controller. The cool command signal, generated by the cabin temperature, causes the bleed air modulating valve to close. When the butterfly valve in the modulating valve is less than 6° of rotation from being fully closed, the compressor clutch is engaged and the cooling cycle is initiated. If the outside air temperature is less than 50°F, the compressor clutch will be prevented from engaging by the low temperature air conditioner shutoff switch. The compressor clutch will not engage also if a Low N_1 engine speed condition exists. As the cabin temperature lowers, the modulating valve will begin to receive commands from the cabin temperature controller to open in varying increments, thereby modulating the cool air from the evaporator with bleed air heat. When the cabin air temperature is cool enough to allow

the modulating valve to rotate to 22° from the closed position, power will be interrupted to the compressor clutch and the cooling cycle will be stopped. The compressor clutch will not engage again until the modulating valve closes to within 6° of the closed position.

A. Aft Evaporator Blower

If maximum cooling effect is desired, the aft blower must be turned on. The aft evaporator blower is controlled by a relay energized through it's own switch on the right inboard subpanel. Operation of the aft evaporator blower is in no way controlled by any other component of the environmental control system.

B. Pressure Cutout Switches

Should an event occur which would cause the discharge pressure from the compressor to increase beyond a safe limit, the high pressure cutout switch will interrupt power to the compressor clutch, cycling the compressor out of operation. Conversely, should an event occur which would cause the suction pressure to drop below a safe limit, a low pressure cutout switch will open the circuit to the clutch, cycling the compressor out of operation. Once the compressor has been cycled out by either pressure switch, the latching relays in the clutch control module must be reset. The reset switch is located in the forward baggage compartment right hand subfloor. When illuminated, a lockout light on the reset switch assembly indicates that a pressure switch has cycled the compressor out. A pair of lights on a printed circuit board indicates which condition cycled the compressor out, high pressure or low pressure.

C. Hot Gas Bypass

A temperature sensing switch on the forward evaporator closes on decreasing temperature of the evaporator at 33°F and energizes a hot gas bypass solenoid valve which shunts hot gas from the compressor discharge, around the condenser and thermostatic expansion valve, directly into the forward evaporator for warming it. The switch opens at 45°F on rising temperature of the evaporator to de-energize the hot gas bypass solenoid and allowing the refrigerant flow to return to normal flow. This prevents the forward evaporator condensate from freezing and accumulating on the evaporator coils.

D. Condenser Blower

The condenser blower motor is energized through a relay controlled by the landing gear down lock switch in the grounding circuit of the relay coil. The down lock switch is actuated by the down lock portion of the gear extension system when the landing gear is lowered. When the gear is retracted, the grounding circuit to the blower motor relay is opened. A cool command from the compressor clutch control module supplies the necessary switching information to the condenser blower relay.

E. Condenser Blower Actuator

The condenser cooling actuator, located in the ram air duct to the condenser, opens a door which allows air to flow to the condenser coils and cooling them. The actuating motor is controlled by a relay which, when energized by the cool command, reverses the polarity of the current and drives the door open. When the cool command is stopped, the relay de-energizes and reverses the polarity to the motor, again driving the door closed.

F. Pressure Relief Valve

A pressure relief valve located on the high pressure line from the compressor to the condenser just before the line passes through the bulkhead at FS 70.0 is installed for the purpose of protecting the system should the system be inadvertently overcharged. The relief valve is set to open at 450 ± 10 psig.

3. SERVICING THE COOLING SYSTEM

The system capacity for Freon with an evacuated system is 120 ounces.

For servicing procedures refer to SERVICING THE AIR CONDITIONING SYSTEM (Ref. Chapter 12-10-00).

AIR CONDITIONING
COOLING
TROUBLESHOOTING

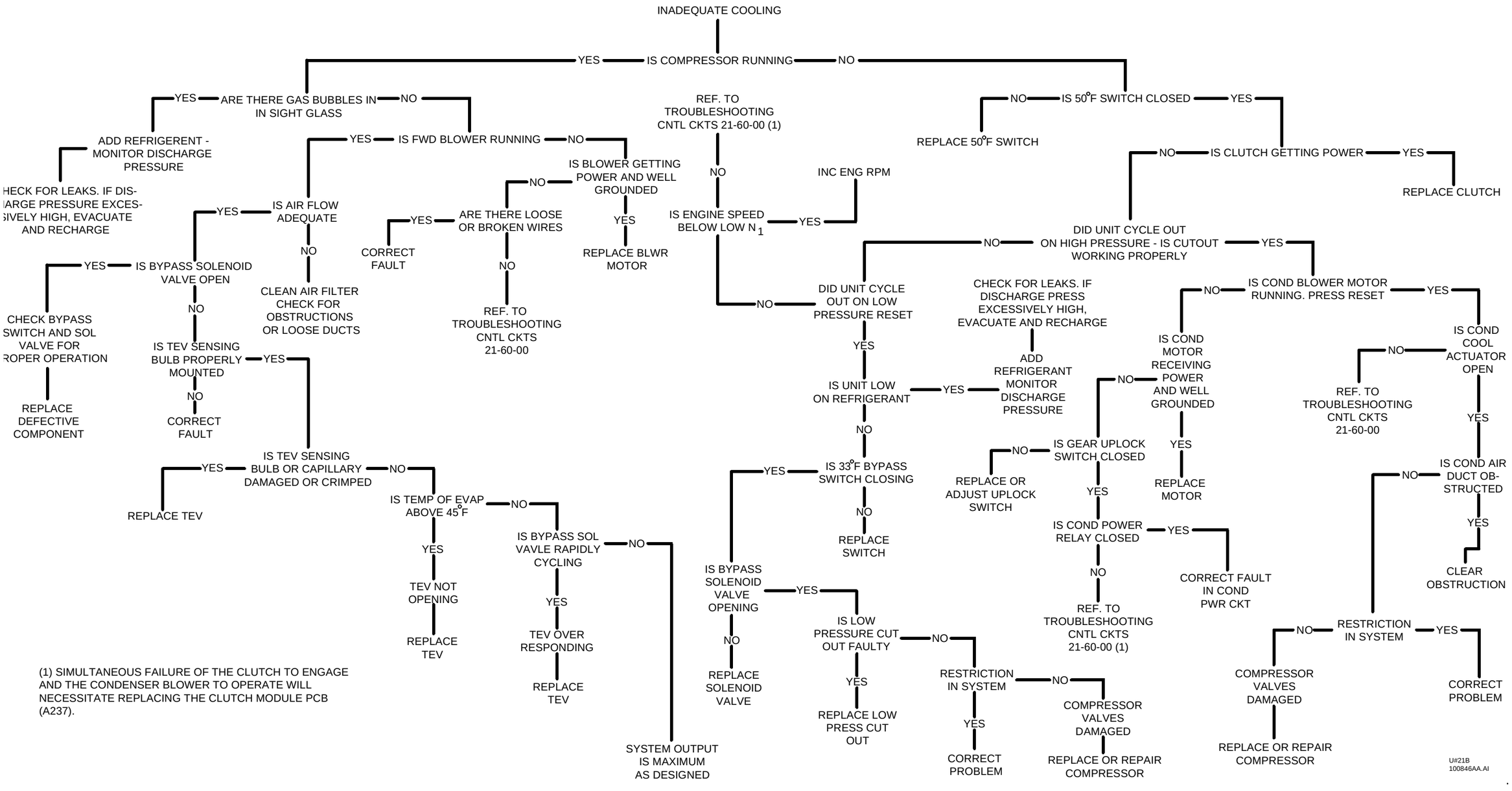
1. AIR CONDITIONER

Troubleshooting - Air Conditioner (Prior to U-165, Except U-50)

INDICATION	POSSIBLE CAUSE	REMARKS
1. Belt slipping.	a. Loose.	a. Adjust.
	b. Overcharge.	b. Correct charge.
	c. Air in system.	c. Evacuate and recharge.
2. Excessive belt wear.	a. Pulleys not in line.	a. Align pulleys.
	b. Belt too tight.	b. Adjust or replace.
	c. Pulley groove wrong size.	c. Replace.
	d. Belt width wrong.	d. Replace.
3. Broken belt.	a. Check all causes above.	a. Replace.
4 Excessive vibration of unit.	a. Overcharge.	a. Correct charge.
	b. Air in system.	b. Evacuate and recharge.
	c. Mount or compressor bolts loose.	c. Tighten.
	d. Drive pulley loose.	d. Tighten.
5. Noisy unit.	a. Defective compressor.	a. Replace.
	b. Defective belts.	b. Replace.
	c. Bad bearing in compressor.	c. Replace compressor.
6. Hissing in evaporator case.	a. Low charge.	a. Correct charge.
7. Chatter or knock in evaporator.	a. Defective expansion valve.	a. Replace.
8. Inadequate cooling.	a. Refrigerant low.	a. Check for leaks and recharge.
	b. 35° Thermal switch malfunction.	b. Replace.
	c. Expansion valve malfunction.	c. Replace.
	d. Solenoid valve malfunction.	d. Replace.
	e. Evaporator stopped up.	e. Clean out obstruction.
9. Air conditioners will not operate in the AUTO mode but will function in the MANCOOL mode.	a. Misadjustment of heater control box.	a. Adjust.
	b. Malfunction of temperature sensing elements.	b. Replace.
	C. Malfunction of control box.	C. Replace.

Troubleshooting - Air Conditioner
(Prior to U-165, Except U-50) (Continued)

INDICATION	POSSIBLE CAUSE	REMARKS
10. A i r conditioner runs constantly in either the AUTO or MAN COOL modes.	a. Malfunction of temperature sensing elements.	a. Replace.
	b. Thermal switch will not allow air conditioning to cut in at 50° or below.	b. Replace.
11. Air conditioner inoperative in either AUTO or MANUAL.	a. Circuit breaker open.	a. Reset.
	b. Over pressure switch open.	b. Reset.
	c. Both 35° thermal switches open.	c. Replace.
	d. Refrigerant power relay defective.	d. Replace or clear contacts.
	e. Compressor defective.	e. Replace.
	f. Both by-pass valves stuck open.	f. Replace.
	g. Dirty filters in pressure side of expansion valve.	g. Remove and clean with dry air.
	h. Receiver dryer dirty.	h. Remove and clean with dry argon or nitrogen.
12. Air conditioner starts immediately after heater cycles off (Auto model).	a. Defective time delay relay if it actuates in less than 5 seconds.	a. Replace.
13. Air conditioner runs constantly in AUTO.	a. Shorted contacts on the time delay relay.	a. Replace.
	b. Defective temperature sensor.	b. Replace.
	c. Defective temperature control box.	c. Correct.



(1) SIMULTANEOUS FAILURE OF THE CLUTCH TO ENGAGE AND THE CONDENSER BLOWER TO OPERATE WILL NECESSITATE REPLACING THE CLUTCH MODULE PCB (A237).

Figure 101
Troubleshooting - Cabin Air Cooling (U-165 and After)

AIR CONDITIONING
COOLING
MAINTENANCE PRACTICES

1. REFRIGERATIVE AIR COOLING SYSTEM

The air conditioning system is similar to many home and automotive units (Ref. Figures 201 and 202). The belt-driven compressor, which is coupled by a magnetic clutch, compresses the refrigerant to a high pressure, high temperature gas. This gas passes through the condenser where cooling air removes heat from the gas, condensing it to a liquid state. The liquid passes through the receiver-dryer where any moisture or foreign material is removed from the system. The refrigerant flows to the expansion valves where it is metered into each evaporator at a rate which allows all the liquid to return to a gas. The heat required for evaporation is absorbed from the cabin air passing over the evaporator coils. After passing through the evaporators, the refrigerant returns to the compressor at a reduced pressure.

2. PRECAUTIONARY SERVICE MEASURES

Before any service is attempted which requires opening of refrigeration pipes or units, the person doing the work should be thoroughly familiar with the instructions on servicing the system. He should follow very carefully these instructions when performing the tasks that will maintain this system in a proper functioning order.

The major reasons for these measures are for safety and to prevent dirt and moisture from entering into the system. Dirt contaminant may cause leaky valves or wear in the compressor. Moisture may not only freeze into ice at expansion valves, but can also cause the formation of hydrochloric or hydroflouric acids in the system.

All subassemblies are shipped, sealed and dehydrated. They are to remain sealed until just prior to making connections. They should be at room temperature before uncapping. (This prevents condensation of moisture from the air that enters the system).

If for any reason, caps are removed but the connections are not made, then the tubes and other parts should not remain unsealed for more than 15 minutes. Reseal connections if period is to be longer.

Compressors are shipped with 10 to 11 ounces at 525 viscosity oil and charged with a mixture of Refrigerant-12 and dry nitrogen to provide an internal pressure at slightly above atmospheric pressure.

All precautions should be taken to prevent damage to fittings or connections. Even minute damage to a connection could cause it to leak. Any fittings getting grease or dirt on them should be wiped clean with a cloth dampened with alcohol. Do not use chlorinated solvents such as trichloroethylene for cleaning agents, for they are contaminants. If dirt, grease or moisture gets inside pipes and cannot be removed, the pipe will have to be replaced. Use a small amount of clean refrigeration oil (525 viscosity) on all tube joints and dip the packing in this oil before assembling the joint. This will help in making a leak-proof joint.

WARNING: Due to air quality-control regulations being enacted in the United States, R-12 refrigerant cannot be vented into the atmosphere. When performing maintenance on the vapor cycle system where refrigerant R-12 can escape from the system, evacuate the system with a recovery or recycle servicing unit that will salvage the refrigerant.

A face shield should be worn when servicing the lines; refrigerant coming in contact with the eyes can cause the loss of sight. Also wear gloves and avoid breathing any vapors.

Do not smoke when servicing the system with R-12, for refrigerant converts to a highly toxic gas when exposed to an open flame.

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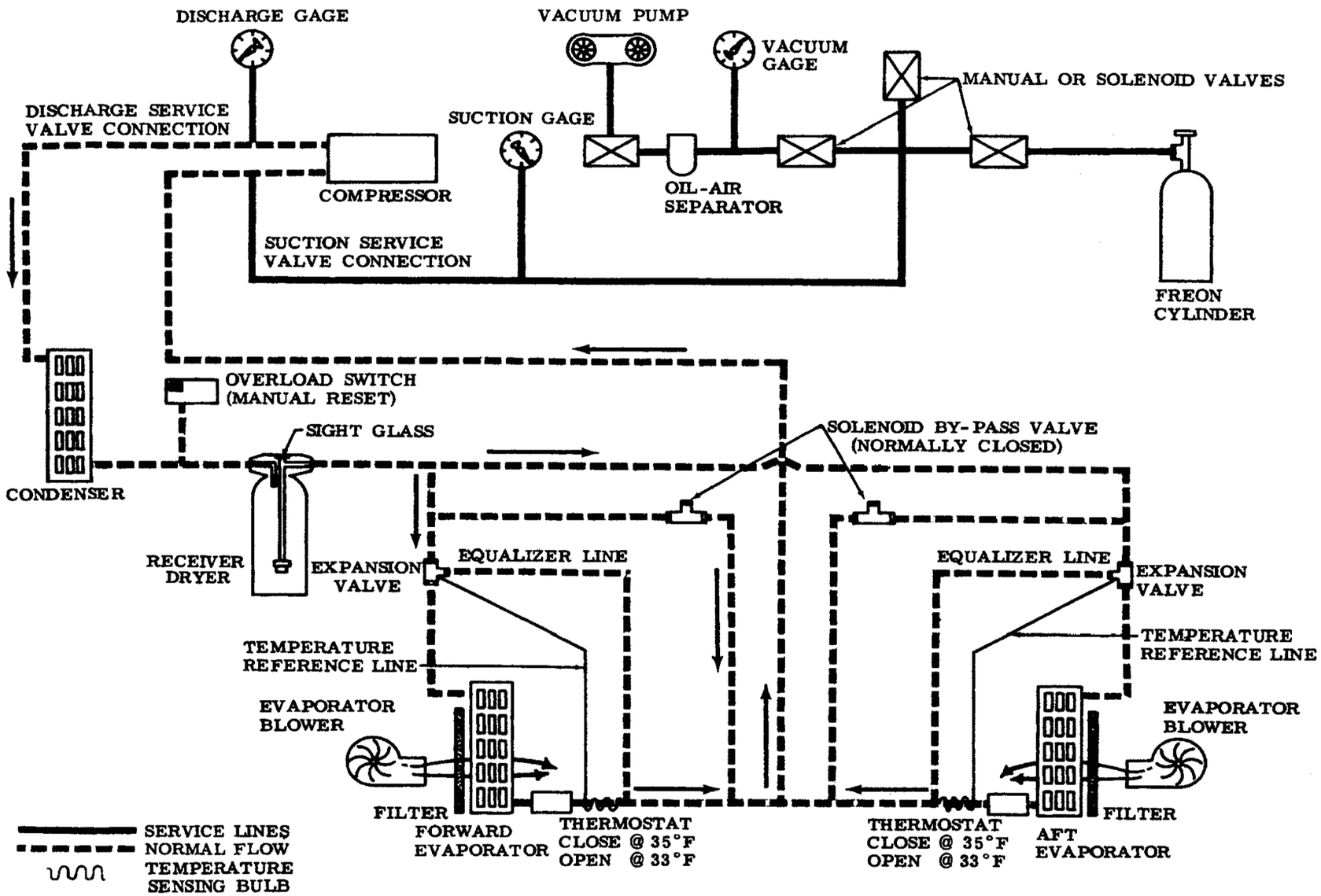

Figure 201
Cooling System Schematic (Prior to U-165, except U-50)

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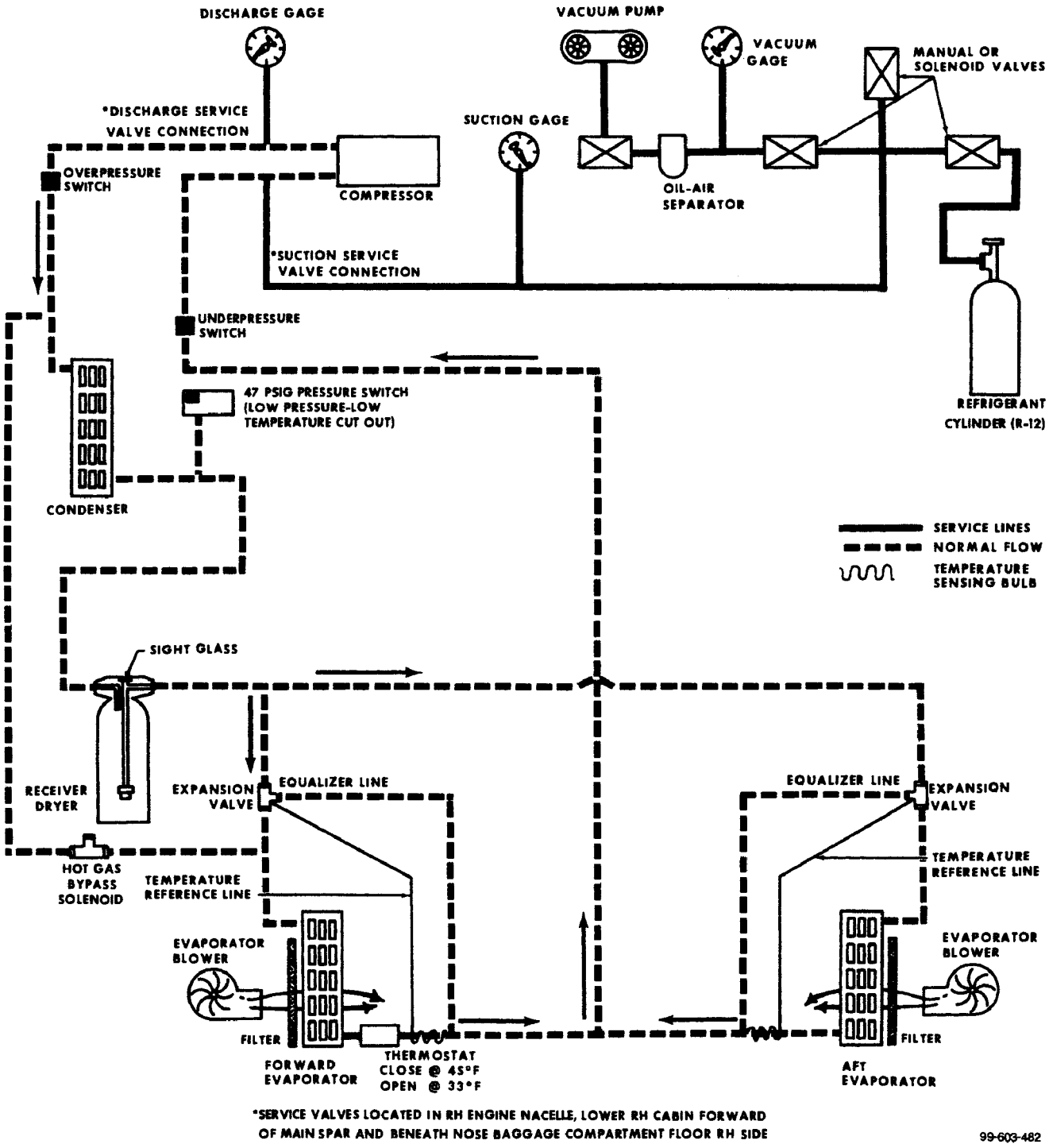

Figure 202
Cooling System Schematic (U-50, U-165 and After)

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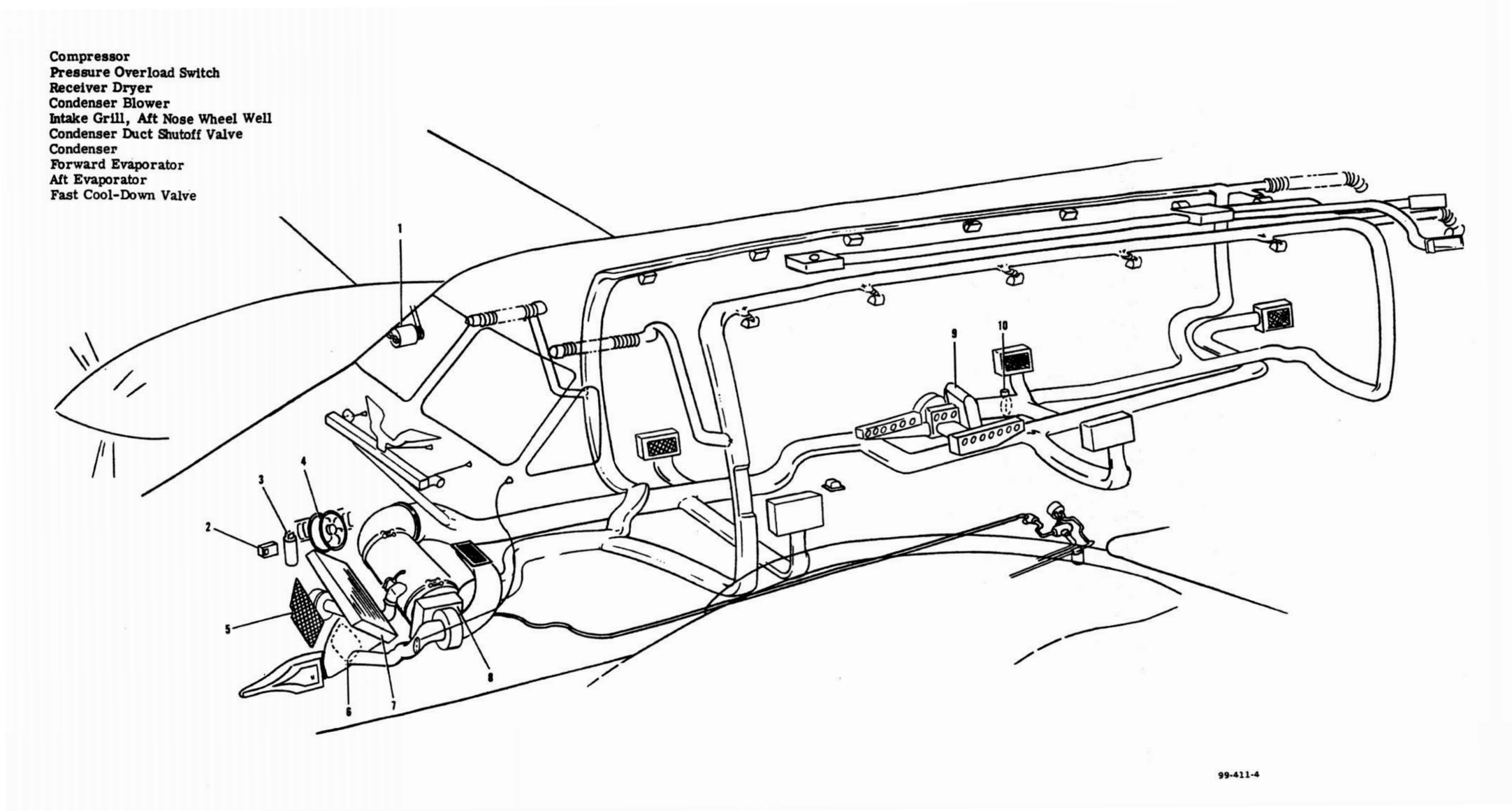

Figure 203
Air Conditioning Installation

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3. SYSTEM LEAK DETECTION

NOTE: It is permissible to add 8 ounces of Trace, a red dye, to the suction side of the refrigerant system to detect leaks. A red film will appear in the areas of leakage.

A reduction of system cooling ability or the presence of bubbles in the refrigerant may indicate a partial loss of refrigerant. Open the right hand forward baggage compartment door and check for bubbles in the sight glass under the plexiglass cover in the floor of the baggage compartment at FS 57. The sight glass should be checked during operation at maximum available ambient and cabin temperatures. Streams of bubbles past the glass or foam in the glass indicates an inadequate refrigerant quantity. If a loss of refrigerant is suspected; an inspection of the system plumbing should be carried out to locate the source of the leak. Large leaks may be located by the appearance of oily spots where oil has been carried out by escaping refrigerant. Smaller leaks, which are much more difficult to locate, may be detected by detergent bubbles, or an electronic detector. In order to use the above detectors, the system must contain a partial charge and be operating. The detergent test is accomplished by covering a suspected area with soap suds and watching for bubbles. An electronic detector has a probe which is moved along the plumbing to pick up escaping refrigerant. R-12 is heavier than air so the probe should be held below the lines. When the electronic probe picks up the presence of escaping refrigerant, the detector emits a flashing light or a high pitched squeal. If a leak is detected at a fitting the use of soft copper flare gaskets in the flare fitting is permissible if the connection does not seal by the normal method.

4. TORQUE PROCEDURE FOR WIGGINS COUPLING

Correct torquing procedures for Wiggins couplings throughout the air conditioning system are as follows: Lubricate packings and internal coupling surfaces with the air conditioning system fluid. Tighten the coupling nuts hand tight and safety wire.

5. CHECKING SYSTEM OIL QUANTITY

If a refrigerant leak has occurred, or a possibility exists that the system oil quantity is low, the quantity may be checked as follows: Operate the system for at least five minutes. Shut down for one to five minutes and, while the compressor is still hot, crack the compressor sump fitting. A flow of oil indicates adequate oil in the system; nearly dry gas indicates that the system is low on oil and the compressor must be removed from the system in order to check the oil content. Remove the drain plug and drain all the oil into a beaker while tipping the compressor back and forth and turning the compressor shaft. Add between 4 and 6 ounces of clean 525 viscosity refrigerant oil to the compressor and reinstall the drain plug.

Replacement compressors are shipped with approximately 10 ounces of oil in the compressor. Completely drain the oil from the new compressor and replace it with the same volume of oil that was drained from the old compressor if that volume is between 4 and 6 ounces. If the volume of oil drained from the old compressor is less than 4 ounces, then add 6 ounces of oil to the completely drained compressor.

A. Depressurizing

With a service unit attached slowly open the high pressure valve to allow a slow escape of refrigerant from the system. If oil accompanies the refrigerant, the refrigerant is escaping too rapidly. When hissing ceases, close the valve.

B. Evacuating the System

Using a service unit with a vacuum pump capable of obtaining an absolute pressure of 125 microns or less and a pressure gage capable of indicating an absolute pressure of 125 microns or less; evacuate the system as follows:

(1) Attach the LO service hose to the suction port.

(2) Attach the HI service hose to the discharge port.

(3) Start the vacuum pump, then throttle the pump down by adjusting the vacuum valve to control the turbulence of the oil in the pump.

(4) As the pump-down progresses and the turbulence reduces, gradually open the vacuum and HI and LO valves until they are fully open for maximum evacuation efficiency.

(5) Crack the ballast valve slightly until the pressure is down to about 200 microns, then close the valve to permit continuing the pump-down.

(6) The system must be evacuated to 125 microns or lower.

C. Charging the System

CAUTION: *Use only R-12 refrigerant (52, Table 6, 91-00-00) to charge the air-conditioning system.*

The system should be charged when:

(1) There are bubbles in the refrigerant as seen through the sight glass. If the OAT is above 75°F, the presence of bubbles in the refrigerant indicates that the refrigerant quantity is low.

NOTE: The presence of bubbles when the OAT is below 75°F is not an indication that the refrigerant quantity is low. Refer to the temperature versus pressure graph as an indication of proper charge. Refer to Table 201.

(2) Leaks have been detected in the system.

(3) Air has entered the system.

(4) Components that carry refrigerant have been replaced.

The vapor cycle system should be serviced by a qualified air conditioning service mechanic. Use only R-12 air conditioning refrigerant (52, Table 6, 91-00-00). Refer to the temperature versus pressure chart as an indication of proper charge. Refer to Table 201. It is recommended that the service unit used for charging be equipped with a supply cylinder heated to maintain sufficient pressure to force refrigerant into the system without operating the compressor. If a heated cylinder is not available, the system may be charged without operating the compressor by allowing refrigerant to flow into the system until equilibrium pressure is reached. After equilibrium pressure is reached, allow the compressor to engage with the RH engine operating at approximately 65% N_1. Add refrigerant vapor to the suction side of the system.

Refrigerant should be added in vapor form to prevent compressor damage. Operate the compressor until no bubbles are seen in the sight glass then add approximately 8 ounces of additional refrigerant. The total capacity of the vapor cycle system is 120 ounces of R-12 refrigerant.

Table 201
Temperature vs. Pressure

Ambient Temp (°F)	Compressor Suction Pressure (PSIG)	Compressor Discharge Pressure (PSIG)
60 to 70	25 to 35	120 to 160
70 to 80	30 to 40	150 to 200
80 to 90	30 to 50	170 to 230
90 to 100	35 to 60	200 to 260
100 to 110	40 to 60	250 to 310

6. COMPRESSOR

If a refrigerant leak has occurred, or a possibility exists that the system oil quantity is low, the quantity may be checked as follows: Operate the system for at least five minutes. Shut down for one to five minutes and, while the compressor is still hot, crack the compressor sump fitting. A flow of oil indicates adequate oil in the system; nearly dry gas indicates that the system is low on oil and the compressor must be removed from the system in order to check the oil content. Remove the drain plug and drain all the oil into a beaker while tipping the compressor back and forth and turning the compressor shaft. Add between 4 and 6 ounces of clean 525 viscosity refrigerant oil to the compressor and reinstall the drain plug.

Replacement compressors are shipped with approximately 10 ounces of oil in the compressor. Completely drain the oil from the new compressor and replace it with the same volume of oil that was drained from the old compressor if that volume is between 4 and 6 ounces. If the volume of oil drained from the old compressor is less than 4 ounces, then add 6 ounces of oil to the completely drained compressor.

A. Removal

(1) Remove the lower nacelle panel which contains the compressor servicing door.

(2) Disconnect the electrical leads to the compressor magnetic clutch.

(3) Loosen the compressor discharge service connection line just enough to allow all pressure to bleed off slowly. After the pressure has bled off, disconnect the discharge service connection line and the suction service connection line from the compressor.

(4) Release the tension on the belt tension adapter. Remove the bolt attaching the belt tension adjuster to the compressor.

(5) Remove the drive belts.

(6) Remove the bolts securing the compressor to its mount assembly.

(7) Remove the compressor.

(8) Drain the oil from the compressor. Measure and record the amount collected.

B. Installation

(1) If installing a new compressor, completely drain the oil from the compressor (Ref. Figure 204).

NOTE: If the volume of oil drained from the existing compressor is between 4 and 6 ounces replace it with the same volume. If the volume of oil drained from the existing compressor is less than 4 ounces, add 6 ounces of oil to the completely drained compressor.

If catastrophic failure caused oil loss in the existing compressor, perform the EVACUATING THE SYSTEM, CLEANING SYSTEM and CHARGING THE SYSTEM procedures.

(2) Add the same volume of new oil (6, Table 6, 91-00-00) to the compressor, as was drained and measured, from the existing compressor. Refer to COMPRESSOR REMOVAL procedure.

(3) Install compressor to mount assembly and secure with attaching bolts. Torque to 95 to 110 inch-pounds.

(4) Install drive belts.

(5) Install belt tension adjuster.

(6) Perform the COMPRESSOR BELT TENSION ADJUSTMENT procedure.

NOTE: Torque belt tension adjuster attaching bolts to 60 to 85 inch-pounds. Torque drive belt housing bolts to 25 to 30 inch-pounds.

(7) Install the discharge service connection line and suction service line to the compressor. Refer to SERVICING THE AIR CONDITIONING SYSTEM in Chapter 12-10-00 for servicing procedures.

(8) Connect electrical leads to the compressor magnetic clutch.

NOTE: The gap between the friction faces of the pulley and clutch should be 0.022 to 0.057 inch.

(9) If the compressor mount support has been removed, use shims (8) to locate the support mounting surfaces in the plane of the gearbox housing within 0.002 inch when the attaching nuts are torqued (Ref. Figure 204).

(10) Install the lower nacelle panel.

7. COMPRESSOR BELT

A. Removal

(1) Open the upper cowling on the right nacelle to gain access to the compressor drive belt. The upper cowling may be disconnected at its hinges and removed if desired. Open the compressor access door.

(2) Remove the bolts attaching the drive belt housing to the compressor mount assembly.

(3) Release the tension on the belt tension adjuster and then disconnect the linkage at one end.

(4) Remove both belts.

B. Installation

NOTE: Always replace belts in pairs, never individually. If only one belt is replaced, excessive loads will be applied to the new belt, shortening its life.

(1) Install both belts.

(2) Connect belt tension adjusters. Torque attaching bolts to 65 to 85 inch-pounds. Torque drive belt housing bolts to 25 to 30 inch-pounds.

(3) Install drive belt housing to compressor mount assembly.

(4) Close compressor access door and upper cowling.

C. Tension Adjustment

After 50 hours operating time, a belt will stretch to its normal operating length. The belt tension should be checked at this time and adjusted (by turning the linkage from the compressor to the mount) so that a 3 pound force, applied at midspan, will deflect each belt 0.17 inch.

8. CONDENSER

A. Removal

(1) Remove the lower nacelle panel which contains the compressor servicing door.

(2) Loosen the compressor discharge service connection line just enough to allow all pressure to bleed off slowly out of the entire system.

(3) Remove the nose baggage compartment floor which covers the condenser and remove the ram air duct.

(4) Disconnect the condenser inlet and outlet lines.

(5) Remove the top and bottom condenser attaching bolts and screws. Some of the attaching bolts along the top of the condenser are accessible through the nose wheel well.

(6) Lift the condenser out.

B. Installation

(1) Install condenser and tighten attaching bolts and screws.

(2) Install condenser inlet and outlet lines.

(3) Install the nose baggage compartment floor.

(4) Tighten the compressor discharge service connection line. Refer to SERVICING THE AIR CONDITIONING SYSTEM in Chapter 12-10-00.

(5) Install lower nacelle panel.

9. CONDENSER BLOWER

A. Removal

(1) Remove the nose baggage compartment floor over the condenser blower.

(2) Disconnect the electrical leads from the motor.

(3) Remove the bolts securing the blower base to its mounting base.

(4) Remove the floor support angle.

(5) Lift the blower out of its compartment.

B. Installation

(1) Install condenser blower into its compartment.

(2) Install the floor support angle.

(3) Install the blower base to mounting bracket bolts.

(4) Connect electrical leads to the motor.

(5) Install nose baggage compartment floor.

10. FORWARD VENTILATION BLOWER

A. Removal

(1) Remove the nose baggage compartment floor section covering the forward ventilation blower to gain access for removal.

(2) Disconnect the electrical leads to the blower.

(3) Disconnect the ram air door cable.

(4) Remove the bolts securing the motor mount to the bulkhead.

(5) Remove the floor support angle on the blower support.

(6) Disconnect the differential pressure switch sense line.

(7) Lift the blower out of its compartment.

B. Installation

(1) Mount ventilation blower in its compartment.

(2) Connect differential pressure switch sense line.

(3) Install floor support angle to blower support.

(4) Install motor mount to bulkhead bolts.

(5) Connect ram air door cable.

(6) Connect electrical leads to blower.

(7) Install nose baggage compartment floor.

11. AFT VENTILATION BLOWER

A. Removal

(1) Remove the seat, carpet and access plates over the evaporator and blower at FS 256.25.

(2) Disconnect the electrical leads to the motor.

(3) Remove the bolts attaching the blower cage outlet to the aft bulkhead.

(4) Remove the bolts attaching the motor to its mounting bracket.

(5) Remove the floor support angle on the blower support.

(6) Disconnect the differential pressure switch sense line.

(7) Lift the blower out of its compartment.

B. Installation

(1) Install the ventilation blower into its compartment.

(2) Connect the differential pressure switch sense line.

(3) Install the floor support angle on the blower support.

(4) Install bolts attaching the motor to its mounting bracket.

(5) Install the bolts attaching the blower cage outlet to the aft bulkhead.

(6) Connect electrical leads to the motor.

12. FORWARD EVAPORATOR

A. Removal

(1) Remove the lower nacelle panel which contains the compressor servicing door.

(2) Loosen the compressor discharge service connection line just enough to allow all pressure to bleed off slowly out of the entire system.

(3) Remove the section of nose baggage compartment floor covering the forward evaporator in order to gain access for removal.

(4) Remove the metal clamp securing the thermostat to the evaporator outlet line.

(5) Disconnect the inlet, outlet and bypass lines from the evaporator.

(6) Remove the bolts attaching the outboard edge of the evaporator to the bulkhead.

(7) Move the evaporator outboard just enough to clean its inboard track and center the evaporator under the access opening.

(8) Lift the evaporator straight up from the compartment.

B. Installation

(1) Mount the evaporator in its compartment.

(2) Install the bolts attaching the outboard edge of the evaporator to the bulkhead.

(3) Connect the inlet, outlet and bypass lines to the evaporator.

(4) Secure the thermostat to the evaporator outlet line with the metal clamp.

13. AFT EVAPORATOR

A. Removal

(1) Remove the lower nacelle panel which contains the compressor servicing door.

(2) Loosen the compressor discharge service connection line just enough to allow the pressure to bleed off slowly out of the entire system.

(3) Remove the seat, carpet and access plate on the right hand side of the cabin aisle at FS 266.50.

(4) Remove the metal clamp securing the thermostat to the evaporator outlet line.

(5) Disconnect the inlet, outlet and bypass lines.

(6) Remove the bolts attaching the outboard edge of the evaporator to the bulkhead.

(7) Move the evaporator outboard just enough to clean its inboard track and center the evaporator under the access opening.

(8) Lift the evaporator straight up from the compartment.

B. Installation

(1) Mount the evaporator in its compartment.

(2) Install the bolts attaching the outboard edge of the evaporator to the bulkhead.

(3) Connect the inlet, outlet and bypass lines to the evaporator.

(4) Secure the thermostat to the evaporator outlet line with the metal clamp.

(5) Install the seat carpet and access plate.

(6) Tighten the compressor discharge service connection line. Refer to SERVICING THE AIR CONDITIONER SYSTEM in Chapter 12-10-00.

(7) Install lower nacelle panel.

14. EVAPORATOR FILTER

A. Replacement

The evaporator filter should be replaced every 300 hours or more frequently if operating in heavy dust or smoke conditions. To gain access to the filters, refer to the applicable evaporator access opening in Chapter 12. Replacement filters are installed with the lattice backing against the evaporator coils and taped to the evaporator case.

15. CLEANING SYSTEM

The system can be flushed by using refrigerant R-11 and dry nitrogen.

CAUTION: *Do not use solvent, as this will cause contamination of the refrigerant oil.*

16. QUILL SHAFT

The system can be flushed by using refrigerant R-11 and dry nitrogen.

A. Lubrication

The compressor drive quill shaft must be lubricated every 600 hours (Ref Figure 204).

NOTE: Engines that are in compliance with Pratt & Whitney Service Bulletin 1354 have provisions to lubricate the engine end of the quill shaft automatically. However, the pulley drive end must still be manually lubricated at 600-hour intervals.

(1) Open the RH inboard cowling door.

(2) Cut the safety wire and relieve the tension on the tension adjuster. Remove the tension adjuster bolts and remove the tension adjuster.

(3) Cut the safety wire and remove the drive belt housing and pulley support bolts from the drive belt housing and pulley support. Remove the drive belt housing and pulley support, compressor drive belt, and compressor drive pulley.

(4) Cut the safety wire and remove the forward and aft pivot bolts. Note the number of compressor aligning shims in their respective locations to aid in aligning the compressor to the compressor mount when installing the compressor.

NOTE: Secure the compressor to prevent excessive strain on the connecting lines.

(5) Cut the safety wire and remove the seven compressor mount bolts securing the compressor mount to the engine and compressor mount support.

(6) Remove the compressor mount and quill shaft. Exercise care not to lose the spring at the end of the quill shaft or the two seals at the engine end of the mount.

(7) Apply lubricant (111, Table 6, 91-00-00) to the ends of the quill shaft with a brush.

NOTE: If the engine is in compliance with Pratt and Whitney Service Bulletin 1354, only the drive pulley end of the quill shaft requires the lubricant.

(8) Install the quill shaft in the compressor mount.

CAUTION: *Assure that the mount seals and quill shaft spring are in place prior to installation of the mount. If the existing seals are worn or damaged, install new seals.*

(9) Install the compressor mount and compressor mount support. Torque the compressor mount bolts to a value of 40 to 50 inch-pounds. Safety wire all but the lower bolt in place with MS20995C32 safety wire.

(10) Install the compressor on the compressor mount and, with the compressor aligning shims installed in the same sequence as they were removed in Step (4), snug the compressor forward and aft pivot bolts down. Do not fully torque the bolts at this time.

(11) Install the compressor drive pulley and compressor drive belt.

(12) Prior to installing the drive belt housing and pulley support, measure the shims for a dimension of 0.010 ± 0.002 inch to provide a clearance of 0.010 ± 0.002 inch between the spacer and bearing (Ref. Figure 204). The shims may be reused if they are not damaged.

(13) Install the drive belt housing and pulley support and torque the drive belt housing and pulley support bolts to 25 to 30 inch-pounds.

(14) Safety wire the drive belt housing and pulley support bolts in place with MS20995C32 safety wire.

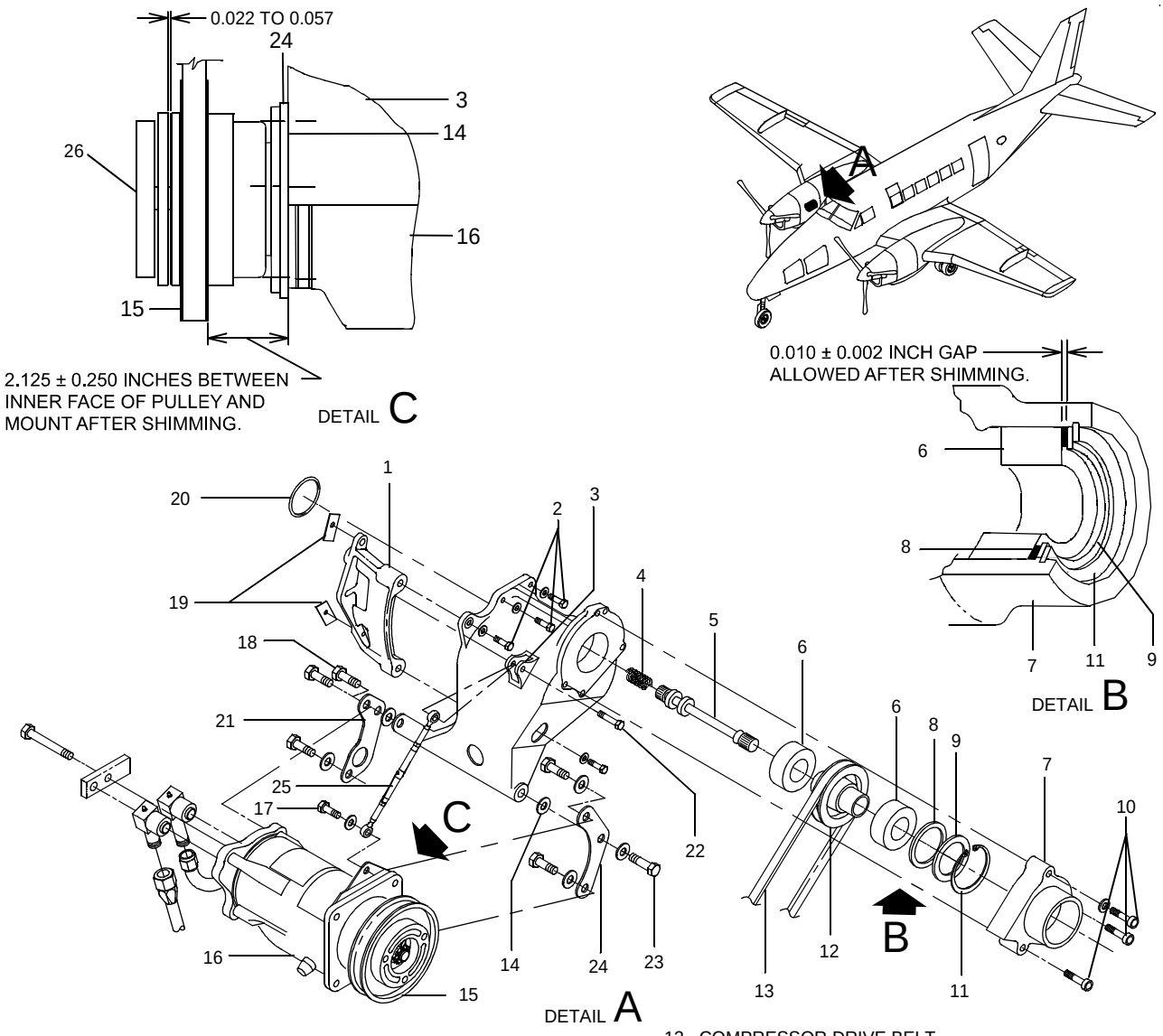
(15) Install the tension adjuster and adjust the compressor drive belt until a force of three pounds applied at the belt midspan deflects it 0.17 inch.

(16) Safety wire the tension adjuster with MS20995C32 safety wire.

(17) Torque the tension adjuster bolts to a torque value of 65 to 85 inch-pounds.

(18) Torque the forward and aft pivot bolts to a torque value of 95 to 110 inch-pounds. To ensure a dimension of 2.150 ± 0.025 inch between the inner face of the compressor pulley and the compressor mount as shown in Detail C, compressor aligning shims may be removed or added as required.

(19) Safety the forward and aft pivot bolts in place with MS20995C32 safety wire.



1. COMPRESSOR MOUNT SUPPORT
2. COMPRESSOR MOUNT BOLTS
3. COMPRESSOR MOUNT
4. SPRING
5. QUILL SHAFT
6. BEARING
7. DRIVE BELT HOUSING & PULLEY SUPPORT
8. SHIM
9. SPACER
10. DRIVE BELT HOUSING & PULLEY SUPPORT BOLTS
11. RETAINING RING
12. COMPRESSOR DRIVE PULLEY
13. COMPRESSOR DRIVE BELT
14. COMPRESSOR ALIGNING SHIM
15. PULLEY FRICTION FACE
16. COMPRESSOR
17. TENSION ADJUSTER BOLT
18. FORWARD PIVOT BOLT
19. SHIMS
20. SEAL
21. FORWARD PLATE
22. TENSION ADJUSTER BOLT
23. AFT PIVOT BOLT
24. AFT PLATE
25. TENSION ADJUSTER
26. ELECTROMAGNETIC CLUTCH

U#21B
034122AA.AI

Figure 204
Compressor Installation

AIR CONDITIONING
TEMPERATURE CONTROL
DESCRIPTION AND OPERATION

1. U-50, U-165 AND AFTER

The heart of the environmental control system in the C99 Airliner Series Airplane is the highly sophisticated cabin temperature controller. It operates on the balancing wheatstone bridge principle. Two thermal sensitive resistors in the cabin air ductwork make up one-half of the bridge circuit, while the cabin temperature selector potentiometer and the thermally sensitive circuitry in the cabin temperature controller make up the other half of the bridge circuit. With a falling cabin temperature, the bridge will become unbalanced toward the cabin half of the bridge, causing the current to flow from the duct half of the bridge circuit; this current is interpreted as a heat command by the modulating valve which opens in response, allowing more bleed air to be mixed with the air supplied by the blower. With a rising cabin temperature, the bridge will become unbalanced toward the duct half of the bridge, causing the current to flow from the cabin half of the bridge circuit; this current is interpreted as a cool command by the modulating valve which closes in response, allowing less bleed air to be mixed with the air supplied by the blower.

This continuous modulation of cabin air causes changes in the resistance of the temperature sensing circuitry which rebalances the bridge. Automatic control of the cabin heating and cooling systems occurs only when the cabin temperature mode switch is in the AUTO position.

In either of the manual modes, cabin temperature can be manually controlled through the manual temperature control switch. Holding the switch in the lNCR position will raise cabin temperature by applying a simulated heat command to the heat side of the modulating valve, driving it open. The DECR position will lower the cabin temperature by applying a simulated cool command to the cool side of the modulating valve, driving it closed.

The 6 degree switch in the modulating valve initiates the energizing signal which doses the cool command relay on the environmental control relay card assembly. The 22° switch will keep the relay energized once the modulating valve has opened beyond 6 degrees. until the 22° limit is reached. Energizing current is conducted through the closed contacts of the 50°F low temperature relay, which is being energized through the 50 degree F low temperature switch.

The new cod command, taken from the 50°F low temp relay contacts, is used as input to the compressor dutch control module, as well as energizing current for the condenser cooling actuator polarity reversing relay. This new cool command is either on or off, as opposed to the modulated commands generated by the cabin temperature controller.

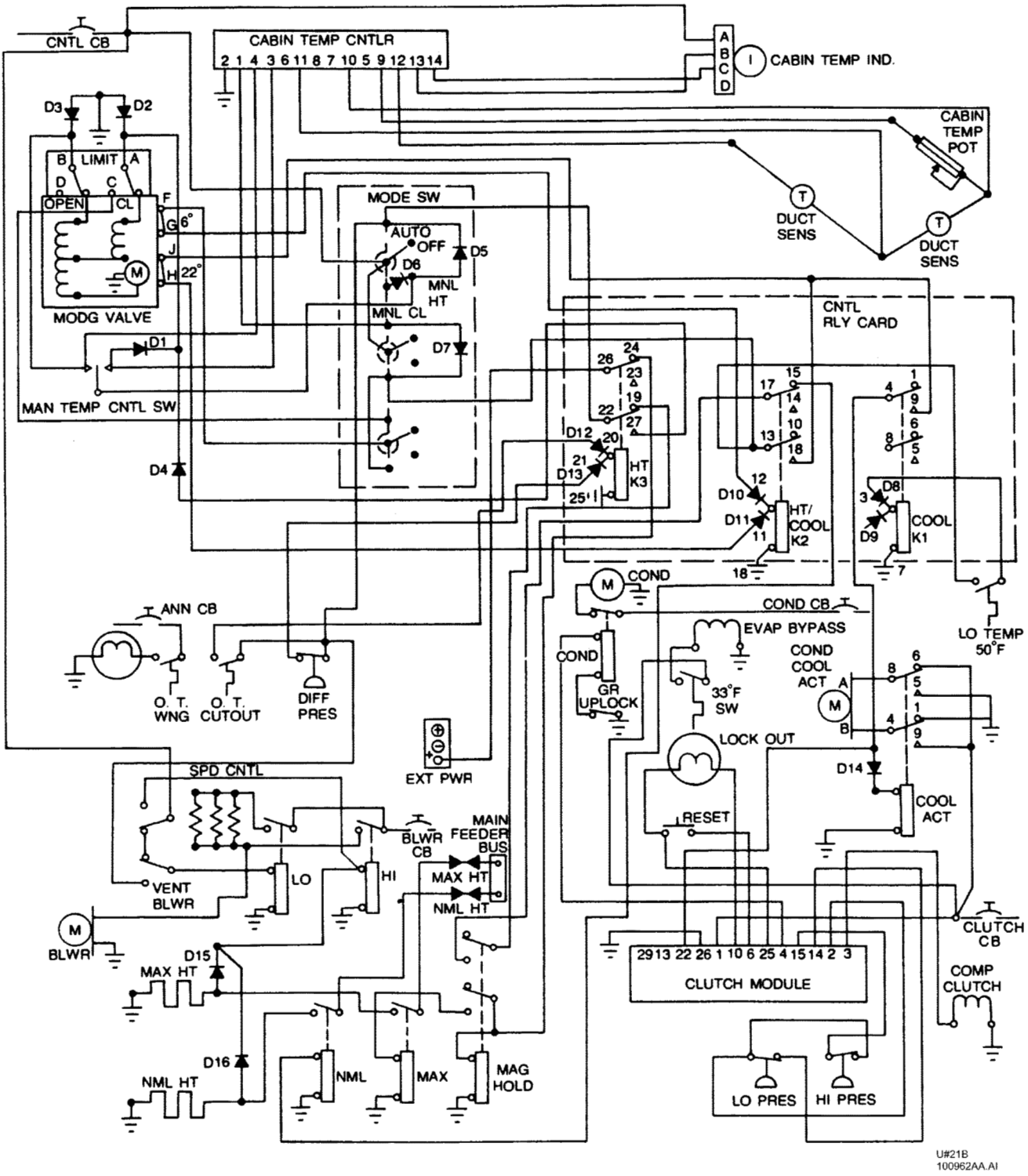


Figure 1
Air Conditioning Control System Schematic

AIR CONDITIONING
ENVIRONMENTAL SYSTEM
TROUBLESHOOTING

1. ENVIRONMENTAL SYSTEM CONTROL CIRCUITS (U-50, U-165 AND AFTER)

Due to the complex design of the environmental system's controlling circuitry, a test unit was designed which can be assembled and used to analyze and troubleshoot the system. Whenever the environmental system will not operate in auto or manual modes, use of this environmental system test unit, in conjunction with the information in Chapter 21 of the current C99 Wiring Diagram Manual and Maintenance Manuals may assist with the troubleshooting of this system. Instructions and materials required to fabricate the test box are outlined later in this chapter. This test unit cannot be used to troubleshoot the cabin temperature controller; this procedure is discussed later in this chapter. Use of this test unit will allow accomplishment of the following:

(1) One person monitoring of voltage throughout the control system on a single display.

(2) Built-in intermittency detector for each test point.

(3) Adapter harnesses are provided to allow quick adaption to the airplane without having to cut wire bundle ties, or to hunt for a specific wire in the bundle or to probe any of the wires.

This test unit is a small box with 34 lights, laid out upon a single schematic panel; 18 of the lights are electrically in parallel with eighteen different preselected test points in the circuit. When illuminated, visual detection of the presence of voltage of that point is allowed in the circuit where the light bulb is found in the schematic.

A row of 16 lights below the circuit schematic is for the voltage detection circuit. Each bulb is in series with a silicone controlled rectifier to ground and each detection circuit is in parallel with 16 of the 18 test points that are being monitored. Voltage intermittences of the millisecond range can be detected with this test set. Once the system under test is energized, the intermittency detection lights are lit by depressing a normally open switch that momentarily applies 28 VDC to the gates of all the S.C.R.S. Should the current at any test point cease momentarily at any time during system operation, the intermittency detection light for that test point will extinguish and will not illuminate when voltage is resumed to the circuit. The detection circuit must be reset by the operator.

This test box is coupled to the airplane under test with one harness. This harness attaches to the test box with a connector and branches off to three separate leads which individually are teed between a male and female connector that plugs in series with existing connectors in the airplane. This method of attaching test leads to the airplane takes only a few minutes to prepare an airplane for testing and about the same amount of time to put is back into service. The following information indicates what is occurring in the system as each light illuminates. Lights 1 through 9 represent the air conditioner circuit. Lights 10 through 16 represent the normal and maximum heat circuits.

Test Indication Guide

Light Number	Indication
1-	Voltage is present at Pin 13 of the P228 connector (A228 relay card) and on the common terminal of the K2 (A228 relay). Voltage here indicates that either auto or manual mode has been selected.

Test Indication Guide (Continued)

Light Number	Indication
2-	Voltage is present at Pin 18 the P228 connector (A228 relay card). Voltage at this point indicates that the K2 (A228 relay) has pulled in and has traversed through the relay to one of the normally open contacts of the K1 (A228 relay card) relay and for one side of the 22° switch
3-	Voltage is present at Pin 4 of the P228 connector (A228 relay card). Voltage present indicates that the K1 (cool) relay (A228 relay card) has pulled in. This relay pull s in when the 50° OAT switch (47 PSI) S167 is closed and opens up when S167 switch is open.
4-	Voltage is present at Pin 22 of the P237 connector on the (A237) clutch module, having passed through Pin 1A of the plug P315 and through a terminal splice at Pin 3 of plug P225. This voltage is the command to activate the time delay circuit in the clutch module (A237) for air conditioner clutch actuation.
5-	Voltage is present at Pin 1 of the P237 connector on the A237 clutch module from the 5 AMP air conditioner clutch circuit breaker. This is the voltage that pulls in the compressor clutch 10 seconds after the clutch module receives a command from the K1 relay (A228 relay card).
6-	Voltage is present at Pin 3 of the P237 connector in the A237 clutch module. This voltage is an output of the clutch module PCB (A237) and is the voltage that energizes the compressor clutch.
7-	Voltage is present at Pin 12 of the P228 connector (A228 relay card). This voltage energizes the coil of the K2 relay (A228 relay card) and originated as a cool command out of the cabin temp control box, passing through the modulating valve only when valve is fully closed (full cool position), through the mode switch, and through the 6° of travel from the full closed position on the modulating valve.
8-	Voltage is present at Pin 11 of the P228 connector, (A228 relay card). This voltage indicates that the 22° (degrees of travel) switch is closed and it provides latch in power for the K2 relay (A228 relay card). The 22° switch remains closed from 0 to 22° of travel from the full cool (closed) position of the modulating valve.
9-	Voltage is present at Pin 3 of the P228 connector (A228 relay card). This voltage pulls in the K1 relay (A228 relay card) and comes through the closed contacts of the switch S167 whenever refrigerant pressure is 47 PSI or higher (approximately 50°F).
10-	Voltage is present at Pin 22 of the P228 connector (A228 relay card). Voltage should be present whenever mode switch is in auto, manual heat or manual cool position.
11-	Voltage is present at Pin 19 of the P228 connector (A228 relay card). Voltage indicates that the K3 relay (A228 relay card) is not engaged. This relay is the electric heat disconnect relay and will not engage to remove power from the normal and max heat control relays whenever there is loss of differential pressure in the vent system or an overtemp signal.

Test Indication Guide (Continued)

Light Number	Indication
12-	Voltage is present at Pin 17 of the P228 connector (A228 relay card). To get voltage, the magnetic hold electric heat switch must be in either normal or max heat position.
13-	Voltage is present at Pin 15 of the P228 connector (A228 relay card). Voltage here will engage the coil of the normal heat relay and is available only when the K2 relay (A228 relay card) is de-energized (not asking for the air conditioner to run).
14-	Voltage is present at Pin 24 of the P228 connector (A228 relay panel). Voltage source is the external aux. power unit, through a 5 AMP circuit breaker, then through the de-energized contacts of the K3 (A228 relay panel) electric heat disconnect relay.
15-	Voltage is present at Pin 20 of the P228 connector (A228 relay card). Voltage here would indicate that the 350°F duct over temp cutout switch S173 has closed due to temperatures above 350°F. This voltage will engage the electric heat disconnect relay and remove power from both the normal and max heat control relays.
16-	Voltage is present at pin 21 of the P228 connector (A228 relay card). Voltage here would indicate a loss of differential pressure in the vent system and would engage the electric heat disconnect relay and remove power from both the normal and max heat control relays.
COOL-	Voltage is present at Pin 3 of the P191 connector. This is output voltage from the automatic cabin temperature control box driving the modulator valve in the cool (closed) direction.
HOT-	Voltage is present at Pin 4 of the P191 connector. This is output voltage from the automatic cabin temperature control box driving the modulator valve in the hot (open) direction.

2. TEST BOX

A. Fabrication

The following is a list of materials required and wiring schematics for fabrication of the test box:

Materials Required

Part or Connector P/N	No. Required	Pins or Socket P/N	No. Required	Connects to Nomenclature
3-582151-3*	1 Ea.	66026-2*(accept two wires) 66010-2* (accepts one wire)	13 6	A228 PCB
3-582151-3	1 Ea.	66026-2* 66010-2*	4 9	A237 PCB
3-582152-3*	1 Ea.	Pins Included	-	P237
3-582152-3*	1 Ea.	Pins Included	-	P238

Materials Required (Continued)

Part or Connector P/N	No. Required	Pins or Socket P/N	No. Required	Connects to Nomenclature
206043-3*	1 Ea.	66101-1* 66105-4*	2 8	E160 Temp Control Box
206044-1*	1 Ea.	66103-1*	10	P191
205840-3*	1 Ea.	205202-4*	19	Test Harness
205839-3*	1 Ea.	205201-5*	19	Test Box
206070-1*	3 Ea.	- - -	- -	Backshell For 206043-3, 205939-3 & 206044-1
99-364206-1	2 Ea.	- - -	- -	P.C. Board Blank
GE C6A	16 Ea.	- - -	- -	S.C.R.
D158-100-5	34 Ea.	018-6000-000-209	34	Lights
601-4300 (P/N of Grayhill)	1 Ea.	- - -	- -	Switch
18, 20 and 22 gage electrical wire - Quantity as required.				
1,000 OHM Resisters - Quantity - 32				
1N5400	2 Ea.			Diode

* Part Number of AMP Incorporated, Harrisburg, PA. 17105

B. Special Tools Required

The following is a list of materials required and wiring schematics for fabrication of the test box:

Materials Required

Socket or Pin P/N	Crimper No.	Insertion and Extraction Tool*
66010-2	90083	465l99-1
66105-4, 66101-1, 66103-4	90067	Insert - 91002-1 Extract - 305183
205202-4, 205201-5	90265-1	91067-2

* Part Number of AMP Incorporated, Harrisburg, PA. 17105

C. Assembly Notes

(1) Use the same gage wire in test box harness as is called out in the Wiring Diagram Manual for each circuit. As an example, the wire that comes off of Pin 4 of P228 (21-61-01) Pg. 3 in the C99 Wiring Diagram Manual) is H96A22. This is a 22 gage wire. Therefore, 22 gage wire should be used for the portion of the harness that connects into that circuit.

(2) The key pin for the A237 air conditioning clutch control P.C. Board is Pin 27. Break off Pin 27 of the connector (P/N 3-582152-3) to be installed on the blank P. C. Board (P/N 99-354206-1).

(3) The key pin for the A228 relay P.C. Board is Pin 28. Break off Pin 28 of the (P/N-582152-3) connector to be installed on blank P.C. Board P/N 99-364206-1.

(4) The following are instructions for assembling the test connections for the A237 and A228 P.C. Boards.

(a) Attach one each P/N 3-582152-3 connector to each blank P.C. Board (P/N 99-364206-1) with two bolts and two nuts (obtained locally). The screws should be approximately one inch long. This will allow sufficient length to attach an additional section of board as a protective cover for the soldered connections.

(b) Cut the blank P.C. Board to an approximate size of 1 x 4 inches.

(c) Solder pins of the connector to the P.C. Board.

(d) Insert the ends of the approximate size wire in the correct holes of the P.C. Board and solder them in place.

(e) Cut an additional 1 x 4 inch section of blank P.C. Board and install on existing screws as a guard plate over the soldered connections.

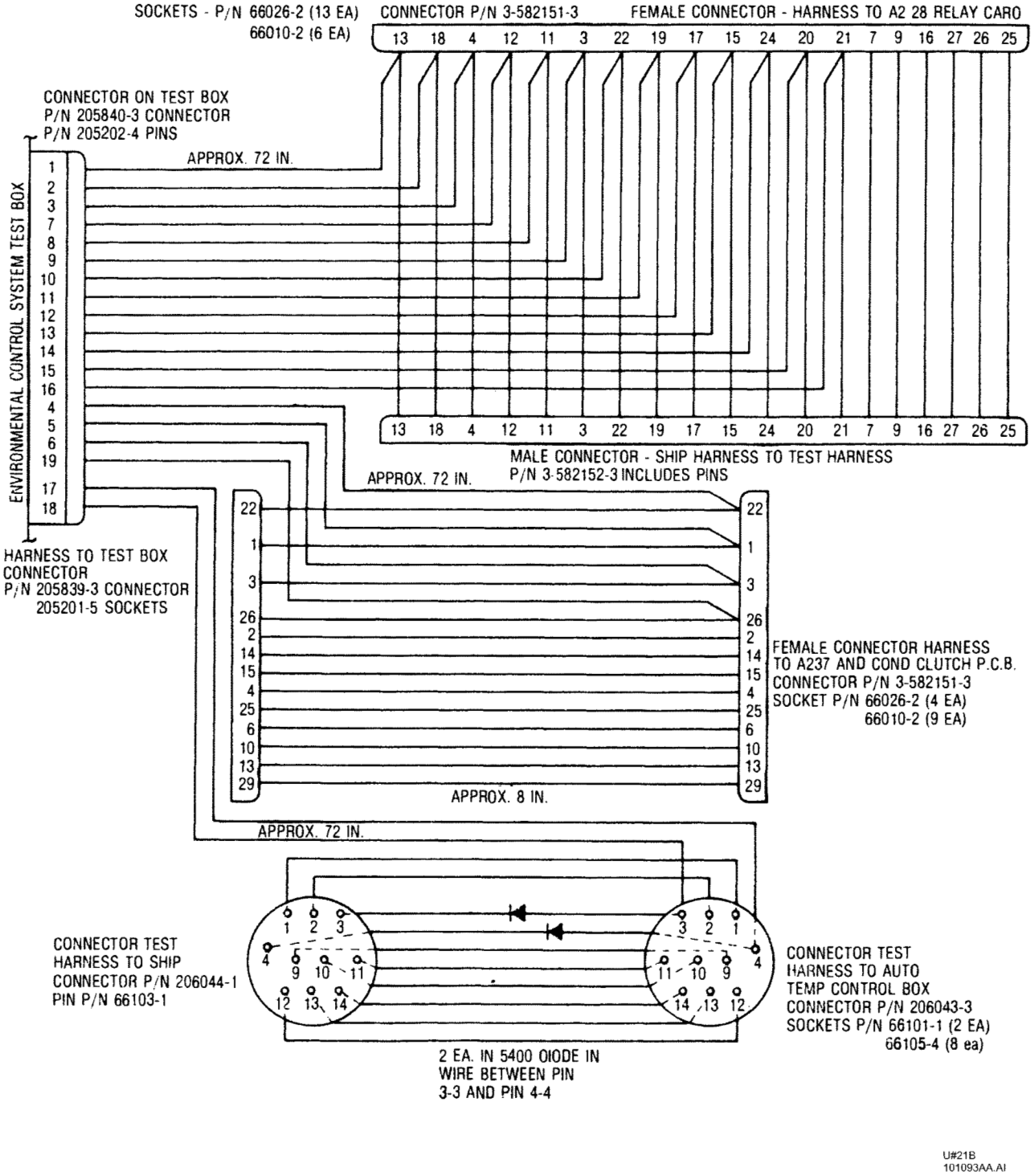


Figure 101
Schematic for Environmental Control System Test Harness

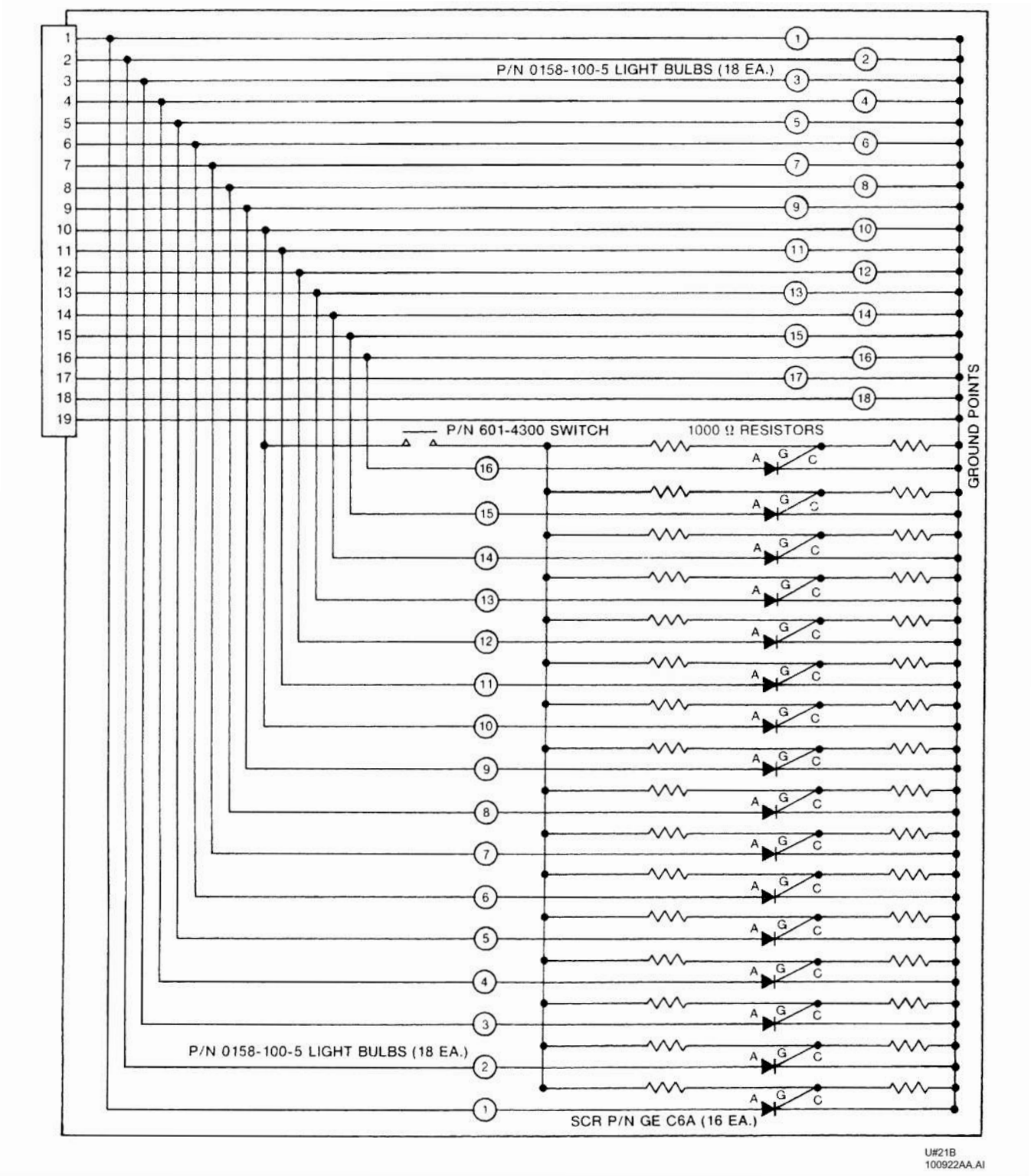


Figure 102
Wiring Schematic for Environmental Control System Test Box

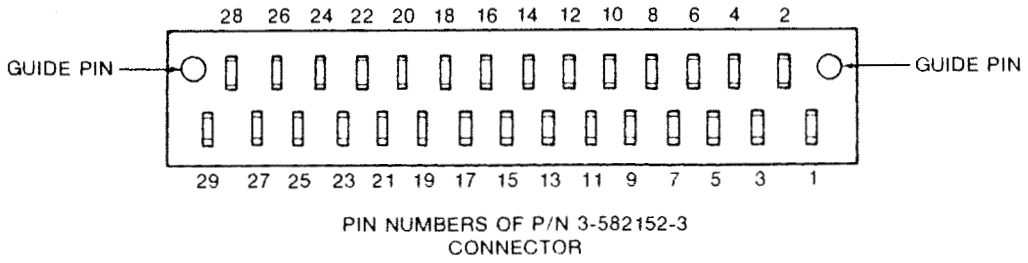


Figure 103
P.C. Board Connector

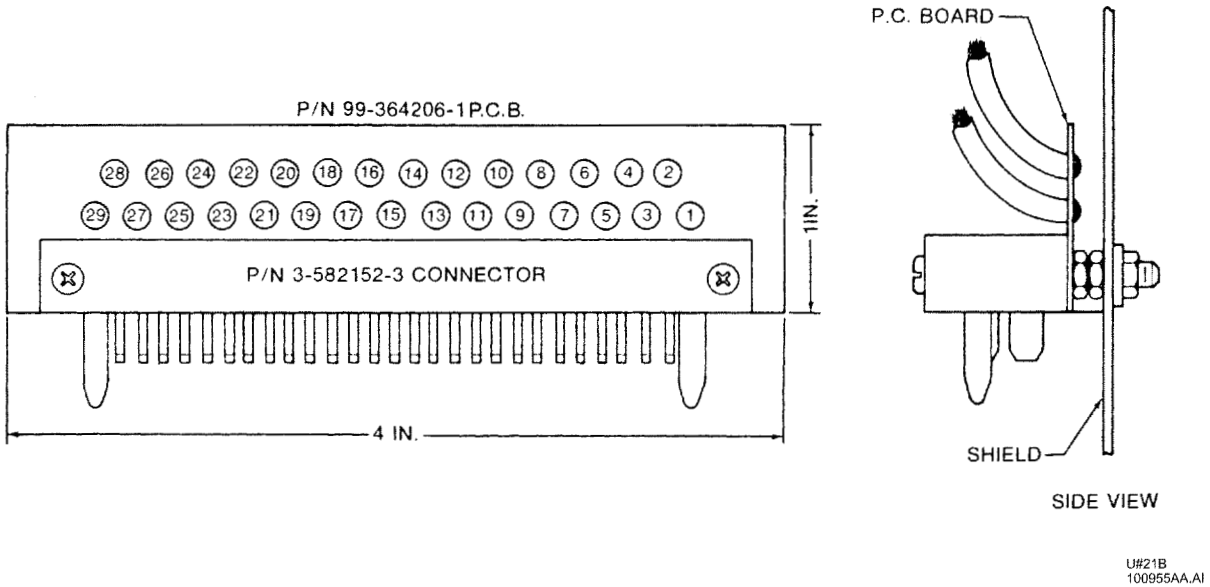


Figure 104
Construction of P.C. Board Connector Adapter

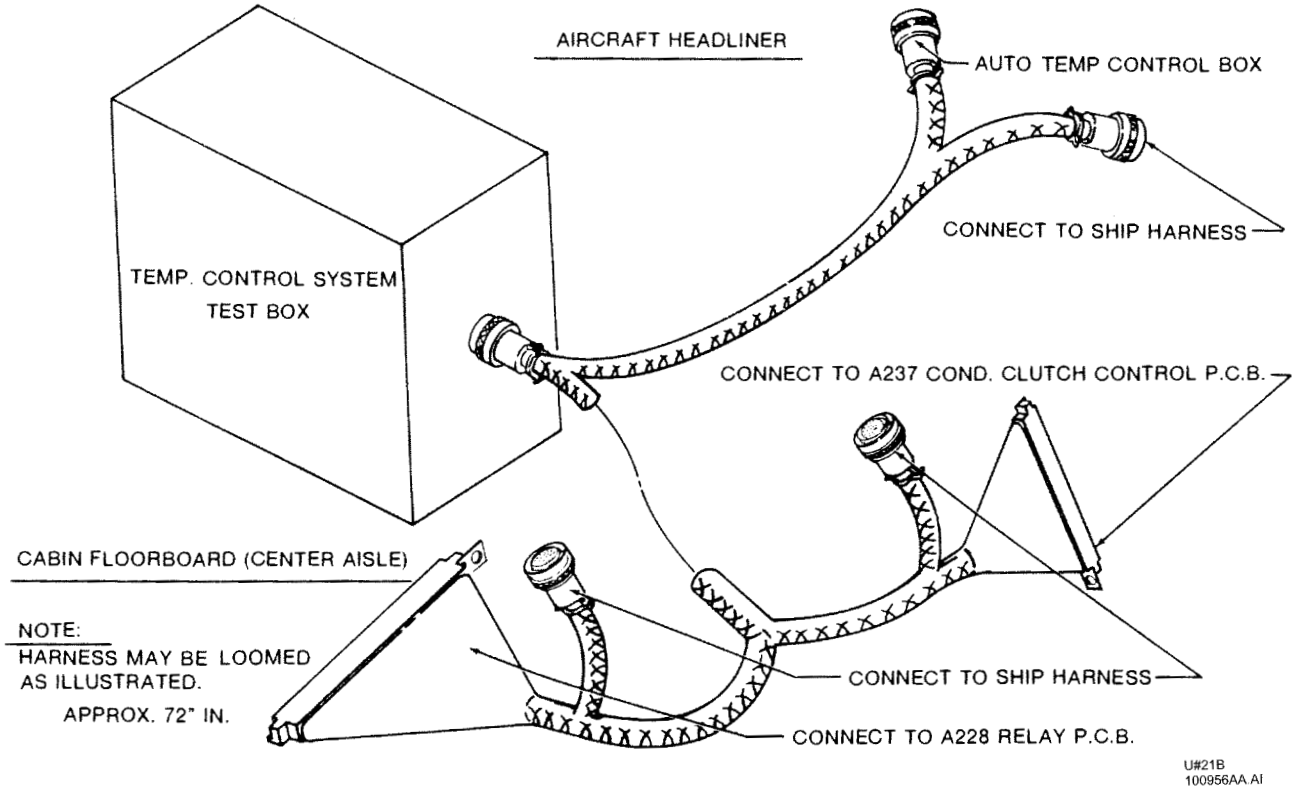


Figure 105
Diagram of Test Box and Harness

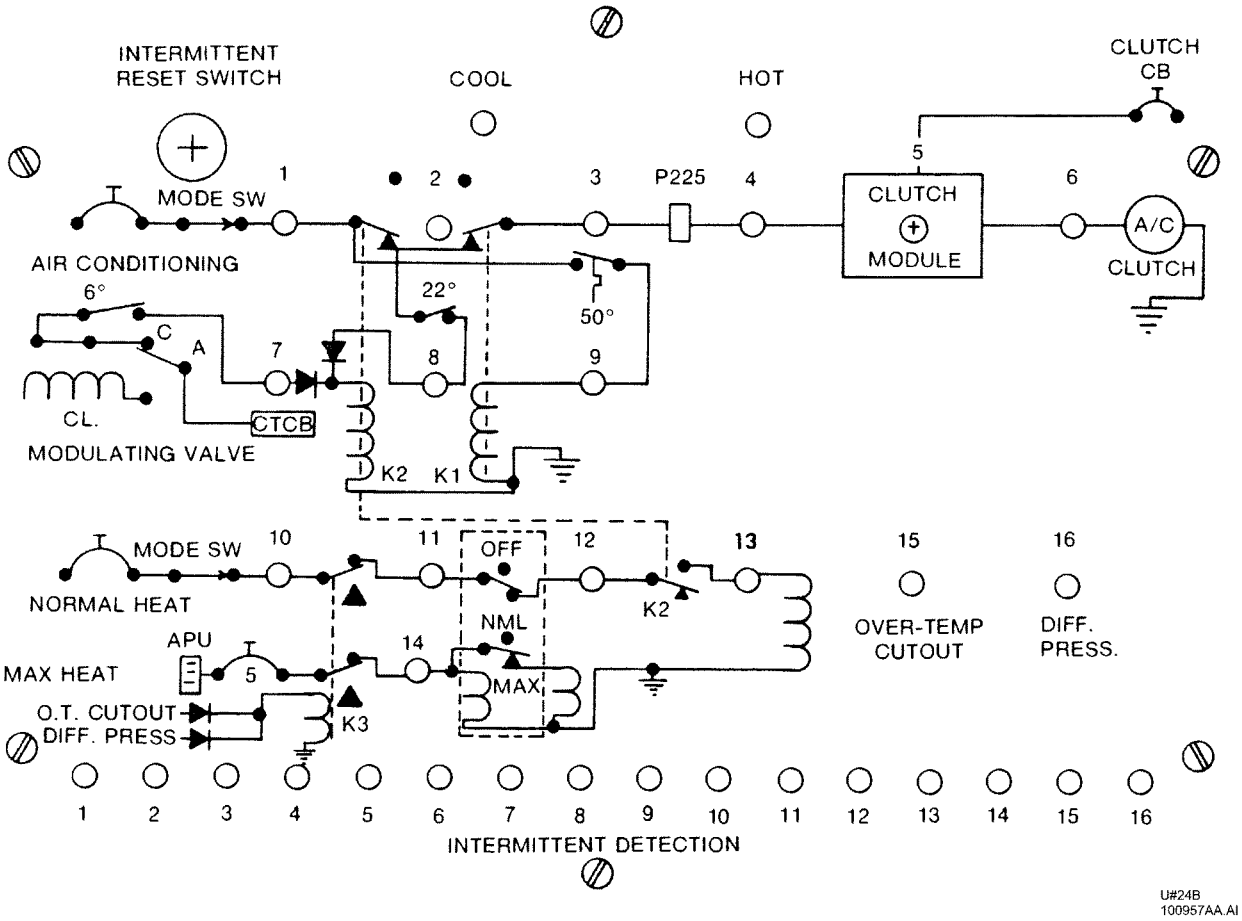


Figure 106
Layout of Environmental Control System Test Box Panel

3. AIR DUCT TEMPERATURE SENSING ELEMENT

A. (U-50, U- 165 and After)

If it has been confirmed that the environmental system is not responding to automatic control, but is responding to manual control, it may be assumed that the fault will be found in either the air duct temperature sensing elements or the cabin temperature controller. Since a higher reliability factor may be found in the controller, it is suggested that the sensing elements be checked first.

Two sensing elements are installed in the lower cabin air ductwork downstream of the electric heater. The resistive function of the elements has been standardized by the use of tungston alloy filaments. One of the elements has been thermally insulated to provide a response which lags behind the other nonthermal insulated element. This activity provides a heat anticipating function in order to prevent overshoot of cabin heating and uncomfortably wide temperature ranges.

A procedure for checking the sensing elements for resistive response is as follows:

NOTE: In order for the resistance measurements of the elements to comply with the reading taken from the graph in Figure 107, it is imperative that the temperature be accurately determined at the location of the elements.

(1) Disconnect the electrical plug from the temperature controller.

(2) Using the graph in Figure 107, determine the correct resistance for the element relative to the temperature measurement taken at the element location.

(3) Using a sensitive precision ohmmeter (digital preferred) measure the resistance between pins 10 and 11. At the cabin temperature controller connector, this measurement should correspond to line AB on the graph. Measuring between 11 and 12 should produce a value corresponding to line CD on the graph. Line AB represents the thermally lagged element.

(4) If the measured resistance from the sensing elements does not agree with the resistance taken from the graph ±1, the affected element should be replaced.

(5) Should the operation of the environmental system be erratic or intermittent when the system is under automatic control, loose internal connections may be indicated. The following procedure can be used to confirm an erratic fault in the sensing elements:

(a) Connect an auxiliary power unit to the airplane. Place the mode temperature switch in MNL HT. Set the electric heat switch in the MAX position. Hold the manual temperature control switch in the INCR position for 30 seconds.

(b) Monitor the resistive response of the element between pins 11 and 12 while the element temperature is increasing. The resistance should not increase above 200 ohms. Any erratic behavior of the meter is characteristic of a defective element.

(c) Connect the ohmmeter between pins 10 and 11. Hold the manual temperature control in the DECR position for 30 seconds. Monitor the meter while the elements are cooling for any erratic behavior. The cooling process can be hastened by placing the forward vent blower switch in high speed.

(6) If no erratic behavior is observed in the resistive response of either of the sensing elements, a test of the cabin temperature controller should be conducted in order to confirm that the fault is indeed in the controller.

4. CABIN TEMPERATURE CONTROLLER

A. (U-50, U- 165 and After)

If, as mentioned earlier in this chapter, the environmental system does not respond as designed with automatic control, but does operate normally under manual control, the fault will be found in either the cabin temperature controller or in the air duct sensing elements. If it has already been ascertained that the air duct sensing elements are operating according to design, the cabin temperature controller may be evaluated using the following procedure:

The cabin temperature controller contains temperature sensitive circuitry; therefore the following procedure should be carried out in a room where the temperature is between 65°F and 85°F.

A test unit (Ref. Figure 108) was designed to be used for troubleshooting the cabin temperature controller. All resistance values must be exactly adhered to in order to assure the proper performance of the test unit.

(1) The components to be used in constructing the test unit are as follows:

NOTE: R1 and R2 may be selected within the range of 100 ohms to 120 ohms. The difference in value between the two resistors must be 5 ohms to 10 ohms.

(a) A 28 ± 0.1 vdc power supply capable of delivering a minimum current of 2 amperes.

(b) A 107 ohm (± 1%), 1/8 watt resistor (R1).

(c) A 115 ohm (± 1%), 1/8 watt resistor (R2).

(d) Two 25 ohm (± 5%), 50 watt resistors (R3 and R4).

(e) A 100 ohm (± 5%), 0.5% minimum resolution, wire wound potentiometer (TR1).

(f) Two 28 vdc, 40 ma, MS25231-1819 lamps (DS1 and DS2).

(g) An air temperature gage (M1), P/N 50-380026-1 or equivalent.

(h) AMP Incorporated P/N 206043-3 connector, 66538-1 pins, and 206043-3 connector, 66538-1 pins, and 206070-2 strain relief.

(2) Measure the ambient temperature at the inlet to the control box with the air temperature gage (M1).

(3) Connect the controller to the test unit.

(4) Turn the potentiometer fully counterclockwise and turn on the power supply. The DS1 lamp should illuminate.

(5) Turn the potentiometer clockwise (TR1) until it is approximately midway between where lamps DS1 and DS2 are pulsing on.

(6) Turn the potentiometer (TR1) counterclockwise until lamp DS1 starts pulsing on, then continue turning TR1 until lamp DS1 illuminates continuously.

(7) Turn the potentiometer (TR1) clockwise until lamp DS1 goes off, then continue turning TR1 until DS2 begins to pulse on. Continue turning TR1 until DS2 illuminates continuously.

(8) Check that the direction of air flow through the control box is as indicated in Figure 108.

(9) Turn the power supply off and disconnect the controller from the test unit.

If the cabin temperature controller does not perform exactly as described above, the controller may be assumed to be defective and must be replaced.

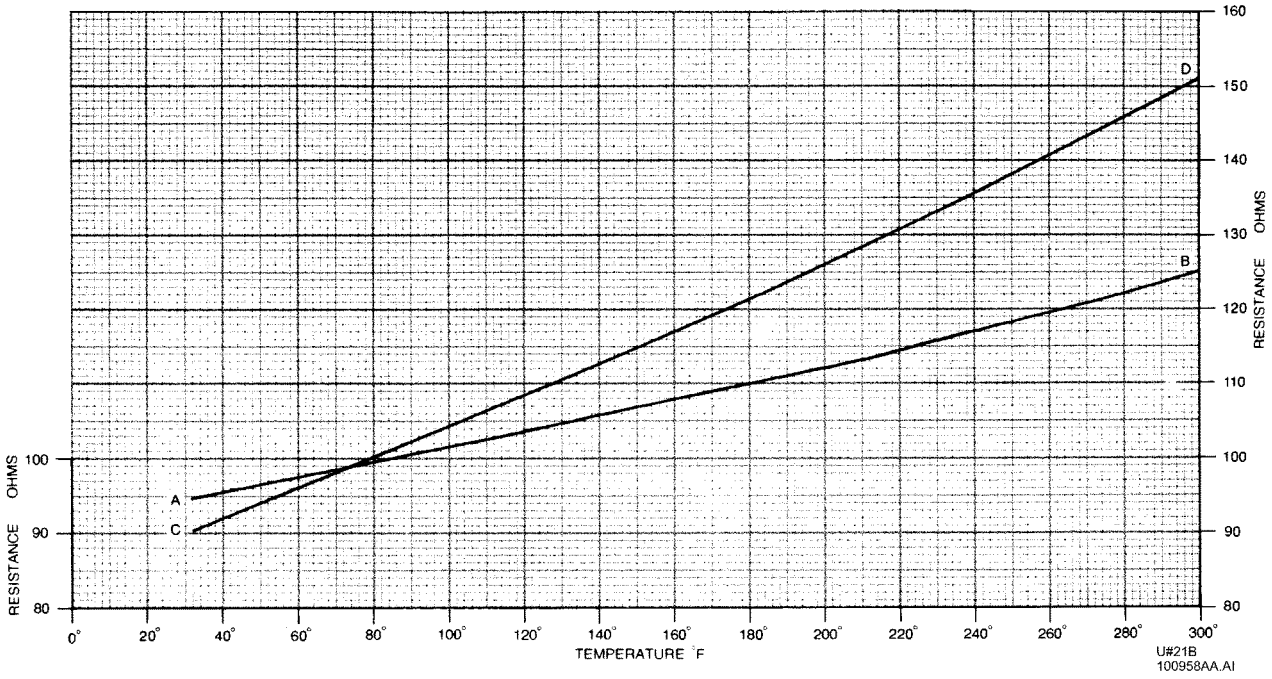


Figure 107
Air Duct Temperature Sensors

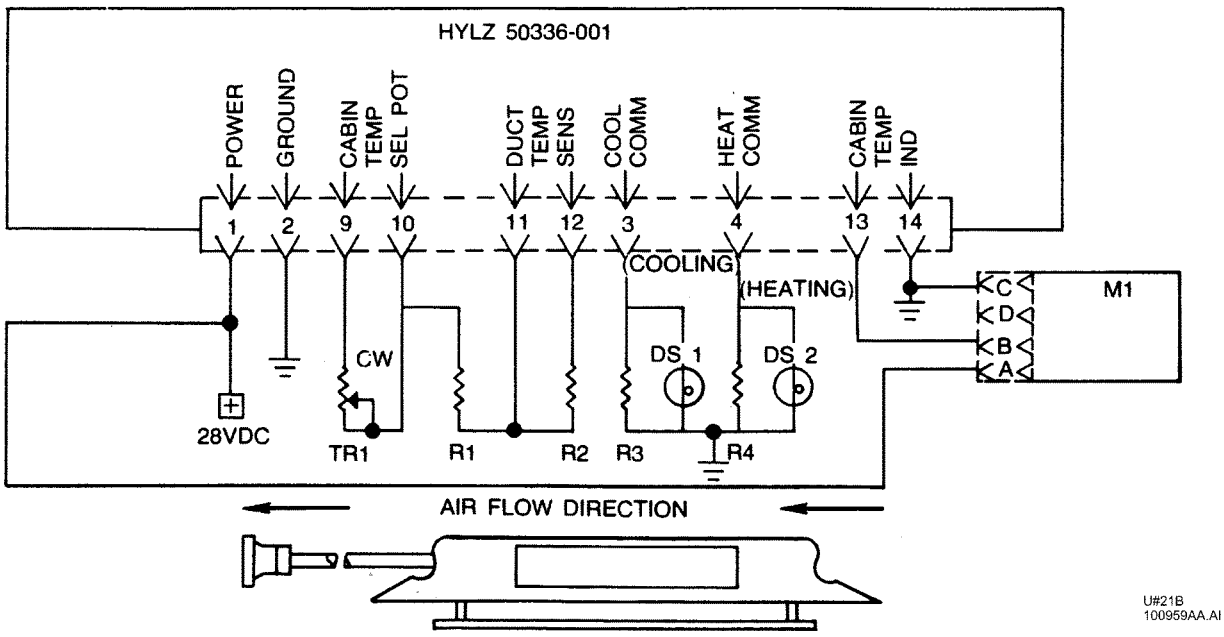


Figure 108
Cabin Temperature Controller Test Unit (U-50, U-165 and After)

Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 22 - AUTO FLIGHT

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AUTO FLIGHT
GENERAL
DESCRIPTION AND OPERATION

1. AUTOPILOT

A. H-14

Primary components of the autopilot pneumatic system are shown in Figure 201. Air pressure picked up in the bleed air system is admitted to the autopilot pneumatic system through a solenoid valve controlled by a flight controller. An air filter protects the servos from any impurities entering the system. The system pressures are maintained at 14 ± 0.5 psi for the rudder servo and 8 ± 0.5 psi for the elevator and aileron servos. Servo actuators, powered by the air pressure, move the aircraft control surfaces. Servos can be overpowered without damage to the unit.

B. Hawker Beechcraft New-Matic Autopilot

The Hawker Beechcraft New-matic autopilots operate on an electro-pneumatic concept. Electronic circuitry is employed for the navigational beam detection, magnetic heading direction and turns. Pressure servos are used for the control actuators. The system is completely non-tumbling. Yaw and roll detection is made by a tilted gyro mounted in the instrument panel. Turns or beam following is made by rotating a valve sleeve by a torquing movement proportional to the voltage imposed upon it. This unit also supplies an output voltage proportional to the turning rate that is used for dip compensation and nose-up signals during turns. The pitch control system does not use a gyro for reference, but uses the airspeed, rate of airspeed change and inertial signals to control the elevator through the pitch servos. Pitch trim employing the horizontal stabilizer must be employed manually with the trim control switches on the control wheels. An altitude hold sensing unit work in conjunction with the pitch control to sustain a given altitude. A series of manuals including maintenance, service, and installation are standard equipment for all aircraft in which the autopilots are installed. Information contained in these manuals should be referred to when service is required.

C. Sperry SPZ-200B Autopilot

The Sperry SPZ 200 B autopilot is a fully integrated flight control system consisting of a three axis autopilot and flight director system. The system provides a flight director display steering command which may be used during the fully coupled flight director/autopilot mode, or during manual flight. The fully independent autopilot may also be used without flight director information in the manual mode. The yaw axis may be engaged independently of roll and pitch for use as a yaw damper.

The aileron servo is mounted beneath the center floorboards aft of the main spar. A control cable runs from a capstan connected to the servo to either one or both sides of the airplane, depending on the system installed, and connected to the aileron control cables. When the autopilot is engaged, the servo rotates causing aileron deflection in a direction dependent on the direction of servo rotation.

The rudder servo is located in the back of the aft electronics bay. A servo control cable is wrapped around a capstan, which is connected to the servo, and is connected to either one or both rudder control cables. When the yaw axis is engaged, the servo rotates in response to the autopilot signals, causing rudder deflection in a direction of servo rotation.

The elevator servo is located in the back of the aft electronics bay close to the rudder servo. A servo control cable is wrapped around a capstan, which is connected to the servo, and is connected to either one or both elevator control cables. When the autopilot is engaged, the servo rotates causing elevator deflection in a direction of servo rotation.

NOTE: Improper servo cable tension may cause unstable or erratic response to autopilot signals.

AUTO FLIGHT
AUTOPILOT
TROUBLESHOOTING

1. AUTOPILOT SYSTEM

NOTE: This procedure applies specifically to the Hawker Beechcraft New-Matic B-7 autopilot, but in general, may also be used for the B-5 autopilot. Manuals noted in the Supplementary Publications list and the appropriate test sets as listed in those manuals will aid further in troubleshooting procedures.

A. Roll Axis

Troubleshooting - Autopilot

INDICATION	POSSIBLE CAUSE	REMARKS
1. Insufficient or excessive pressure indicated on aircraft system gage.	a. Leak in aircraft pressure system.	a. Check all lines and fittings for breaks, looseness, kinks, etc.
	b. Regulator valve improperly.	b. Adjust as outlined in Chapter 36 for pneumatic on Chapter 37 for vacuum.
	c. Regulator improperly adjusted.	c. Adjust as outlined in Chapter 36 for pneumatic or Chapter 37 for vacuum.
	d. Faulty pressure pump.	d. Replace pump.
	e. Ambient air filter clogged.	e. Clean or replace.
	f. Clogged system filter.	f. Check as outlined in Chapter 36 for pneumatic or Chapter 37 for vacuum and replace as necessary.
2. Aircraft hunts or recovers slowly from turn in one direction.	a. Regulator valve improperly adjusted.	a. Adjust as outlined in Chapter 36 for pneumatic or Chapter 37 for vacuum.
	b. Loose aircraft primary cables or excessive friction in and/or rudder cables, pulleys, bell cranks or loose servo cables.	b. Check security of attachment, binding, etc. and adjust as outlined in Chapter 27.
	c. Leak in servo or servo lines.	c. Check for leaks.
	d. Obstruction in servo lines	d. Check for foreign matter.
	e. Faulty turn coordinator gyro.	e. Replace turn coordinator.
3. Autopilot sluggish.	a. Low system pressure setting.	a. Check system filters and adjust as outlined in Chapter 36 for pneumatic on Chapter 37 for vacuum
4. Aircraft turns continuously on basic stabilization. (Controller OFF)	a. Aircraft out of trim or improperly rigged.	a. Trim aircraft or check controls for proper rig as outlined in Chapter 27.

Troubleshooting - Autopilot (Continued)

INDICATION	POSSIBLE CAUSE	REMARKS
	b. Loose primary cables or excessive friction in cables and system. Loose servo cable.	b. Check security of attachment, binding, etc., and adjustment as outlined in Chapter 27.
	c. Defective turn coordinator gyro.	c. Replace turn coordinator gyro.
	d. Leak in servo or servo line.	d. Check for servo or line leaks.
5. Aircraft rate of turn too fast or too slow.	a. Improper regulator adjustment.	a. Adjust regulator as outlined in Chapter 36 for pneumatic or Chapter 37 for vacuum.
	b. Turn coordinator faulty.	b. Replace turn coordinator.
6. Continuous control wheel oscillation in smooth air.	a. Turn coordinator faulty.	a. Replace turn coordinator.
	b. Improper gyro speed or excessive pressure in system.	b. Adjust system pressure as outlined in Chapter 36 for pneumatic or 37 for vacuum.
7. No turns or turns in one direction only, in response to turn control or on all modes of navigation coupler operation.	a. Faulty turn coordinator.	a. Replace turn coordinator.
	b. Faulty controller/amplifier.	b. Replace controller/amplifier.
8. Aircraft rolls in one direction only, either left or right.	a. Servos improperly phased.	a. Plumb as noted on Figure 205.
	b. Turn coordinator not plumbed properly.	b. Plumb as noted on Figure 205.
9. Aircraft turns in the wrong direction in CAP and TRK modes.	a. Nav input signal reversed.	a. Reverse connectors to VOR.
10. No aircraft response from navigation coupler in any mode, ground check shows electrical.	a. Faulty turn coordinator gyro.	a. Replace turn coordinator.
	b. Obstruction in pressure lines.	b. Check for foreign matter.
11. Aircraft fails to turn to and hold magnetic headings.	a. Faulty magnetic heading sensor.	a. Replace magnetic heading sensor.
	b. Faulty heading selector resolver.	b. Replace controller/amplifier.
	c. Faulty controller/amplifier.	c. Replace controller/amplifier.
12. Magnetic headings consistently high or low.	a. Heading sensor misaligned in aircraft.	a. Check for proper installation.
	b. Heading azimuth dial shifted on shaft.	b. Tighten screw and recalibrate.
	c. Improper adjustment of controller/ amplifier.	c. Calibrate for the magnetic cardinal points.

Troubleshooting - Autopilot (Continued)

INDICATION	POSSIBLE CAUSE	REMARKS
13. Cardinal headings inaccurate.	a. Controller/amplifier improperly adjusted.	a. Calibrate for the magnetic cardinal points.
	b. Leak in servo system.	b. Check for leaks.
	c. Low primary pressure.	c. Adjust system as outlined in Chapter 36 for pneumatic or Chapter 37 for vacuum.
14. Cardinal headings accurate but intermediate headings inaccurate.	a. Faulty heading sensor.	a. Replace the heading sensor.
	b. Faulty controller/amplifier.	b. Replace controller/amplifier.
15. Insufficient or no control in CAP and TRIC modes.	a. Faulty controller/amplifier.	a. Replace controller/amplifier.
	b. Faulty omni converter.	b. Replace omni converter.
	c. Insufficient signal from omni.	c. Repair or replace omni indicator.
	d. NAV SENS improperly adjusted.	d. Readjust.
16. Localizer approach is either sluggish or too sensitive.	a. Loc gain is set high or low.	a. Adjust localizer gain.
17. No electrical output left or right on controller/amplifier test jacks.	a. No A+ input or improperly grounded.	a. Check A+ and ground.
	b. Defective controller/amplifier or power supply.	b. Replace controller/amplifier or power supply.
18. Output only one way on controller/amplifier test jacks.	a. Defective controller/amplifier.	a. Replace controller/amplifier.
19. No output on HDG mode on controller/amplifier test.	a. Defective controller/amplifier, harness, or heading sensor.	a. Replace controller/amplifier, harness, or heading sensor.
20. Heading output on two reciprocal headings, but not on the other two.	a. Defective sensor, harness or faulty controller/amplifier.	a. Replace heading sensor, or check harness. Replace controller/amplifier.
21. "0" output when in CAP, TRK, and APP mode, with nav signal.	a. Defective nav switching console; or no nav information; or defective controller/amplifier.	a. Check nav input leads. Replace controller/amplifier.
22. Output voltages in CAP mode decays to "0" voltage.	a. Wrong nav input signals.	a. Check wiring.
	b. Defective switching console (if installed).	b. Repair or replace console.
	c. Dirty input signal (AC volts).	c. Check indicators.
23. Voltage output in MAN, CAP, TRK, and APP mode, but none in HDG mode.	a. Polarization pins reversed on.	a. Reverse pin (Ref. Figure 205) heading sensor plug.

Troubleshooting - Autopilot (Continued)

INDICATION	POSSIBLE CAUSE	REMARKS
24. Nav indicator needle deflects left or right when controller/ amplifier or radio is turned on.	a. One of the components is shorted to ground.	a. Check for shorts.
25. Low or high intercept angle.	a. Incorrect setting on controller/ amplifier.	a. Adjust intercept angle.
	b. Low or high voltage output on nav indicators.	b. Check nav indicators to manufacturer's specs.

B. Pitch Axis

Troubleshooting - Autopilot

INDICATION	POSSIBLE CAUSE	REMARKS
1. Pitch channel will not center up electrically.	a. Defective pitch/altitude sensor or amplifier.	a. Check on Test Set TS-108 or replace one at a time.
2. Altitude channel will not center up electrically.	a. Defective pitch/altitude sensor or amplifier.	a. Check on Test Set TS-108 or replace one at a time.
3. Altitude hold solenoid valve will not actuate.	a. Pressure switch on servo control valve out of circuit.	a. Check for faulty switch and replace if necessary.
	b. Defective solenoid valve.	b. Replace solenoid valve.
	c. Defective altitude switch on controller/amplifier.	c. Check continuity (Ref. Figure 205).
4. Servo control valve will not center.	a. Improper pressure adjustment.	a. Adjust as outlined in Chapter 36 for pneumatic or Chapter 37 for vacuum.
	b. Sticky valve.	b. Replace valve.
5. Output voltage is inadequate.	a. Pitch/altitude amplifier sensor or harness shorted or improperly wired.	a. Ref. Figure 205, run continuity check and check for shorts.
6. Pressure switch will not make contact when pressure is on.	a. Defective pressure switch or not set at proper pressure.	a. Replace pressure switch.
7. Output voltage one way only on pitch and altitude channels.	a. Servo control valve shorted to ground.	a. Replace valve.
8. System will maintain trimmed configuration even though centered electrically.	a. Servo control valve not pneumatically centered.	a. Disconnect electrical power. Center valve pneumatically by use of differential gage to $\pm$ 0.4 in. Hg.
	b. Leak in servos or improperly rigged.	b. Check for leaks and rig.
	c. Leak in pitch/altitude sensor.	c. Replace sensor.

Troubleshooting - Autopilot (Continued)

INDICATION	POSSIBLE CAUSE	REMARKS
9. System will not respond to airspeed changes.	a. Primary pressure not set properly.	a. Adjust as outlined in Chapter 36 for pneumatic or Chapter 37 for vacuum.
	b. Pitot pressure inadequate.	b. Check pitot plumbing.
	c. Decay rate improperly adjusted.	c. Adjust as required.
10. System will not respond to up command adjustment.	a. Defective pitch/altitude amplifier.	a. Replace pitch/altitude amplifier.
	b. No EVT potentiometer output.	b. Replace turn coordinator.
11. System will not respond to altitude gain adjustment.	a. Pitch/altitude amplifier limiter improperly set.	a. Adjust as required.
12. Aircraft has long term oscillation about pitch axis with altitude hold OFF.	a. Decay rate improperly adjusted.	a. Adjust as required.
	b. Pitch altitude gain improperly adjusted.	b. Adjust as required.
	c. Friction in elevator or servo system.	c. Check for friction and correct.
13. Aircraft has short term oscillation about pitch axis.	a. Decay rate too tight.	a. Adjust as required.
	b. Pitch gain too high.	b. Adjust as required.
	c. Primary pressure too high.	c. Readjust as outlined in Chapter 36 for pneumatic or Chapter 37 for vacuum.
14. Aircraft oscillates with altitude hold ON.	a. Altitude gain too high.	a. Adjust as required.
	b. Decay rate improperly adjusted.	b. Adjust as required.
15. Aircraft does not return to altitude when displaced.	a. Altitude hold solenoid inoperative.	a. Replace solenoid.
	b. Leak in altitude system.	b. Check for leaks.
	c. Altitude limiter improperly.	c. Adjust as required.
16. Aircraft descends or ascends continually when system engaged.	a. Servo control valve not phased correctly.	a. Apply positive 6.0 volts (max) to blue lead and verify nose up response.

AUTO FLIGHT
AUTOPILOT
MAINTENANCE PRACTICES

1. H-14 AUTOPILOT

NOTE: Before the autopilot control cables can be rigged, the respective airplane primary control cables must be rigged as specified in Chapter 27.

A. Rigging the Aileron Control System

(1) Place the aileron quadrant in the neutral position and install the rig pin.

(2) Set the aileron servo cable drum in the center of travel.

(3) Rig the aileron servo cables to the tension specified in Figure 202.

(4) Remove the aileron quadrant rig pin.

B. Rigging the Elevator Control System

(1) Place the elevator in the neutral position and install the five rig pins in the fore and aft bell cranks, aft torque shaft and each elevator horn.

(2) Set the elevator servo cable drum in the center of travel.

(3) Rig the elevator servo cables to the tension specified in Figure 202.

(4) Remove the rig pins.

C. Rigging the Rudder Control System

(1) Position the rudder pedals in the aft position and install the rig pin in the control lock holes in the pilot's rudder pedals.

(2) Set the rudder servo cable drum in the center of travel.

(3) Rig the rudder servo cables to the tension specified in Figure 202.

(4) Remove the rig pin.

D. Horizontal Stabilizer Control

The horizontal stabilizer is electrically actuated and controlled by a trim control box mounted adjacent to a computer in the aft fuselage area. The computer senses the desired trim position and actuates the stabilizer accordingly. The trim control box is connected to the stabilizer standby trim actuator and computer with electrical leads. A test switch is mounted on the pedestal placarded TRIM UP, TRIM DOWN for preflight of the trim control box.

2. MAINTENANCE AND REPAIR

Maintenance, repair, troubleshooting, and testing procedures are described in the H-14 Maintenance and Overhaul Manual (Refer to Supplementary Publications in Chapter 20). Located in the aft cabin area close to the rear bulkhead is an access door which provides access for adjustment of the autopilot.

NOTE: When test air pressure is required, remove the line between the bleed air connection port and the autopilot solenoid valve and apply 18 psi of air pressure to the inlet side of the solenoid valve.

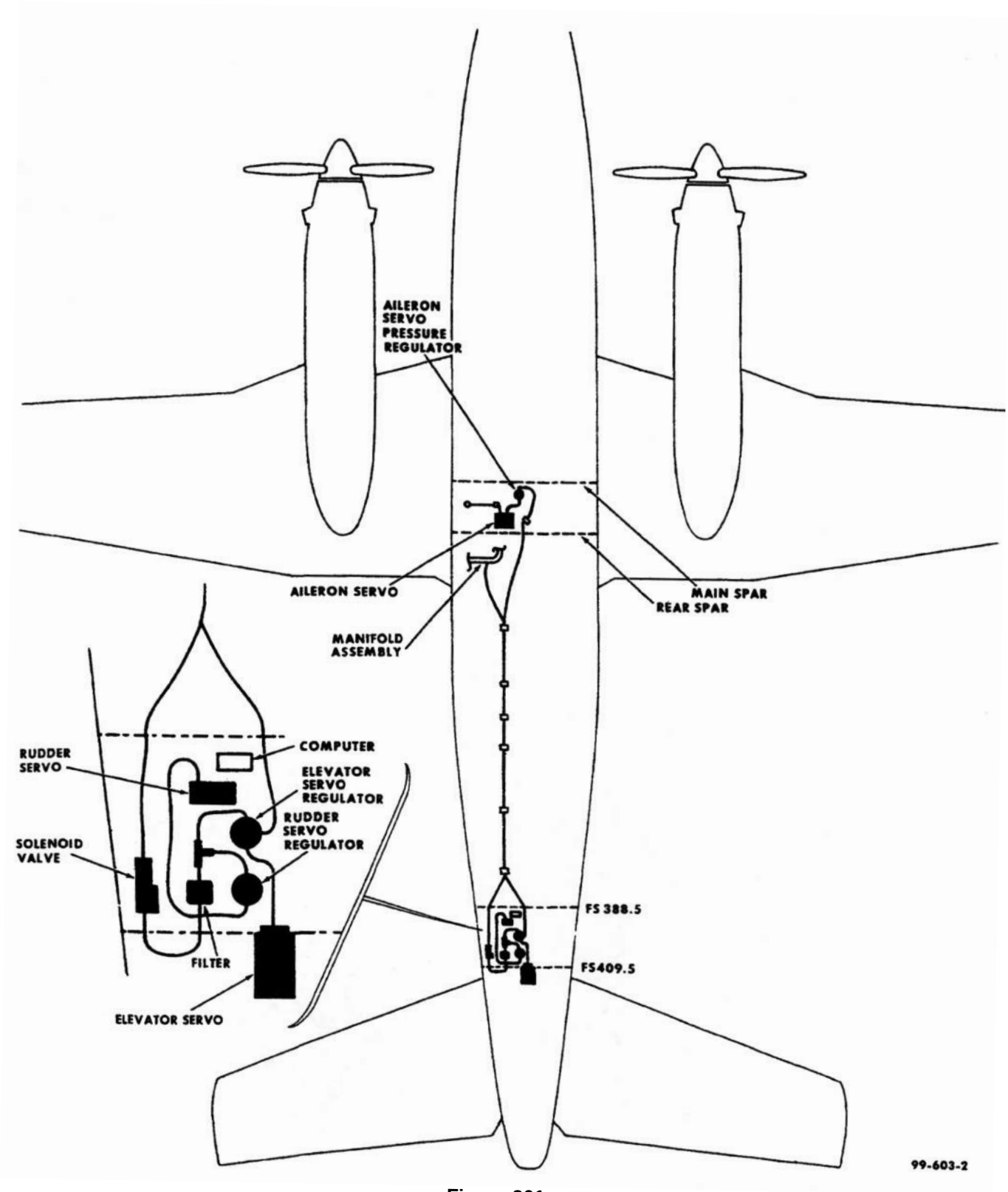

Figure 201
H-14 Autopilot System

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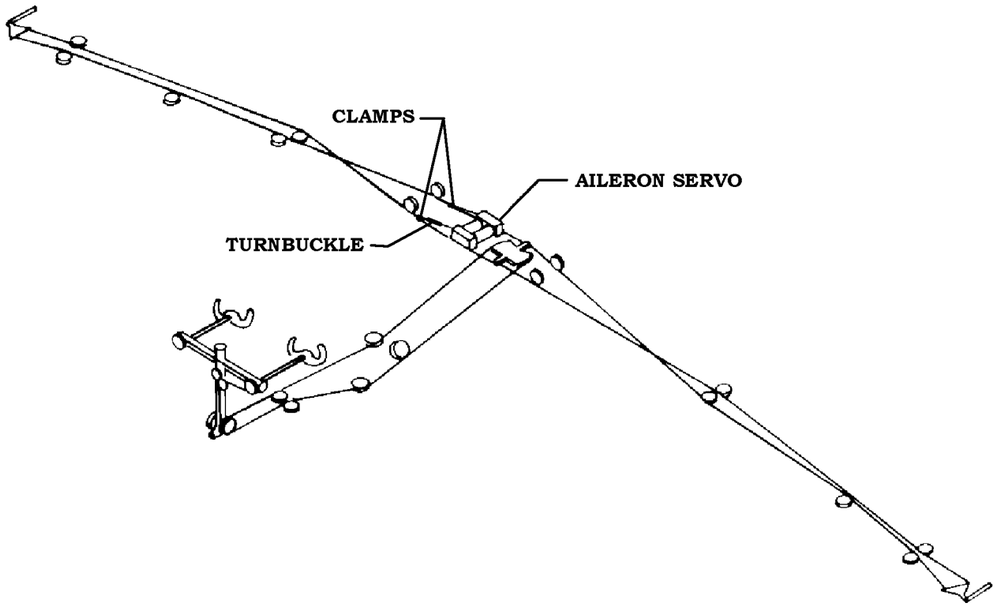

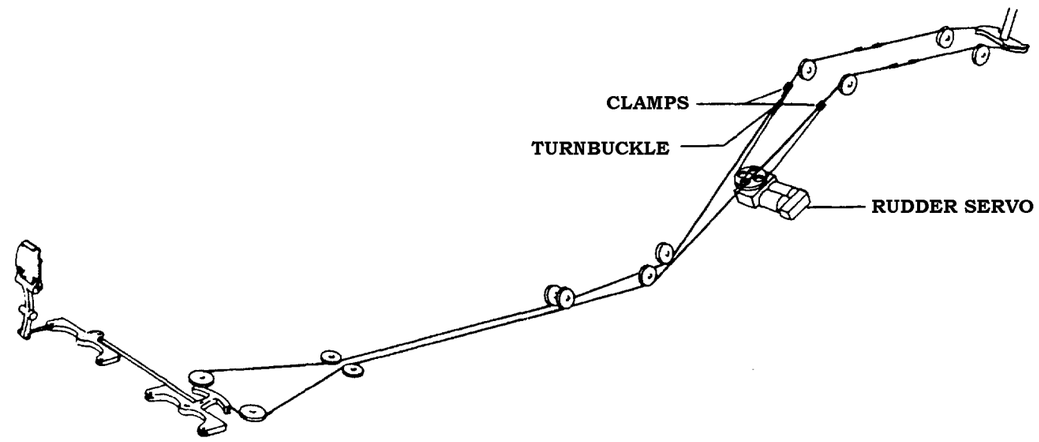

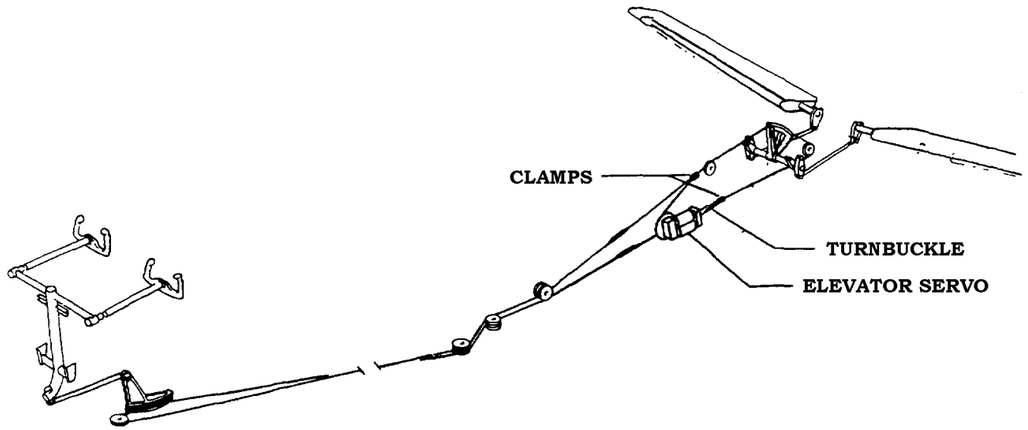

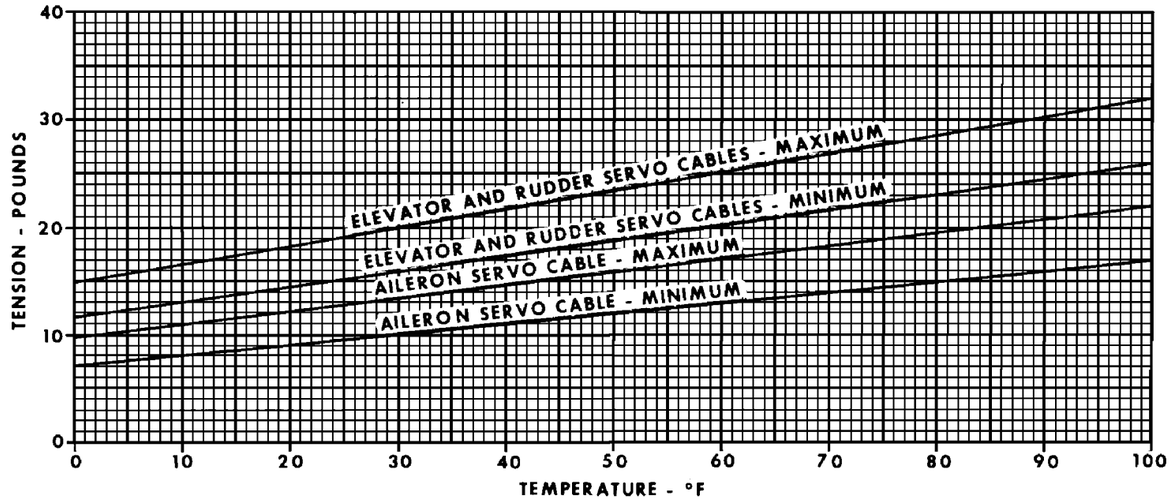

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Figure 202
H-14 Autopilot Cable Rigging and Tension Graph

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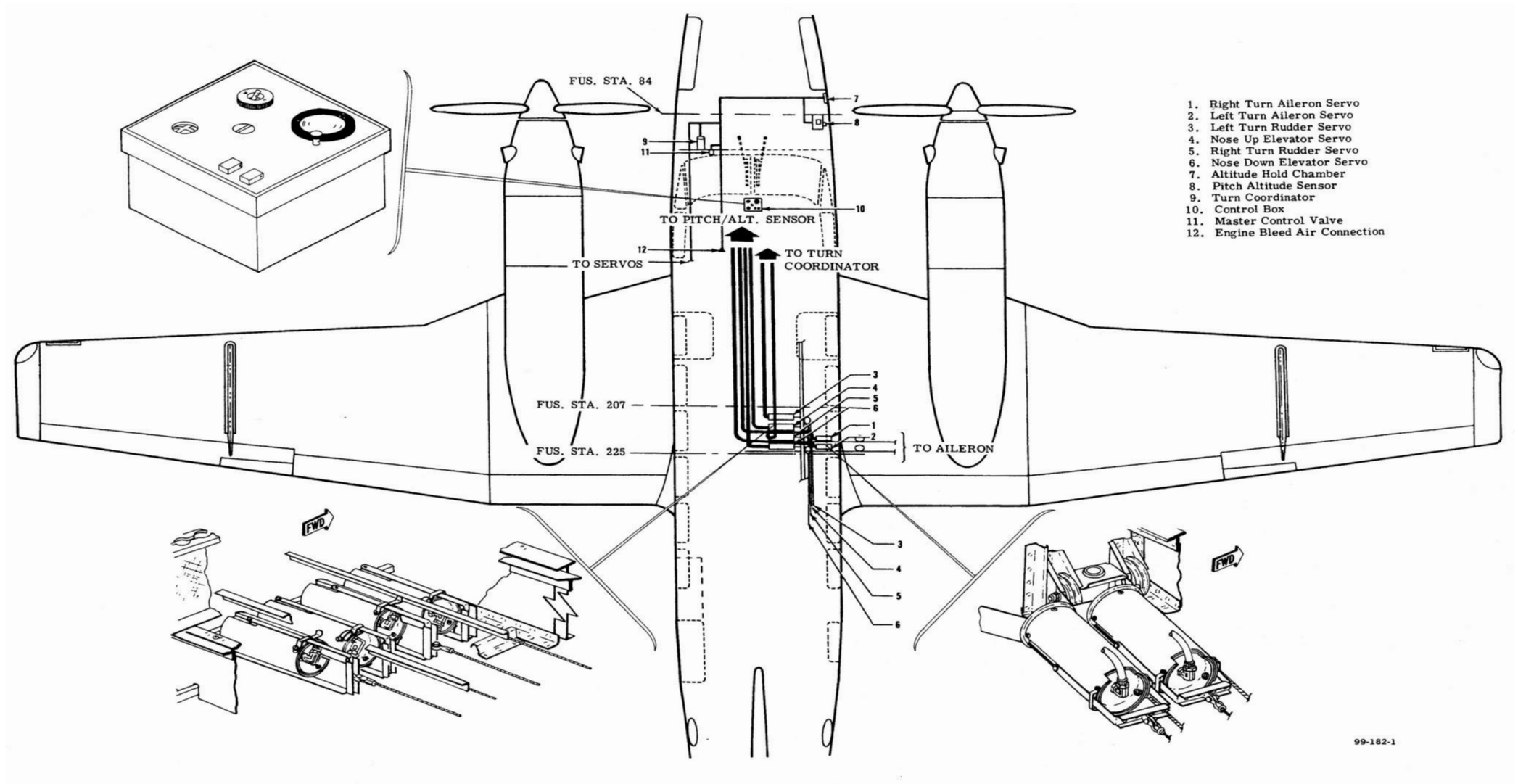

Figure 203
B5P Autopilot Installation

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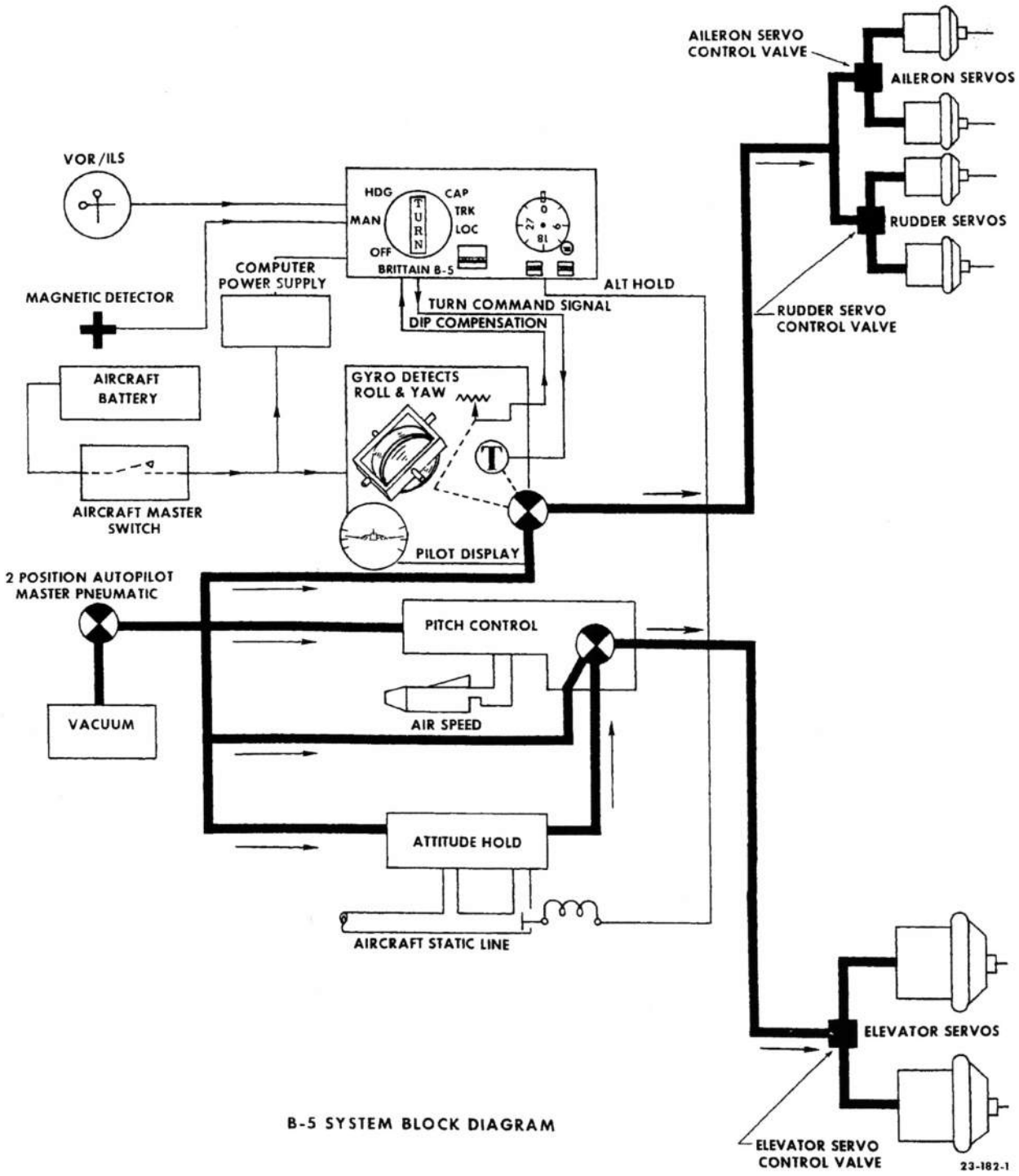

Figure 204
Hawker Beechcraft New-Matic B-5 Autopilot System

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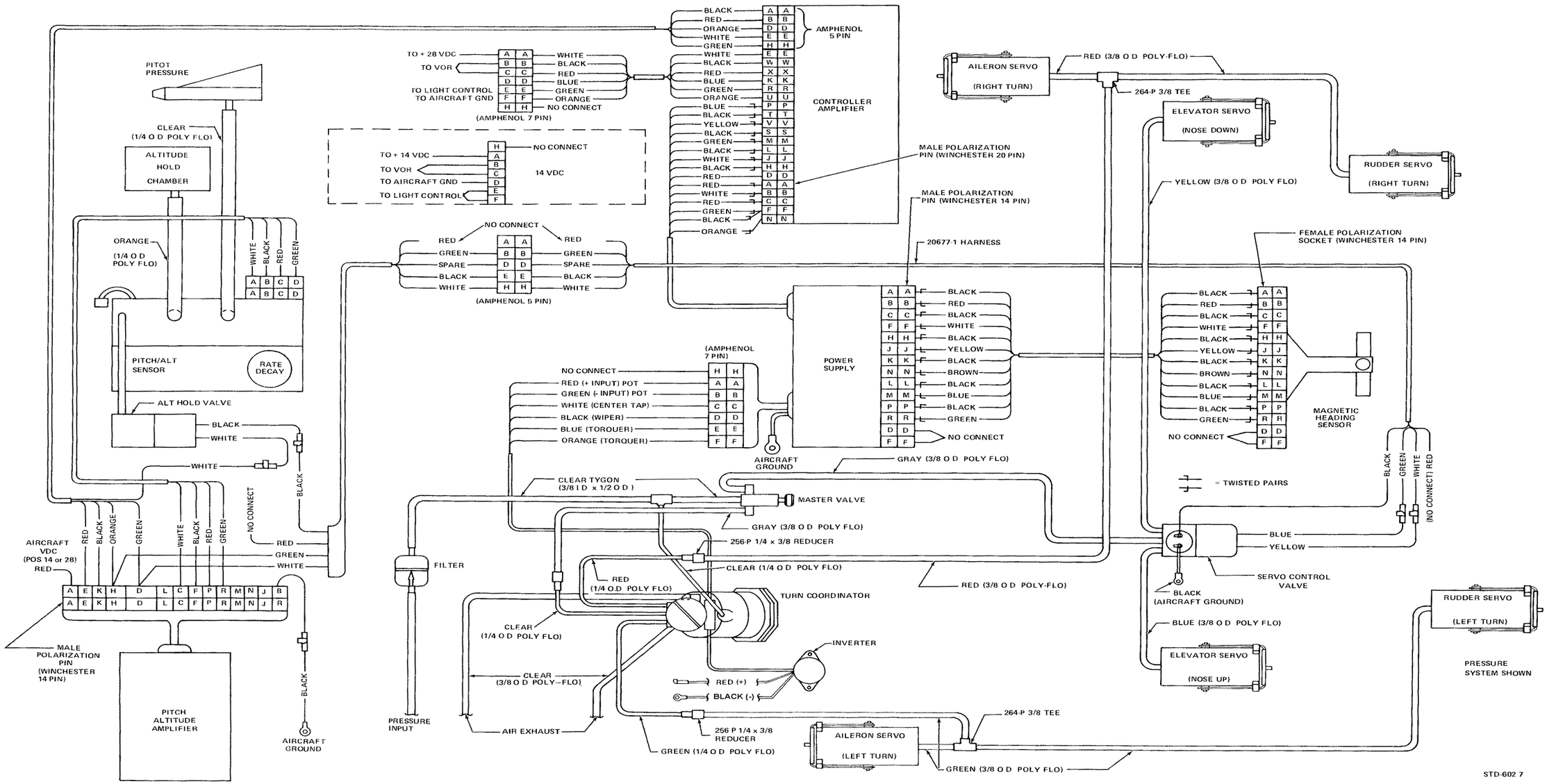

Figure 205
Hawker Beechcraft New-Matic B-7 System Block Diagram

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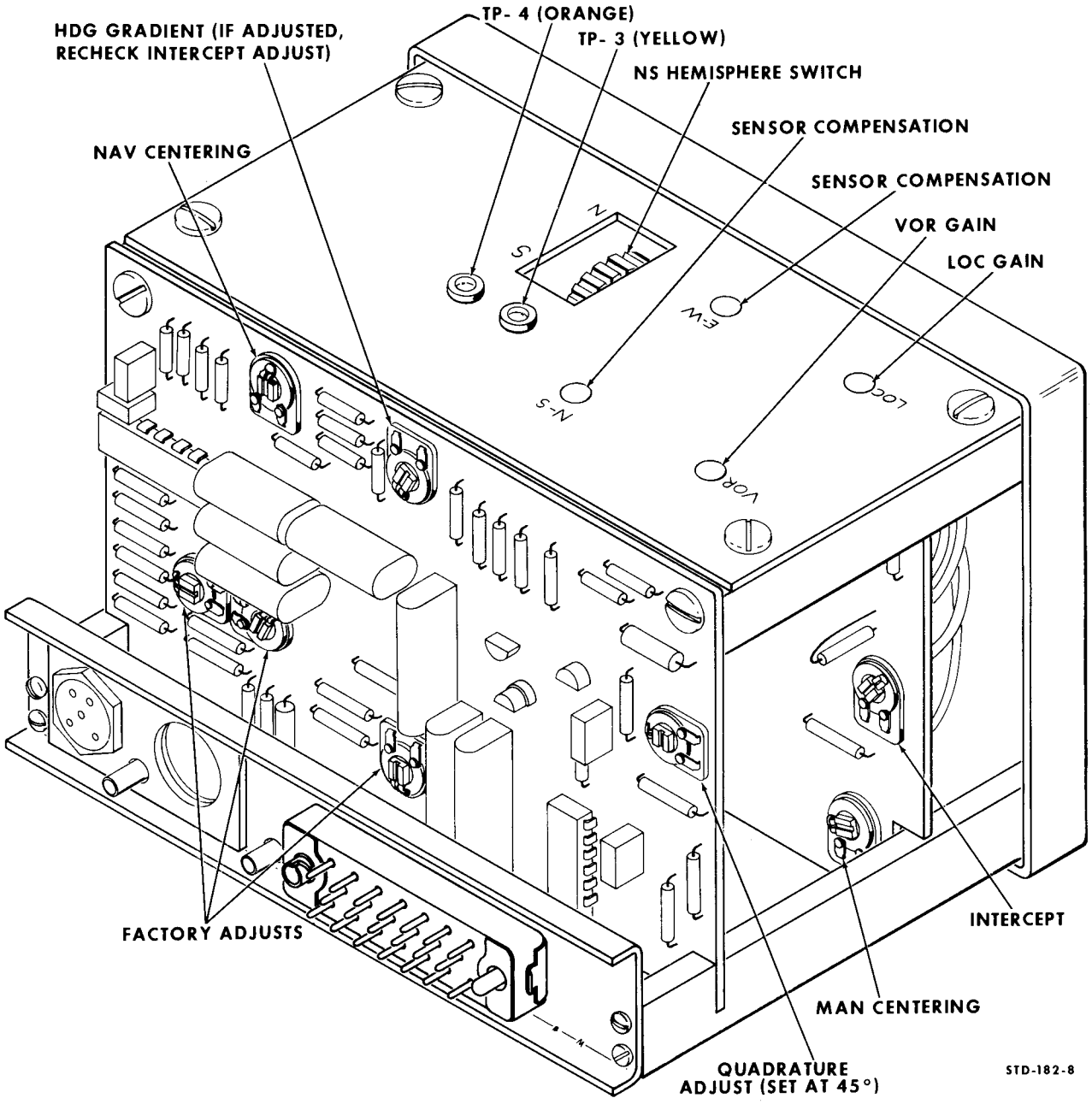

Figure 206
Hawker Beechcraft New-Matic B-5 and B-7
Heading Lock/Navigation Coupler System Adjustment Points

3. SPERRY SPZ-200B AUTOPILOT

Refer to SPZ-200B Flight Control System Maintenance Manuals. Sperry-Rand Flight Systems, P.O. Box 21111, Phoenix, Arizona 85064.

A. Rigging

NOTE: Before the autopilot control cables can be rigged, the respective flight control cables must be rigged as specified in Chapter 27. The tension is to be set with the applicable control in neutral position. After the tension has been set, work the applicable control surface and check the tension and reset if necessary. The servo cable tension can then be set to the values as specified in the applicable cable tension graph, Figure 207.

(1) Rig the aileron servo cables as follows:

(a) Place the aileron quadrant in the neutral position and install the rig pin.

(b) Set the aileron servo cable capstan in the center of travel.

(c) Set the aileron servo cables to the tension specified in the applicable cable tension graph, Figure 207.

(d) Remove the aileron quadrant rig pin.

(2) Rig the elevator servo cables as follows:

(a) Place the elevator in the neutral position and install the five rig pins in the fore and aft bellcranks, aft torque shaft and each elevator horn.

(b) Set the elevator servo cable capstan in the center of travel.

(c) Set the elevator servo cables to the tension specified in the applicable cable tension graph, Figure 207.

(d) Remove the rig pins.

(3) The horizontal stabilizer is electrically actuated and controlled through a trim control box mounted adjacent to the computer in the aft fuselage area. The computer senses the desired trim position and actuates the stabilizer accordingly. The trim control box is connected to the stabilizer standby trim actuator and computer with electrical leads. A test switch, placarded TRIM UP. TRIM DOWN, is mounted on the pedestal for preflight testing of the trim control box.

(4) Rig the rudder servo cables as follows:

(a) Position the rudder pedals in the aft position and install the rig pin in the control lock holes in the pilot's rudder pedals.

(b) Set the rudder servo cable capstan in the center of travel.

(c) Set the rudder servo cables to the tension specified in the applicable cable tension graph, Figure 207.

(d) Remove the rig pin.

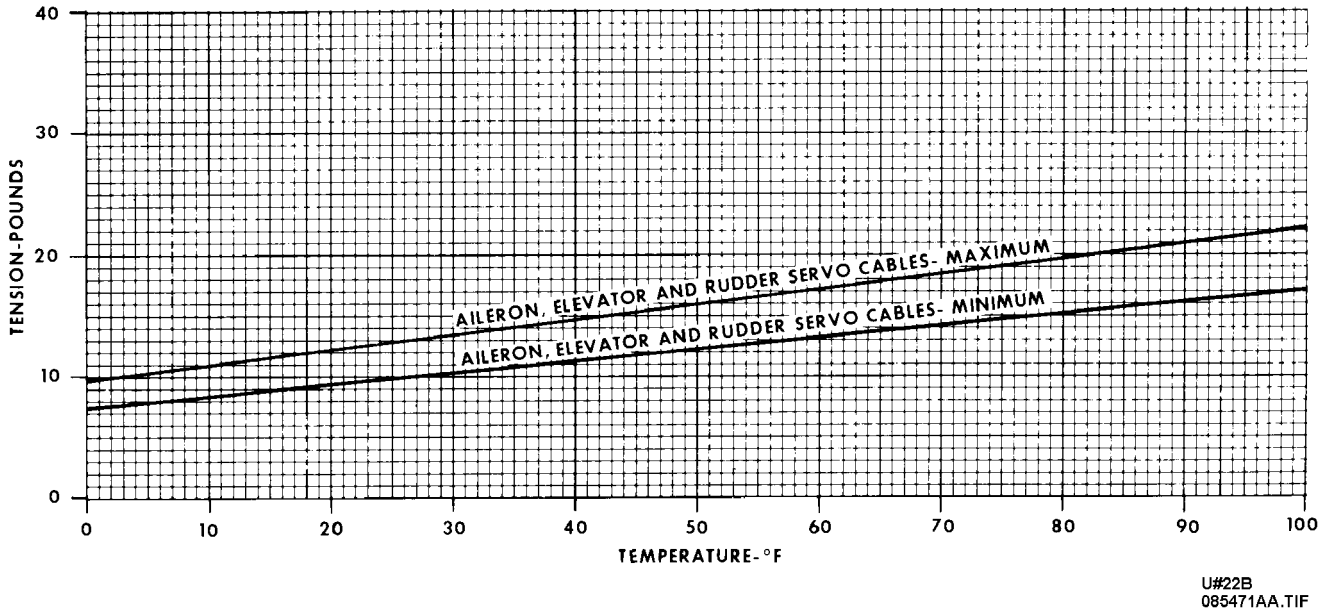

Figure 207
SPZ-200B Autopilot

4. AILERON SERVO CABLE

A. Removal

(1) Prepare the airplane for cable rigging as noted in AILERON CONTROL SYSTEM RIGGING, Chapter 27-10-00.

(2) Remove the clamps and bolts securing the cables to the forward aileron control cable.

(3) Unwind and remove the servo cable from the capstan.

B. Installation

(1) Prepare the airplane for cable rigging as noted in AILERON CONTROL SYSTEM RIGGING, Chapter 27-10-00.

(2) Attach the cable to the capstan.

(3) Wrap the cable around the capstan until it is full.

(4) Attach the servo cable to the forward aileron cable, using the attachment clamps and bolts.

(5) Rig the servo cables as noted in AUTOPILOT CONTROL SYSTEM RIGGING.

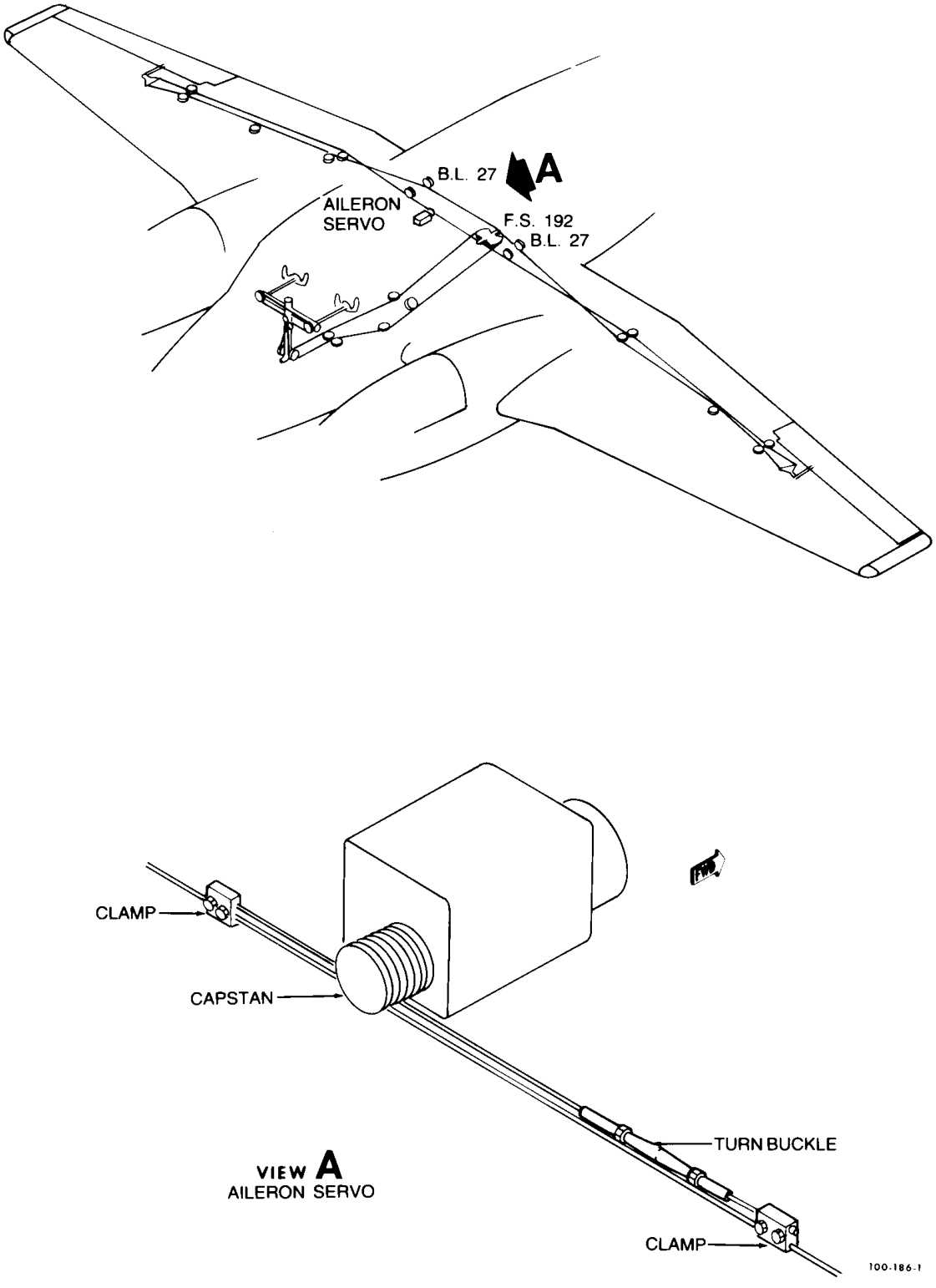

Figure 208
SPZ-200B Aileron Servo and Cable Installation

5. ELEVATOR SERVO CABLE

A. Removal

(1) Prepare the airplane for cable rigging as noted in ELEVATOR CONTROL SYSTEM RIGGING, Chapter 27-30-00.

(2) Remove the bolts securing the servo cable ends to the attachment links on the elevator cables.

(3) Unwind and remove the servo cables from the capstan.

B. Installation

(1) Prepare the airplane for cable rigging as noted in ELEVATOR CONTROL SYSTEM RIGGING, Chapter 27-30-00.

(2) Attach the servo cable to the capstan.

(3) Wrap the cable with the turnbuckle approximately 1 1/2 turns in a clockwise direction around the capstan.

(4) Wrap the other cable approximately 1 3/4 turns counterclockwise around the capstan.

(5) Pass the turnbuckle end of the servo cable around the pulley and attach it to the down elevator cable with the attaching link and bolt.

(6) Pass the other end of the servo cable around the pulley and attach it to the up elevator cable with the attaching link and bolt.

(7) Rig the servo cables as noted in AUTOPILOT CONTROL SYSTEM RIGGING.

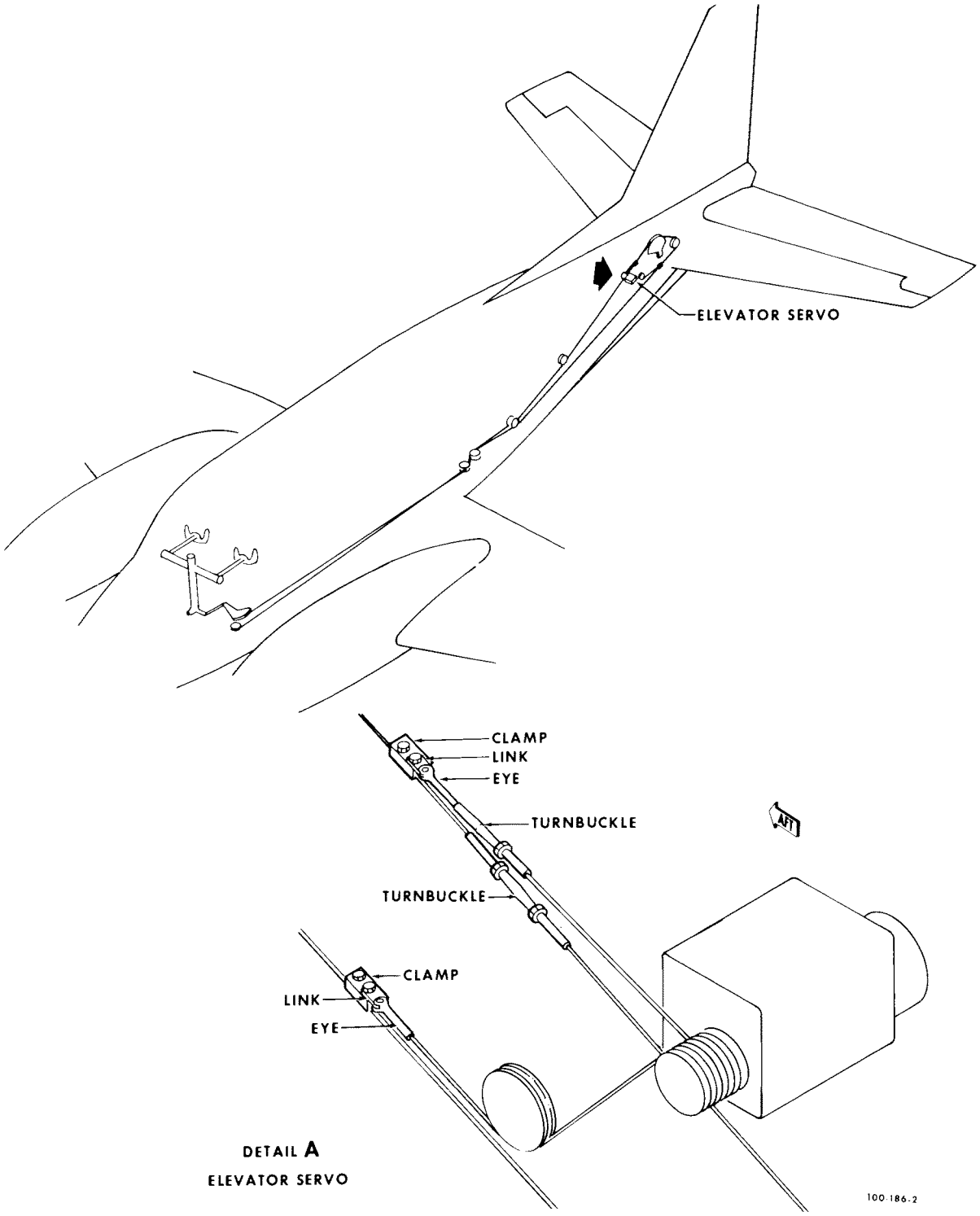

Figure 209
SPZ-200B Elevator Servo and Cable Installation

6. RUDDER SERVO CABLE

A. Removal

(1) Prepare the airplane for cable rigging as noted in RUDDER RIGGING, Chapter 27-20-00.

(2) Disconnect the servo cables from the left rudder cable by removing the bolts from the attaching links.

(3) Unwind the servo cable and remove it from the capstan.

B. Installation

(1) Prepare the airplane for cable rigging as noted in RUDDER RIGGING, Chapter 27-20-00.

(2) Attach the servo cables to the capstan.

(3) Wrap the servo cable with the turnbuckle around the capstan approximately 1/2 turn in the counterclockwise direction.

(4) Wrap the other servo cable around the capstan approximately 1 1/2 turns in a clockwise direction.

(5) Connect the servo cable with the turnbuckle to the left aft rudder cable with the attaching link and bolt.

(6) Pass the other servo cable around the pulley and connect it to the left forward rudder cable with the attaching link and bolt.

(7) Rig the servo cables as noted in AUTOPILOT CONTROL SYSTEM RIGGING.

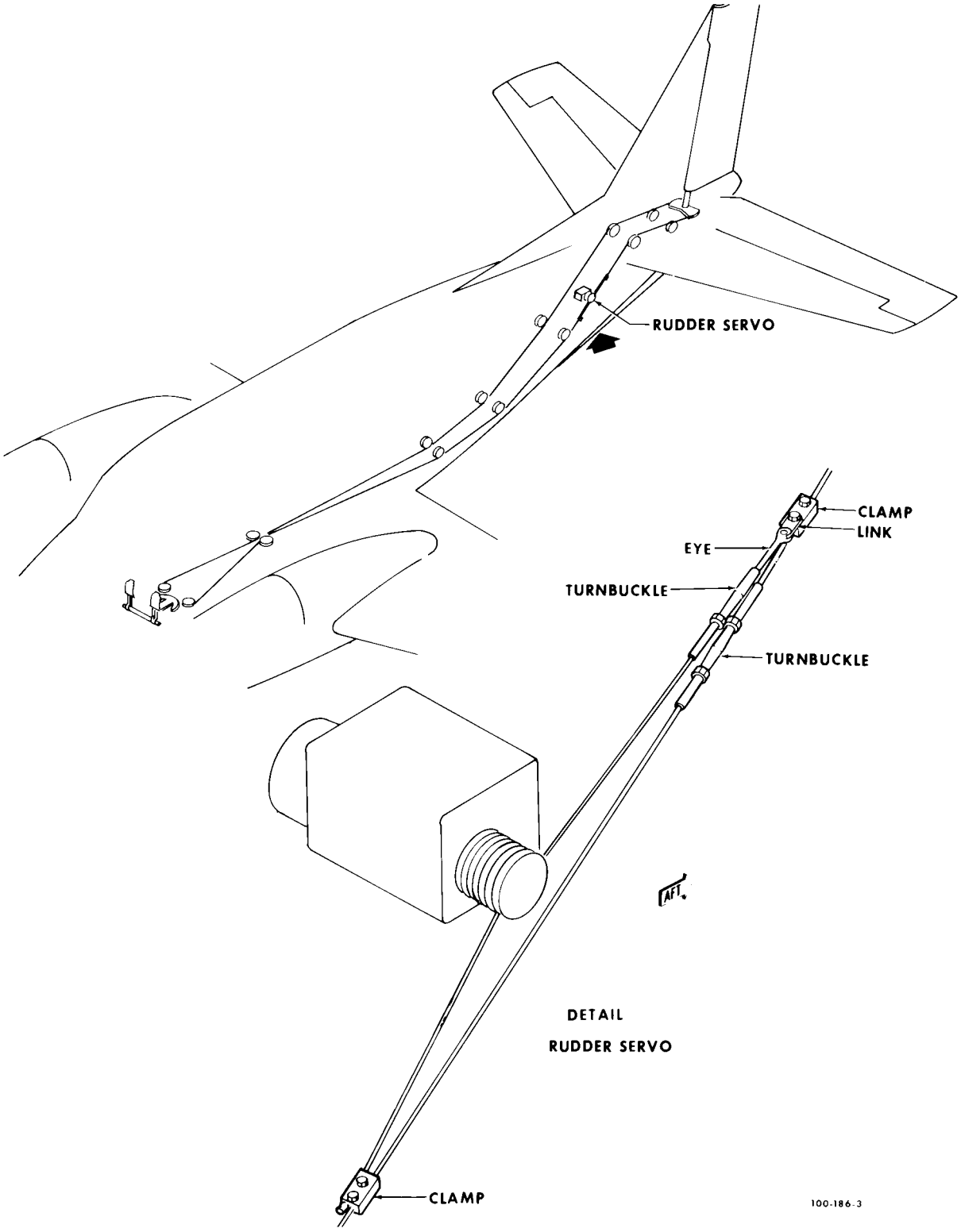

Figure 210
SPZ-200B Rudder Servo and Cable Installation

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CHAPTER 23 - COMMUNICATIONS

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23-60-00

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General 101
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COMMUNICATIONS
STATIC DISCHARGING
DESCRIPTION AND OPERATION

1. GENERAL

A static electrical charge may build up on the surface of the airplane while it is in flight. This electrical charge, if retained, can cause electrical interference in radio and avionics equipment operation it is also dangerous to personnel disembarking from the airplane after landing and to personnel servicing the airplane. To dissipate this electrical build up, static wicks are installed on the airplane, three on the aileron, rudder and elevator, two on the wing tip and one on the ventral fin.

COMMUNICATIONS
STATIC DISCHARGING
TROUBLESHOOTING

1. GENERAL

NOTE

CAUTION: *If precipitation static effects are evident or suspected or if a lightning strike has occurred perform the INSPECTION OF STATIC WICKS procedure in the Maintenance Practices section.*

NOTE: Check the static wicks using a megohmmeter with a minimum test voltage of 500 volts to measure resistance of the static discharger wicks. Test voltages less than 500 volts will not generate sufficient leakage rates to provide an accurate reading.

To ensure the best possible function of the static dischargers, the resistance between the base of the static dischargers and the surface to which each static discharger is attached must be less than 0.0100 ohm, and the resistance between the control surfaces on which static dischargers are installed and the airplane structure must be less than 0.0025 ohm.

All static dischargers on the airplane MUST be of the same type and of the same vendor.

COMMUNICATIONS
STATIC DISCHARGING
MAINTENANCE PRACTICES

1. STATIC WICK

Static wicks, three each on the elevator, ailerons and rudder, two each on the wing tips and one on the ventral fin, are removed and installed in the same manner.

A. Removal

(1) Remove the two attaching rivets.

(2) Remove the static wick from the airplane.

B. Installation

(1) Clean around the static wick area by removing all anodic film, grease, oil, paints, lacquer, metal finishes or other high resistance properties with 3M sandpaper, 600 grit or equivalent, and solvent (41, Table 1, 91-00-00). The mating surfaces must be smooth and contoured so that maximum surface area is in actual contact.

NOTE: Acceptable substitutes for the preceding may be used in accordance with MIL-B-5087B or the surface may be prepared in accordance with MIL-M-3171C.

Dissimilar materials are not to be used in intimate contact unless suitably protected against electrolytic corrosion. Whenever it is necessary that any combination of such metals be assembled, an interposing material compatible to each should be used.

(2) Install the static wick on the airplane using two rivets.

(3) Refinish the surface area around the wick attachment point with the original finish or a clear lacquer conforming to MIL-L-6806.

C. Inspection

The static wicks should be inspected every 300 hours to ensure proper discharging capabilities.

NOTE: Check the static wicks using a megohmmeter with a minimum test voltage of 500 volts to measure resistance of the static discharger wicks. Test voltages less than 500 volts will not generate sufficient leakage rates to provide an accurate reading.

To ensure the best possible function of the static dischargers, the resistance between the base of the static dischargers and the surface to which each static discharger is attached must be less than 0.0100 ohm, and the resistance between the control surfaces on which static dischargers are installed and the airplane structure must be less than 0.0025 ohm.

All static dischargers on the airplane MUST be of the same type and of the same vendor.

Check the static wicks using the following procedure:

CAUTION: *Electrical shock can result if the megohmmeter is improperly used. Refer to the applicable manufacturers operating handbook for proper operating procedures.*

(1) Connect a test lead from the megohmmeter to the tip of the static wick.

(2) Connect the other test lead from the megohmmeter to the base of the static wick.

(3) Place the megohmmeter power switch to ON and observe the meter indication. A reading of between 1 to 100 megohms indicates the static wick is satisfactory. A reading of less than 1 or in excess of 100 megohms indicates the static wick is unsatisfactory and must be replaced.

NOTE: If optional rigid yellow Dayton-Granger static dischargers (P/N 16305 or P/N 16307) are used, they should have a resistance reading from the tip to its base, in the range of 8 to 200 megohms.

If static wicks (P/N 35-5010-3) are installed, they should have a resistance reading from the tip to its base in the range of 470 ± 20% kilo-ohms.

(4) Place the megohmmeter power switch to OFF and disconnect the test lead from the static wick tip and its base.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 24 - ELECTRICAL POWER

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ELECTRICAL POWER
GENERAL
DESCRIPTION AND OPERATION

1. ELECTRICAL POWER

NOTE: The information contained in this chapter is effective for the C99 airplanes (serials U-50, U-165 and After) ONLY. The wiring diagrams pertinent to the C99 are found in the Model C99 Wiring Diagram Manual P/N 99-590030-5. Refer to the Model 99 Airliner Series Wiring Diagram and Electrical System Manual P/N 99-590028A, for maintenance and wiring diagram information for prior airplane serials.

Direct current for the electrical systems of the Model 99 Airliner Series airplanes is supplied by a 24 volt 34 ampere-hour battery and two 30 volt, 250 ampere starter-generators. Alternating current for certain engine instruments and for avionics is supplied by either one of two inverters.

Electrical system repair methods used on the Model 99 Airliner Series airplanes must be made in accordance with the Federal Aviation Administration's Aircraft Inspection and Repair Manual AC43.13-1A and with the Aircraft Alterations Manual AC43.13-2. Any components replaced and any wire, cable, or terminals used in the maintenance of the electrical system must be of airplane quality. Any solder connections must be made in an approved manner. Any solderless terminals or splices used must be applied with tooling specified by the supplier.

A. AC Generation

The AC power for the avionic equipment and the ac powered engine instruments is supplied by either one of two inverters. The inverters are installed in the wing center section outboard of each nacelle. The inverter operation is controlled by an inverter select switch on the left subpanel. Selection of either inverter actuates the inverter power relay installed near that inverter to supply the inverter with DC power. Inverter select relays provide the necessary switching to permit the operating inverter to supply the 26 vac instrument power and the 115 vac power for the test jack and for the inverter warning relay. A voltage/frequency meter is installed to monitor the inverter output. The meter is located below and to the left of the standard turn and slip indicator. The meter monitors frequency, except when the button in the lower left corner of the meter is pressed it monitors voltage.

The inverter select relays are energized when the number one inverter is selected. The inverter warning relay is energized to prevent the illumination of the inverter warning annunciator when the selected inverter is supplying 115 vac.

The inverter select relays and the inverter warning relay are located on the forward right equipment panel. Pictorial coverage of the installation panels is provided in Chapter 39.

The inverter system described here is the standard installation (Ref. Figure 1). The circuit diagram in the Wring Diagram Manual provides the circuit routing of the DC and AC power for the standard airplane instrumentation.

B. DC Generation

A 24-volt battery and two 30-volt starter-generators connected in parallel provide the airplane’s electrical power. The battery is located in the right wing center section forward of the main spar. The battery is air cooled.

Battery power is routed through a battery relay to a pair of starter relays and to the external power bus. The starter relays are located in their respective wing center section inboard of the engine nacelles. The battery relay is mounted in the battery box. The battery power is then routed to the battery bus located on the DC Power Distribution Panel Assembly located below the cabin center aisle floor forward of the main spar.

DC power from the generators is routed through the generator line contactors that are controlled by the generator control panels to provide the reverse current protection. The generator line contactors and the loadmeter shunts are mounted on the DC Power Distribution Panel Assembly.

C. Starter and Ignition

The starter and ignition circuits provide power to motor the engine and for ignition during engine starts. By selection of the STARTER ONLY mode, the starter can be used for motoring the engine to clear the engine of excess fuel.

At any time the starter/generator is being used for motive power, the starter relay is energized and a fuel purge solenoid valve is energized open to allow fuel to pass from the fuel control back into the tanks. This relieves fuel pump pressure and purges air from the fuel control unit.

An annunciator readout is illuminated to indicate when power is applied to the ignitors. More detailed information on the ignition circuits will be found in Chapter 71.

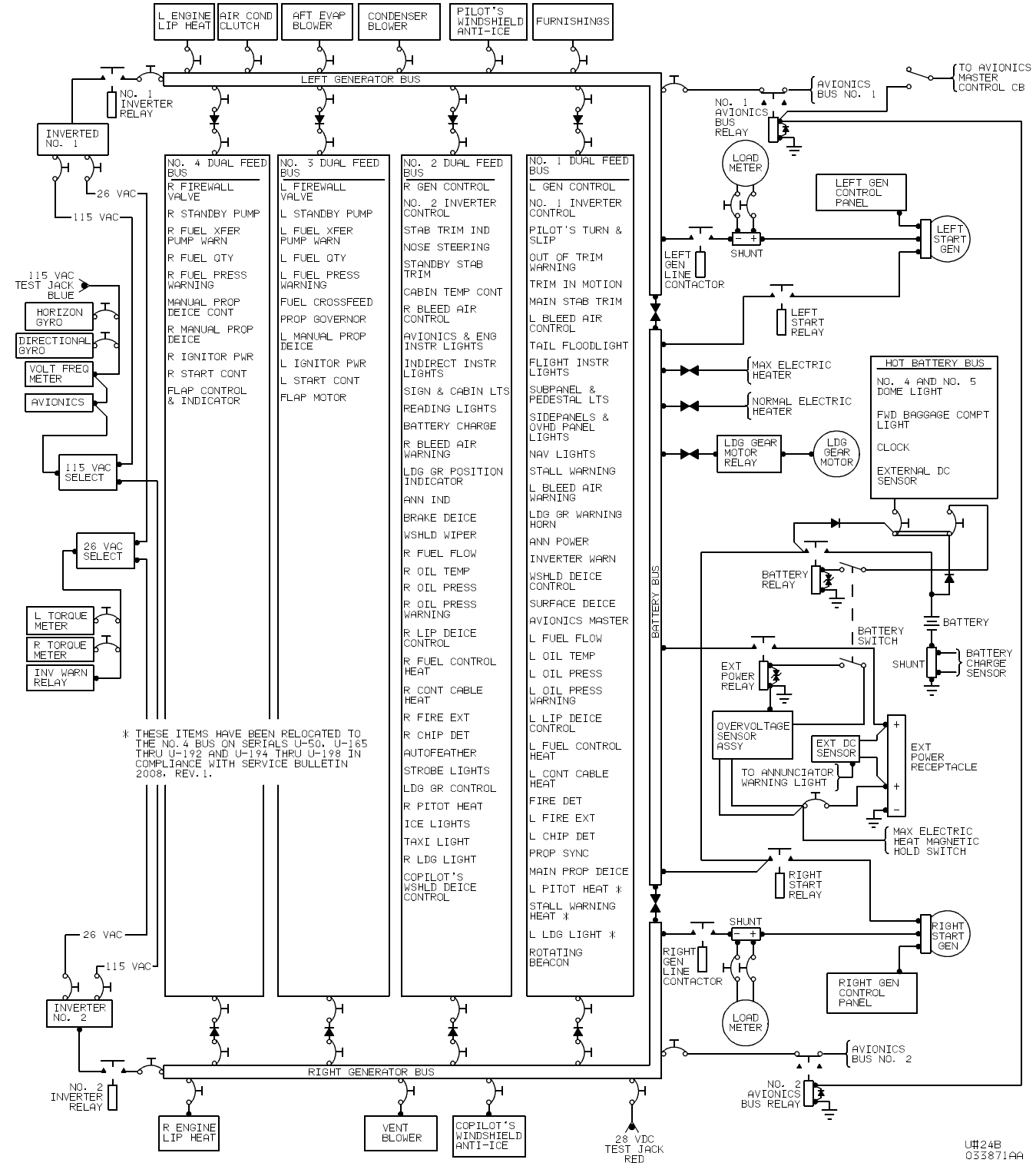

Figure 1
Electrical Power Distribution Schematic

ELECTRICAL POWER
DC GENERATION
DESCRIPTION AND OPERATION

1. GENERAL

The generator phase of operation is controlled by the generator switches located under the master switches gang bar with the battery switch. Any time the start switch is closed, the generator control relay is energized and opens the shunt field of the generator. If the generator is taken off the line while the generator is running, the generator control switch must be placed in RESET in order to bring the generator back on the line.

The generator system consists of DC starter-generators, line contactors, generator control panels, generator control relays, field grounding relays, control switches, a voltmeter, load meters and annunciator lights. Proper operation of each of these components is essential to obtain satisfactory generator operation. Many different component failures can result in similar generator system responses, making it virtually impossible to isolate a faulty component merely by analyzing the system response.

The generator provides electrical power to recharge the airplane battery and operate the airplane electrical loads. These loads require a constant voltage source for proper operation. The generator output is maintained at a constant level by controlling the shunt field excitation. The interpole and compensating windings of the generator are in series with the armature and provide a voltage proportional to the generator current.

Generator operation is controlled by generator control panels mounted beneath the center aisle floor forward of the main spar. The generator control panels provide voltage regulation, generator load sharing, differential voltage and reverse current sensing and control, overvoltage and overexcitation protection and the field flash circuit.

A. Voltage Regulation

The generator output voltage is sensed at the generator side of the generator line contactor. The control panel supplies the generator field excitation current required to supply the electrical load and maintain a bus voltage of 28.25 $\pm$ 0.25 vdc.

B. Generator Control

When the generator switch is turned off, the power to the line contactor coil is interrupted, thus removing the generator from the bus. At the same time, generator voltage is applied to the overvoltage test circuit of the control panel to trip the voltage circuit, de-exciting the generator.

The equalization sensing and the generator interpole and compensating windings are grounded when the generator is switched OFF, provided the respective starter circuit is not in operation. The grounding is provided when the field grounding relay is energized. Battery power is then routed from the generator control circuit breaker through the generator control switch and the normally closed contacts of the generator control relay to the coil of the field grounding relay.

During the start of an engine, the generator control relay is energized, thus interrupting the coil circuit of the field grounding relay.

The equalization of the generators is accomplished by utilizing the voltage developed across the generator interpole and compensating windings. The paralleling circuit includes the LH and RH generator control panels. Equal load sharing is dependent upon equal resistance of the generator

windings and of the external circuitry. The generator control panels are designed to control the generators and the load shared within 10% of the rating of one generator for total loads greater than 25%.

C. Reverse Current Protection and Differential Voltage Control

The reverse current protection and differential voltage control are provided by the generator control panels. Bus voltage is sensed at the bus side of the line contactor. The generator output voltage is sensed at the generator side of the line contactor. Whenever the generator is operating and the control switch is placed in the RESET position, the generator output voltage will rise to the regulated voltage. When the generator switch is placed in the ON position, a voltage output from the control panel will close the line contactor to connect the generator to the bus, if the generator voltage is greater than bus voltage or within 0.3 volts below bus voltage.

When the generator field becomes underexcited for any reason, or when the generator slows down to the point where it can no longer maintain a positive load, the generator will begin to draw current from the airplane bus. This current, termed "reverse current", passes through the interpole and compensating windings of the generator. The voltage developed by this current is sensed by the generator control panel. When the voltage exceeds 0.2 volts, the generator control panel removes the voltage from the coil of the bus contactor, permitting the contactor to open, removing the generator from the airplane bus.

D. Overvoltage Protection

Overvoltage protection is provided by the generator control panel In the event the output voltage reaches 32.5 vdc, the overvoltage protection portion of the generator control panel will trip an internal field relay and open the coil circuit of the bus contactor. The overvoltage generator is thus de-excited and isolated from the airplane bus.

Should a circuit fault occur which would supply generator output or bus voltage to the generator field, or should the voltage regulation of one of the control panels fail, the affected generator would attempt to assume the full electrical load. If the bus voltage were to rise above 28.25 vdc, reverse current would begin to flow in the normally regulated generator and it would be removed, temporarily, from the bus. Should the resultant voltage increase above 32.5 vdc in the affected generator, it will be removed from the bus also and the non-affected generator can be reconnected to the bus. The resultant bus voltage would depend upon the generator speed, the electrical load and the nature of the fault.

E. Overexcitation Protection

Overexcitation protection is provided by the generator control panel. This portion of the control panel will activate in the event the generator load and speed conditions are such, or the nature of the fault is such that the generator voltage starts to increase without control, but does not go into overvoltage. Should the generator load unbalance reach the designed limitation value, the circuitry will be activated to remove the affected generator from the bus.

F. Field Flash Circuit

When the generator switch is placed in RESET, the generator residual voltage is applied through a low resistance circuit, by-passing the regulator, until the generator voltage builds up high enough for the voltage regulator to effectively control the generator. Any time the generator control panel has been tripped for overvoltage, or the generator has a low residual voltage, RESET must be used in order to bring the generator on the line.

ELECTRICAL POWER
DC GENERATION
TROUBLESHOOTING

1. STARTER/GENERATOR AND CONTROLS

The generator control switch directs battery voltage and generator voltage to the control panel for OFF, RESET and ON conditions. A voltmeter and the loadmeter are connected on the generator side of the line contactors to provide a cockpit indication of generator voltage and load as well as ongoing system performance information.

The GEN OUT lights in the annunciator are connected to the line contactor coil and provide an indication of whether or not the generators are on the line. It is possible to troubleshoot this system by monitoring each of the control panel inputs for proper results during various modes of operation of a normal system.

The Table (Chart 101) contains detailed information on continuity and voltage measurements that may be obtained at the generator control panel of a normal system. The continuity checks should be performed with the control panel disconnected, the battery off, external power off and the engines off.

The resistance values shown in the Table are approximate values. It is not expected that these exact values will be obtained during checkout, they should be used as a guide to locate open and short circuits in the airplane wiring.

Usually, it should not be necessary to check each of the inputs given in the table to isolate a defective component. Checking the ground wires and one or two specific circuits is normally sufficient. The troubleshooting chart lists the observed trouble and probable cause and identifies the pins of the generator control panel which should be monitored as well as the expected voltages. Many times the description of the system malfunction is sufficient to indicate which circuits to check.

A. Voltage Check

A test jack on the circuit breaker panel provides for sampling DC voltage during routine checks. If voltage adjustments are to be made, then voltage measurements should be taken at the bus side of the line contactors according to the procedure outlined under VOLTAGE ADJUSTMENTS.

B. Overvoltage Check

Operation of the overvoltage function of the generator control panel can be checked by observing the voltmeter or monitoring pin B of the generator control panel when the generator is turned off. If the overvoltage function of the generator control panel is not operating, the generator will be disconnected from the bus, but operating voltage will still be indicated on the panel mounted voltmeters. This check can be routinely monitored by the flight crew during normal engine shutdown.

C. Generator Control Panel Test Unit

A Generator Control Panel Test Unit Assembly should be used to gain access to the individual inputs and outputs of the generator control panel. A wiring schematic (Figure 101) provides adequate information for building the test unit.

The wiring harness plug is disconnected from the generator control panel and connected to the receptacle of the test unit. The plug from the test unit is inserted into the receptacle of the generator control panel, thereby completing the series attachment of the test unit to the control panel. Do not connect the control panel when checking continuity.

A sensitive multimeter or volt/ohm meter, digital being preferred, capable of measuring voltages accurate to within one percent, is connected to the test unit by way of the banana jacks on the face of the unit, proper polarity of these connections must be closely observed as both positive and negative values will be measured. The alligator clip from pin G should be positively grounded to the airplane structure.

Individual inputs or outputs are selected by rotating the knob on the rotary switch to the desired switch position. A test switch in the unit applies a ground signal to pin E to check the operation of the equalizer circuit. A push-to-test light provides information concerning the operation of the generator control relay during engine start.

All voltage measurements are made with one generator on and a 50% load. Since maximum generator output is 250 amperes, then enough electrical equipment should be turned on to establish approximately 125 amperes of load on the electrical system. Refer to the electrical load utilization chart in section 24-50-00 for load requirements of the various electrical systems and components.

When checking the generator equalizer circuit for operation, a voltage drop of two or three volts, at pin B is an indication that the generator equalizer circuit is operating properly.

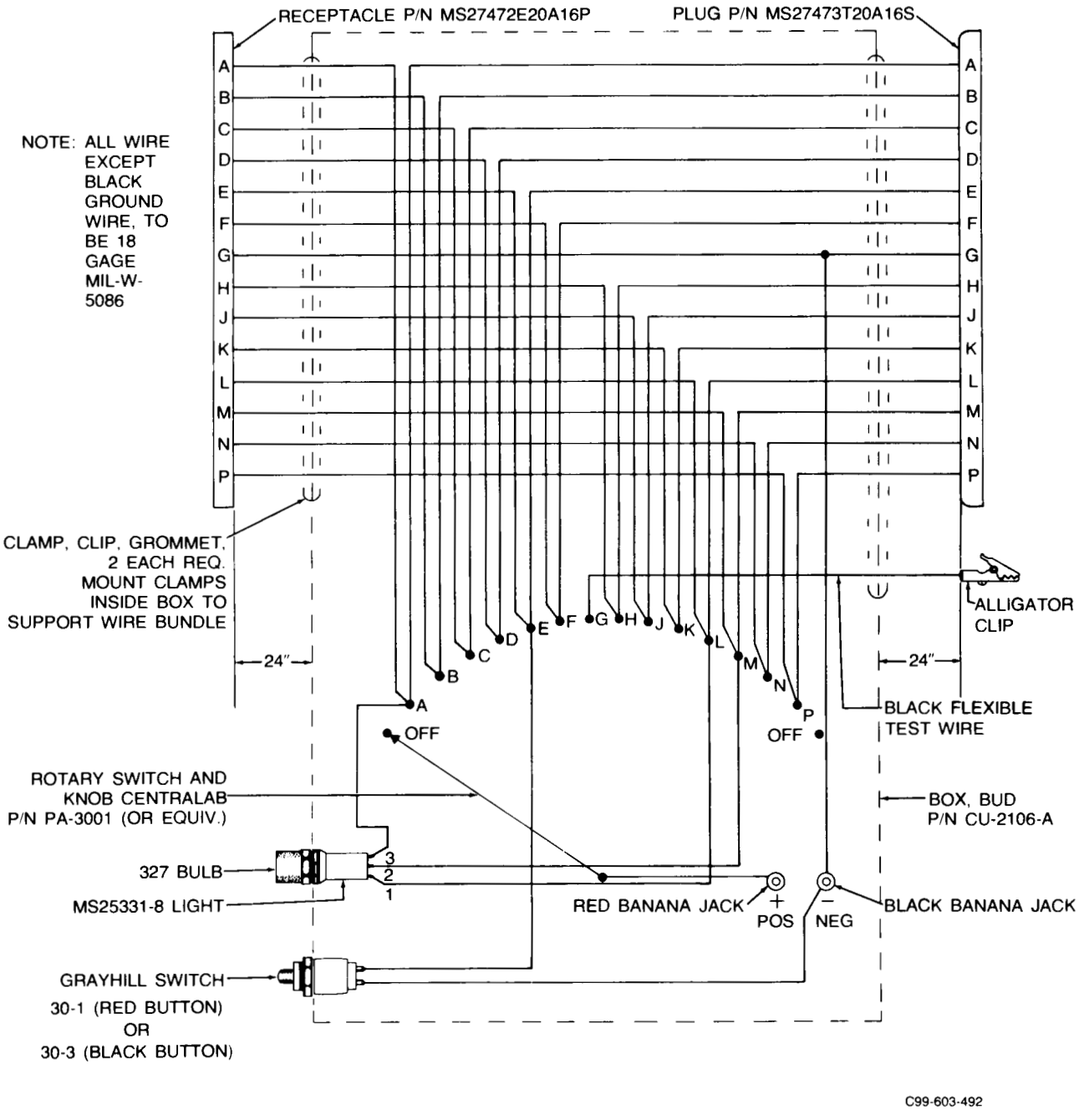

Figure 101
Generator Control Test Assembly Unit

Table 101
Troubleshooting - Voltage and Continuity Values
Generator Control Panel

CONN PIN LETTER	FUNCTION	CONTINUITY CHECK(1)			VOLTAGE CHECK		
		BATTERY OFF			ENG @ 70% LOAD @ 50%		
		GEN SWITCH POSITION			GEN SWITCH POSITION		
		OFF	ON	RESET	OFF	RESET	ON
A	Bus Voltage Signal Input	50 (2) ∞ (3)	50 (2) ∞ (3)	50 (2) ∞ (3)	26	26	28
B	Gen Voltage Signal Input	0 (4) ∞ (5)	0 (4) ∞ (5)	0 (4) ∞ (5)	1 (8)	28 (9)	28
C	Line Contactor Control Power Input	∞	0 (4) ∞ (5)	∞	0	0	28
D	Interpole Voltage Signal Input	0	0	0	0	0	-0.7 (12)
E	Equalizer	∞	∞	∞	0	0	-0.7 (13)
F	Battery Input (Reset)	∞	∞	50 (6) ∞ (7)	0	26	0
G	Ground Signal	0	0	0	0	0	0
H	Line Contactor Control Output	60	60	60	0	0	28
J	Regulator Power Input	0 (4) ∞ (5)	0 (4) ∞ (5)	0 (4) ∞ (5)	0	28 (9)	28
K	Gen Start-Up Power Input (Reset)	∞	∞	0 (4) ∞ (5)	0	28 (9)	0
L	Power Ground	0	0	0	0	0	0

Table 101
Troubleshooting - Voltage and Continuity Values
Generator Control Panel

CONN PIN LETTER	FUNCTION	CONTINUITY CHECK(1)			VOLTAGE CHECK		
		BATTERY OFF			ENG @ 70% LOAD @ 50%		
		GEN SWITCH POSITION			GEN SWITCH POSITION		
		OFF	ON	RESET	OFF	RESET	ON
M	Regulator Power Output	2	2	2	0	4 (10)	6 (10)
N*	Remote Trip	∞	∞	∞	0	28	28
P	Overvoltage	0 (4) ∞ (5)	∞	∞	1 (8)	11 (11)	11 (11)

* Not used on C99

( ) Indicates a note to be found on next page

NOTES:

1. Continuity check should be made with battery off, external power off, engines off and control panel disconnected.
2. 10 amp control panel circuit breaker closed. Resistance may vary drastically due to bus loads.
3. 10 amp control panel circuit breaker open.
4. Generator resistance, may vary due to load and circuit breaker resistance
5. 15 amp generator field circuit breaker open.
6. 10 amp generator control circuit breaker closed.
7. 10 amp generator control circuit breaker open.
8. Generator residual voltage.
9. Generator voltage should stabilize at 28.25 ± .25 vdc after about 1 second.
10. Switching ouput, the average voltage will vary with generator speed, load.
11. Internal voltage in control panel.
12. For generator current the interpole voltage is negative with respect to ground. The voltage will change with generator current.
13. For single generator, operation will be equal to pin "D"

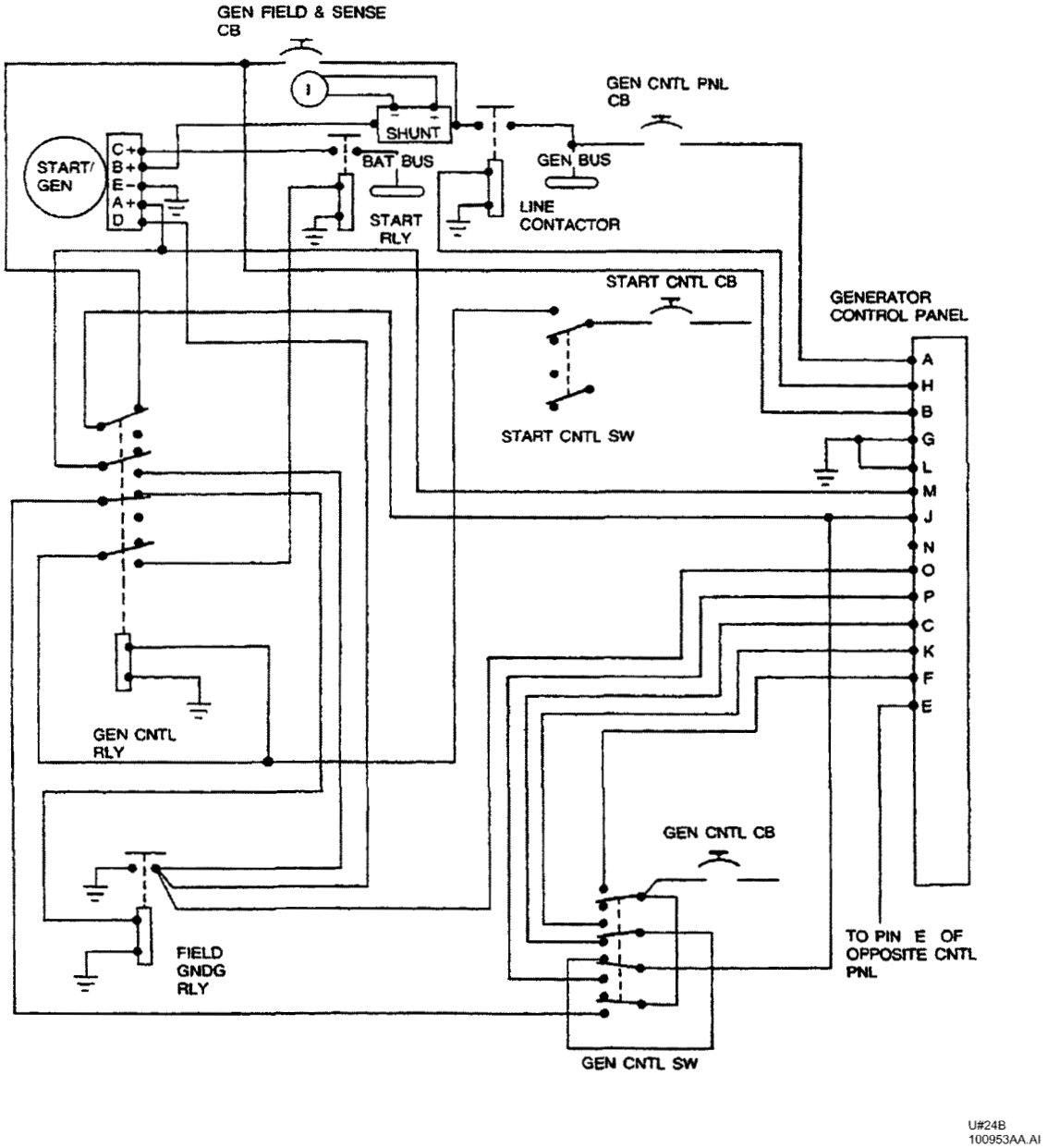

Figure 102
DC Generation and Control Schematic

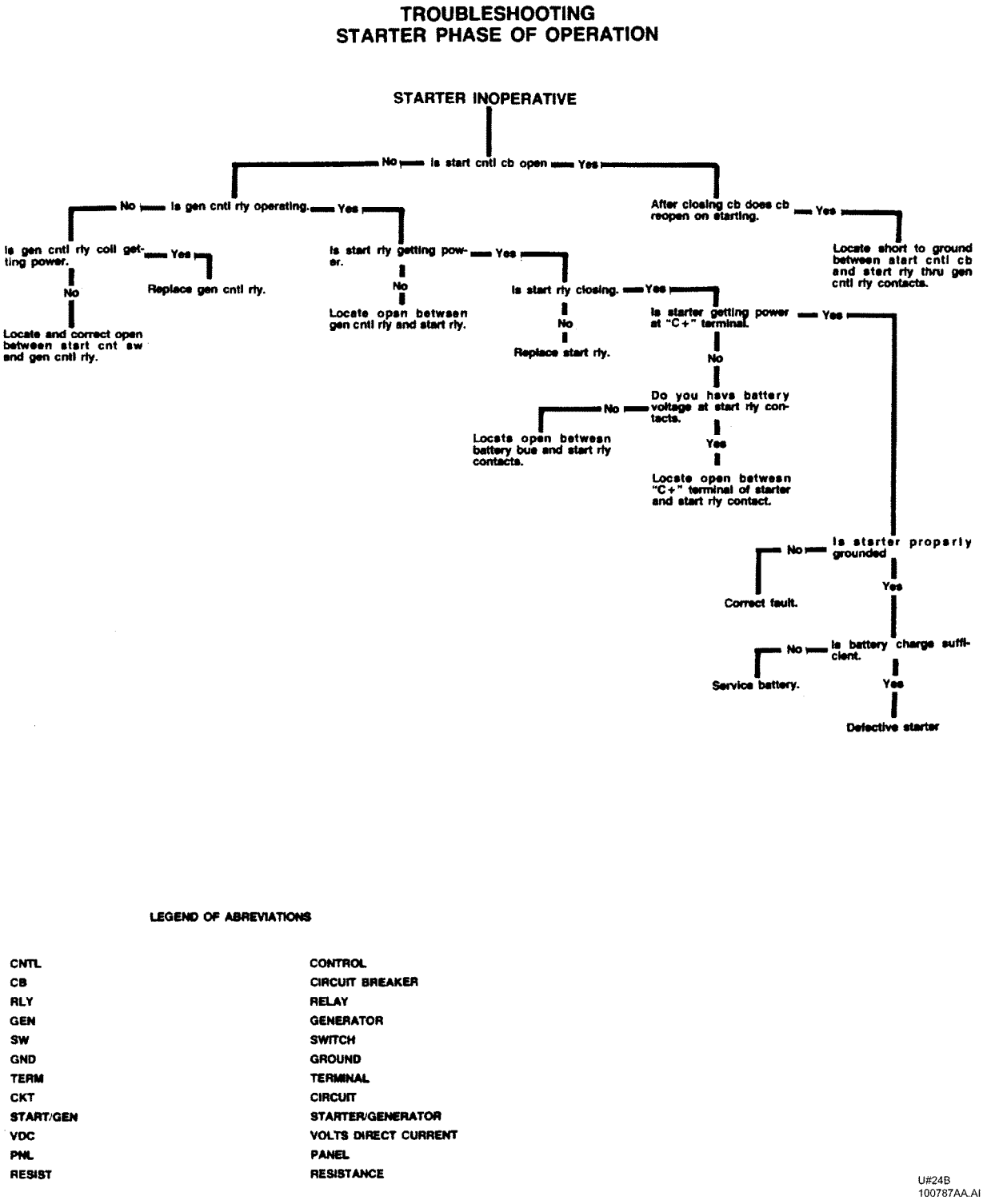

Figure 103
Troubleshooting - Starter Phase of Operation

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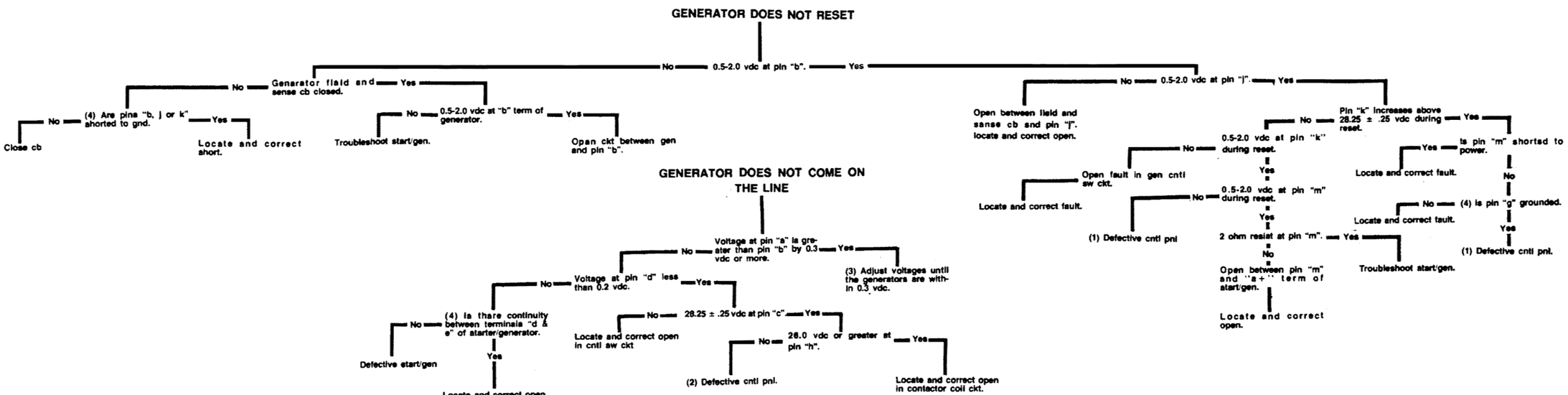
TROUBLESHOOTING
GENERATOR PHASE OF OPERATION (~)



(7) All checks are made with only one generator on the line at a time.

(6) Remove ground bracket from firewall. Disassemble snd clean. Small amounts of corrosion between bracket and firewall may add slight resistances to ground circuit. Repeat procedure on opposite engine.

(5) The test switch on the test unit applies a ground signal to the equalizer circuit of the cntl pnl (pin a) suppressing generator output voltage approximately 2.0 vdc.

(4) All resistance and continuity checks are to be made with engines off, battery off and cntl pnl disconnected.

(3) Refer to "Generator Voltage Adjustment".

(2) If pin "h" is shorted to gnd, correct fault before replacing cntl pnl.

(1) Failure of the generator control relay to properly short the shunt field of the generator may produce transient voltage spikes at pin "m" of the control panel during start. The magnitude of the spikes may be sufficient at times to permanently damage the internal circuitry of the control panel. A dimly illuminated lamp on the test unit during start is a normal indication. Should the lamp flash brightly when starts are initiated and the start relay closes, a voltage transient has been sensed at pin "m" of the control panel. Another bright flash of the lamp when the start is terminated and the start relay opens may occur during this check.

NOTE

At any time repairs or checks are made on the generators or control panels, overvoltage and reverse current protection circuits should be checked for operation as follows.

a. Remove the generators from the line one at a time by placing the generator control switches in OFF. If voltmeter readings do not return to zero, the overvoltage protect circuits are not functioning properly.

b. Reset the generators and bring them back on the line. Shut down both engines leaving the battery and both generators ON. As the engines slow down and reverse current begins to flow into the generators, the generators will be removed from the line. Loadmeter readings will be zero and the generator out lights will be on, indicating reverse current protect circuits are functioning properly. If the reverse current protect circuits are not operating properly, both generator isolation limiters must be checked for damage. If correct voltage indications are found at pins "a and d" of the control panels, the control panels are not providing reverse current protection and are defective.

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Figure 101
Troubleshooting - Generator Phase of Operation

ELECTRICAL POWER
DC GENERATION
MAINTENANCE PRACTICES

1. STARTER-GENERATOR

A. Removal

(1) Disconnect the electrical leads from the unit and remove the blower cover from the aft end of the generator.

(2) Loosen the quick-disconnect clamp securing the starter-generator to the mounting adapter and remove the unit.

CAUTION: *It is mandatory that the starter-generator be fully supported from the time the retaining clamp is loosened until the unit is removed from the engine The starter- generator must never be allowed to support its own weight through the spline shaft engagement If this precaution is not observed, damage to the shaft shear section will result.*

B. Installation

(1) Install a new packing on the starter-generator drive shaft.

(2) Position the starter-generator on the mounting adapter and secure it in place with the quick-disconnect clamp.

CAUTION: *It is mandatory that the starter-generator be fully supported from the time the unit is placed in position until the clamp is installed and properly torqued The starter-generator must never be allowed to support its own weight through the spline shaft engagement If this precaution is not observed, damage to the shaft shear section will result.*

(3) Close the clamp hinge over the T-bolt. Check with a mirror to make certain the clamp groove fully captures both the flange on the quick disconnect pad and the flange on the starter-generator around its entire circumference.

(4) When the clamp is properly positioned and the hinge and T-bolt are closed, tighten the T-bolt to a torque of 50-inch pounds.

(5) Connect the electrical leads to the starter-generator as indicated in the wiring diagram manual.

(6) Install the blower cover on the aft end of the starter-generator. Tighten the retaining nuts of the blower cover to 25 inch-pounds.

(7) After the installation is complete, run the engine at idle speed for at least two minutes. Shut down the engine and recheck the clamp retaining nut for proper torque.

NOTE: If the torque drops to 50° of the value in Step (5), check the clamp for proper engagement with the starter-generator and quick-disconnect pad, then repeat Steps (3), (4) and (5).

C. Brush Replacement

The generator brushes are provided with a diagonal groove from one side of the contact surface to a point on the opposite side which corresponds to the maximum permissible wear point (Ref. Figure 201). Brush wear can be estimated by the position of the groove on the contact surface. The brushes should be inspected each 100 hours of operation and replaced if the wear limit will be reached prior to the next inspection. The brushes must be replaced every 500 hours.

NOTE: Cored brushes used in the 99 Airliner require preliminary seating and run-in. Refer to Chapter 24 of the Beech 99 Airliner Series Component Maintenance Manual, Publication Number 23700, Revision 5 or subsequent. Cored brushes may cause the commutator to appear grooved. This is normal and should not be interpreted as cause for turning down or replacing the commutator.

2. GENERATOR VOLTAGE

A. Adjustment

The generator output voltage should be checked at each periodic inspection and after replacement of generator system components, or whenever the system has been repaired. Voltage measurements should be made at the DC test jack, located under the sub-panel to the right of the pedestal, or at the bus side of the generator contactors. This check should be done with a minimum of load on the system and engine rpm at 70%. The isolation limiter busses and generator contactors are located on the power distribution panel under the center aisle floorboard aft of the partition between the cockpit and cabin. The generator control panels and generator field grounding relays are located on a panel under the center aisle floorboards immediately aft of the power distribution panel.

(1) Start both engines in the manner described in the applicable Pilot's Operating Handbook.

CAUTION: *If cross-starts are attempted, damage to the 325-ampere isolation limiters may result.*

(2) Turn both generators ON. Operate the engines at any desired speed for a period sufficient to permit battery recharge after the engine starts. This will allow the generator system components time to stabilize.

(3) Connect a precision portable voltmeter with a 0 to 50 vdc scale (known to be accurate within one percent) to the bus or the DC test jack.

(4) Turn the RH generator OFF and record voltage output of the bus as supplied by the LH generator. This voltage should be 28.25 ± 0.25 vdc.

(5) If deviant voltages are observed, they may be adjusted by means of a voltage adjustment potentiometer provided on the side of each generator control panel. Turning the potentiometer clockwise will increase the voltage output of the generator while turning the potentiometer counterclockwise will decrease the output voltage.

(6) Turn the RH generator ON. Observe that the RH generator comes on the line, as indicated by a loadmeter, and that the LH generator stays on stays on the line. Turn the LH generator OFF.

(7) Repeat the above procedure to adjust the voltage of the RH generator while keeping the LH generator off the line.

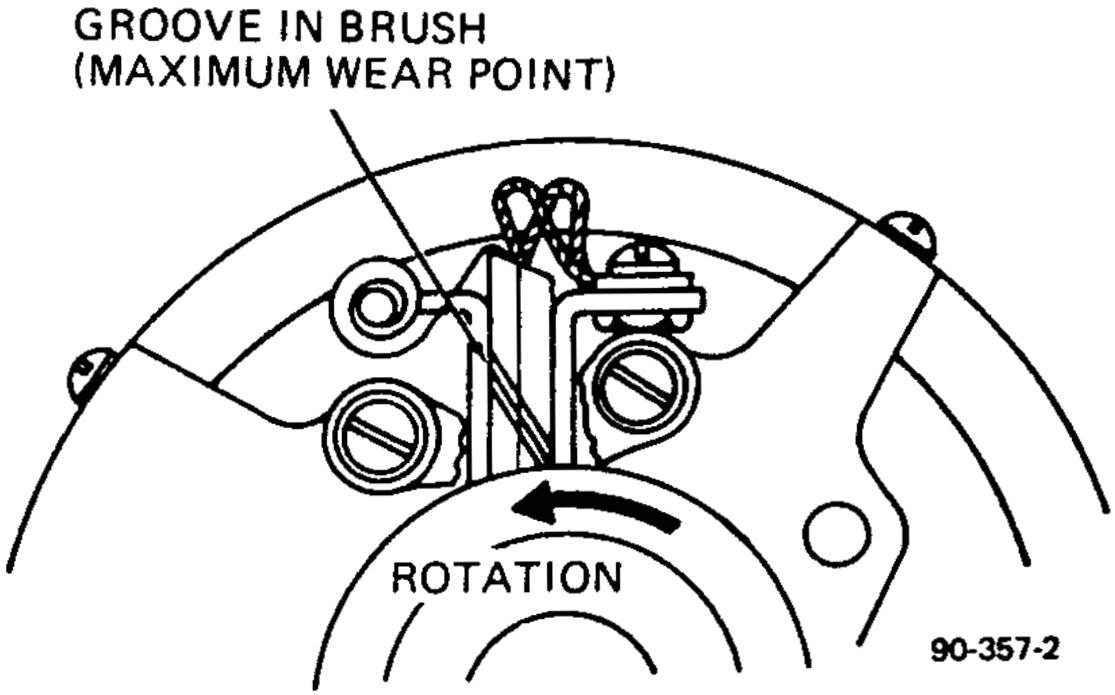

Figure 201
Starter - Generator Brush Wear Limits

ELECTRICAL POWER
BATTERY POWER
DESCRIPTION AND OPERATION

1. GENERAL

Either a Saft or a General Electric nickel-cadmium vented cell battery may be installed in the Model 99 Airliner Series airplane. The battery is rated at 24 volts dc and 34 ampere hours capacity.

Each nickel-cadmium battery consists of a steel case containing a number of identical and individual cells connected to each other in series and fitted side-by-side in the battery case. Interconnection between the cells is via high conductivity link bars. These are held in place on each cell by nuts on threaded external terminals. The end cells are connected by either a solid or a flexible link to a terminal on one face of the battery case or to a terminal that extends through the battery cover. The design of nickel-cadmium batteries allows for replacement of the individual cells in the event one cell should be damaged. All batteries are ventilated to permit cooling during operation and to allow the escape of gases that are produced during overcharge.

Nickel-cadmium battery facilities must be separate from lead-acid battery facilities. Nickel-cadmium batteries use a highly alkaline solution of potassium hydroxide as the electrolyte. Chemically, this is just about the exact opposite of the sulfuric acid solution used in lead-acid batteries.

Anything associated with lead-acid batteries, acid fumes included, should never come in contact with a nickel-cadmium battery or its electrolyte. Even traces of sulfuric acid entering a nickel-cadmium battery can result in permanent damage.

Contamination of the electrolyte with tap water, acids or other non-compatible substances will result in poor performance or complete failure of the battery. Battery operation with damaged, missing, or loose vent caps will cause the potassium carbonate (formed by a chemical reaction between carbon dioxide and potassium hydroxide) level to increase, resulting in depressed battery discharge voltages. Oxygen will also enter the cell and discharge the negative plate.

ELECTRICAL POWER
BATTERY POWER
TROUBLESHOOTING

1. BATTERY SYSTEM

Troubleshooting - Battery System

TROUBLE	POSSIBLE CAUSE	REMARKS
1. No power indicated with battery master switch ON.	a. Battery discharged or defective.	
	b. Open circuit between battery and master switch.	b. Check continuity.
	c. Master switch defective.	c. Check switch for operation.
	d. Defective relay.	d. Check relay operation.
2. Power on with master switch in OFF position.	a. Master switch defective.	a. Check switch for operation.
	b. Relay contacts stuck.	b. Check and replace relay if necessary.

2. BATTERY

Troubleshooting - Battery

TROUBLE	POSSIBLE CAUSE	REMARKS
1. Apparent loss of capacity.	a. Cells unbalanced.	a. Equalize cell voltage by performing full capacity discharge cycle.
	b. Electrolyte level too low.	b. Charge, adjust electrolyte level, and capacity test.
	c. Charging rate too low in airplane.	c. Check and adjust airplane charging system.
	d. Too little usage or shallow discharges.	d. Refer to a. above.
2. Complete failure to operate.	a. Loose or broken lead.	a. Repair or replace.
	b. Loose or disengaged terminals in battery.	b. Repair or replace any damaged hardware, capacity test.
	c. Battery not charged.	c. Charge and capacity test.
	d. Cell open internally.	d. Replace defective cell and capacity test.

Troubleshooting - Battery (Continued)

TROUBLE	POSSIBLE CAUSE	REMARKS
3. Excessive spewage (crystalline deposits on outside of cells).	a. Excessive charge rate.	a. Clean the battery, recondition, adjust the electrolyte level and capacity test.
	b. Electrolyte level too high.	b. Clean cell.
	c. Vent caps loose or broken.	c. Clean cell, replace or tighten vent cap.
	d. Cracked cell case.	d. Replace cell, clean battery.
4. Cell cases distorted.	a. Excessive charge rate.	a. Clean the battery, recondition, adjust the electrolyte level and capacity test. Adjust voltage regulator of airplane.
	b. Cell with internal short.	b. Replace defective cell, recondition and capacity test.
	c. Plugged vent caps, minor explosion.	c. Disassemble, replace defective parts, clean the battery, recondition and capacity test.
5. Unequal voltages among cells.	a. Cells unbalanced.	a. Equalize cell voltages (perform a full capacity discharge cycle), capacity test.
6. Foreign matter within cells.	a. Impure or acid-contaminated water.	a. Such cells will not normally respond to charging. They will show up as unbalanced cells and must be replaced.
7. Frequent addition of water.	a. Unbalanced cells.	a. Equalize cell voltages.
	b. Leaky or defective cells, damage to packing or vent cap.	b. Replace defective parts and inspect for electrolyte leakage. Clean, recondition and level electrolyte.
8. Burn marks on connectors.	a. Loose connectors.	
9. Overheating of intercell connector.	a. Loose or dirty intercell connectors.	a. Disassemble, clean, reassemble and properly torque intercell connectors, capacity test.
10. Foam or bubbling during charging.	a. Oil or grease contamination in the electrolyte.	a. Replace defective cells.
	b. Low concentration of electrolyte.	b. Recondition, replace cells that continue to foam.
11. Below normal output.	a. Battery switch left ON.	a. Recharge and capacity test.
	b. Voltage regulator set too low.	b. Recharge and capacity test. Reset voltage regulator.

Troubleshooting - Battery (Continued)

TROUBLE	POSSIBLE CAUSE	REMARKS
11. Below normal output. (Cont'd)	c. Internal connection links loose.	c. Torque, recharge and capacity test.
	d. External connector burned or pitted.	d. Clean or replace, recharge and capacity test.
	e. Defective or reversed cells.	e. Replace, recharge and capacity test.
	f. Cell case current leakage.	f. Disassemble, clean and recondition, replace any defective cells. and capacity test.

ELECTRICAL POWER
BATTERY POWER
MAINTENANCE PRACTICES

1. BATTERY

The airplane is equipped with a 24 volt nickel-cadmium battery. To obtain optimum service from the nickel-cadmium battery, proper and regular maintenance of the battery MUST be performed.

CAUTION: *Methods of servicing lead-acid batteries do not apply for the servicing of nickel-cadmium batteries.*

A Systematic Battery Maintenance Program should be established and carefully followed:

CAUTION: *Rings, metal watch bands. and identification bracelets should be removed prior to battery servicing Metal articles will, if allowed to contact intercell connectors of opposite polarity, fuse themselves to the connectors and severe burning will result.*

(1) The battery should be removed from the airplane for service.

(2) A log of the services performed on each battery should be maintained.

(3) The battery should be removed from the airplane and serviced after 100 Flight Hours or 30 days, whichever occurs first. If the ambient temperatures are above 90°F or the time between engine starts averages less than 30 minutes, the duty cycle should be reduced.

(4) The log of battery services performed should be evaluated to determine the need to service the battery at the above recommended intervals or to extend the intervals if justified. Accurate water consumption data is a valid barometer to use for adjustment of the servicing intervals.

Since the proper battery servicing requires two days, an additional battery (or batteries) will be required where airplane utilization warrants. For additional information on battery maintenance, refer to the Operating and Service for General Electric Nickel-Cadmium Vented-Cell Batteries (BBD-DEM-235). If the airplane is equipped with a SAFT battery, refer to the Nickel-Cadmium Aircraft Battery Maintenance Manual (DC3-01-78-3176-5A) for additional maintenance information. Advisory Circular AC 00-33. printed by Department of Transportation, Federal Aviation Administration, is another good source of battery maintenance information.

A. Maintenance Log

Customers are advised to keep a complete and up-to-date maintenance log on each battery. This information will help determine the source of any battery problems and will assist in substantiating warranty claims. A sample format for a service log is given in Figure 201.

B. Removal

(1) Remove the battery access panel located on the upper right hand center section between the fuselage and nacelle to gain access to the wing mounted battery.

(2) Cut the safety wire and remove the battery quick- disconnect.

(3) Cut the safety wire and loosen the hold-down wing nuts. Push the wing nuts aside to clear the battery hold-down bar.

(4) Disconnect the battery vent tubing.

(5) Lift the battery out of the well.

C. Pre-installation Instructions For Nickel-Cadmium Batteries

Unless otherwise indicated by a red warning tag, a nickel-cadmium battery is shipped in fully discharged state and contains the proper amount of electrolyte.

Observe the following precautions to ensure maximum performance and protect battery warranty:

(1) Do not remove the shorting strap until prepared to charge the battery. Batteries from which the shorting strap has been removed (for even a short period of time) must be considered in an unknown state of charge condition and must be completely discharged prior to charging and installation.

(2) Inspect batteries shipped from the factory for shipping plugs in the vent holes of each of the battery cells The blunt aluminum screws that serve as shipping plugs must be removed prior to operation of the battery. The Bunson valves, included with the battery in a separate plastic bag, should then be screwed into the vent cap assembly in place of the screw plugs. The Bunson valves will release excessive pressure from gas accumulation to prevent cell rupture.

NOTE: On batteries not equipped with the screw-type plugs and Bunson valves, remove the shipping plugs and clean the filler-cap vent plugs as noted under BATTERY CLEANING. Retighten the cell vents with the vent plug wrench included with the battery.

(3) Check the terminal screws securing the cell links for tightness. Refer to the service sheet furnished with the battery or to the manufacturer's maintenance manual for the proper torque value.

(4) Before charging, determine that all cells are properly installed by making a cumulative voltage check.

(5) After determining the battery is in good physical condition and is properly assembled, it should be charged as outlined under BATTERY CHARGING, in this chapter, and the electrolyte level adjusted.

Catalog No. _____ Serial No. _____

Date Installed _____ On _____

REMOVAL DATE AND REASON	GENERAL CONDITION			END OF CC CHARGE VOLTAGE RANGE		ELECTROLYTE LEVEL	ELECTRICAL LEAK CHECK	CAPACITY	REMARKS
M – Maintenance F – Failure (Indicate Type)	Case and Cover	Hardware and Liners	Cells and Vents	Lowest Reading Minimum 1.50	Highest Reading Maximum 1.70				Indicate: Average water added, hardware or cell replacements, burns or discolorations, etc.

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Figure 201
Battery Service Log

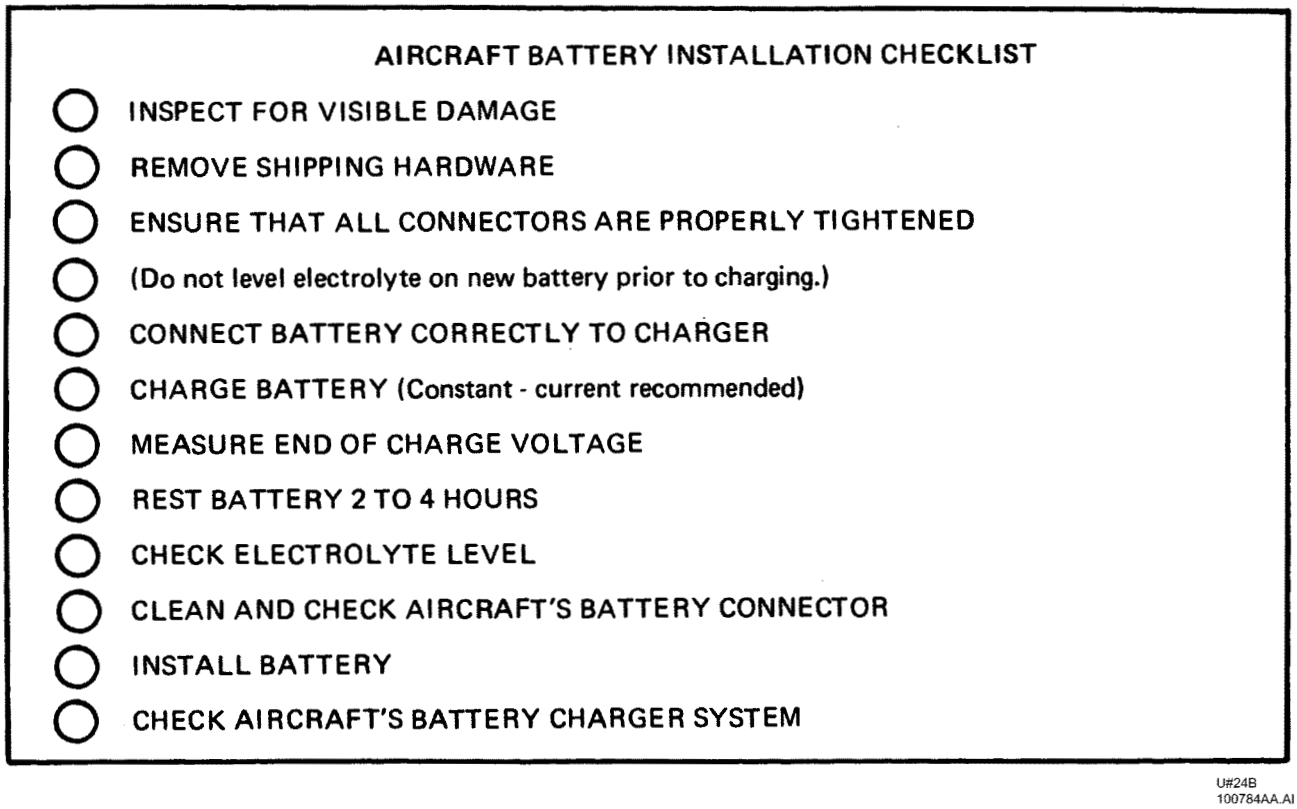

Figure 202
Pre-installation Checklist

D. Installation

(1) Place the battery in the well and connect the vent tubing.

(2) Install the hold-down bar and secure it in place with the wing nuts, then safety the wing nuts.

(3) Install the quick-disconnect and safety it.

(4) Install the access panel on the center section.

E. Charging

The two basic methods of charging nickel-cadmium batteries are the constant potential and constant current methods. Variations of the two basic methods may be incorporated in automatic equipment.

WARNING: Complete servicing of the battery is required if the battery is subjected to more than 32 volts for 2 minutes.

Special Notes on Charging.

The following special comments are made with respect to charging nickel-cadmium batteries:

(1) Charging is most efficient at battery temperatures between 40°F and 80°F.

(2) Two or more batteries may be charged in parallel on a constant potential charging bus, provided the charging equipment has the proper current producing capability.

(3) Do not charge batteries in parallel using the constant current method.

(4) Do not charge individual cells unless the plastic case is supported on each side. A special frame may be built to fit the cell, or two boards or plates may be placed on each side of the cell and held together by a C-clamp. The sides of the cell must be kept flat during charging.

(5) Perform necessary inspection, cleaning and repairs before charging.

(6) Do not energize charging equipment until after the battery has been connected to the charging circuit.

(7) When charging a battery in the shop, a thermometer should be placed so that the bulb is below and between the top of the cells. Do not place on charge any battery that has a temperature of 100°F or higher.

(8) The foam sometimes seen in cells during charging does not indicate a defect. Foaming usually occurs after water is added and will disappear after a few cycles of operation.

The recommended charging procedures and charge rates are given in the manufacturer's maintenance manual or on the service sheet furnished with the battery.

F. Capacity Reconditioning

The capacity of a nickel-cadmium battery does not decrease appreciably with age. However, there can be a temporary loss of capacity under certain duty cycles. A temporary loss of capacity is normally an indication of imbalance between cells. Imbalance can be caused by differences in temperature, charge efficiency, self-discharge rate, etc. The purpose of reconditioning is to restore a battery to its full

capability and to prevent premature damage and failure. Effective reconditioning requires specific procedures for certain periods of time. No step in the procedure can be eliminated nor can any time period be shortened and still yield effective battery reconditioning.

G. Frequency of Reconditioning

Due to the variables involved in usage, it is impossible to establish a time interval for reconditioning that will cover all batteries. Until service experience dictates otherwise, a visual and electrolyte check of the battery should be made after the first 50 hours of flight. If the condition of the battery is normal and the level of electrolyte in the battery is satisfactory, schedule the initial reconditioning at 100 hours. Repeat the reconditioning procedure outlined below at 100-hour intervals until servicing experiences justifies a change.

NOTE: The log of battery services performed should be evaluated to determine the need for servicing, the battery at the above recommended intervals or extending the intervals. Accurate water consumption data is a valid barometer to use for adjustment of the servicing intervals.

H. Reconditioning Procedure

(1) Discharge the battery at a current equal to or less than the one-hour rate. Short out each cell as it drops below 0.5 volts. The cells may be shorted by dips or by wires having clips on each end. Allow the shorts to remain on the cells for a minimum of 16 hours, and preferably for 24 hours.

(2) Remove the shorting clips and charge for 24 hours at 2.7 amperes (34 ampere-hour batteries). After approximately 5 minutes of charge, measure the individual cell voltages. If any cell voltage is greater than 1.50 volts, add distilled water. The amount of water required is approximately 1 cc per rated ampere-hour capacity, for example, a 34 ampere-hour cell may require about 34 cc of water at this time.

(3) After approximately 10 minutes of charge, remeasure the cell voltages. Replace any cell that measures below 1.20 volts or above 1.55 volts.

(4) Continue charging for 20 hours. After 20 hours of charging, measure and record the individual cell voltages. If necessary, add distilled water to level the cell 1/8 to 1/4 inch above the baffle. Do not remove any electrolyte from the cells.

(5) Measure and record the voltage at 24 charge hours and compare with the 20 hour reading. If the 24 hour voltage reading is below the 20 hour reading by more than 0.04 volts, replace the cell. Also replace any cell that measured below 1.50 volts at 24 charge hours. After removing the battery from charge, measure the electrolyte temperatures. If the electrolyte temperature of any cell is greater than 30°F above the ambient, replace that cell.

I. Electrolyte Level Adjustment

Although the electrolyte level in the nickel-cadmium battery varies with the state of charge, it should be visible above the bottom of the baffle when the battery is fully charged. When the state of charge of the battery is low, the plates absorb some of the electrolyte, then release it as the battery is recharged. The electrolyte level on any battery must be adjusted after a full charge and a two to four hour rest on open circuit. Check the electrolyte level of the battery (fully charged) in the following manner:

CAUTION: *Never use acid or tools contaminated with acid during this adjustment, for both bodily injury and equipment damage may result. If possible, use equipment reserved for nickel-cadmium batteries. If lead-acid battery equipment must be used, remove all possible acid contamination with a sodium bicarbonate solution and rinse. Even minute traces of acid can damage a nickel-cadmium battery.*

(1) Remove the battery from the airplane.

(2) With the battery removed from the airplane, remove the filler cap vent plug on each cell one cell at a time.

(3) Insert a transparent tube (approximately 6 inches long and 1/4 inch in diameter) perpendicularly into the filler well until the open end rests lightly on the cell baffle, then place the index finger over the top open end and withdraw the tube.

CAUTION: *Do not push down, for the light material of the baffle will give enough to result in a false indication of the electrolyte level.*

(4) The electrolyte level of a fully charged battery should be between 1/8 and 1/4 inch above the bottom of the baffle. If the level of liquid in the tube exceeds 1/4 inch, remove the excess with a syringe or squeeze bottle. If the level of the electrolyte is less than 1/8 inch above the bottom of the baffle, add DISTILLED WATER with a syringe or squeeze bottle.

CAUTION: *Tap water contains minerals, chlorines, softening agents, and other foreign materials which will contaminate a storage battery and shorten its life.*

WARNING: The battery may be damaged if the proper procedure is not followed when adding water to the cells.

(5) Clean and install, the filler cap, vent cap, vent plugs and check the battery terminal links for tightness. Discolored links or melted nylon around cell terminals indicate loose link connections.

CAUTION: *If water or electrolyte is spilled into the battery container, the resultant electrolyte corrosion may cause battery failure. The battery case must be cleaned as instructed in BATTERY CLEANING.*

J. Electrical Leakage Check

The self (internal) discharge rate of a vented nickel-cadmium aircraft cell is in the order of C/1000 when fully charged. C in this formula represents capacity in ampere-hours. This is about 34 ma for 34 ampere-hour batteries. The only pertinent measure of external leakage is the rate of discharge caused by the leakage. The rate is significant only when it approaches the rate of internal leakage. Therefore. external leakage need be considered excessive only when an ammeter shorting the battery positive or negative terminal to the battery case indicates 34 ma or more on 34 ampere-hour batteries. Any current less than the preceding limits indicates a magnitude of leakage that has a negligible effect on battery performance.

Perform the following test to determine if external leakage is sufficient to necessitate cleaning the battery. Set the range selector of a multi-range ammeter to the 500 ma scale or higher (a low cost meter is recommended to preclude possible damage to an expensive precision meter). Connect the positive terminal of the ammeter to the positive battery terminal and the negative ammeter terminal to the battery case. Decrease the ammeter current range to obtain a readable value of current and record the value. Perform the same measurement at the negative battery terminal by connecting it to the negative terminal of the ammeter and connecting the positive terminal of the ammeter to the battery case. If the

current reading at either terminal is more than 34 ma on 34 ampere-hour batteries, the battery should be cleaned. This test should be made again after the battery has been completely cleaned and charged. If the current measured is again more than 34 ma on 34 ampere-hour batteries, it may be assumed that one of the cells has a seal leak. That cell may be found by measuring connector-to-battery case voltages. The lowest voltages will occur at the connectors on each side of the defective cell. A cell found leaking in this manner should be replaced.

K. Capacity Check

(1) Discharge the battery at a rate of 14.9 (on 34 ampere-hour batteries) until an average voltage of one volt per cell is reached. Measure the time required for the battery to reach that discharged state. Any battery that discharges to one volt per cell in 84 minutes or less should be given another reconditioning (deep) cycle.

(2) After the second reconditioning cycle, recheck the battery capacity by discharging at the rate used in Step (1). Measure the individual cell voltages after 84 minutes of discharge. If any cell is below one volt, replace it.

L. Cleaning and Inspection

(1) Scrub each cell with a 5% solution of boric acid in water, but take great care to prevent the solution from entering the cell.

(2) Wash each cell off under running water and dry with an air hose or clean absorbent towel.

(3) Inspect each cell for defects such as cracks, holes, or burn spots. Replace defective cells with new or rebuilt ones.

(4) Make sure that battery hardware is clean and in good mechanical condition. Wash the hardware. liners, case, cover, and other associated parts in a warm soapy solution to remove accumulated dirt and carbonate deposits. Use a stiff brush to remove heavy deposits. After washing, rinse the parts free of soap and spread them out to dry.

(5) Remove corrosion preventive from connectors, screws, nuts, and washers with alcohol or by degreasing.

(6) Wash vent caps thoroughly with hot water and no soap.

(7) After the parts are dry, sort out damaged or heavily corroded pieces. Scrap any links having burns, bends, or defective nickel plating. If a link is tarnished at the terminal connection, it should be polished with a wire brush. It is recommended that new terminal screws and nuts be used to ensure optimal electrical connection.

(8) Check the battery receptacle for burns, cracks, and bent or pitted terminals. Defective receptacles can overheat, cause arcing, and decrease output voltage to result in premature battery failure.

(9) Scrap bent or torn battery cases and covers that are beyond repair.

(10) Replace or repair loose or damaged cover gaskets and cell holddown bars.

NOTE: Refer to the maintenance manual of the battery manufacturer for additional details on battery disassembly and assembly.

ELECTRICAL POWER
BATTERY MONITOR
DESCRIPTION AND OPERATION

1. GENERAL

A battery charge current monitor is installed to provide a visual indication of abnormal battery charge current. The system provides an indication when the conditions exist for a thermal runaway of the nickel-cadmium battery. The battery may then be removed from the charging system before further battery damage occurs. The most common cause of the thermal runaway is damage to the gas barrier between the plates resulting from overcharging the battery at a high rate at high temperatures. Once it has sustained gas barrier damage, the battery is subject to thermal runaway when charged on a constant voltage charging system over a period of time. Normally a high rate of charge current will have a heating effect only when the overcharging current is gradually increasing, however, a high discharge current usually has a heating effect.

During normal operation, the idle current of the battery is less than one amp. It increases significantly above the normal level when the battery is charged at an elevated temperature or from a high charge voltage. A high idle current increases water consumption and may destroy the gas barrier (cellophane separator) between the plates. Once a battery has sustained damage to the gas barrier, it will have a high idle current and be subject to thermal runaway. The battery monitor system provides an indication of the high charge current resulting from high battery temperature, high charging voltage, or gas barrier damage.

The battery monitor system consists of a shunt in the negative lead of the battery, a battery charge current detector, and a BATTERY CHARGE light in the annunciator panel. The shunt is located adjacent to the battery box in the right center section, while the detector module is mounted on the electrical/electronic module beneath the center aisle floor immediately aft of the partition between the cockpit and cabin. The charge current detector circuit amplifies the signal from the shunt and provides a 28 volt output for the BATTERY CHARGE light when the signal exceeds the trigger level of 1.5 $\pm$ 0.2 mv. This trigger level corresponds to 7.5 $\pm$ 1 amp with the 250-amp shunt used for 34 to 45 ampere-hour batteries. The circuit will reset when the signal decreases 0.3 mv below the trigger level to provide 1.5 amp hysteresis. There is a six second time delay to override momentary recharges when heavy loads are switched on.

The system will illuminate the BATTERY CHARGE light in the annunicator during battery recharge to provide a self- test of the system. Following an engine start, the BATTERY CHARGE light will illuminate and remain on for about five minutes until the charge current decreases to the current detector reset level as the battery approaches full charge. The time the light remains on will exceed five minutes if the trigger level of the current detector is too low or if the battery is in a low state of charge, has been previously discharged at a low rate, has a low charge voltage per cell, has gas barrier damage or is at a low temperature.

After the BATTERY CHARGE light extinguishes, it should remain off unless either a thermal runaway has begun or the battery idle current increases in response to an increase in electrical system voltage. Such a voltage increase normally results from poor generator paralleling or load switching. The BATTERY CHARGE light may illuminate for short intervals as the battery recharges when the generator speed is increased above cut-in speed.

ELECTRICAL POWER
BATTERY MONITOR
MAINTENANCE PRACTICES

1. BATTERY MONITOR

A. Calibration Check

Calibration of the battery monitor system can be checked on the airplane by simulating a charge current sufficient to trigger the current detector circuit. A circuit to accomplish this function is shown in Figure 201. It is recommended that system calibration be checked at regular intervals when the battery is removed for service. Check system calibration as follows:

NOTE: The signal level is very low and susceptible to resistance unbalance in the shunt signal leads. All connections in these leads must be clean and tight to prevent improper system calibration and erratic operation.

(1) Disconnect the battery and connect a variable resistance and ammeter to the battery connector (Ref. Figure 201).

(2) Use an external power source or energize one generator and the battery relay to apply 28 volts to the bus of the airplane electrical system, then adjust the variable resistor until the battery charge current detector triggers.

(3) Monitor the voltage at pin 7 on the current detector to determine when the circuit triggers. The voltage at pin 7 will switch from two volts to ten volts when the circuit triggers, then six seconds later the BATTERY CHARGE light in the annunciator should illuminate.

(4) As the current is reduced to the reset level, the voltage at pin 7 should switch back to two volts when the BATTERY CHARGE light extinguishes.

B. Functional Check

Use a 28-volt, 2-amp power supply in the circuit shown in Figure 202 to bench test the current detector for proper operation as follows:

(1) With 28 volts applied to the current detector circuit and the signal current adjusted to a minimum, check for a reading of two volts at pin 7 of the current detector and that the indicator light remains off.

(2) Increase the signal current until pin 7 switches to a reading of 10 volts. The signal current should be 0.75 $\pm$ 0.1 amp and the indicator light should illuminate within 6 seconds.

(3) Decrease the signal current until the voltage at pin 7 switches back to two volts. The current should be approximately 0.15 amp less than the trigger voltage and the light should extinguish.

(4) Replace the current detector assembly if found defective.

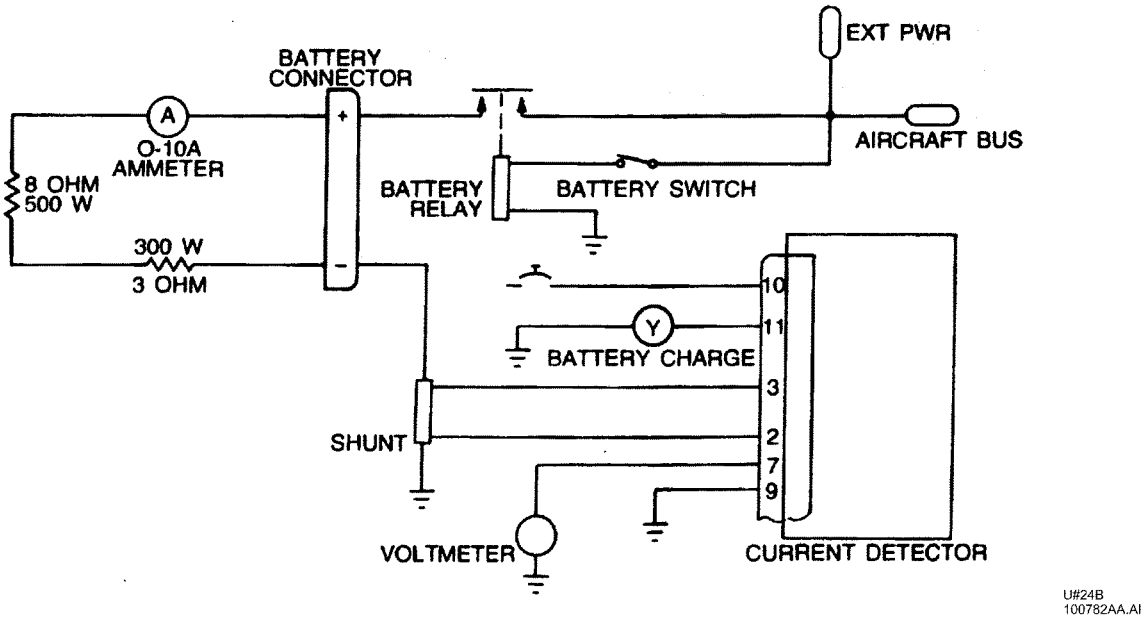

Figure 201
Calibration Circuit

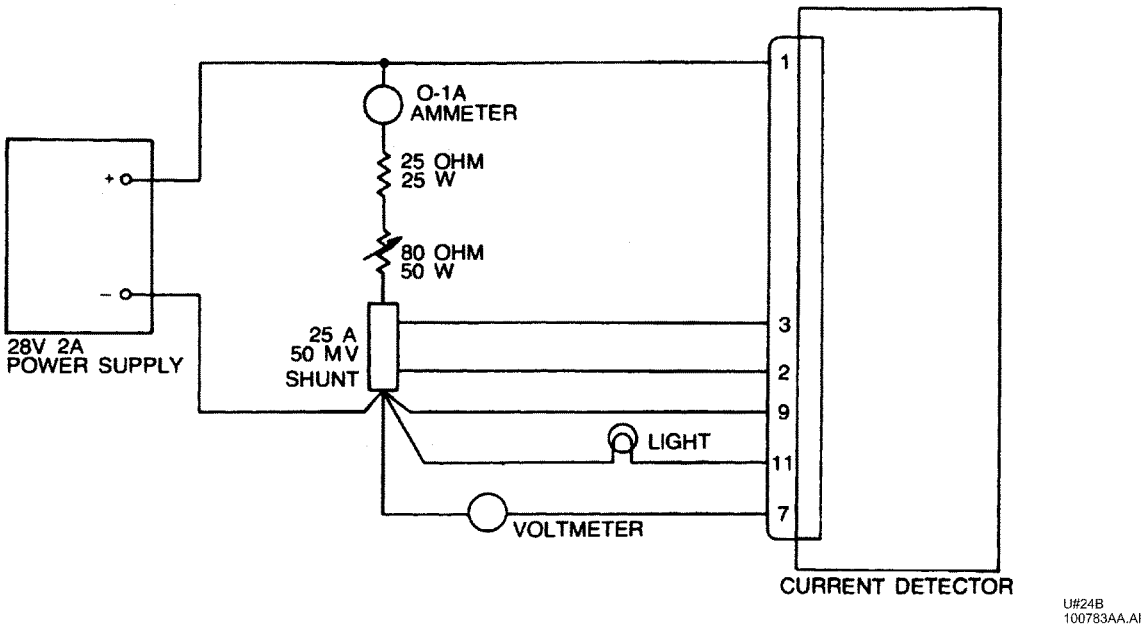

Figure 202
Functional Test Circuit

2. BATTERY GAS BARRIER

A. Check

It is imperative that the condition of the battery gas barrier be checked whenever the battery is removed and serviced to determine the acceptability of the battery from the standpoint of thermal stability. Test the battery as follows:

(1) Overcharge a fully charged battery for four hours with a constant current of 0.1 of the one hour capacity rating of the battery. This current must be CONSTANT rather than a pulsing average of 0.1C.

(2) Check individual cell voltage of the battery while still charging it at the end of the four hour period.

(3) Any cell with a voltage of less than 1.50 volts has gas barrier damage and should be replaced.

ELECTRICAL POWER
EXTERNAL POWER
DESCRIPTION AND OPERATION

1. GENERAL

The external power receptacle is designed for use with an auxiliary ground power unit having a standard AN plug. An external dc sensor in the external power installation illuminates a caution light, EXT PWR, in the annunciator panel.

Polarity protection is provided by the external power relay. A voltage of +24 to +28 vdc applied to the small pin of the external power receptacle energizes the external power relay through one half of the battery switch. A diode in the coil circuit of the external power relay prevents closing the relay if a negative voltage is applied to the small pin of the external power receptacle.

ELECTRICAL POWER
EXTERNAL POWER
MAINTENANCE PRACTICES

1. GENERAL

A ground power unit, for engine starts or for use during ground operation without the engines running, must be capable of supplying load requirements without excessive voltage drop. The unit must be capable of delivering up to 1000 amperes for 1 second and be capable of delivering up to 300 amperes continuously at 24 to 30 volts. Use of an inadequate ground power unit can cause a voltage drop below the dropout voltage of the external power relay and the starter relay. This will result in relay chatter and/or welded contacts. By the same token, a continuous load in excess of 350 amperes will damage the external power relay and power cables of the airplane.

CAUTION: *The output setting must not exceed 1000 amperes on external power sources with a higher current-carrying capability. Any current in excess of 1000 amperes may overtorque the drive shaft of the starter-generator or produce heat sufficient to shorten the life of the unit.*

Observe the following precautions when using an external power source:

(1) Use only an auxiliary power source that is negatively grounded. If the polarity of the power source is unknown, determine the polarity with a voltmeter before connecting the unit to the airplane.

(2) Before connecting an external power unit, turn off all radio equipment, the generator switches and the battery switch.

CAUTION: *Voltage is required to energize the Avionics Master Power relays to remove the power from the avionics equipment. Therefore, never apply external power to the airplane without first applying battery voltage. If the battery is removed from the airplane connect an external battery parallel to the external power unit prior to energizing the power unit.*

(3) After the external power plug is connected, turn on the battery switch before turning on any other equipment to protect transistorized equipment against transient voltage spikes.

CAUTION: *The battery may be damaged if exposed to voltages higher than 30 volts for extended periods of time.*

(4) If the ground power unit does not have a standard AN plug, check the polarity of the plug. The positive lead from the ground power unit must connect to the center post, the negative lead must connect to the front post, and a positive voltage of 24 to 28 vdc must be applied to the small polarizing pin of the airplane's external power receptacle.

ELECTRICAL POWER
AC ELECTRICAL LOAD DISTRIBUTION
DESCRIPTION AND OPERATION

1. GENERAL

The electrical load distribution system provides the means of supplying ac and dc power to the various airplane systems. The 26 vac engine instrument power is supplied to the torquemeter system, directional gyro and attitude gryo at the Right Circuit Breaker Panel Assembly. The dc power from the battery and generators is distributed to the load systems from the DC Power Distribution Panel Assembly under the center aisle floor immediately forward of the main spar.

The DC Power Distribution Panel Assembly (refer to Chapter 39) is arranged as a dual bus. Two 325-ampere limiters provide isolation of the left and right generator systems and the battery system in the event of a ground fault in either of the generator systems, or in the battery system. Two dual fed buses in the Right Circuit Panel Assembly and the Left Inboard Subpanel and two dual fed bus loops in the Fuel Panel supply power to load circuits. Each of the four dual fed bus loops receives power from a limiter on the left bus and a limiter on the right bus of the DC Power Distribution Panel Assembly. Isolation from a ground fault in either bus is provided by a pair of isolation diodes for each dual fed bus loop.

A battery bus located in the lower right center section, below the battery box is hot wired to provide power to the cabin entry and loading lights, baggage pod light, pilots control wheel clock and map lights, and forward baggage loading lights.

A. Voltmeter

The dc voltage of the electrical system is indicated by the voltmeter. Two 325-ampere limiters isolate the two feeder buses and enable the voltmeter to read the dc bus voltage on either of the buses should a fault occur. The dc voltage measured by the meter does not provide an accurate enough indication of bus voltage to be used for generator voltage adjustment.

B. Loadmeter

Two loadmeters, LH and RH, sense the voltage developed across the loadmeter shunts resulting from the generator output current. The voltage developed across the shunts is directly proportional to generator output and is 50 millivolts at full generator output. The loadmeter sensing leads are protected by 5-ampere circuit breakers.

2. AC POWER DISTRIBUTION

Alternating current is supplied to the engine gages, ac gyros, and avionics by either the No. 1 or the No. 2 inverter. The inverters provide 26 vac and 115 vac power to supply the ac equipment on the airplanes.

The 26 vac power is supplied to the airplane engine instruments through the 26 volt inverter select relay. The 115 vac 400 Hz power is supplied to the ac gyros through the 115 volt inverter select relay, and to the avionics equipment through an avionics inverter select relay. The two 26 and 115 vac select relays are dc controlled relays energized by the application of 28 vdc from the No. 1 inverter control circuit breaker through the inverter select switch. The relays are energized with the inverter selection switch in the No. 1 position ONLY. The inverter select switch also supplies 28 vdc input voltage to the selected inverter. An avionics inverter select relay is energized by the application of 115 vac from the No. 1 inverter In the inverter No. 2 mode, the avionics equipment receives 115 vac from the No. 2 inverter through the normally closed contacts of the avionics inverter select relays.

Should the selected inverter fail, or the ac power to the systems be lost, an INVERTER warning indicator on the annunciator panel will be illuminated. The indicator is illuminated by the application of 28 vdc from the inverter warn circuit breaker through the normally closed contacts of the inverter fail relay to the annunciator. The inverter fail relay is an ac controlled relay energized by power from the ac bus. When the relay is energized, the 28 vdc is removed from the annunciator INVERTER warn indicator and the light is extinguished.

A Volt/Frequency indicator provides the means for monitoring the output voltage and frequency of the selected inverter.

A. AC Test Jacks

The 115 vac may be sampled at the 115 vac (blue) test jack located on the bottom of the subpanel, to the right of the pedestal. The test jack samples the 115 vac from either the No. 1 or No. 2 inverter, through the inverter select relay. Inverter installations required by certain optional avionics installations utilize the standard wiring but may require additional components such as power relays and transformers. Refer to the package of avionics drawings furnished with the individual airplane for these deviations.

The inverter access panels located in the wings outboard of the nacelle and the circuit breaker panel cover should be removed prior to troubleshooting the inverters.

ELECTRICAL POWER
AC ELECTRICAL LOAD DISTRIBUTION
MAINTENANCE PRACTICES

1. DUEL BUS FEEDER DIODES

The dual bus feeder diodes should be inspected at 600 flight hour intervals as instructed under PERIODIC INSPECTION OF DUAL BUS FEEDER DIODES. Whenever the dual bus has been modified or extensive repairs have been made that could result in the dual bus loops being interconnected or open, it is necessary to perform a DUAL BUS CONFORMITY INSPECTION as outlined in this section. This will ensure that the original design of the dual bus is maintained.

A. Periodic Inspection

An open or shorted dual bus feeder diode cannot be detected during the normal operation of the airplane electrical system. Should a malfunction occur which would cause a bus isolation limiter to open, such as a ground fault on a generator bus, an open dual bus feeder diode could not supply power to its respective dual bus loop. A shorted diode would not isolate its respective dual bus from a ground fault. The inspection procedure outlined here will ensure the dual bus capability. The inspection may be performed with either battery power or an auxiliary ground power unit connected to the external power receptacle. A suggested inspection procedure follows:

CAUTION: *Voltage is required to energize the Avionics Master Power relays to remove the power from the avionics equipment. Therefore, never apply external power to the airplane without first applying battery voltage. If the battery is removed from the airplane, connect an external battery parallel to the external power unit prior to energizing the power unit.*

NOTE: Refer to Chapter 39 in this Maintenance Manual for assistance in component location.

(1) Remove all electrical power from the airplane.

(2) Remove the center aisle floorboards, covers, etc. as necessary to gain access to the Power Distribution Panel Assembly.

(3) Remove the Left Isolation Limiter.

(4) Turn the battery master switch ON.

(5) Bus voltage SHOULD be present on the Right Isolation Bus. This may be verified by attempting to operate the No. 2 inverter. The No. 2 inverter SHOULD operate. Operation of the No. 2 inverter will verify continuity of the Right Isolation Limiter.

(6) The following annunciator readouts will verify bus voltage to each of the four dual fed buses: L BL AIR FAIL - No 1 bus, R - BL AIR FAIL - No. 2 bus, L FUEL PRESS - No 3 bus; R FUEL PRESS - No 4 bus.

NOTE: The reference designators for the components in the following Steps refer to the C99 Wiring Diagram Manual.

(7) If any of the annunciators do not illuminate, locate the component which is open and correct it. The open component can be located by the following Steps:

(a) If the L BL AIR FAIL annunciator does not illuminate, the Primary Right Feeder Circuit Breaker for No. 1 bus (A145CB2), RH Feeder Diode for No. 1bus (A144CR2), the Secondary

Right Feeder Circuit Breaker for No. 1 bus (A146CB163); or the Left Bleed Air Warning Circuit Breaker (A146CB145) would be open.

(b) If the R BL AIR FAIL annunciator does not illuminate, the Primary Right Feeder Circuit Breaker for No. 2 bus (A145CB4), the RH Feeder Diode for No. 2 Bus (A144CR4), the Secondary Right Feeder Circuit Breaker for No. 2 bus (A146CB164) or the Right Bleed Air Warning Circuit breaker (A146CB146) would be open.

(c) If the L FUEL PRESS annunciator does not illuminate, the Primary Right Feeder Circuit Breaker for No. 3 bus (A145CB6), the RH Feeder Diode for No. 3 bus (A143CR3), the Secondary Right Feeder Circuit Breaker for No. 3 bus (A124CBl) or the Left Fuel Pressure Warning Circuit Breaker (A124CB19) would be open.

(d) If the R FUEL PRESS annunciator does not illuminate, the Primary Right Feeder Circuit Breaker for No. 4 bus (A145CB8), the RH Feeder Diode for No. 4 bus (A143CRl), the Secondary Right Feeder Circuit Breaker for No. 4 bus (A124CB2) or the Right Fuel Pressure Warning Circuit Breaker (A124CB20) would be open.

(8) After correcting the open component, repeat the test.

(9) Attempt to operate the No. 1 inverter. It SHOULD NOT operate. If the No. 1 inverter DOES operate, at least one of the dual bus feeder diodes is shorted. Detection of the shorted diode (diodes) may be accomplished by opening the following dual bus feeder circuit breakers one at a time: LH No. 1, LH No. 2, LH No. 3 and LH No. 4. Observe which circuit breaker, the opening of which, stops the inverter. The corresponding diode will be shorted. After replacement of the shorted diode, repeat the test.

(10) Remove all electrical power from the airplane.

(11) Replace the Left Isolation Limiter. Remove the Right Isolation Limiter (Fl).

(12) Turn the Battery Master Switch ON.

(13) Bus voltage SHOULD be present on the Left Isolation Bus. This may be verified by attempting to operate the No. 1 inverter. The No. 1 inverter SHOULD operate. Operation will verify continuity of the Left Isolation Limiter.

(14) The following annunciator readouts will verify bus voltage to each of the four dual fed buses: L BL AR FAIL - No. 1 bus, R BL AIR FAIL - No. 2 bus; L FUEL PRESS - No 3 bus, R FUEL PRESS - No. 4 bus.

(15) If any of the annunciators do not illuminate, locate the component which is open and correct it. The open component can be located by the following Steps:

(a) If the L BL AIR FAIL annunciator does not illuminate, the Primary Left Feeder Circuit Breaker for No. 1 bus (A145CBl), the LH Feeder Diode for No. 1 bus (A144CRl), the Secondary Left Feeder Circuit Breaker for No. 1 bus (A146CB165) or the Left Bleed Air Warning Circuit Breaker (A146CB145) would be open.

(b) If the R BL AIR FAIL annunciator does not illuminate, the Primary Left Feeder Circuit Breaker for No. 2 bus (A144CB3), the Left Feeder Diode for No. 2 bus (A144CR3), the Secondary Left Feeder Circuit Breaker for No. 2 bus (A146CB166) or the Right Bleed Air Warning Circuit Breaker (A146CB146) would be open.

(c) If the L FUEL PRESS annunciator does not illuminate, The Primary Left Feeder Circuit Breaker for No. 3 bus A145CB5), the LH Feeder Diode for No. 3 bus (A143CR4), the Secondary Left Feeder Circuit Breaker for No. 3 bus (A124CB29) or the Left Fuel Pressure Warning Circuit Breaker (A124CB19) would be open.

(d) If the R FUEL PRESS annunciator does not illuminate, the Primary Left Feeder Circuit Breaker for No. 4 bus (A145CB7), the LH Feeder Diode for No. 4 bus (A145CR2), the Secondary Left Feeder Circuit Breaker for No. 4 bus (A124CB30) or the Right Fuel Pressure Warning Circuit Breaker (A124CB20) would be open.

(16) After correcting the open component, repeat the test.

(17) Attempt to operate the No. 2 inverter. It SHOULD NOT operate. If the No. 2 inverter DOES operate, at least one of the dual bus feeder diodes is shorted. Detection of the shorted diode (diodes) may be accomplished by opening the following dual bus feeder circuit breakers one at a time: RH No. 1; RH No. 2, RH No. 3 and RH No. 4. Observe which circuit breaker the opening of which, stops the inverter. The corresponding diode will be shorted. After replacement of the shorted diode, repeat the test.

(18) Remove all electrical power from the airplane.

(19) Replace the Right Isolation Limiter.

2. DUAL BUS CONFORMITY

A dual bus conformity inspection should be performed whenever modifications to the Right Circuit Breaker Panel Assembly or the Left Inboard Subpanel bus system have been made, or whenever repairs have been made that may result in the dual bus loops being interconnected or open. Either the airplane battery or an auxiliary ground power unit may be used when conducting this inspection. A suggested inspection procedure follows:

A. Inspection

Perform each Step of the following procedure and closely observe the results. Check for open circuits if voltage is not present where the procedure indicates voltage should be present. Check for improper connections of wires or buses and for interconnection of the bus loops at such places as switch and circuit breaker terminals, bent bus bars, etc. if voltage should not be present.

The presence or absence of voltage in the following Steps may be determined either by the ability or inability to operate circuits in the dual bus loops under test, or by the absence or presence of voltage as measured by a voltmeter. Certain circuits have been chosen as indicators because of ease of operation Other circuits may be chosen after reference to the proper circuit diagram in the Wiring Diagram Manual.

(1) Turn the battery master switch ON. Open the LH and RH No. 1, LH and RH No. 2, LH and RH No. 3, and LH and RH No. 4 dual bus feeder circuit breakers.

(2) Close the RH No. 1 dual bus feeder circuit breaker. The L BL AIR FAIL annunciator readout SHOULD illuminate. The R BL AIR FAIL, L FUEL PRESS and R FUEL PRESS annunciator readouts SHOULD NOT illuminate.

(3) Open the RH No. 1 and close the LH No. 1 dual bus feeder circuit breakers. L BL AIR FAIL annunciator readout SHOULD illuminate. The RH BL AIR FAIL, L FUEL PRESS and R FUEL PRESS annunciator readouts SHOULD NOT illuminate. The correct indication ensures continuity of the No. 1 bus, and isolation from the No. 2, No. 3 and No. 4 buses.

(4) Open the LH No. 1 and close the RH No. 2 dual bus feeder circuit breakers. The R BL AIR FAIL annunciator readout SHOULD illuminate. The L BL AIR FAIL, L FUEL PRESS and R FUEL PRESS annunciator readouts SHOULD NOT illuminate.

(5) Open the RH No. 2 and close the LH NO. 2 dual bus feeder circuit breakers. The R BL AIR FAIL annunciator readout SHOULD illuminate. The L BL AIR FAIL, L FUEL PRESS and R FUEL PRESS annunciator readouts SHOULD NOT illuminate. The correct indication ensures continuity of the No. 2 bus, and isolation from the No. 1, No. 3 and No. 4 buses.

(6) Open the LH No. 2 and close the RH No. 3 dual bus feeder circuit breakers. The L FUEL PRESS annunciator readout SHOULD illuminate. The L BL AIR FAIL, R BL AIR FAIL and R FUEL PRESS annunciator readouts SHOULD NOT illuminate.

(7) Open the RH No. 3 and close the LH No 3 dual bus feeder circuit breakers. The L FUEL PRESS annunciator readout SHOULD illuminate. The L BL AIR FAIL, R BL AIR FAIL and R FUEL PRESS annunciator readouts SHOULD NOT illuminate. The correct indication ensures continuity of the No. 3 bus, and isolation from the No. 1, No. 2 and No. 4 buses.

(8) Open the LH No. 3 and close the RH No. 4 dual bus feeder circuit breakers. R FUEL PRESS annunciator readout SHOULD illuminate. The L BL AIR FAIL, R BL AIR FAIL and L FUEL PRESS annunciator readouts SHOULD NOT illuminate.

(9) Open the RH No. 4 and close the LH No. 4 dual bus feeder circuit breakers. The R FUEL PRESS annunciator readout SHOULD illuminate. The L BL AIR FAIL, R BL AIR FAIL and L FUEL PRESS annunciator readouts SHOULD NOT illuminate. The correct indication ensures continuity of the No. 4 bus, and isolation from the No. 1, No. 2 and No. 3 buses.

(10) Turn the battery master switch OFF and close all circuit breakers.

3. ELECTRICAL LOAD UTILIZATION TABLES

The following Tables provide information pertaining to the capacity of the generator for supplying the electrical load on the airplane while maintaining a full charge on the battery. Since the airplane is equipped with two 30 volt, 250 ampere starter-generators, the total load shall not exceed 90% (450 amperes) of the total generating capacity. When an item of equipment functions at various times in different systems, the load per unit value listed in the Table represents the highest value required to operate that particular unit in the various systems in which it functions.

A. Electrical Load Utilization

Table 201
Aircraft Loads - A.C.

EQUIPMENT		Unit No.	OPER. RANGE FREQ.			Loads Rqd Per Unit Total Watts	Total VARS	Power Factor
Description	Identification		Volt	Hz	VA			
Inv Warn Relay	50-380048-7	1	115	400	12.00	9.60	7.20	0.80
L Torque Press Ind	100-384048-3	1	25	400	12.00	2.16	11.70	0.18
R Torque Press Ind	100-384048-3	1	25	400	12.00	2.16	11.70	0.18
Volt Freq Meter	101-384078-1	1	115	400	0.84	0.81	0.20	0.97
ADF #1	ADA-650	1	115	400	6.50	1.30	6.37	0.20
RMI #1	KI-226	1	115	400	13.00	13.00	0	1.00
RMI #2	KI-226	1	115	400	13.00	13.00	0	1.00
Compass #1	C-14-43	1	115	400	40.00	32.00	24.00	0.80
Autopilot	SPZ-200B	1	115	400	40.00	32.00	24.00	0.80
Flight Director	SPI-400	1	115	400	11.5	10.35	5.00	0.90
Gyro Horizon	GH-14	1	26	400	33.00	31.68	9.24	0.96
Total AC Load								

POWER FACTOR FOR THIS INSTALLATION IS 0.83

FORMULA USED: $pt = \frac{Watts}{\sqrt{(WATTS)^2 + (VARS)^2}}$

B. Electrical Load Utilization U-50, U-165 and After

DC Equipment Load Requirements

LOAD SOURCE	CRUISE LOAD (AMP.)	LOAD SOURCE	CRUISE LOAD (AMP.)
Hot Bus	0.51	L Generator Bus	14.625
Cross Tie Bus	0	R Generator Bus	30.005
No. 1 Duel Fed Bus	42.47	No.1 Avionics Bus	15.80
No. 2 Duel Fed Bus	35.65	No.2 Avionics Bus	15.28
No. 3 Duel Fed Bus	1.60		
No. 4 Duel Fed Bus	1.91		
Total Airplane Load			**153.56**

DC Equipment Load Requirements

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
HOT BUS				
Entry and Loading Lights	2	0.52		3
FWD Baggage Light	1	0.52		3
Pilots Clock Light	1	0.01	0.01	
Battery Relay	1	0.50	0.50	
External DC Sensor	1	0.12		3
TOTAL			**0.51**	
CROSS TIE BUS				
Ldg Gr Motor	1	150.0	0	7
Normal Elec Heat	1	35.7	0	8
Max Elec Heat	1	35.7	0	8, 3
TOTAL			**0**	
NO. 1 DUAL FED BUS				
Lower Beacon	1	3.00	3.00	
Upper Beacon	1	3.00	3.00	
*L- Ldg Light	2	8.96	0	7, 14
*L Pitot Heat	1	4.29	4.29	
Prop De-Ice Timer	1	0.44	0	8
Prop De-Ice Boots	1	18.00	0	8
L Chip Det Warn Light	1	0.04	0	9
Control Amp	2	0.05	0.10	
Fire Ext Handle Light	4	0.04	0	
L Pwr Ctrl Cable Heat	1	2.00	2.00	
L Airline Heat	1	6.00	6.00	
L Lip De-Ice Ctrl Relay	1	0.35	0	8
L Lip De-Ice Test Relay	1	0.09	0	8
L Oil Press Warn Light	2	0.04	0	9
L Oil Temp Ind	1	0.10	0.10	

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
NO. 1 DUAL FED BUS (Continued)				
L Oil Press Ind	1	0.10	0.10	
L Fuel Flow Transmitter	1	0.36	0.36	
Avionics Master Relay	2	0.35	0	12
Inv Out Warn Light	2	0.04	0	9
Ann Master Warn Light	8	0.04	0	9
Ann Master Caut Light	8	0.04	0	9
Ann Module Power		0.18	0.18	
Ldg Gr Warn Flasher	1	0.30	0	7, 14
Ldg Gr Warn Horn	1	0.20	0	7, 14
Silence Relay	1	0.04	0	7, 14
L Bleed Air Fail Warn	2	0.04	0	9
Stall Warn Horn	1	0.20	0	
Stall Warn Trans	1	0.30	0.30	
Stall Warn Heat Rly	1	0.09	0	3
L Nav Light	1	0.89	0.89	
R Nav Light	1	0.89	0.89	
Tail Nav Light	1	1.02	1.02	
Ovhd Flood Light	1	0.30	0.30	
Sidepanel Lts Dim Ctrl	1	0.04	0.04	
R CB Panel Lights	78	0.024	1.87	
Ovhd Panel Lt Dim Ctrl	1	0.04	0.04	
Ovhd Panel Lts	34	0.024	0.82	
Fuel Qty Ind Lts	2	0.048	0.10	
Fuel Panel Lts	50	0.024	1.20	
Ind Instr Lts Dim Ctrl	1	0.04	0.04	
Subpanel Lt Pedestal Lt	126	0.024	3.02	
Pilots Gyro Instr Lt	4	0.04	0.16	
Copilots Gyro Instr Lt	4	0.04	0.16	
Pilots Flt Instr Lt	9	0.04	0.36	
Copilots Flt Instr Lt	8	0.04	0.32	

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
NO. 1 DUAL FED BUS (Continued)				
OAT Lt	1	0.04	0.04	10
Mag Comp Lt	1	0.04	0.04	
Pilots Flt Instr Lt Dim	1	0.04	0.04	
Copilots Flt Instr Lt Dim	1	0.04	0.04	
L Bleed Air Valve	1	1.00	1.00	
L Bleed Air Off Lt	2	0.04	0	9
L Pneu Bleed Air Valve	1	0.80	0	20
A Up Rly Stab Trim	1	0.50	0.50	22
A Dn Rly Stab Trim	1	0.50	0	22
B Up Rly Stab Trim	1	0.35	0.35	22
B Dn Rly Stab Trim	1	0.35	0	22
Main Trim Release Rly	1	0.50	0	
Main Trim Motor	1	4.00	4.00	
Trim Ind Ctrl Box	1	0.15	0.15	
Ldg Gr & Trim Warn Rly	1	0.09	0	7
Ldg Gr & Trim Warn Horn	1	0.20	0	7
Pilots Turn & Slip Ind	1	0.25	0.25	
26 VAC Select Rly	1	0.09	0.09	
115 VAC Select Rly	1	0.09	0.09	
Inv Relay	1	0.35	0.35	
L Field Gnd Relay	1	0.50	0	19
* Stall Warn Heat	1	3.00	3.00	8
Prop Sync Kit	1	1.00	1.00	
Prop Sync Caution Lt	2	0.04	0	9
L Fire Ext Monitor	1	0.02	0	
L Fire Ext	1	3.00	0	
L Fire Ext Switch Lt	2	0.04	0	
Tail Flood Light	2	2.68	5.36	
Pilots Anti-Ice Rly	1	0.35	0	8
Windshield Anti-Ice Ctrl	1	0.02	0	8

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
NO. 1 DUAL FED BUS (Continued)				
Deice Dist Valve	1	1.10	0	8
L Brake Deice On Lt	2	0.04	0	8
TOTAL No. 1 Duel Fed Bus				
NO. 2 DUAL FED BUS				
L Autofeather Rly	1	0.04	0	7
R Autofeather Rly	1	0.04	0	7
L Autofeather Dump Valve	1	1.20	0	7
R Autofeather Dump Valve	1	1.20	0	7
L Autofeather Lt	2	0.04	0	9
R Autofeather Lt	2	0.04	0	9
R Chip Det Warn Lt	2	0.04	0	9
R Fuel Pwr Ctrl Heat	2	2.00	4.00	
R Airline Heat	1	6.00	6.00	
R Lip De-Ice Ctrl Rly	1	0.35	0	8
R Lip De-Ice Test Rly	1	0.09	0	8
R Oil Press Warn Lt	2	0.04	0	9
R Oil Temp Ind	1	0.10	0.10	
R Oil Press Ind	1	0.10	0.10	
R Fuel Flow Trans	1	0.36	0.36	
Engine Instr Lt	24	0.04	0.96	
Pilots Clock Lt	1	0.04	0.04	
Pilots Map Lt	1	0.30	0.30	
Copilots Map Lt	1	0.30	0.30	
Avionis Panel Lt	40	0.024	0.96	
Eng Instr Lt Dim Ctrl	1	0.04	0.04	
Radio Lt Dim Ctrl	1	0.04	0.04	
Ind Instr Lts	9	0.17	1.53	
Volt Loadmeter Lt	4	0.024	0.10	
Subpanel Lt Dim Ctrl	1	0.04	0.04	

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
NO. 2 DUAL FED BUS (Continued)				
Dome Lt	5	0.52	2.50	
Cabin Sign Lt	10	0.04	0.40	
Ordinance Warn Chime	1	0.18	0.18	
Reading Lt	15	0.30	4.50	
Batt Chg Mon Module	1	0.02	0.02	
Batt Chg Caution Lt	2	0.04	0	9
R Bleed Air Fail Lt	2	0.04	0	9
Ldg Gr Dn Lock Lt	4	0.04	0	7
Ldg Gr Handle Lt	2	0.04	0	7
Ann Dim Ctrl	1	0.02	0.02	
R Bleed Air Valve	1	1.00	1.00	
R Bleed Air Off Lt	2	0.04	0	
R Pneu Bleed Air Valve	1	0.80	0	20
Speed Ctrl Rly	1	0.35	0	
Vent Blower Pwr Rly	1	0.35	0.35	
Cabin Temp Ctrl	1	0.35	0.35	
Temp Ctrl Rly	3	0.04	0	8
Bleed Air Bypass	1	2.00	2.00	
Cabin Temp Ind	1	0.10	0.10	
Over Temp Warn Lt	2	0.04	0	9
Stby Trim A Up Rly	1	0.09	0	22, 15
Stby Trim A Dn Rly	1	0.09	0	22, 15
Stby Trim B Up Rly	1	0.09	0	22, 15
Stby Trim B Dn Rly	1	0.09	0	22, 15
Stby Trim Motor	1	2.00	0	15
Nose Gr Strg Disc Act	1	2.00	0	7
Stab Pos Ind	1	0.01	0.01	
No. 2 Inv Pwr Rly	1	0.35	0	16
R Field Gnd Rly	1	0.50	0	13
Windshield Wiper Motor	1	5.2	0	8

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
NO. 2 DUAL FED BUS (Continued)				
Liquid Level Sensor	1	0.03	0.03	
Ldg Gr Sel Valve	1	2.00	2.00	
Ldg Gr Dn Lock Sol	1	0.09	0	7
Ldg Gr Rem Lim Rly	1	0.60	0	14
Hyd Fluid Low Lt	2	0.04	0	9
R Pitot Heat	1	4.29	4.29	
R Ice Lt	1	1.43	0	8
L Ice Lt	1	1.43	0	8
Taxi Lt	1	8.93	0	3
R Ldg Lt	2	8.93	0	14
Brake Deice Ctrl Mod	1	0.03	0	8
L Brake Deice Valve	1	0.80	0	8
R Brake Deice Valve	1	0.80	0	8
R Fire Ext Mon	1	0.02	0	
R Fire Ext	1	3.00	0	
R Fire Ext Switch Lt	2	0.04	0	
Strobe Lt Pwr Sup	1	3.50	3.50	
R Brake Deice On Lt	2	0.04	0	8
Copilots Clock Lt	1	0.04	0.04	
Norm Elec Heat Rly	1	0.60	0	8
Aft Evap Blwr Rly	1	0.35	0	8
TOTAL No. 2 DUEL FED BUS			**35.65**	
NO. 3 DUAL FED BUS				
L Firewall Valve	1	2.00	0	
L Boost Pump	1	8.50	0	11
L Transfer Valve	1	1.42	1.42	
L Nac Not Full Lt	2	0.04	0	9
L Fuel Qty Ind	1	0.03	0.03	
Fuel Qty Sel Rly	1	0.15	0.15	

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
NO. 3 DUAL FED BUS (Continued)				
L Fuel Press Lt	2	0.04	0	9
L No Transfer Lt	2	0.04	0	9
Fuel X-Feed Lt	1	2.00	0	21
Fuel X-Feed Valve	2	0.04	0	9
Prop Speed Reset Sol	2	0.66	0	
L Prop Deice Boots		18.00	0	8
L Ignitor	1	1.00	0	2, 13
L Ignition Lt	2	0.04	0	2, 13
L Ignitor Pwr Rly	1	0.09	0	2, 13
L Purge Valve	1	0.09	0	2, 13
L Gen Ctrl Rly	1	0.10	0	2, 13
L Start Rly	1	4.50	0	2, 13
Start Ctrl Rly	1	0.04	0	2, 13
Flap Motor	1	28.0	0	7
TOTAL No. 3 DUEL FED BUS			**1.60**	
NO. 4 DUAL FED BUS				
R Firewall Valve	1	2.00	0	
R Boost Pump	1	8.50	0	11
R Transfer Valve	1	1.42	1.42	
R Nac Not Full Lt	2	0.04	0	9
R Fuel Qty Ind	1	0.03	0.03	
R Fuel Qty Sel Rly	1	0.15	0.15	
R Fuel Press Lt.	2	0.04	0	9
R No Fuel Transfer Lt	2	0.04	0	9
Prop Deice Override Rly	2	0.50	0	8
R Prop Deice Boots		18.00	0	8
R Ignitor	1	1.00	0	2, 13
R Ignition On Lt	2	0.04	0	2, 13
R Ignitor Pwr Rly	1	0.09	0	2, 13

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
NO. 4 DUAL FED BUS (Continued)				
Purge Valve	1	0.09	0	2, 13
R Gen Ctrl Rly	1	0.10	0	2, 13
R Starter Rly	1	4.50	0	2, 13
Starter Ctrl Rly	1	0.04	0	2, 13
Flap System Con	1	0.12	0.12	
Flap Motor Rly	1	1.25	0	7
Flap Pos Feedback Pot	1	0.06	0.06	
Flap Ctrl Handle Pot	1	0.06	0.06	
Flap Pos Transmitter	1	0.06	0.06	
Flight Hour Meter	1	0.01	0.01	
TOTAL No. 4 DUEL FED BUS			**1.91**	
L GENERATOR BUS				
L Tip Heater	1	17.55	0	8
L Gen Loadmeter	1	0.005	0.005	
Cigarette Lighter	1	6.20	6.20	
No. 1 Inverter	1	16.00	8.42	
Hot Gas Bypass Sol	1	1.20	0	8
Cond Actuator Motor	1	1.00	0	8
N_1 Speed Switch	1	0.01	0	8
Cond Blower Rly	1	0.35	0	8
Air Cind Clutch	1	1.80	0	8
Cond Blower Motor	1	55.00	0	3
Aft Evap Blower Motor	1	20.00	0	8
Pilots Wshld Anti-Ice	1	28.30	0	8
Copilots Anti-Ice Rly	1	0.35	0	8
TOTAL L GENERATOR BUS			**14.625**	

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
R GENERATOR BUS				
Vent Blower Motor	1	30.00	30.00	
R Generator Loadmeter	1	0.005	0.005	
R Lip Heater	1	17.55	0	8
No. 2 Inverter	1	16.00	0	5
Copilots Wshld Anti-Ice	1	28.3	0	
TOTAL R GENERATOR BUS			**30.005**	
NO. 1 AVIONICS BUS				
No. 1 Comm	1	0.50	0.50	
No. 1 Nav	1	0.80	0.80	
Audio, Pilot	1	1.48	1.48	
No.1 Glideslope	1	0.51	0.51	
No.1 DME	1	1.10	1.10	
Autopilot	1	7.85	7.85	
Flt Director	1	2.06	2.06	
No.1 Transponder	1	1.50		
RNAV	1	0.65	0.65	
RNAV Blower Motor	1	0.55	0.55	
Radio Relay	1	0.30	0.30	
TOTAL NO. 1 AVIONICS BUS			**15.80**	
NO. 2 AVIONICS BUS				
No. 2 Comm	1	0.50	0.50	4
No. 2 Nav	1	0.80	0.80	
Audio, Copilot	1	1.48	1.48	
No. 1 ADF	1	1.00	1.00	
No. 1 Marker Beacon	1	1.20	1.20	
No. 2 Glideslope	1	0.51	0.51	
No. 2 Compass	1	2.60	2.60	
Radar	1	3.50	3.50	

DC Equipment Load Requirements (Continued)

EQUIPMENT DESCRIPTION	NO. UNITS	CONT. LOAD AMP. UNIT	CRUISE LOAD (AMP.)	NOTE
NO. 2 AVIONICS BUS (Continued)				
No. 2 Transponder	1	1.50	1.50	
Interphone	1	0.19	0.19	
Paging Amp	1	2.00	2.00	
TOTAL NO. 2 AVIONICS BUS			**15.28**	

* These items have been relocated to the No. 4 Dual Fed Bus on serials U-50, U-201 and after and airplanes in compliance with Service Instruction 1231.

NOTES

1. All resistance loads calculated at 80% of normal bus voltage and lamp loads calculated at 87% full load current.
2. Starting loads are not tabulated in cruise load as they should be ground loads only.
3. Used only on ground.
4. Limit transmissions to three minutes total when operating on battery only.
5. Only one inverter is used at a time.
6. Half of each dual feed bus is supplied by each generator bus when both generators are operating.
7. Used during takeoff and landing only.
8. Cruise load calculated (for night VFR conditions) without air conditioning, radiant heat, or deice equipment.
9. Annunciator warning lights on only when waming or caution legends are energized.
10. Controlled by momentary contact switch.
11. Used only when engine driven boost pump fails.
12. Relays energized when avionics master switch is OFF.
13. Used only during starting.
14. Used five minutes before landing.
15. Standby trim not used while main trim is operational.
16. Only one operates at a time.
17. Annunciator indicator light and battery charge power.
18. Only one transmits at a time.
19. Relay energized when generator switch turned off.
20. Valve energized when bleed air switch turned off.
21. Used only when one engine fails.
22. One A relay and one B relay operates at the same time.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 25 - EQUIPMENT/FURNISHINGS

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EQUIPMENT/FURNISHING
FLIGHT COMPARTMENT
MAINTENANCE PRACTICES

1. SEAT

A. Removal

CAUTION: *Before removing the pilot's seat, lower the seat to its lowest position and reposition the left armrest so it will not strike and/or damage the components of the fuel control panel during seat removal.*

(1) Remove the seat stops at the end of each track.

(2) Release the fore and aft adjustment lock.

(3) Move the seat forward until it clears the mounting tracks.

B. Installation

(1) Place the seat in position and align the seat guides with the mounting tracks.

CAUTION: *Before sliding the seat into place, lower the seat to its lowest position so the armrest will not strike the fuel control panel during installation of the seat.*

(2) Release the fore and aft adjustment lock and slide the seat into the desired position. Engage the fore and aft lock, making certain it holds the seat securely in place.

(3) Replace the seat stops on the end of each track.

2. SEAT TRACK

A. Wear Limits

Excessive wear of the seat attach components and seat tracks will cause the crew seat to become unstable. Wear limits of the components have been established to determine when component replacement is necessary. These components include the seat channel on the seat base, seat lock pins and the seat tracks. The maximum wear limits are illustrated in Figure 201. Replacement of worn parts must be accomplished if the limits are exceeded.

(1) The seat base must be replaced if measurements of the seat channel are less than the minimum dimensions in Detail A.

(2) Replace the lock pins if the diameter is less than 0.220 inch or pin engagement into the seat track is less than 0.11 inch as shown in Detail D.

(3) The seat tracks must be replaced if measurements of the tracks are less than the minimum dimensions shown in Figure 201. A slight gouge in the top surface (Detail B), caused by the lock pin, is not detrimental to the operation of the seat. The lock pin hole may be elongated in a forward and aft direction to a maximum of 0.315 inch long, measured at the 0.06 inch depth, before track replacement is necessary. The diameter of the holes must not exceed 0.270 inch below 0.06 inch depth.

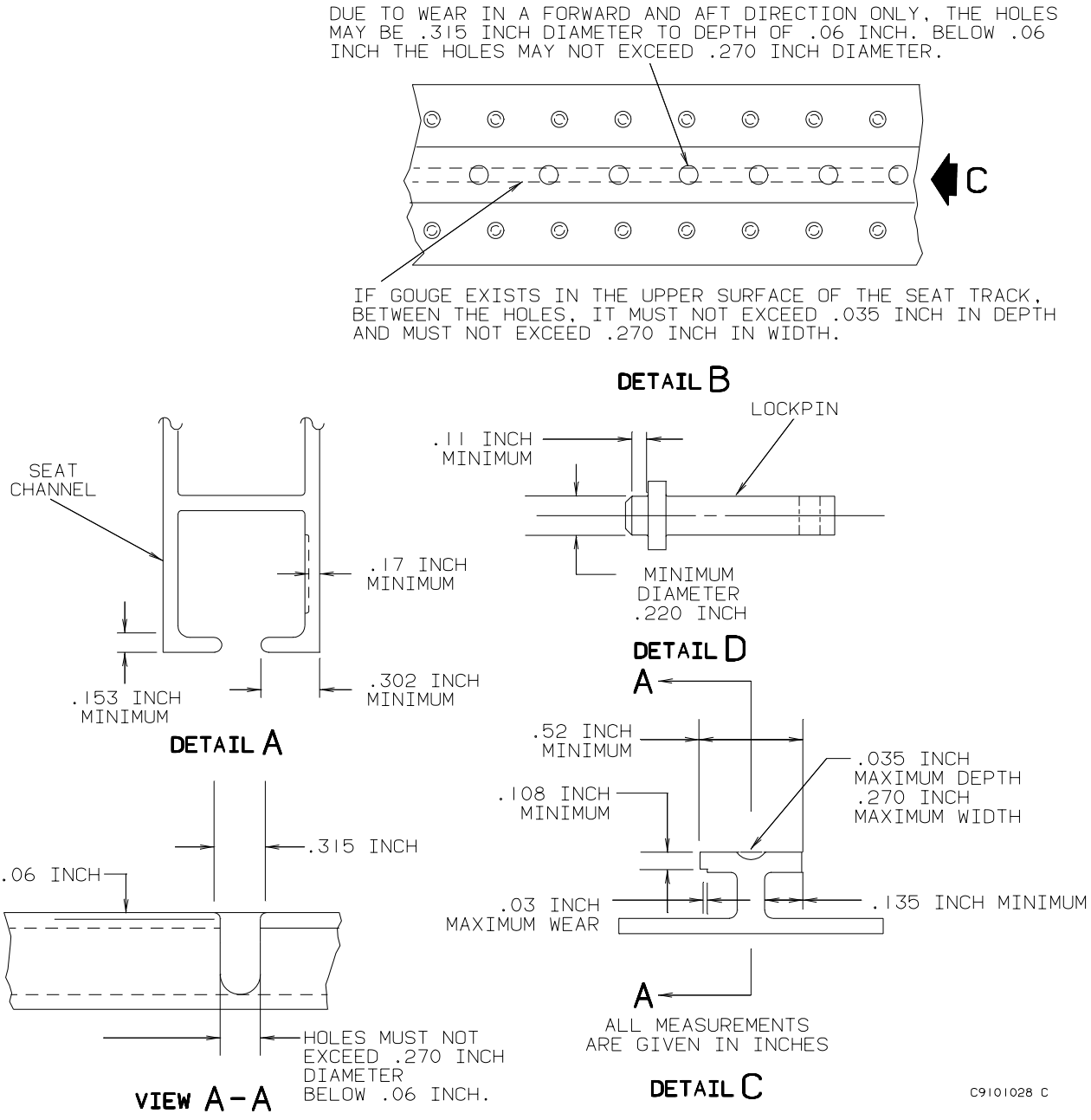

Figure 201
Flight Compartment Seat Track Wear Limits

EQUIPMENT/FURNISHING
PASSENGER COMPARTMENT
MAINTENANCE PRACTICES

1. SEAT (ELEVEN FORWARD SEATS)

A. Removal

(1) Using a screwdriver, simultaneously lift the two locking pins upward until they clear the seat track.

(2) Move the seat forward or aft approximately one inch to allow clearance for the four guide pins.

(3) Lift the seat from the seat tracks.

B. Installation

(1) Place the seat in the desired position on the seat track and place the four guide pins in the seat track gutter.

(2) Move the seat forward or aft approximately one inch, and remove the tool to allow the two locking pins to engage the seat track gutter.

2. FOLD UP SEAT

A. Removal

(1) Remove the screws at the forward and aft chair/sidewall attach brackets at the RH sidewall.

(2) At the bottom of the chair legs (four places), press in on the button of the retainer, rotate 90° and lift the seat assembly from the floor mounting brackets.

NOTE: If the airplane is to be used in a cargo configuration, disengage the seat legs from the floor mounting brackets as outlined in Step (2) of PASSENGER SEAT REMOVAL (FOLD UP). Fold the seats to the sidewall and secure with the seat retaining strap.

B. Installation

(1) Position the chair legs (four places) into the floor mounting brackets, press in on the button of the retainer and rotate 90°. Release the button to lock the seat in place.

(2) Install the screws at the forward and aft chair/sidewall attach brackets at the RH sidewall.

3. AFT CABIN FOLD UP SEAT

A. Removal

(1) Remove the two screws from the chair/sidewall attach bracket.

(2) At the bottom of the chair legs (two places), lift and rotate the retainer. Lift the seat assembly from the floor mounting brackets.

NOTE: If the airplane is to be used in a cargo configuration, disengage the seat legs from the floor mounting brackets as outlined in Step (2) of PASSENGER SEAT REMOVAL (AFT CABIN FOLD UP). Fold the seats to the sidewall and secure with the seat retaining strap.

B. Installation

(1) Position the chair legs (two places) into the floor mounting brackets and rotate the retainer to lock the seat in place.

(2) At the sidewall, install the screws at the forward and aft chair/sidewall attach bracket.

4. SIDEWALL FOLDING SEAT (U-1 THRU U-164, EXCEPT U-50)

This passenger seating arrangement provides a quick change from passenger to cargo to passenger/cargo arrangement. The folding seat is mounted to a floor track in the floorboard and to side mounting rings in the bulkhead. The seat is held to its floor mounting track by a plunger on each leg. Rotating the knurled knob on each leg retracts the plunger permitting the seat to be lifted from the track and folded against the airplane sidewall. The seat can be secured in the folded position by attaching a leather strap to the seat leg and to the equipment ring mounted in the airplane sidewall below the windows.

A. Removal

(1) Remove the connecting bolts from the side mounting rings on the airplane sidewall.

(2) Rotate the plunger retract knobs at the base of the forward and aft seat legs and lift the seat clear.

B. Installation

(1) Place the seat in position and rotate the plunger retract knobs at the base of the forward and aft seat legs.

(2) Install the connecting bolts from the side mounting rings on the airplane sidewall.

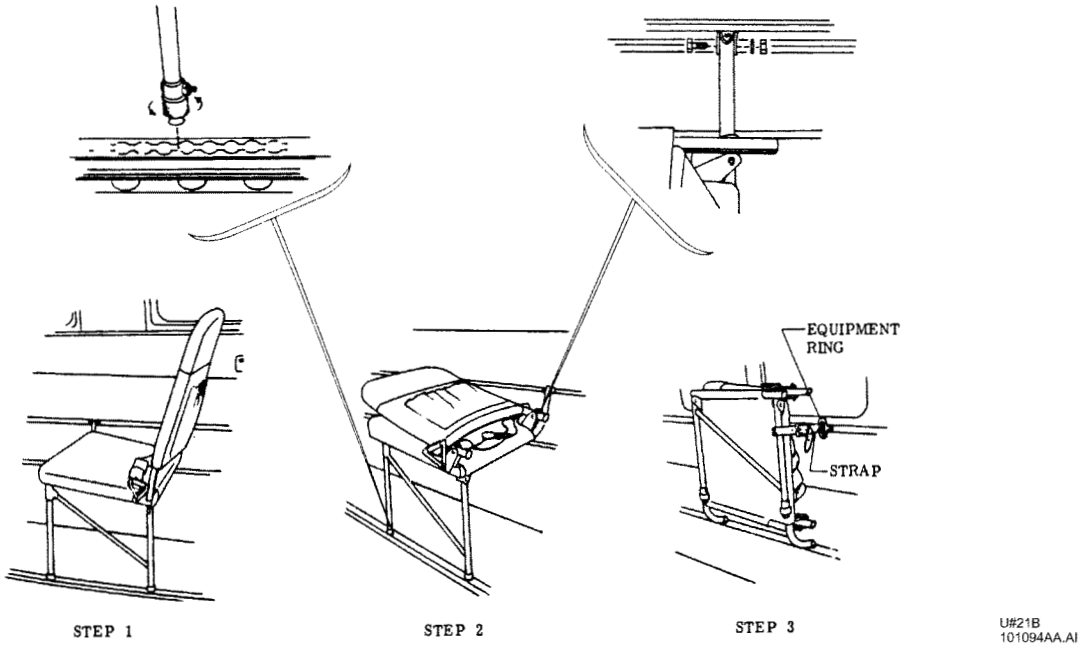

Figure 201
Passenger Seat (Sidewall Folding) (U-1 thru U-164, except U-50)

5. PASSENGER SEAT TRACK

A. Wear Limits

As the seat tracks wear, their strength is reduced. Figure 202 indicates acceptable wear limitations which have been established for the cabin seat tracks.

Replace the seat track if the noted minimum dimensions are not met. Measure the 0.071 inch dimension approximately 0.03 inch in from the edge of the flange.

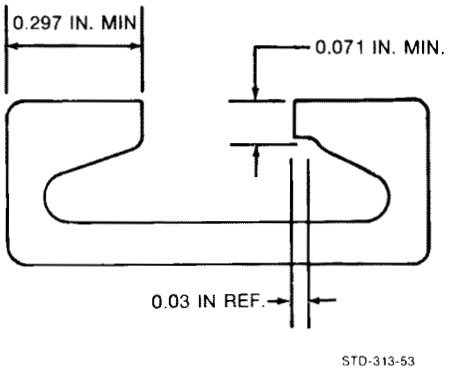

Figure 202
Passenger Seat Track Wear Limits

EQUIPMENT/FURNISHINGS
EMERGENCY
DESCRIPTION AND OPERATION

1. EMERGENCY LOCATOR TRANSMITTER

Airplanes U-50 and U-149 and After are factory equipped with an emergency locator transmitter (ELT) to assist in the tracking and recovery of the airplane and crew in the event of a crash or an emergency landing. Airplanes U-149 thru U-164 are equipped with Collins/Communications Components Corporation ELTs. Narco ELTs are installed on airplanes U-50, U-165 and After and on U-149 thru U-164 equipped with Kit No. 101-3046-1.

The ELT is mounted on the right side of the fuselage at a point just aft of FS 395. An access hole with a spring loaded cover is installed adjacent to the transmitter to provide access for manual activation of the ELT. An antenna for the ELT is mounted on top of the fuselage under the vertical stabilizer at a point just forward of FS 409.50.

The output frequencies of the ELT are 121.5 and 243.0 MHz, simultaneously. Range is approximately line of sight. The ARM-OFF-ON switch is located on the transmitter, and controls the operation of the set. The ON position turns the set on for testing and the ARM position actuates the set to operate automatically upon impact. A reset switch, located on the forward end of the transmitter, resets the transmitter in the event the impact switch is accidentally triggered. Airplanes U-149 thru U-164 with Kit No. 101-3039-1 installed have a remote REARM-ARM-XMIT switch located adjacent to the ELT in the aft fuselage. The XMIT position turns the set on for testing and the ARM position actuates the set to operate automatically upon impact. The REARM position resets the transmitter in the event the impact switch is accidentally triggered. Airplanes with Kit No. 101-3046-1 installed have a remote switch, placarded ARM- XMIT, located adjacent to the ELT in the aft fuselage. The remote ARM-XMIT switch is a momentary switch that enables manual activation of the ELT for testing while the unit is in the airplane.

EQUIPMENT/FURNISHING
EMERGENCY
MAINTENANCE PRACTICES

1. EMERGENCY LOCATOR TRANSMITTER

A. Maintenance

Maintenance on the ELT is normally limited to replacing the battery. The following is a list of the various conditions which warrant battery replacement:

(1) Visual inspection shows signs of leakage, corrosion or unsecure leads.

(2) Elapsed replacement date noted on the battery case (this date represents 50% of the useful life of the battery).

NOTE: The useful life of the battery is the length of time which the battery may be stored without losing its ability to continuously operate the ELT for 48 hours.

(3) After any emergency use.

(4) After one cumulative hour of use.

(5) After operation of unknown duration.

(6) If transmitter is stored in an area where the temperature is normally above 38°C (100°F), the battery life will be shortened.

CAUTION: *Avoid storage of batteries at temperatures in excess of 55°C (130°F).*

The information on battery life and replacement is included in the data furnished with each ELT, and is usually placarded on the battery.

NOTE: Replacement batteries should be obtained only from ELT manufacturers, airplane manufacturers or other acceptable suppliers, since the condition and useful life of over-the-counter batteries, such as those sold for flashlights, portable radios, etc., are usually unknown.

CAUTION: *The ELT switch should not be turned ON unless the ELT is connected to its associated antenna or a 50-ohm dummy load.*

B. Testing

Generally, tests will be performed following maintenance or repair of ELTs, other than battery replacement, to determine their operational capability. Testing the ELT, if improperly done, could trigger false alerts and create frequency jamming, and may interfere with the reception of a bonafide emergency transmission. Federal Communications Commission regulations require that this testing be performed in a screened or shielded test room, or in a test enclosure that will hold the self contained ELT unit with the antenna fully extended.

CAUTION: *The ELT switch should not be turned ON unless the ELT is connected to its associated antenna or a 50-ohm dummy load.*

Operational testing of installed ELTs may be accomplished as follows:

NOTE: Tests should not be longer than three audio sweeps. One audio sweep may be defined as amplitude modulating the carrier with an audio frequency sweeping downward over a range of not less than 700 Hz, within the range 1600 to 300 Hz, and a sweep repetition rate between two and four Hz. Tests should be conducted only in the first five minutes of any hour. If the operational tests must be made at a time not included within the first five minutes after the hour, the tests should be coordinated with the nearest FAA tower or flight service station.

(1) Turn COMM-1 ON and tune to 121.5 MHz.

(2) Turn the COMM-1 audio switch to the SPEAKER position and place the volume control in the center of its range.

(3) Turn the ELT ARM-OFF-ON switch to ON and monitor ELT signal. On airplanes equipped with a remote switch (located in the aft fuselage on the RH side or on the instrument panel), the remote switch may be momentarily flipped to the XMIT position and the ELT signal monitored.

NOTE: A distinctive downward sweeping tone should have been heard from the monitoring receiver during the test. If the tone was heard, the ELT is functioning properly. If there is no audible tone (assuming that the VHF transceiver is operational), the battery is probably disconnected, outdated or discharged.

(4) Place the ARM-OFF-ON switch on the ELT to the OFF position. If the remote switch is being utilized to test the unit, the switch should be released so it will return to the ARM position. The audio signal should disappear completely.

(5) Place the switch in the ARM position. There should be no audio signal present.

NOTE: If the signal is heard, the impact switch has probably been activated and should be reset.

(6) Firmly press the reset switch on the front of the ELT and listen to ensure the audio signal disappears from COMM-1.

2. NARCO BATTERY

A. Replacement

(1) Place the ARM-OFF-ON switch on the ELT in the OFF position.

(2) Disconnect the antenna cable from the ELT. Disconnect the remote switch wiring, if installed, from the terminals on the ELT (Ref. Figure 201).

(3) Unlatch the mounting strap and remove the ELT from the airplane.

(4) Extend the portable antenna.

CAUTION: *Exercise extreme care in extending the portable antenna and handling the control head during the battery replacement to avoid damage to the antenna or the plastic tab on the antenna.*

(5) Remove the four screws attaching the control head to the battery casing and slide the control head and battery case apart. The battery connection leads are approximately 3 inches long.

NOTE: Do not remove the sealant on the inside lip of the battery pack or a water tight seal will not be made when the ELT unit is reassembled.

(6) Disconnect the battery by unsnapping the battery terminals from the bottom of the transmitter PC board. Discard the old battery.

NOTE: Inspect for, and properly treat, any corrosion that may be indicated in the area when the battery is replaced.

WARNING: DO NOT discard the battery in fire.

(7) Connect the terminals of the new battery to the bottom of the transmitter PC board.

(8) Using a stick, apply a bead of sealant (supplied with each battery pack) around the area of the control head which is joined with the battery case when reassembled.

NOTE: This sealant provides a water tight seal when the unit is assembled.

(9) Insert the control head section into the battery case being careful not to pinch the wires, and install the four attaching screws. Wipe any excess sealant from the outside of the unit.

NOTE: If the four screw holes do not line up, rotate the battery case 180° and reinsert.

(10) Stow the portable antenna.

CAUTION: *Exercise extreme care in order to avoid damage to the antenna or the plastic tab on the upper end.*

(11) Install the transmitter in the airplane and secure the mounting strap.

(12) Connect the fixed antenna cable to the ELT. Ensure that the plastic contact separator is inserted between the portable antenna contact and the portable antenna (Ref. Figure 201).

NOTE: With the contact separator not in place a very weak signal may be transmitted. This signal may be strong enough for a functional test but too weak for emergency use.

(13) Connect the remote switch wiring, if installed, to the terminals on the ELT.

(14) Press the RESET button and place the ARM-OFF switch on the ELT in the ARM position.

(15) The new replacement date should be marked on the ELT in a visible area. This will aid in future inspections of the ELT. This date is 50% of the useful life of the battery as defined by the battery manufacturer.

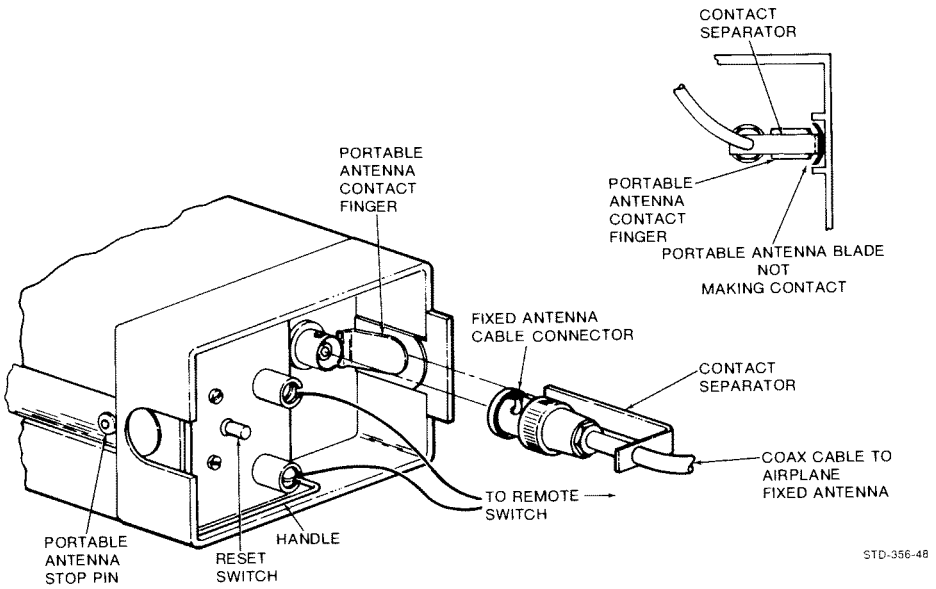

Figure 201
Narco ELT

3. COLLINS/COMMUNICATIONS COMPONENTS CORPORATION BATTERY

A. Replacement

NOTE: The Narco ELT may be installed by Kit No. 101-3046 into airplanes originally equipped with units manufactured by Collins/ Communications Corporation.

(1) Place the ARM-OFF-ON switch on the ELT in the OFF position.

(2) Disconnect the antenna cable and the remote switch wiring, if installed, and remove the ELT from the airplane.

(3) Remove the screws which hold the mounting base on the transmitter and remove the base.

(4) Remove the old battery and disconnect the electrical connector. Discard the old battery.

WARNING: DO NOT discard the battery in fire.

NOTE: Inspect for, and properly treat, any corrosion that may be indicated in the area when the battery is replaced.

(5) Connect a fresh battery and install it in the compartment.

(6) Replace the base and screws.

(7) Install the transmitter in the airplane and attach the antenna cable and remote switch wiring, if installed.

(8) The new replacement date should be marked on the ELT in a visible area. This will aid in future inspections of the ELT. This date is 50% of the useful life of the battery as defined by the battery manufacturer.

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	101 and 102	Nov 1/10
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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 26 - FIRE PROTECTION

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FIRE PROTECTION
DETECTION
DESCRIPTION AND OPERATION

1. FIRE AND SMOKE DETECTION

A. U-1 thru U-164, except U-50

To provide immediate alarm in the event of fire in the engine or nose avionics compartment, a fire and optional smoke detector system is installed (Ref. Figure 1). The basic fire detection system consists of three photoconductive cells in each engine nacelle, two control amplifiers under the center aisle floor just aft of the main spar, two indicator lights on the fire extinguisher T-handles, a test switch on the left subpanel and a circuit breaker on the INDICATORS AND WARNING subpanel on the copilot's right. Optional smoke detection equipment, if installed, includes a smoke detector in the nose avionics compartment and an amplifier adjacent to the fire detector amplifiers in the fuselage.

The flame detectors, sensitive to infrared rays, are positioned in the engine compartments to receive direct and reflected rays, thus viewing the entire compartment with only three cells. Heat level and rate of heat rise are not factors in the sensing method. The cell emits an electrical signal whose potential is proportional to the infrared intensity and ratio of the radiation striking the cell. To prevent stray light rays from signaling a false alarm, a relay in the control amplifier closes only when the signal reaches a preset alarm level. When the relay closes, the lights in the appropriate warning T-handle illuminate. After the fire is extinguished, the cell output voltage drops below the alarm level and the relay in the control amplifier opens. No manual resetting is required to reactivate the detector system.

The smoke detector contains a constantly burning light and a photoconductive cell enclosed in a perforated case. Smoke particles entering the case reflect infrared rays from the light into the cell, which transmits a signal to the smoke detector amplifier. The potential of this signal is proportional to the density of the smoke. When the signal strength is sufficient to close the relay in the amplifier, the light on the annunciator panel lights up the word SMOKE.

The test switch on the subpanel has positions to test each of the detectors. The system may be tested any time on the ground or in flight by rotating the switch through the various positions. The three test positions activate one corresponding set of flame detectors in each nacelle. The SMOKE position will test the smoke detector in the avionics compartment. Failure of a light to glow in any one position indicates trouble in that particular detector circuit rather than a trouble affecting all detectors.

B. U-50, U-165 and After

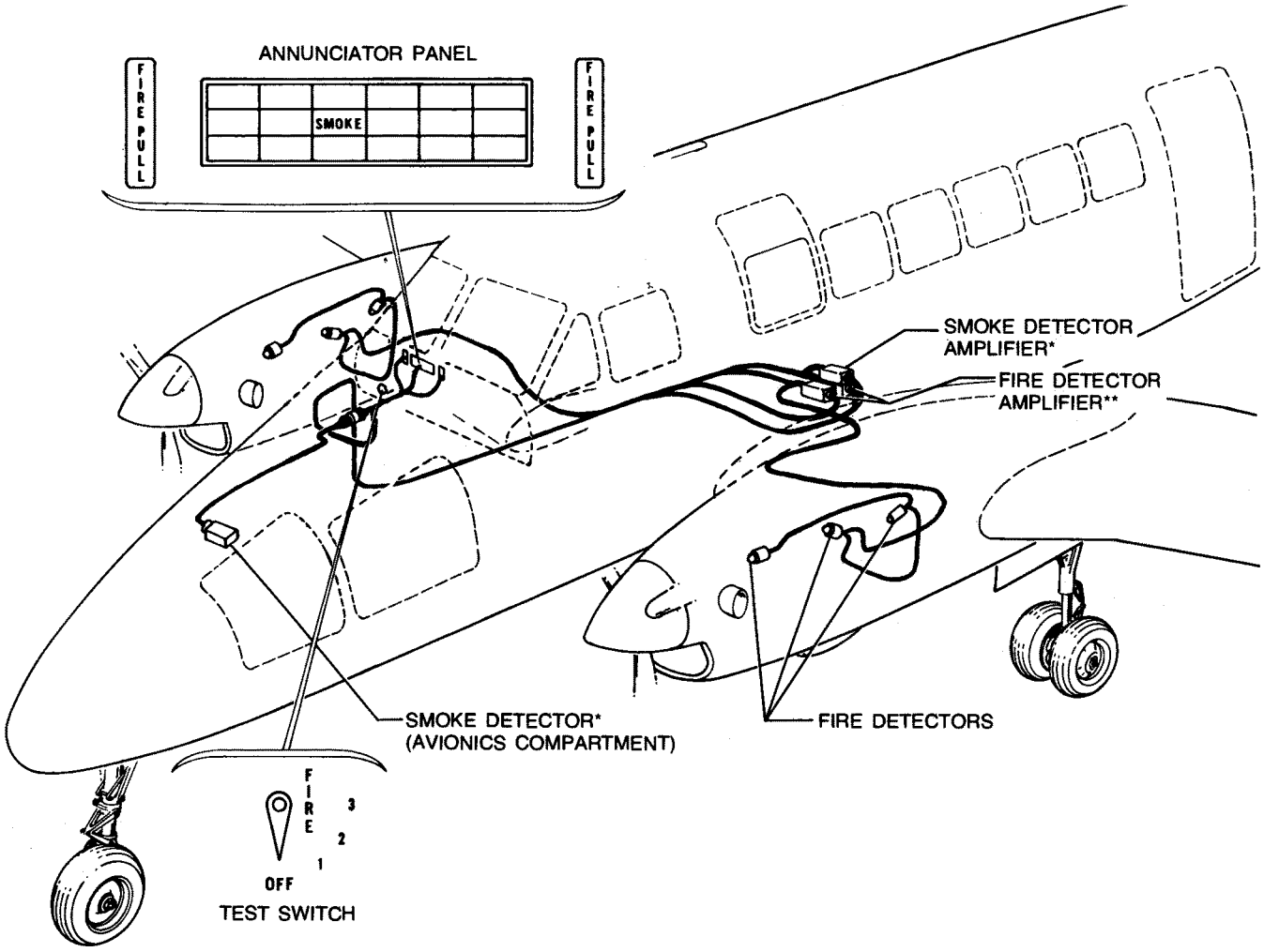
NOTE: On U-50 and U-165 and After, the smoke detection system is not offered.

A fire detection system is installed to provide immediate alarm in the event of a fire in either engine. The fire detection system consists of a forward, upper and lower photoconductive cell in each engine nacelle. In addition, the fire detector system contains two control amplifiers located under the floorboards between the pilot's and copilot's seats just aft of FS 143.000, which is approximately even with the aft edge of the pilot's and copilot's side windows. The system contains indicator lights on each of the fire extinguisher T-handles, a test switch on the copilot's subpanel, a circuit breaker on the copilot's sidewall circuit breaker panel.

The flame detectors, sensitive to infrared rays, are positioned in the engine compartments to receive direct and reflected rays, thus viewing the entire compartment with only three cells. Heat level and rate of heat rise are not factors in the sensing method. The cell emits an electrical signal whose potential is proportional to the infrared intensity and ratio of the radiation striking the cell. To prevent stray light rays from signaling a false alarm, a relay in the control amplifier closes only when the signal reaches a preset

alarm level. When the relay closes, the appropriate fire extinguisher T-handle illuminates. After the fire is extinguished, the cell output voltage drops below the alarm level and the relay in the control amplifier opens. No manual resetting is required to reactivate the detector system.

The test switch on the copilot's subpanel has three positions. Each position of the switch corresponds to one flame detector in both engine nacelles. Position 1 corresponds to the lower rear detector in each nacelle, position 2 to the upper rear detector in each nacelle and position 3 to the forward detector in each nacelle. The system may be tested any time on the ground or in flight by rotating the test switch through the various positions. Failure of the light to glow in any one position indicates trouble in that particular detector circuit, rather than a trouble affecting all detectors.



* THE SMOKE DETECTOR SYSTEM IS NOT OFFERED ON U-50, U-165 AND AFTER.

** ON U-50, U-165 AND AFTER, THE CONTROL AMPLIFIERS ARE LOCATED JUST AFT OF F.S. 143.00, UNDER THE FLOORBOARDS BETWEEN THE PILOT'S AND COPILOT'S SEATS.

99-482-8

Figure 1
Fire and Smoke Detection System

FIRE PROTECTION
DETECTION
TROUBLESHOOTING

1. DETECTION

A. Fire Detection System

Table 101
Troubleshooting Fire Detection System

PROBLEM	PROBABLE CAUSE	REMARKS
(1) Failure of all detectors to test.	a. Warning lamp inoperative.	a. Check circuit continuity and warning bulb.
	b. Defective control amplifier.	b. Replace control amplifier.
(2) Failure of one or more detectors to test.	a. Faulty detector wiring.	a. Check detector white leads to control amplifier and power lead to detector red lead.
	b. Defective detector.	b. Replace detector.
	c. Defective control amplifier.	c. Replace control amplifier.
(3) Failure to recycle after test.	a. 28 volt leakage to signal lead.	a. Check control amplifier signal connector pins A, B, and C for more than 7 volts.
	b. Open detector ground.	b. Check for open in blue lead to ground.
	c. External light causing alarm signal.	c. Check for presence of light bulbs or direct view of setting sun.
	d. Defective detector.	d. Replace detector.
(4) Continuous or intermittent alarm.	a. 28 volt leakage to signal lead.	a. Check control signal connector pins A, B, and C for more than 7 volts.
	b. Open detector ground.	b. Check for open in blue lead to ground.
	c. 28 volt leakage to alarm circuit.	c. Check control amplifier power connector pin C.
	d. Defective control amplifier.	d. Replace control amplifier.
	e. Defective detector.	e. Replace detector.

B. Smoke Detection System (U-1 thru U-164, except U-50)

Table 102
Troubleshooting Smoke Detection System (U-1 thru U-164, except U-50)

PROBLEM	PROBABLE CAUSE	REMARKS
(1) Failure to test.	a. Warning lamp inoperative.	a. Check circuit continuity and warning bulb.
	b. Defective test lamp or beacon lamp in detector.	b. Replace required item.
	c. Faulty wiring.	c. Repair wiring.
	d. Defective detector.	d. Replace detector.
	e. Defective control amplifier.	e. Replace control amplifier.
(2) Failure to recycle after test.	a. 28 volt leakage to signal lead.	a. Check detector pin C for more than 10 volts.

FIRE PROTECTION
DETECTION
MAINTENANCE PRACTICES

1. PROCEDURES

CAUTION: *It is not permissible to splice fire detector wiring. The complete fire detector pigtail must be installed when the unit is replaced.*

A. Functional Test (Bulb or Kit)

The flame detectors can be functionally checked by exposing them to an infrared light source such as a 250 watt infrared bulb.

NOTE: If an infrared bulb is not available, an ordinary 300 watt bulb may be substituted for the check. Apply only enough voltage (approximately 90 volts) to make the bulb glow orange. At higher voltages, the bulb will emit ultraviolet rays that bias the system until the fire detectors cannot be functionally checked.

Activate the system and direct the light at each detector in a subdued light so that external infrared sources do not bias the system, and observe the warning lights in the T-handles. A test kit, P/N 30-502B, is available from Pyrotechtor, Inc., Hingham, Massachusetts, to test the flame detectors and troubleshoot the system.

B. Functional Test (Flame Detector Circuits)

The flame detector circuits may be tested by rotating the test switch on the copilot's subpanel through the 1 - 2 - 3 placarded positions. If the T-handle lights fail to illuminate in one or more of the test positions, check the corresponding flame detector and its circuit. The corresponding flame detectors are as follows:

Test position 1: The lower rear flame detector in each engine nacelle.

Test position 2: The upper rear flame detector in each engine nacelle.

Test position 3: The forward flame detector in each engine nacelle.

FIRE PROTECTION
ENGINE BLEED AIR WARNING SYSTEM
DESCRIPTION AND OPERATION

1. FIRE AND SMOKE DETECTION

A. Prior to U-165, except U-50 and Airplanes in Compliance With SI 0722-456

If the bleed air line should rupture, the warning system will provide a visual signal at the annunciator panel, allowing the bleed air heat system to be shut off before the hot escaping air can damage the surrounding structure.

The system consists of a 2-amp circuit breaker on the copilot's environmental group subpanel, a BLEED AIR LINE FAIL light on the annunciator panel, a pressure-sensitive switch pressurized by instrument air pressure and poly-flo tubing routed adjacent to the bleed air line. The poly-flo tubing is plugged at both ends; directly aft of the left engine firewall and under the floorboards aft of the pilot's rudder pedals. An electric, pressure-sensitive switch teed off of the poly-flo line, is located under the left hand seat track at FS 207.00 (even with the second cabin window). The pressure switch is held in a normally open position by air pressure from the bleed air pressure instrument system. Should the bleed air line rupture, the heat from the escaping bleed air will melt the poly-flo tubing, allowing the air in the bleed air failure line to be released. The resultant loss in pressure will cause the contacts of the pressure switch to close, thereby illuminating the BLEED AIR LINE FAIL light on the annunciator panel. Each time the system is activated, the poly-flo tubing in the area where the damage occurred must be replaced when the bleed air line is replaced.

B. U-50, U-165 and After

The engine bleed air warning system provides a visual indication of a rupture in the bleed air lines so that the affected engine bleed air can be shut OFF before the heat of the escaping air damages the skin and structure adjacent to the break in the line. The bleed air lines are accompanied in close proximity by plastic poly-flo tubing from the engines to the cabin. Excessive heat on the plastic tubing, caused by a ruptured bleed air line, will cause the tubing to fail.

The system has two pressure switches mounted on a channel beneath the floorboards on the airplane's left side, approximately even with the emergency exit hatch. The two switches are pressurized by air tapped just as it exits the pneumatic pressure regulator and relief valve. Refer to Figure 1 for the routing of the poly-flo tubing. When the tubing melts and the tubing pressure drops to 1.5 ± 0.5 psi or below, the normally open switch in the line will close, causing a circuit to be completed to the respective (left or right) BL AIR FAIL light in the warning annunciator panel. When the indication of bleed air failure becomes evident, the bleed air for that side must be turned off by placing the respective lever lock bleed air valve switch on the copilot's subpanel in the INST & ENVIR OFF position. When the switch is placed in this position, the engine firewall bleed air shutoff valve closes, permitting no engine bleed air past the engine firewall. Each time the system is actuated, the poly-flo tubing in the area must be replaced when the bleed air line is repaired.

On each warning system (right or left), the poly-flo tubing is plugged in three places. The right warning system is plugged in two places just aft of the engine firewall and is plugged adjacent to the bleed air plumbing at FS 94.00. The left warning system is plugged at the same relative location as the right system.

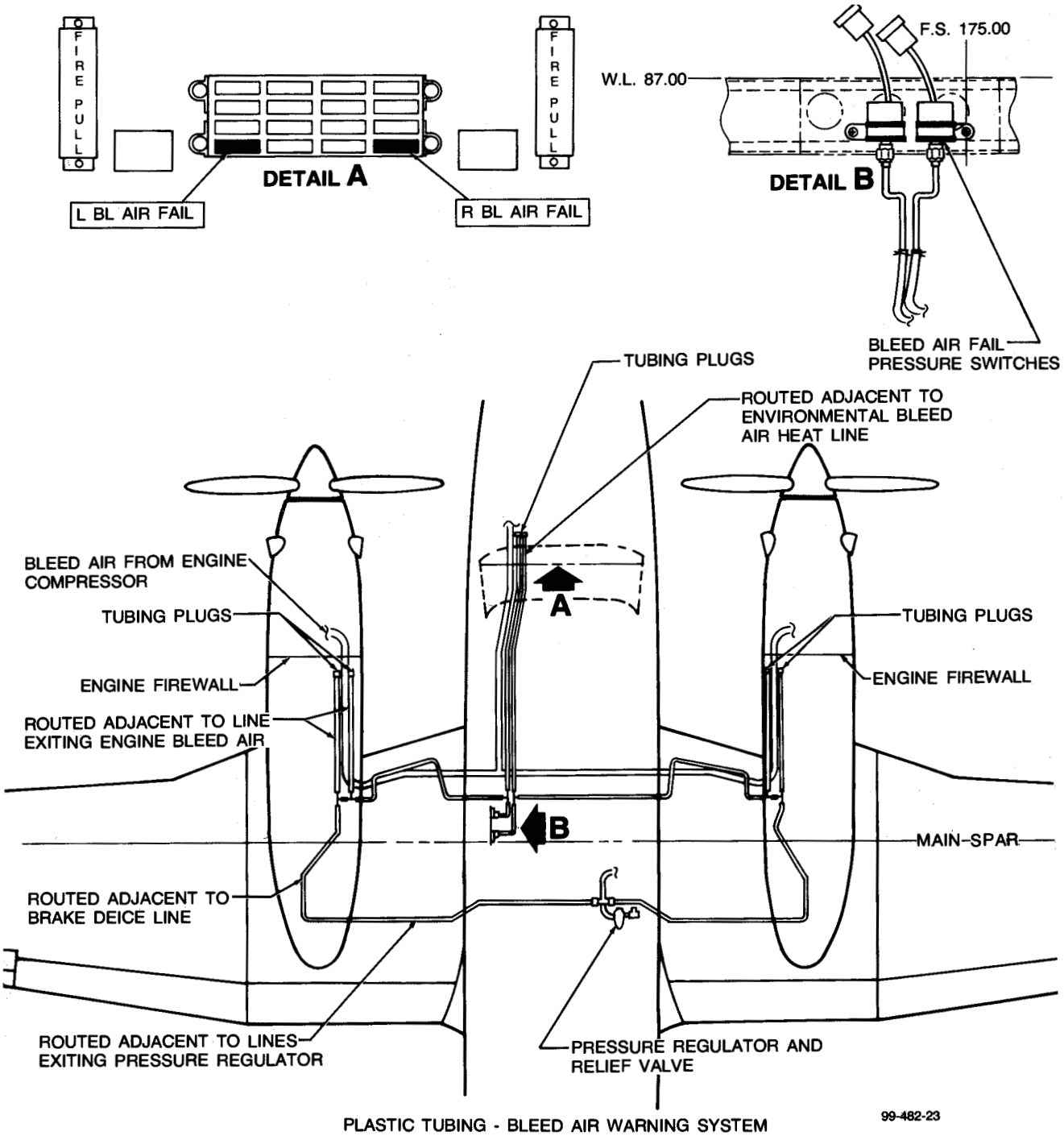


Figure 1
Bleed Air Warning System

FIRE PROTECTION
ENGINE BLEED AIR WARNING SYSTEM
MAINTENANCE PRACTICES

1. BLEED AIR WARNING SYSTEM

A. Maintenance Precautions

The plastic bleed air warning system tubing must be routed to maintain a clearance of two to four inches when following pneumatic and bleed air heat lines, except in the wheel well areas where the spacing may be 1.5 inches from the pneumatic lines.

The plastic bleed air warning system tubing must follow the bleed air and pneumatic lines and be located so that it will be below the horizontal centerline of the bleed air lines.

CAUTION: *When replacing the plastic warning system tubing, use only water as a lubricant.*

B. Operational Check

(1) Remove the inspection panel from between the left seat tracks, forward of the main spar to gain access to the two bleed air fail pressure switches.

(2) Using a tee fitting, plumb in a regulated air source between the left pressure switch and its related tube.

(3) Disconnect and cap the line running to the right bleed air fail pressure switch.

(4) Turn the airplane battery master switch ON. Both bleed air failure lights (red) should illuminate.

(5) Apply air pressure to the left side of the system (approximately 5 psi). The left fail light should go out. If the light stays on, check for a leak in the tubing. (Remember, both sides are connected through a common tube in the pneumatic manifold.) If the left light stays on but the right light goes out, then the system is wired incorrectly and should be rewired.

(6) Slowly decrease the pressure until the left fail light comes on. Note this pressure. The light should come on at 1.5 $\pm$ 0.5 psi. If the switch is out of tolerance, it should be replaced.

(7) Cap off the left side, connect the air source to the right side and repeat the procedure to check the right side of the system.

(8) After completing the test, both sides should be reconnected and the inspection plate should be reinstalled.

(9) Remove electrical power from the airplane.

2. BLEED AIR WARNING PRESSURE SWITCH (PRIOR TO U-165, EXCEPT U-50)

A. Removal

(1) To gain access to the switch, remove the seat, carpeting, and access cover in the area of FS 207.00 (between the second and third forward cabin windows along the left side of the fuselage).

(2) Identify, tag and disconnect the electrical leads and clamps securing the switch and fittings to the bulkhead. Install protective caps on all connectors.

(3) Remove the switch from the bleed air line fitting.

B. Installation

(1) Install the switch to the bleed air line fitting.

(2) Remove the protective caps from the connectors. Install the electric leads and clamps securing the switch and fittings to the bulkhead. Remove the tags.

(3) Install the access cover, carpeting, and seat.

3. BLEED AIR WARNING PRESSURE SWITCH (U-50, U-165 AND AFTER)

A. Removal

(1) Gain access to the pressure switches. They are located on a channel under the floorboards on the airplane left side, approximately even with the emergency exit hatch.

NOTE: The forward switch mounted on the channel is the left engine bleed air pressure switch; the aft switch is the right engine air pressure switch.

(2) Identify, tag and disconnect the electrical connector from the pressure switch to be removed. Install protective caps on all connectors.

(3) Loosen the fitting at the base of the pressure switch and cap the fitting to prevent the entry of foreign matter.

(4) Remove the screw that secures the clamp to the pressure switch on the channel. Remove the pressure switch.

B. Installation

(1) Install the pressure switch on the channel and secure with the clamp and screw.

NOTE: The forward and aft pressure switches are interchangeable; however, the forward pressure switch is the left engine pressure switch and the aft switch is the right engine pressure switch.

(2) Remove the cap from the poly-flo tubing fitting and install the fitting at the base of the pressure switch.

(3) Remove the protective caps from the connectors. Connect the electrical connector at the top of the pressure switch. Remove the tags.

FIRE PROTECTION
EXTINGUISHING
DESCRIPTION AND OPERATION

1. ENGINE FIRE EXTINGUISHER

A. U-1 thru U-164, except U-50

An optional fire extinguisher supply is mounted on brackets at the aft side of the main spar in each wheel well of the main landing gear. Each cylinder is charged with 2-1/2 pounds of Bromotrifluoromethane ($CBrF_3$) fire extinguishing agent and pressurized with dry nitrogen to 450 psi at 70°F. The supply line from the cylinder runs along the outboard side of the nacelle on U-80 and After. Prior to U-80, the supply line from the cylinder on the right engine ran along the inboard side of the nacelle. The fire extinguisher plumbing branches off into a network of spray tubes covering the engine. The system is armed by pulling out the left or right fire extinguisher T-handle. At the same time the handle is pulled, the corresponding firewall shut-off valve closes and the fire extinguisher switch under the handle is illuminated. Pushing the switch causes an explosive squib to release the agent. To ensure operation during cold weather, each supply cylinder is encircled by an electrothermal heater. Any time the temperature drops below 35°F, the heaters are actuated by thermostats mounted adjacent to the supply cylinder in each main landing gear wheel well. The circuit breakers for the extinguisher circuits and the electrothermal heaters are located on the EMERGENCY circuit breaker panel to the right of the copilot.

NOTE: None of the installations after U-164 are equipped with electrothermal heaters.

B. U-50, U-165 and After

An optional fire extinguisher supply is mounted on brackets at the aft side of the main spar in each wheel well of the main landing gear. Each cylinder is charged with 2-1/2 pounds of Bromotrifluoromethane ($CBrF_3$) and pressurized with dry nitrogen to 450 +25/-0 psi at 70°F. The lines from each cylinder run along the right side of each nacelle and branch into a network of spray tubes about the engine to diffuse the $CBrF_3$ in the event of a fire. Four of the spray tube nozzles are positioned to discharge beneath the engine exhaust case; one nozzle is positioned to discharge into the starter generator cooling duct; and two nozzles are positioned to discharge in the engine accessory area.

The system is designed to simultaneously discharge a concentration of fire extinguishing agent for a half second into each of the nacelle fire zones. Each fire extinguisher is actuated through a monitor module by its respective control switch. The monitor module PCB's (A239, A240) are mounted in the aft PCB rack, under the center aisle floorboard, just forward of FS 158.00.

The fire extinguisher control switches used to activate the system are located on the upper center of the instrument panel, above the audio panel. The right and left push-to-actuate switches each incorporate three indicator lights. The red lens, placarded L or R ENG FIRE - PUSH TO EXT, warns of the presence of fire in the engine. An amber lens, placarded D, indicated that the system has been discharged and the cartridge is empty. A green lens, placarded OK, is provided only for the test function for preflight. To discharge the cartridge, pull the firewall T-handle which arms the system, then raise the safety wired, clear plastic cover and press the face of the lens. This is a one shot system and will be completely expended upon activation, The amber D light will illuminate and remain illuminated, regardless of battery switch position, until the pyrotechnic cartridge has been replaced.

The rotary test switch is located on the left side of the copilot's subpanel. Test functions incorporated in the fire detector rotary test switch, LEFT - EXT - RIGHT, placarded to the left of the switch, are provided to test the circuitry of the fire extinguisher pyrotechnic cartridge. During preflight the pilot should rotate the test switch through the two positions and verify the illumination of the amber D light and green OK light.

A gage, calibrated in psi, is supplied on each fire extinguisher supply cylinder for determining the level of charge and should be checked during preflight.

The circuit breakers for the fire extinguishers are located in the ENGINE section of the copilot's circuit breaker panel.

FIRE PROTECTION
EXTINGUISHING
MAINTENANCE PRACTICES

1. ENGINE FIRE EXTINGUISHING

CAUTION: *The engine should be washed internally and externally as indicated under ENGINE WASHING PROCEDURES in Chapter 12-20-00 any time the engine fire extinguisher has been discharged. This will protect the engine against the corrosive effects of the fire extinguishing agent. This precaution is particularly critical on a hot engine.*

Access to the fire extinguisher cylinder is through the wheel wells of the main landing gear (Ref. Figure 201). On U-1 thru U-164, except U-50, each cylinder is charged with 2-1/2 pounds of Bromotrifluoromethane ($CBrF_3$) and pressurized with dry nitrogen to 450 psi at 70°F. On U-50, U-165 and After, each cylinder is charged with 2-1/2 pounds of Bromotrifluoromethane ($CBrF_3$) and pressurized with dry nitrogen to 450 +25/-0 psi at 70°F. Check the pressure gage on each cylinder prior to flight to ascertain that the cylinders are charged to within the pressure limits for the ambient temperature as noted in the table below. If the pressure is not in tolerance, refer to the SUPPLY CYLINDER WEIGHING section of this chapter.

TEMPERATURE IN °F	INDICATED PRESSURE IN PSI
-40°	190 to 240
-20°	220 to 275
0°	250 to 315
20°	290 to 365
40°	340 to 420
60°	390 to 480
80°	455 to 550
100°	525 to 635
120°	605 to 730

2. FIRE EXTINGUISHER SWITCH BULB

A. Replacement

To change the bulbs in the fire extinguisher switch and light combination, proceed as follows:

(1) Remove electrical power from the airplane.

CAUTION: *To avoid inadvertently actuating the switch and discharging the supply cylinder, disconnect the electrical wiring at the supply cylinders when changing the bulbs.*

Install a shunt (shorting device) or jumper wire across the contacts of the explosive squib to prevent accidental activation from stray voltages or static charges whenever the squib is not connected to the airplane wiring.

(2) Identify, tag and disconnect the electrical wiring from the supply cylinder. Insulate all bare wires.

(3) Remove the light capsule from the panel front, which is held in place by spring clips, and replace the bulbs.

(4) After the bulb replacement, slip the light capsule back into place.

(5) Perform the EXTINGUISHER ACTIVATION CHECK procedure.

3. SUPPLY CYLINDER

A. Removal

(1) Remove electrical power from the airplane.

(2) Working through main gear wheel well, identify, tag and disconnect the heater from electrical system at the quick disconnects (if the heater is installed). Insulate all bare wires.

(3) If the heater is installed, unlace the braid holding the heater in place around the supply cylinder.

CAUTION: *Install a shunt (shorting device) or jumper wire across the contacts of the explosive squib to prevent accidental activation from stray voltages or static charges whenever the squib is not connected to the airplane wiring.*

(4) Identify, tag and disconnect the electrical wiring from the supply cylinder. Insulate all bare wires.

(5) Disconnect the tube attached to the bottom of the supply cylinder.

(6) Remove the attaching bolts and remove the cylinder from the mounting brackets.

B. Installation

(1) Ensure electrical power has been removed from the airplane and that the fire extinguisher activation switches are not depressed.

(2) Secure the cylinder to the mounting brackets with the attaching bolts.

(3) Remove the insulation from the wires. Remove the shunt and connect the airplane wiring to supply cylinder terminals. Remove the tags.

(4) Connect the tube to the bottom of the cylinder.

(5) Position the heater in place around the cylinder (on those cylinders equipped with heater installations).

(6) Remove the insulation from the wires. Connect the heater wiring at the quick disconnects leading to ground and to the thermostat mounted adjacent to the supply cylinder (on heater installations). Remove the tags.

(7) Restore electrical power to the airplane.

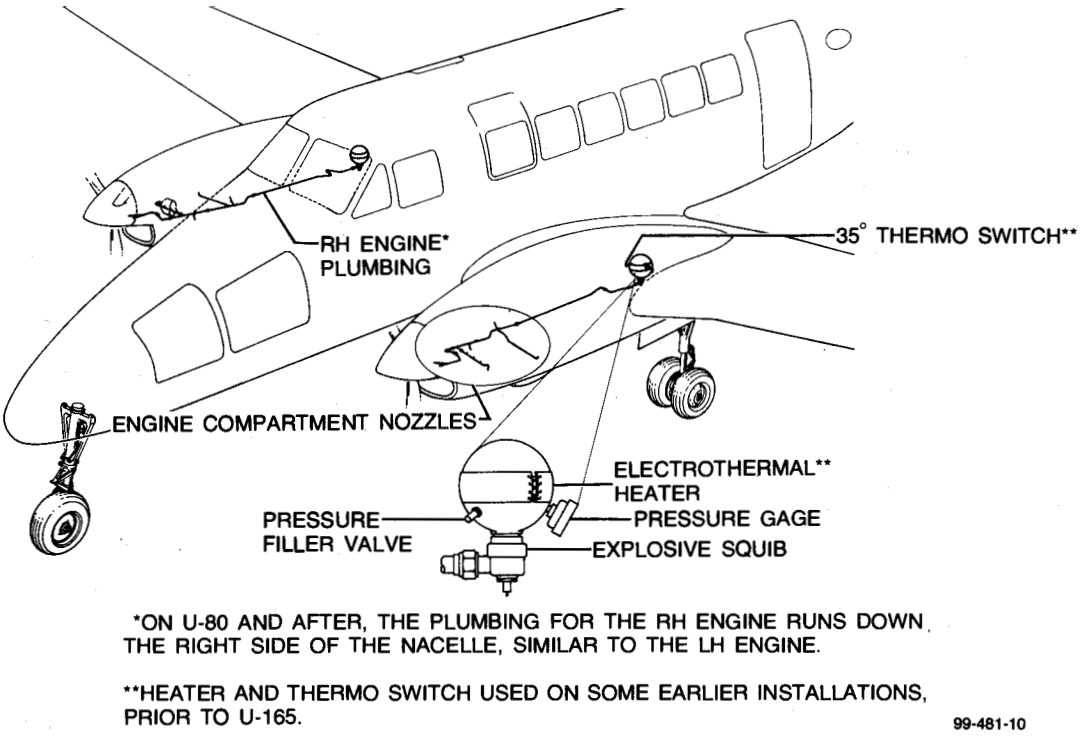


Figure 201
Engine Fire Extinguishing System

C. Weighing

Weigh the cylinders once a year and recharge if necessary. The weight of the fully charged cylinder is marked on a decal attached to the cylinder. Fully discharged supply cylinders or any cylinder weighing less than the specified amount should be returned to a recharging station certified by the supplier (HTL Industries, Inc., 1900 Walker Ave., Monrovia, California) for a proper charge of $CBrF_3$ extinguishing agent. Such stations are located in most of the major cities in this country. Refer to the data provided under the LIST OF CERTIFIED CHARGING STATIONS heading.

4. EXTINGUISHER ACTIVATION

A. Check

(1) Construct a test lamp consisting of one 327 light bulb connected to two 22 gage wires. The wires should be of sufficient length to allow the operator to sit in the cockpit, perform the procedure and view the test lamp connected to wiring in the wheel well. Install an alligator clip on the free end of each wire.

(2) Remove electrical power from the airplane.

CAUTION: *Install a shunt (shorting device) or jumper wire across the contacts of the explosive squib to prevent accidental activation from stray voltages or static charges whenever the squib is not connected to the airplane wiring.*

(3) Working through the main gear wheel wells, identify, tag and disconnect the wiring to the left and right fire extinguisher squibs. Insulate all bare wires.

(4) Left Fire Extinguisher Check

(a) Connect the test lamp leads to the wires disconnected from the squib. Position the test lamp so that it can be viewed by the operator in the cockpit.

(b) Apply electrical power to the airplane.

(c) Pull out the left T-handle.

(d) Press and hold the left fire extinguisher switch. The test lamp should illuminate.

(e) Release the left fire extinguisher switch. The test light should extinguish.

(f) Repeat the actions in Steps (4) (d) and (4) (e) an additional five times.

(g) Press and hold the left fire extinguisher switch. Push in and pull out the left T-handle six times. The test light should extinguish each time the left T-handle is pushed in. The test light should illuminate each time the left T-handle is pulled out. The left T-handle is to end up in the out position with the test lamp illuminated.

(h) Push in the left T-handle. The test lamp should extinguish.

(i) Release the left fire extinguisher switch.

CAUTION: *Ensure the left fire extinguisher switch is not depressed and that the test light is extinguished before continuing with this procedure.*

(j) Remove electrical power from the airplane.

(k) Disconnect the test lamp from the left squib wiring.

(5) Right Fire Extinguisher Check

(a) Connect the test lamp leads to the wires disconnected from the squib. Position the test lamp so that it can be viewed by the operator in the cockpit.

(b) Apply electrical power to the airplane.

(c) Pull out the right T-handle.

(d) Press and hold the right fire extinguisher switch. The test lamp should illuminate.

(e) Release the right fire extinguisher switch. The test light should extinguish.

(f) Repeat the actions in Steps (5) (d) and (5) (e) an additional five times.

(g) Press and hold the right fire extinguisher switch. Push in and pull out the right T-handle six times. The test light should extinguish each time the right T-handle is pushed in. The test light should illuminate each time the right T-handle is pulled out. The right T-handle is to end up in the out position with the test lamp illuminated.

(h) Push in the right T-handle. The test lamp should extinguish.

(i) Release the right fire extinguisher switch.

CAUTION: *Ensure the right fire extinguisher switch is not depressed and that the test light is extinguished before continuing with this procedure.*

(j) Remove electrical power from the airplane.

(k) Disconnect the test lamp from the left squib wiring.

(6) Remove the shunt. Remove the insulation from the wires. Identify and connect the wires to the left and right fire extinguisher squibs. Remove the tags.

5. LIST OF CERTIFIED CHARGING STATIONS

Stations marked with an * symbol are equipped for hermetically-sealed containers.

*Aviation Comprogas Services, 10 Railroad Avenue, East Northport, LI., NY 11731 Phone: 516/368-2203	Bacon Equipment Co., 2709 North Beckley Avenue, Dallas, TX 75208 Phone: 214/742-5871
Protection C.F.H., Inc., 103 Gun Ave., Pointe Claire, Que. H9R 3X2, Canada Phone: 514/694-3980	*CB Enterprises, 10704 Vanowen Street, North Hollywood, CA 91605 Phone: 213/980-2332
*Korean Air Lines, C.P.O. Box 864, Kimpo International Airport, Seoul, Korea	*Japan Air Lines, Hanada International Airport, Tokyo, Japan
John Cameron Aviation, Hanger 16, Bankstown Airport, N.S.W. 2200, Australia Phone: 7055812	Air Support International, Inc., 6753 E. 47th Avenue Dr., Denver, CO 80216 Phone: 303/333-5441
Tym's, 414 West Arbor Vitae, lnglewood, CA 90301 Phone: 213/673-3330 or 678-3330	*United Air Lines, Inc., San Francisco Int'l Airport, San Francisco, CA. 94128
City Sales, U. S. 2 and Lake Antoine Rd., P.O. Box 663, Iron Mountain, MI 49801 Phone: 906/774-3555	*Delta Air Lines, Inc., Jet Maintenance Base, Hartsfield-Atlanta International Airport, Atlanta, GA 30320 Phone: 404/346-6011
Aero Electric, Inc., 113 S. Laura, Wichita, KS 67211 Phone: 316/263-0197	Safety Supply Company, 172 Osborne Street, South Winnipeg, Manitoba, R3J 1Z1, Canada Phone: 204/453-7838
*Deutsche Lufthansa, Weg Beim Jaeger D2000, Hamburg 63, W. Germany	Eastern Aero Marine, 3850 N.W. 25th Street, Miami, FL 33142 Phone: 305/871-4050
Electronique Aerospatiale (EAS), Boite Postale No. 4, Le Bourget - 93, France	Fire Fighter Sales & Service Co., 1721 Main Street, Pittsburgh, PA. 15215 Phone: 412/782-2800
*KLM ~ Royal Dutch Airlines, Schiphol Airport, P.O. Box 770, Amsterdam, The Netherlands	Aero Mercantil, Leaver & CIA, Guaymaral Airport, Bogota, Columbia Phone: 481761 or 488799
Air Asia Co. Ltd., Tainan Airfield, Tainan, Taiwan 700, Republic of China Phone: 234141	*FasOrient (PTE) Ltd., Jalan Kayu, P.O. Box 17, Singapore 28

Stations marked with an * symbol are equipped for hermetically-sealed containers. (Continued)

Forind-Avio SRI., Via Bartolini, 1 20155 Milano, Italy	*Graviner LTD., Poyle Road, Colnbrook Slough, Buckinghamshire 5L3 OHB, England
General Fire Equipment Co., E. 4004 Trent Ave., Spokane, WA 99202 Phone: 509/535-4255	Houston Fire & Safety Equip. Co., 8555 Monroe Road, Houston, TX 77061 Phone: 713/943-0920
Vancouver Welding, 800 Harney St., Vancouver, WA 98660 Phone: 206/696-4605	*HTL Advanced Technology Division, 1800 Highland Avenue, Duarte, CA 91010 Phone: 213/359-9317 or 681-7487 TWX 910-585-3238

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

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FLIGHT CONTROLS
GENERAL
DESCRIPTION AND OPERATION

1. FLIGHT CONTROLS

The flight controls, with the exception of the flaps and the horizontal stabilizer, are conventional cable-operated surfaces requiring no power assistance for normal control by the pilot or copilot. The rudder and ailerons have cable-operated flight adjustable trim tabs. The flaps and horizontal stabilizer have electrically powered actuators controlled by switches on the pedestal (for the flaps) and on the control wheel (for the horizontal stabilizer). Positive stops on the control surfaces limit their travel while traveling stops secured on the cables limit the trim tab movement. Figure 205, 27-10-00, shows the proper method of safetying the trim tab stops to prevent their loosening and moving on the cables.

Since the cables can be disconnected at the turnbuckles, each cable has one right-hand and one left-hand threaded cable end. Proper routing of the pedestal and actuator drums as shown in Figure 203, 27-10-00, ensures against crossing the cables and causing improper trim tab movement. The stops for the horizontal stabilizer are built into the electric actuator motor.

FLIGHT CONTROLS
GENERAL
MAINTENANCE PRACTICES

1. CONTROL COLUMN

Refer to the applicable rigging Procedures for details regarding chain and cable tension, control wheel movement and force, and down-spring force and system movement.

2. CONTROL WHEEL

A. Removal

(1) Disconnect the electrical wiring from the various control wheel switches as necessary.

(2) Remove the taper pin which secures the control wheel adapter to the control column and pull the control wheel and adapter from the control column.

B. Installation

(1) Position the control wheel adapter and control wheel in place on the control column and insert the taper pin that secures the control wheel adapter to the control column while observing the following precautions:

(a) Excessive driving of the taper pin can cause cracks in the control column tubing. A light weight rawhide or nylon mallet should be used to set the taper pin when it is installed.

(b) Torque the taper pin nut 12 to 15 inch-pounds.

(c) With the taper pin properly torqued, the large end of the pin shank should be at least flush with, but not more than 0.06 inch above the surface of the control column.

(d) If the taper pin hole has become enlarged or elongated, it can be taper reamed as necessary, and the next size larger taper pin can be installed.

(2) Connect the electrical wiring from the various control wheel switches as necessary.

3. CONTROL COLUMN CABLE

A. Removal

(1) Disconnect the aileron cables from the control column cable at the turnbuckles.

(2) Paint one tooth of each of the control column actuating sprockets and its corresponding link of chain to ensure proper alignment of the control wheels at installation.

(3) Loosen the cable turnbuckle in the center of the control column horizontal cross member, remove the cable retaining pins, unsafety the chain from the sprockets, and remove the control column cable and chain assembly.

B. Installation

(1) Place the control column cable and chain assembly on the cross member of the control column with the painted links of the chains engaging the corresponding painted sprocket teeth. Install the cable retaining pins and safety wire the chain to the sprocket.

(2) Connect the aileron cables at the turnbuckles on the control column.

(3) Rig the cables as outlined in Aileron Control System Rigging (Ref. 27-10-00).

FLIGHT CONTROLS
AILERON AND TAB
MAINTENANCE PRACTICES

1. AILERON

A. Removal

Only the left aileron incorporates and aileron tab. The aileron tab pushrod connects directly to the actuator mounted forward of the rear spar. Access to the pushrod is through a door in the lower skin. The hinge brackets inserted between the aileron skin and spar are secured with four screws. The aileron pushrod is connected to one hinge bracket to transmit movements to the aileron.

NOTE: On the left aileron, remove the cotter pins, bolts, washers and nuts connecting the horns on the underneath side of the aileron tab to the pushrods from the aileron tab actuator.

(1) Support the aileron in an appropriate manner to prevent damaging the aileron when the screws are removed from the hinge brackets.

(2) Remove the four screws attaching each aileron hinge bracket to the aileron: two screws on the top and two screws on the bottom at each hinge.

(3) While providing support for the aileron, pull the aileron straight away from the wing.

(4) Remove the screws attaching the bonding jumper cable to the aileron.

(5) Move the aileron straight away from the wing.

CAUTION: *On the left aileron, use caution to not damage the aileron skin while moving the aileron tab pushrods through the aileron. Do not disturb the adjustment of the pushrods.*

B. Installation

WARNING: Failure to follow the correct procedure for AILERON INSTALLATION could result in partial/complete loss of the aileron resulting in degradation of flying qualities and or loss of control of the aircraft.

NOTE: The following dimensions and instructions are to be followed during aileron installation procedures:

1. The hinges on the aileron and the hinge brackets on the wing trailing edge should be checked to assure that they are in alignment within 0.06 inch horizontally and 0.03 inch vertically with each other.

2. The gap at the leading edge of the aileron measured between the wing skin trailing edge and the nose of the aileron, deflected 20° towards that edge, shall be 0.30 ± 0.05 inch.

3. The gap between the outboard edge of the flap and the aileron shall be 0.19 +0.06/ -0.13.

4. The gap between the aileron and the wing tip shall be 0.38 ± 0.12 inch.

5. The contour of the aileron may be 0.06 inch above or below the contour of the wing on one or both sides.

6. The aileron and the aileron linkage may have a total of 0.06 inch of lost motion. The bellcrank must be stationary and this measurement be taken at the trailing edge of the aileron.

Any repair, modification, painting or replacement of the aileron or aileron tab requires balancing.

(1) Support the aileron in position and install the bonding jumper cables.

NOTE: On the left aileron, carefully guide the aileron tab pushrods through the aileron so as not to damage the aileron leading edge or skin.

(2) Align the inboard, center and outboard hinges of the aileron in the proper location on the aileron spar, then loosely install two bolts (one bolt in the top and one in the bottom) through each hinge into the nut on the aileron spar.

(3) Pull the aileron to confirm that the bolts are properly installed.

NOTE: If there is no movement when the aileron is pulled on, the bolts are properly installed.

(4) Install and tighten the remaining bolts.

NOTE: The clearance noted in Figure 201 must be maintained between the aileron and the outboard flap.

(5) Install the bolts, washers, nuts and cotter pins securing the aileron tab pushrods to the aileron tab horns.

(6) Check the aileron and tab for proper travel and attachment.

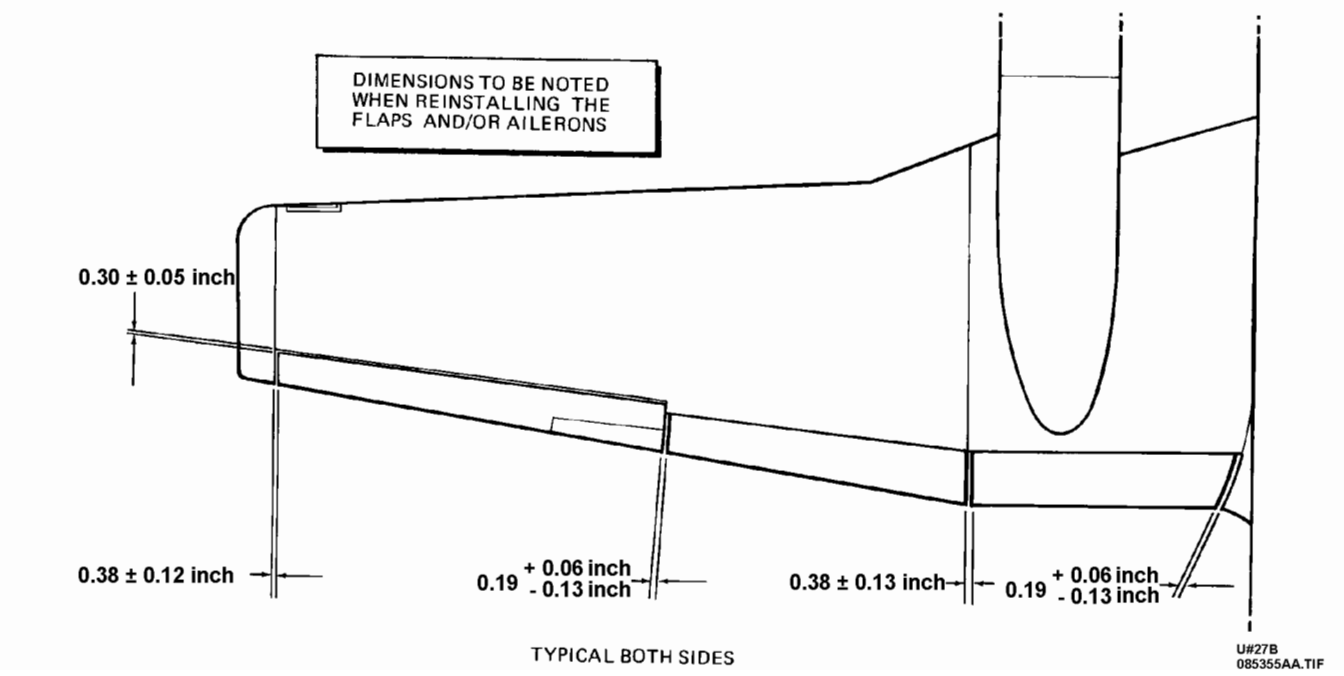

Figure 201
Flap and Aileron Installation

2. AILERON CABLE (FUSELAGE)

A. Removal

(1) Remove the pilot's seat and the left forward passenger seats as necessary.

(2) Remove the left pilot's compartment floorboards, and the center floorboard access plate in the crew compartment, all of the access plates forward of the rear spar on the left side of the forward cabin area and the large access door immediately forward of the rear spar on the right side.

(3) Disconnect the forward ends of the cables at the turnbuckles at the vertical member of the control column.

(4) Disconnect the aileron cable locking plates on the quadrant. Remove the cable retaining pins from all pulley brackets.

(5) Attach lead lines to the forward end of the cables and remove the cables through the aft access opening.

B. Installation

(1) Attach the cables to the lead lines and route them into position.

(2) Install the cable retaining pins in the pulley brackets.

(3) Connect the cables to the quadrant by installing the cable locking plate and attaching screws. Safety the screws with 0.032 inch safety wire.

(4) Connect the cable sections at the turnbuckles on the vertical member of the control column.

(5) Rig the cables as outlined under AILERON CONTROL SYSTEM RIGGING.

(6) Replace all access plates, floor panels and seats.

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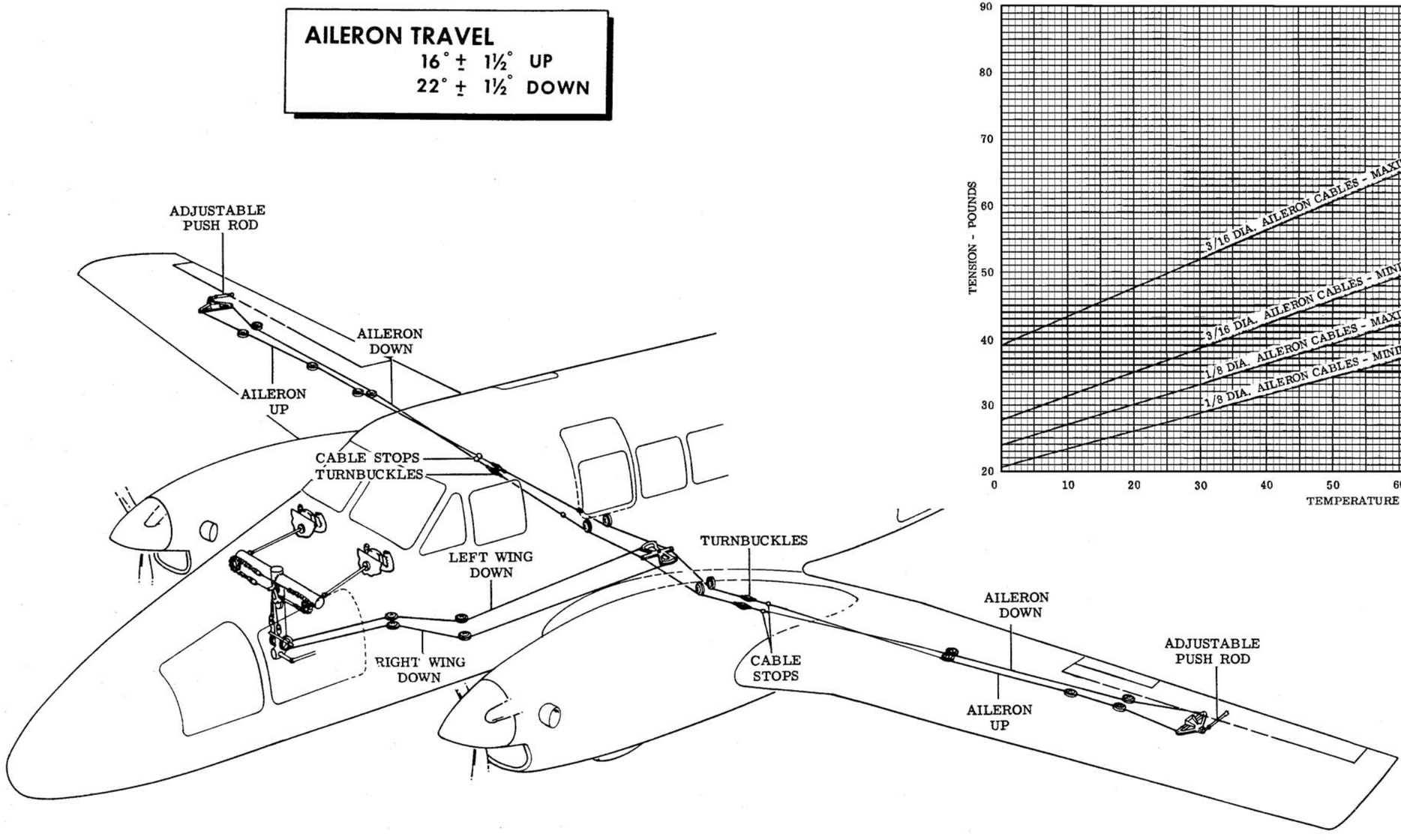

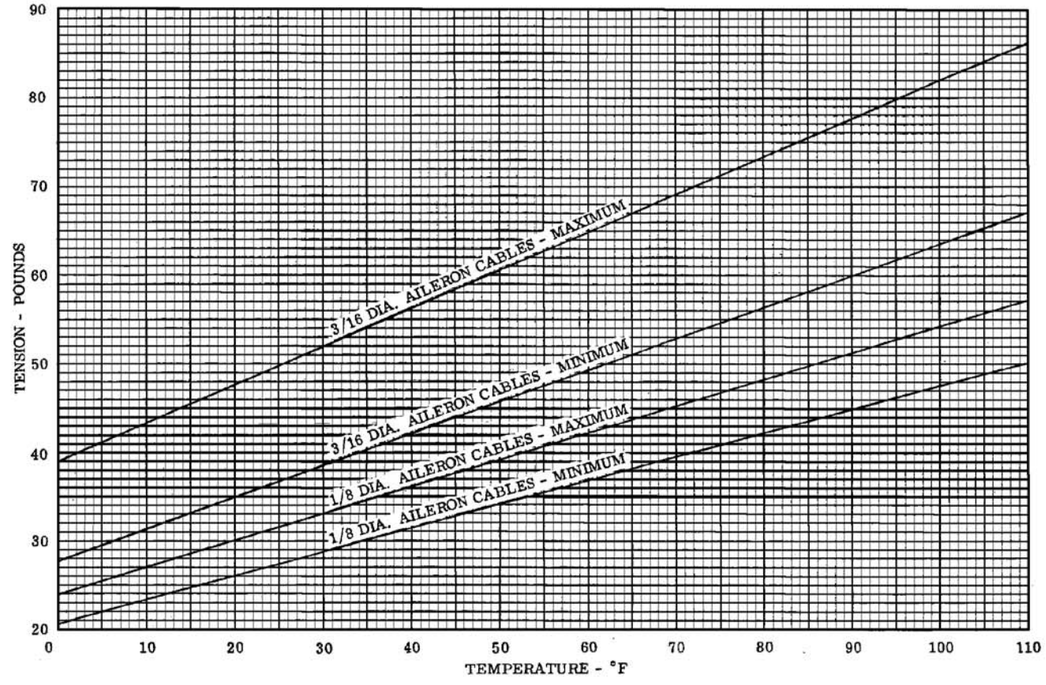

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Figure 202
Aileron Control Cable Rigging

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3. AILERON CABLE (WING)

A. Removal

(1) Remove all inspection plates on the underside of the outboard wing panel and center section immediately forward of the rear spar.

(2) Remove the left passenger seat and all four access panels forward of the rear spar on the left seat deck. Remove the access panel immediately ahead of the rear spar on the right seat deck if the right cables are being removed.

(3) Remove the cable retaining pins.

(4) Disconnect the cables at the turnbuckles in the wheel well and attach lead lines to the cable ends.

(5) Disconnect the outboard cables at the bellcrank and the inboard cables at the quadrant.

NOTE: Before the inboard cables can be disconnected from the quadrant, the opposite cables must be loosened.

(6) Pull the cables through the access openings at the bellcrank and quadrant.

NOTE: Aileron cables from the quadrant to the turnbuckles and from the turnbuckles to the bellcrank are not interchangeable. To avoid incorrect installation, label each cable in relation to its location prior to removal.

B. Installation

(1) Attach the cables to the lead lines and route into position.

(2) Attach the cables to the bellcrank in the outer wing panel and to the quadrant.

(3) Install the cable retaining pins in the pulley brackets.

(4) Connect the aileron control cables at the turnbuckles.

(5) Rig the cables as outlined under AILERON CONTROL SYSTEM RIGGING.

(6) Replace all access covers, floor coverings and seats.

4. AILERON CONTROL SYSTEM

A. Rigging

WARNING: Check for correct direction of aileron travel by moving the control wheel. When the control wheel is turned to the left, the left aileron should move up and the right aileron should move down. When the control wheel is turned to the right, the right aileron should move up and the left aileron should move down.

NOTE: The travel gage used to rig the aileron should be installed at wing station 211.923.

(1) Place the aileron quadrant in the neutral position and install the rig pins in the quadrant, and in the control column lock pin hole (prior to U-165).

(2) Rig the aileron cables that interconnect the pilot's and copilot's controls at the control column to the tension of 10 +15/ -0 pounds. Both control wheels should align within ± 1/2°.

(3) Rig the control column-to-quadrant cables to the 1/8-inch cable tension shown in Figure 202. Adjust the cables so that the control wheels will be in the neutral position when the proper tension is reached.

(4) With the aileron cable disconnected from the wing bellcrank, back off the aileron bellcrank stops all the way.

(5) Adjust the aileron travel to the correct setting by adjusting the bellcrank stop bolts out as necessary to achieve 16° ± 1 1/2° up and 22° ± 1 1/2° down.

(6) Attach the cables to the bellcrank and set in a 2° droop of the aileron by adjusting the turnbuckles in the wheel well. The cable tension shown in Figure 202 can be achieved at this time.

(7) Check the travel of the ailerons to assure correct setting of the bellcrank stop bolts.

(8) Be certain the clearance between the aileron bellcrank and tab cable pulley and the guard retaining bolt head maintains a minimum clearance of 0.12 inch.

NOTE: Visually check that aileron movement corresponds to the control wheel movement. The force required to move the control wheel 10° to the left or right of neutral must be 20 +5/ -10 inch pounds.

5. AILERON TRIM TAB CABLE (WHEEL WELL TO TAB CONTROL)

A. Removal

(1) Remove the pilot's seat and all of the left forward passenger seats ahead of the rear spar.

(2) Remove the left cockpit floorboards and all left access covers ahead of the rear spar.

(3) Disconnect the tab cables at the turnbuckles and connect lead lines to the inboard cable ends. Mark the lines for right and left-hand threads on the cable ends.

(4) Remove the cable retaining pins, fairleads in the fuselage, and the grommets from the inboard wheel well rib.

(5) Pull the cable from the airplane structure and detach the lead lines.

(6) Remove the cable drum support bracket from the pedestal.

(7) Remove the nut and bushing from the drum shaft and remove the drum from the shaft.

(8) Unwind the cable from the drum, drive the cable lock pin from the drum and remove the cable.

B. Installation

(1) Place the exact center of the cable in the slot in the cable drum and insert the lock pin. The cable must be installed as shown in Figure 203, Sheet 1 of 2. Wrap each end of the cable around the drum 2 1/4 turns.

(2) Set the indicator at "0", then install the cable guide and drum with the short shaft and nut.

(3) Install the drum support bracket.

(4) Route the cables, connect the turnbuckles and install the retaining pins, fairleads and grommet.

(5) Perform the AILERON TAB RIGGING procedure in this section and adjust the indicator as necessary.

(6) Replace all access covers, floor coverings and seats.

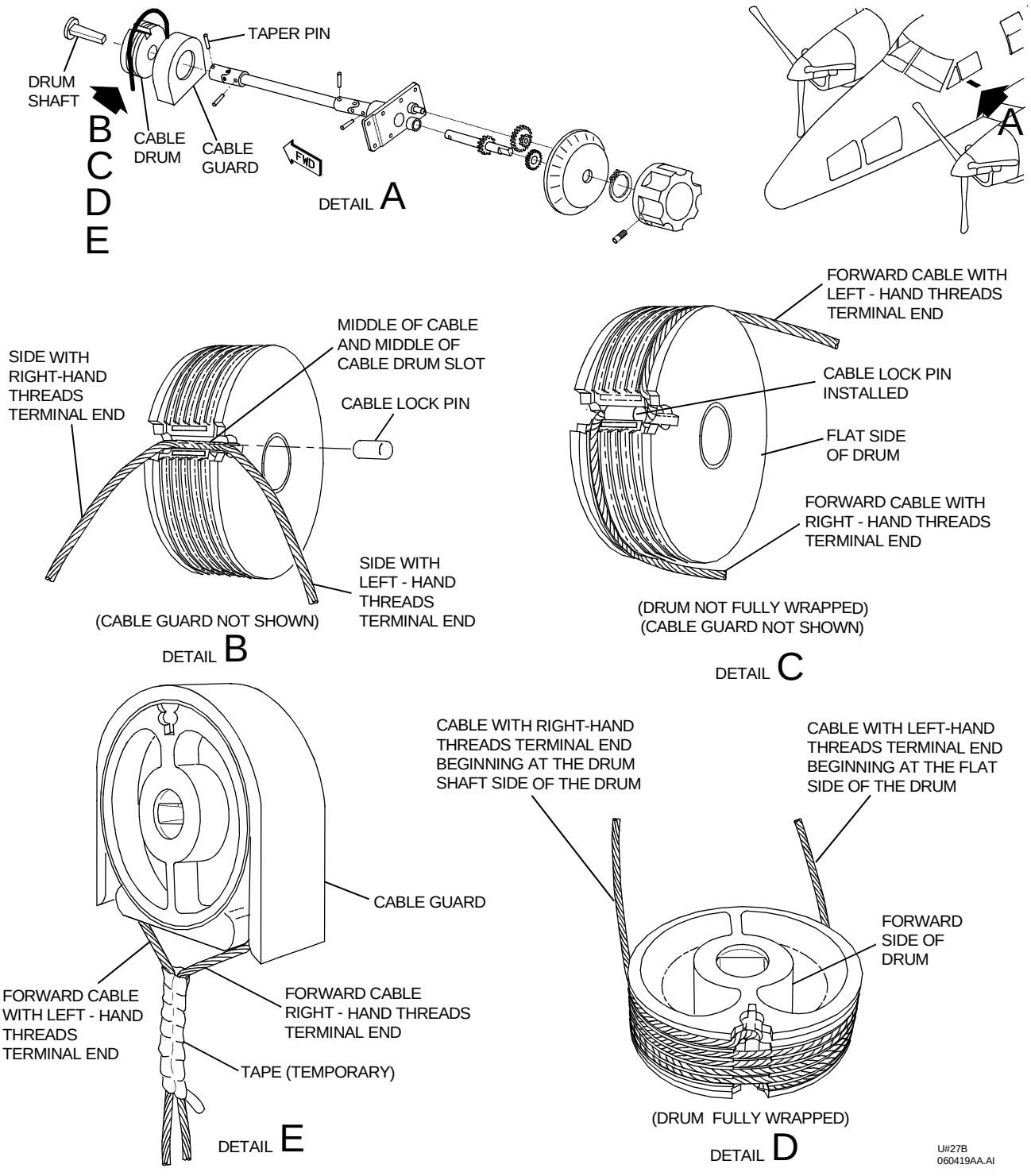


Figure 203 (Sheet 1 of 2)
Trim Tab Cable Winding

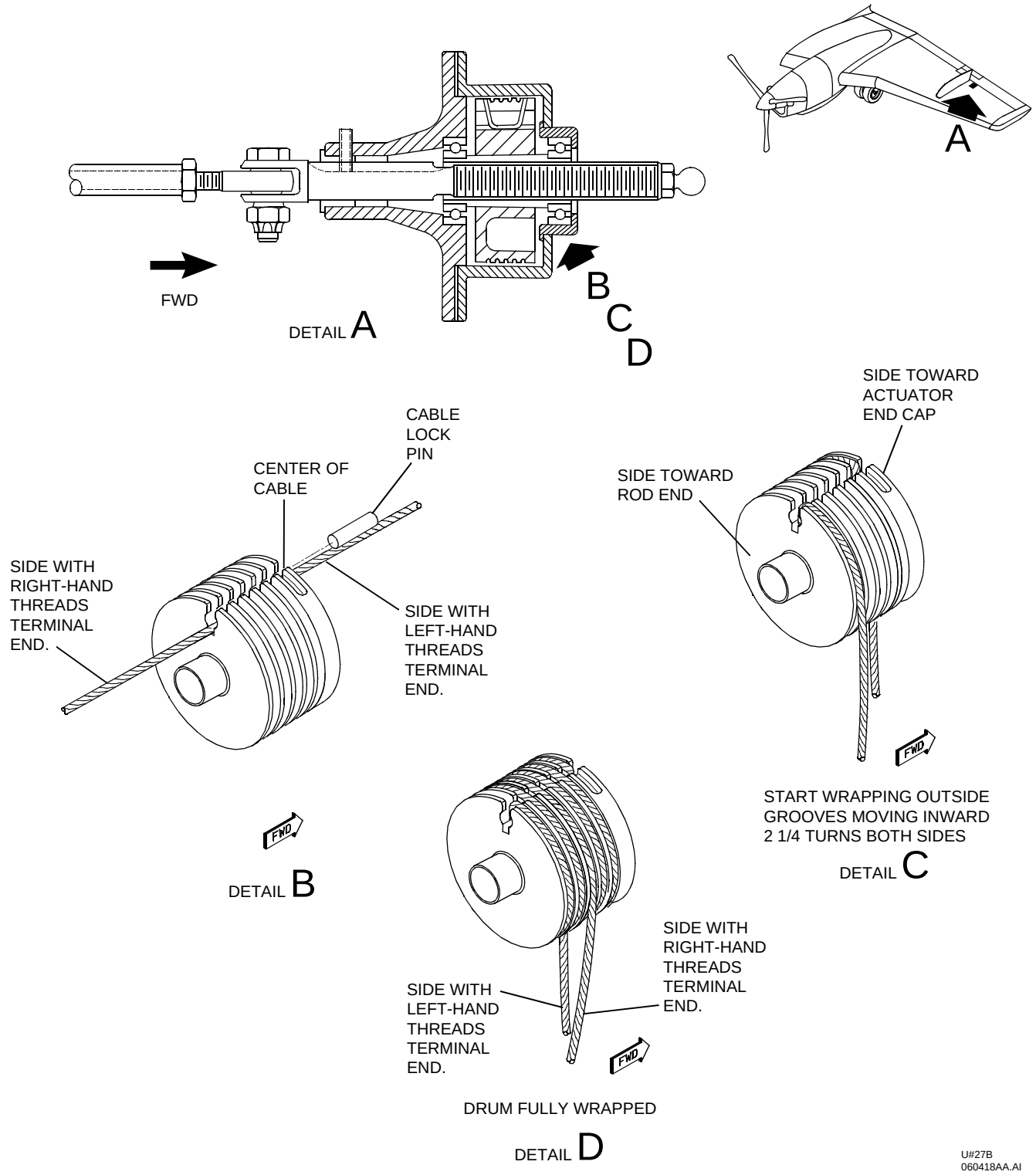


Figure 203 (Sheet 2 of 2)
Trim Tab Cable Winding

6. AILERON TRIM TAB ACTUATOR AND OUTBOARD CABLE

A. Removal

(1) Remove the inspection plates on the underside of the wing immediately forward of the rear spar.

(2) Disconnect the tab control cables at the turnbuckles in the wheel well and attach lead lines to the cables. Mark the lines for left and right-hand threads on the cable ends.

(3) Remove the cable retaining pins from the pulley brackets.

(4) Disconnect the tab push-pull rod at the indicator.

(5) Remove the bolts attaching the actuator to the rear spar.

(6) Remove the actuator and tab control cables from the wing, pulling the cables out through the actuator access hole.

(7) Remove the actuator housing from the actuator.

(8) Remove the cable from the actuator drum.

B. Installation

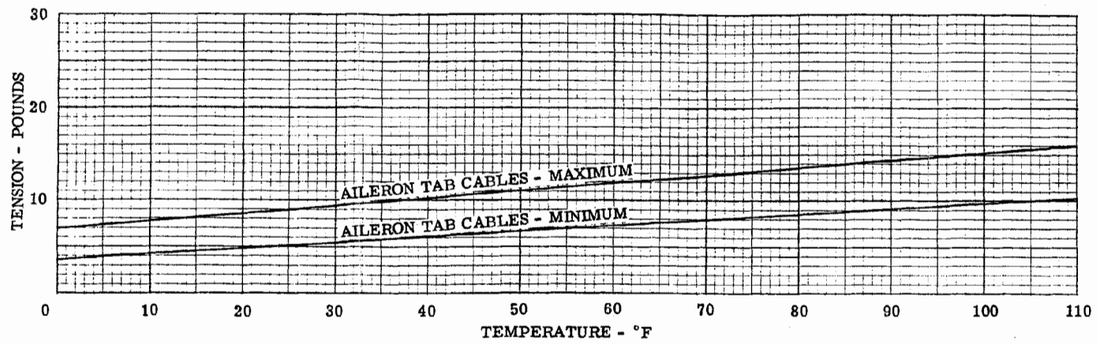
(1) Set the actuator screw at 1/2 full travel.

(2) Place the exact center of the cable in the slot in the actuator drum and install the cable lock pin. The cable must be installed as shown in Figure 203, Sheet 2 of 2. Wrap each end of the cable around the drum 2 1/4 times, then install the actuator housing.

(3) Install the actuator and connect the tab push-pull rod.

(4) Connect the lead lines to the cables and route the cables through the wing. Install the cable retaining pins in the pulley brackets.

(5) Connect the cables at the turnbuckles and rig the tab cables. Adjust the indicator if necessary.



AILERON TRIM TAB TRAVEL

15° ± 1½° UP
15° ± 1½° DOWN

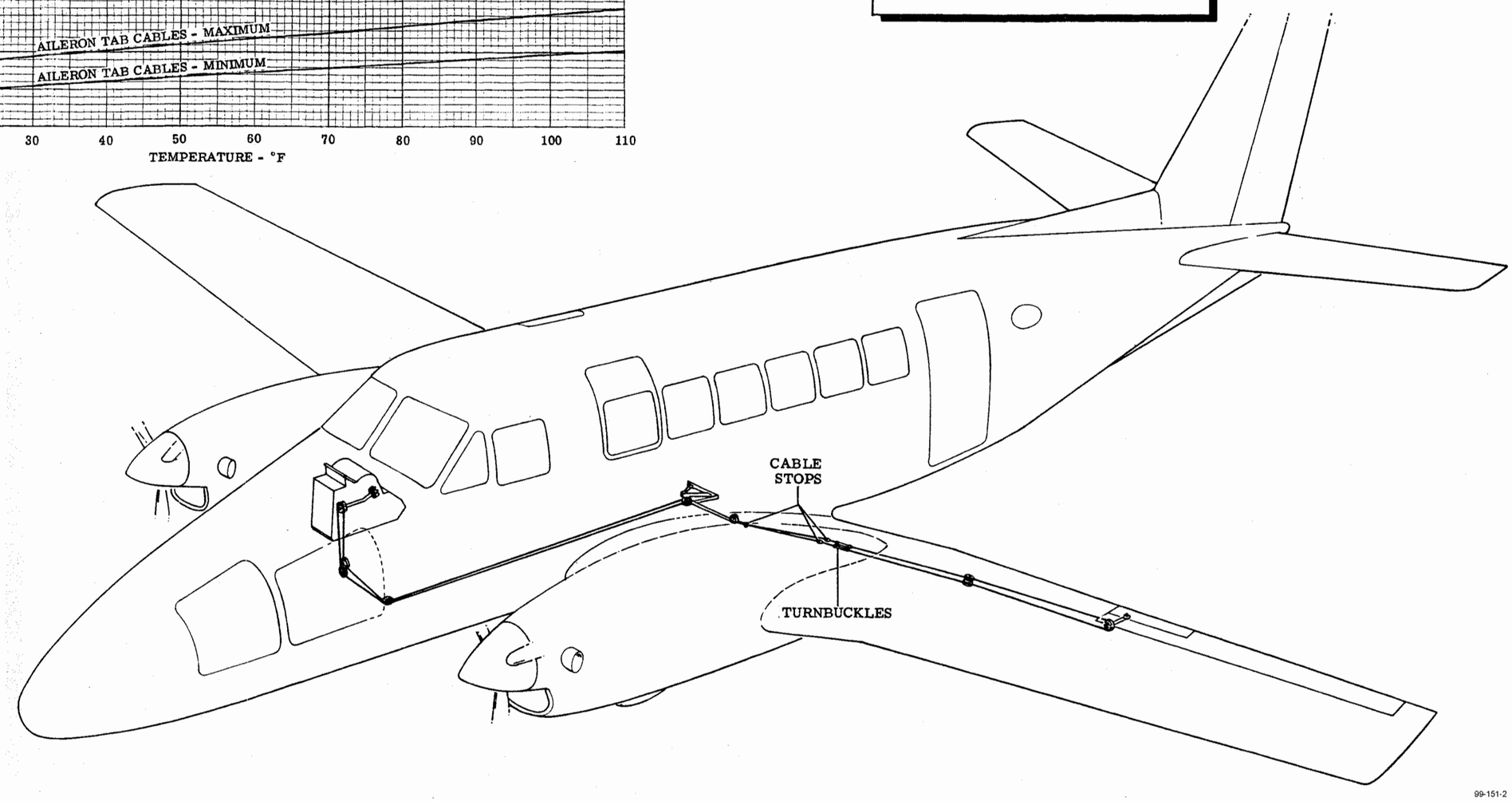


Figure 204
Aileron Trim Tab Rigging and Cable Tension

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7. AILERON TRIM TAB INDICATOR

A. Adjustment

(1) Loosen the setscrew in the side of the trim tab adjusting knob.

(2) Remove the knob from the shaft.

(3) Raise the indicator dial until it turns freely.

(4) Turn the dial to the desired position and lower to engage the reduction gears.

(5) Replace knob and tighten setscrew.

8. AILERON TRIM TAB

A. Rigging

Before rigging the aileron tab, the ailerons must be properly rigged.

NOTE: When the aileron tab is set in the neutral position, ± 1/2° is acceptable with the aileron in either the full up or full down position, provided that the maximum travel limits of neither tab nor aileron is exceeded.

(1) Disconnect the tab from the tab actuator.

(2) Install the aileron control lock.

(3) Set the tab control wheel so that the tab indicator reads “0”.

(4) Rig the cables to the tension shown in Figure 204.

(5) Position the actuator screw in the midpoint of its travel.

(6) Connect the tab to the actuator and adjust the actuator linkage until the tab is in neutral.

(7) Adjust the aileron tab travel to 15° ± 1 1/2° down. Make the adjustments with the cable stops in the left wheel well, then torque the stops to between 40 and 45 inch-pounds and safety as shown in Figure 205.

(8) Perform a visual check to make sure that the aileron tab movement corresponds to the movement indicated on the tab indicator: right wing up, tab down; then left wing up, tab up.

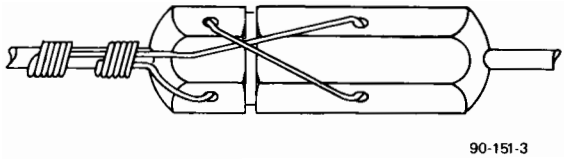


Figure 205
Safetying the Trim Tab Cable Stops

B. Freeplay Check

Visually inspect the aileron tab for any damage, security of hinge attach point and for tightness of the actuating system. Inconsistencies should be remedied prior to checking the freeplay of the tab. This check should be performed every 100 hours to ensure that the trim tab freeplay falls within the prescribed limits.

A check fixture (P/N 45-135030-9/810) or the equivalent as shown in Figure 206, and a push-pull scale, for applying accurate loading to the tabs, are required for making the inspection for freeplay of the tabs.

(1) Securely lock the control surfaces to prevent movement of the ailerons. Set the aileron tabs in the neutral position.

(2) Affix the dial indicator check fixture so that the dial indicator point is 4.12 inches aft of the tab hinge line and on the outboard edge of the aileron tab (Ref. Figure 206).

(3) Apply a small piece of masking tape (for paint protection) 4.92 inches aft of the tab hinge line and along the centerline of the tab actuator. This will be the point of pressure against the tab by the push-pull scale.

(4) Apply another piece of masking tape in the corresponding position on the bottom surface of the tab for the same purpose.

(5) Zero the dial indicator at no load initially. Do not reset during the checking procedure.

(6) With the push-pull scale at the point of masking tape, apply a full three pounds downward load. Record the dial reading as “A”.

(7) Release half the load until a 1.5 pound downward load is obtained. Record the dial reading as “B”.

(8) Apply a full three pound upward load at the masking tape on the bottom surface. Record the dial reading as “C”.

(9) Release half the load until a 1.5 pound upward load is obtained. Record the dial reading as “D”.

(10) Enter the recorded values on a copy of Chart 201 and proceed as follows:

(a) Multiply “B” by 2 and record as “2B”.

(b) Subtract “A” from “2B” and record as “X”.

(c) Multiply “D” by 2 and record as “2D”.

(d) Subtract “C” from “2D” and record as “Y”.

NOTE: The result of “X” and “Y” can be negative numbers. If the sum of “X” and “Y” is negative, that number must be within the limits designated by “E”.

(e) Add “X” and “Y” and record as “E”.

CHART 201 - AILERON TRIM TAB FREE PLAY LIMITS

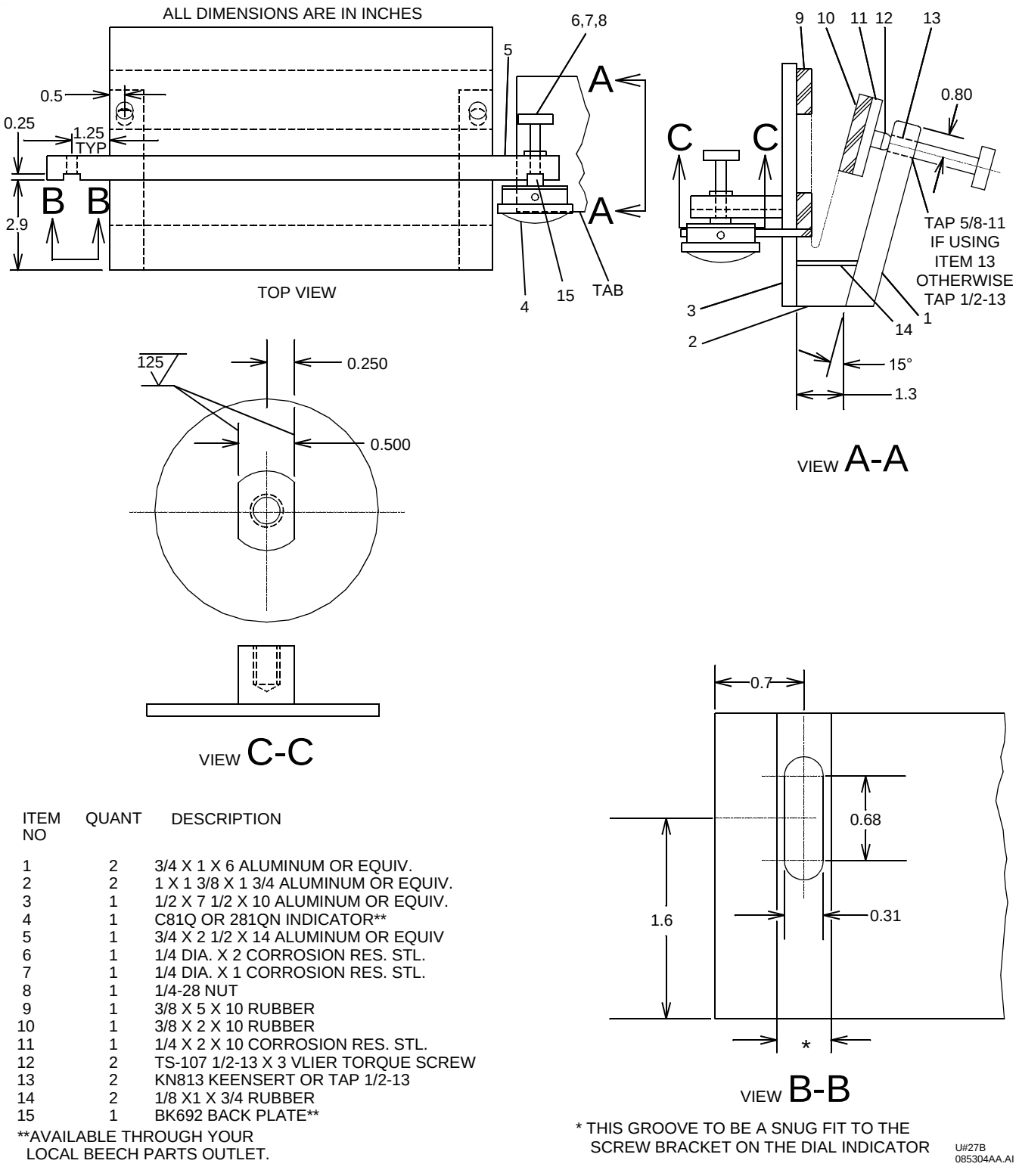
Serial Number: _____ Date: _____

(_____) X 2	= (_____)	- (_____)	= (_____)
B	2B	A	X
(_____) X 2	= (_____)	- (_____)	= (_____)
D	2D	C	Y
(_____)	+ (_____)	= (_____)	
X	Y	E	

(E = 0.034 inch maximum)

(11) If deflection of the tab is within the allowable limits, the tab and its linkage are in good condition.

(12) If the freeplay is excessive, disconnect the trim tab actuator rod and visually inspect the bolts and bushing for indications of excessive wear. Replace excessively worn parts.



ITEM NO	QUANT	DESCRIPTION
1	2	3/4 X 1 X 6 ALUMINUM OR EQUIV.
2	2	1 X 1 3/8 X 1 3/4 ALUMINUM OR EQUIV.
3	1	1/2 X 7 1/2 X 10 ALUMINUM OR EQUIV.
4	1	C81Q OR 281QN INDICATOR**
5	1	3/4 X 2 1/2 X 14 ALUMINUM OR EQUIV
6	1	1/4 DIA. X 2 CORROSION RES. STL.
7	1	1/4 DIA. X 1 CORROSION RES. STL.
8	1	1/4-28 NUT
9	1	3/8 X 5 X 10 RUBBER
10	1	3/8 X 2 X 10 RUBBER
11	1	1/4 X 2 X 10 CORROSION RES. STL.
12	2	TS-107 1/2-13 X 3 VLIER TORQUE SCREW
13	2	KN813 KEENSERT OR TAP 1/2-13
14	2	1/8 X1 X 3/4 RUBBER
15	1	BK692 BACK PLATE**

**AVAILABLE THROUGH YOUR LOCAL BEECH PARTS OUTLET.

* THIS GROOVE TO BE A SNUG FIT TO THE SCREW BRACKET ON THE DIAL INDICATOR

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Figure 206
Fabricating Check Fixture for Tab Deflection

FLIGHT CONTROLS
RUDDER AND TAB
MAINTENANCE PRACTICES

1. RUDDER

A. Removal

(1) Detach the tail cone, disconnect all electrical wiring to the lights, and remove the tail cone.

(2) Disconnect the rudder tab cables at the turnbuckles and remove the tab cable retainer pins.

(3) Remove the bolts securing the rudder torque tube and tab pulley bracket to the rudder bell crank.

(4) Disconnect the bonding cables at the upper and lower hinge points and remove the upper and lower hinge bolts and the rudder.

B. Installation

(1) Align the holes in the rudder and vertical stabilizer hinges and install the bolts attaching the rudder in place.

(2) Connect the rudder bonding cables.

(3) Attach the rudder tab pulley bracket and torque tube to the rudder bell crank with the attaching screws.

NOTE: The gap between the rudder torque tube and the bell crank should be from 0.005 to 0.020 inch prior to installation of the attaching bolts. Laminations may be peeled from the lower rudder hinge shim, located between the torque tube and the bell crank, to obtain this spacing.

(4) Connect the rudder tab cables at the turnbuckles and install the cable retainer pins.

(5) Rig the rudder tab cables as instructed under RUDDER TRIM TAB RIGGING.

(6) Connect all electrical wiring to the lights and install the tail cone.

CAUTION: *Do not operate the airplane with the upper rotating beacon removed, unless a cover plate has been installed over the rotating beacon location and the rudder has been balanced.*

C. Rigging

NOTE: The nose landing gear must be placed on a jack as described under NOSE JACKING in Chapter 07-10-00, and raised off the ground before rigging the rudder. This will permit rudder forces to be measured properly.

(1) Position the rudder pedals in the aft position and install the rig pin in the control lock holes in the pilot's rudder pedals.

(2) Adjust the copilot's rudder pedals to the neutral position with the push-pull tube between the bellcranks.

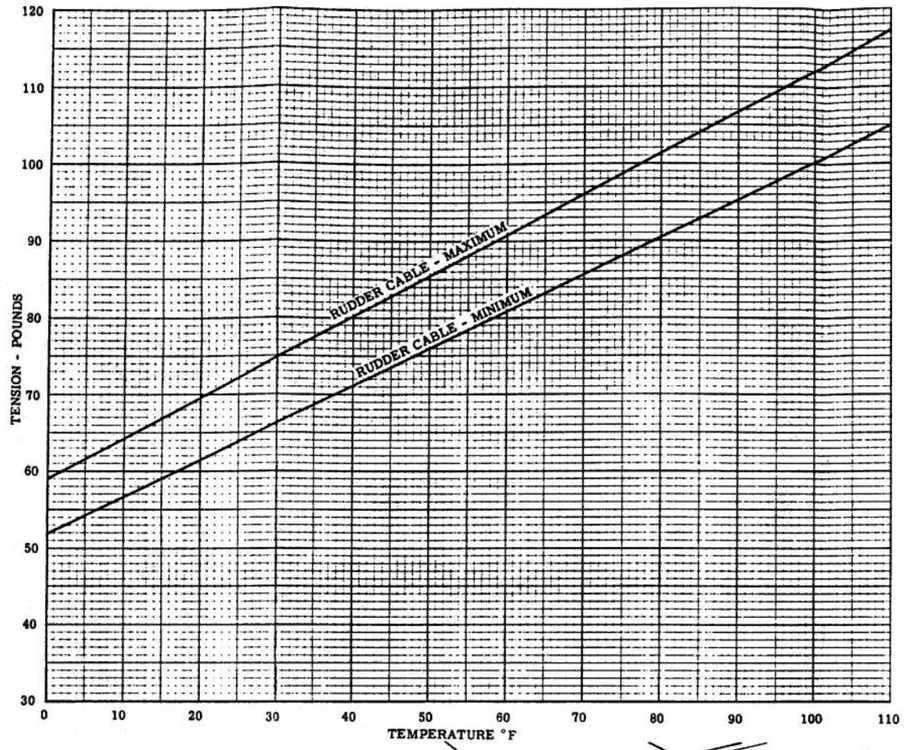
(3) Rig the cables to the tension shown in Figure 201. The adjustment should be made on both cables so that the rudder is in the neutral position, 0° (in line with the airplane centerline).

(4) Remove the rig pin from the pilot's rudder pedals.

(5) Adjust the rudder travel to obtain 26° +1° -0° right and 20° +1° -0° left. The above travel limits are made with the adjusting stops on the rudder lower hinge bracket.

(6) Set the forward left bellcrank stops to maintain 0.150 to 0.210 inch clearance with the rudder against the stop in each direction.

NOTE: Visually check that the rudder movement corresponds to the rudder pedal movement. Use a hand force gage at the gust lock hole in the rudder pedal arm to check the rudder pedal force required for full rudder deflection with the nose gear disconnected and either pair of rudder pedals in both the fore and aft adjustment positions. The maximum rudder pedal force required for full left deflection of the rudder must not exceed 40 pounds. The rudder must begin moving away from full left deflection when a force of six pounds or more is applied to the rudder pedals. Full right deflection of the rudder must be attained with a rudder pedal force of not over 45 pounds. A minimum rudder pedal force of 13 pounds must be required to start the rudder moving from full right deflection. The force on the rudder pedals required to deflect the rudder in either direction must increase or decrease smoothly and steadily without any of the sudden changes that indicate friction or interference in the control system. All readings are to be taken in still air.



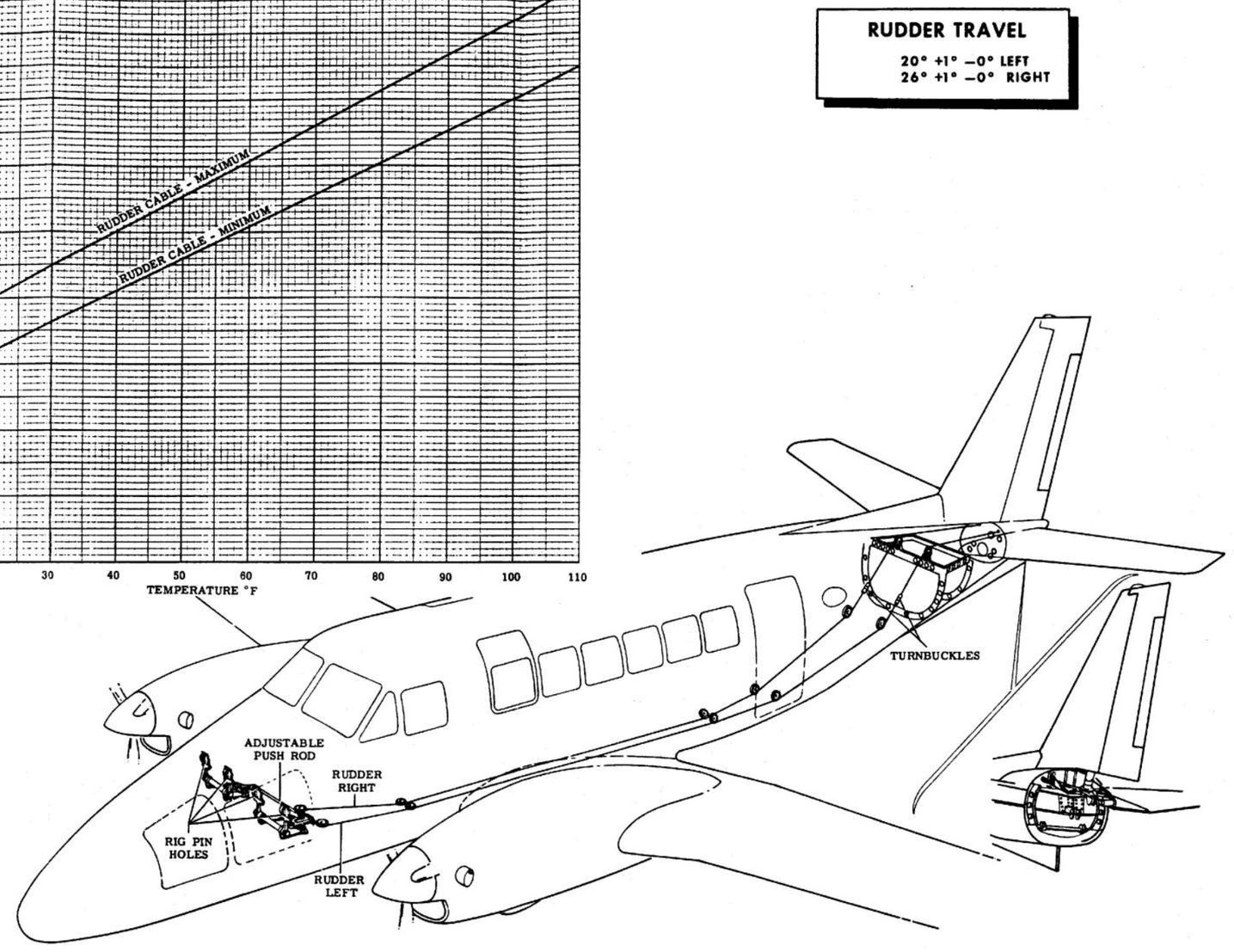


Figure 201
Rudder Control Cable Rigging and Cable Tension Graph

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2. RUDDER CABLE

A. Removal

(1) Remove the pilot's seat, all left passenger seats and the toilet (if equipped).

(2) Remove the pilot's floorboards, all of the floor access plates on the left side of the cabin and aft compartment, the tail section access door and the tail cone.

(3) Remove the cable retaining pins from the pulley brackets.

(4) Disconnect the cables at the turnbuckles and connect lead lines to the forward cables.

(5) Disconnect the forward cables from the bellcrank and remove them through the crew compartment.

(6) Disconnect the aft cables from the rudder bellcrank for removal.

B. Installation

(1) Connect the aft cables to the rudder bellcrank, route the cables into the tail section and install the cable retaining pins in the pulley brackets.

(2) Route the forward cables from the cockpit and connect them to the bellcrank. Install the cable retaining pins in the pulley brackets.

(3) Connect and rig the cables.

(4) Replace the tail cone, access plates, floor coverings and seats.

3. RUDDER PEDAL AND BELLCRANK

A. Removal

(1) Remove the floorboards forward of the pilot's and copilot's seats.

(2) Remove the nonadjustable push-pull tubes connecting the idler arms and bellcranks.

(3) Disconnect the nose steering push-pull rod at the arm to the right of the pilot's right rudder pedal.

(4) Remove the taper pins securing the inboard collars on both rudder pedal cross-shafts.

(5) Disconnect the brake master cylinders from the rudder pedals.

(6) Remove the bolts securing the cross-shaft supports (inboard of each shaft).

(7) Pull the cross-shaft inboard until it is free of the outboard supports.

(8) Lift the rudder pedals up and out.

NOTE: Note the proper position of the idler arms, pedals and collars on the shaft before removing the rudder pedals. These positions must be maintained to assure proper operation when the pedals are installed.

(9) Release tension on the rudder cables and disconnect the cables from the pilot's bell crank.

(10) Remove the adjustable push-pull tube connecting the bellcranks.

(11) Remove the bolts attaching the bellcranks to the brackets.

B. Installation

NOTE: On airplanes U-165 and after, the adjustment lever for the fore and aft position of each rudder pedal has been moved to the inboard side of the pedal for ease of access. Check the rudder pedals for ease of adjustment from one position to the other.

(1) Place the bellcranks in their respective positions and install the attaching bolts.

(2) Install the adjustable push-pull tube connecting the bellcranks.

(3) Place the pilot's and copilot's rudder pedals in position with the collars and idler arms, and slide the shafts into their outboard supports.

(4) Secure the inboard supports with their attaching screws.

(5) Replace the taper pins securing the inboard collars on each of the cross-shafts. Torque the taper pins 15 to 20 inch-pounds.

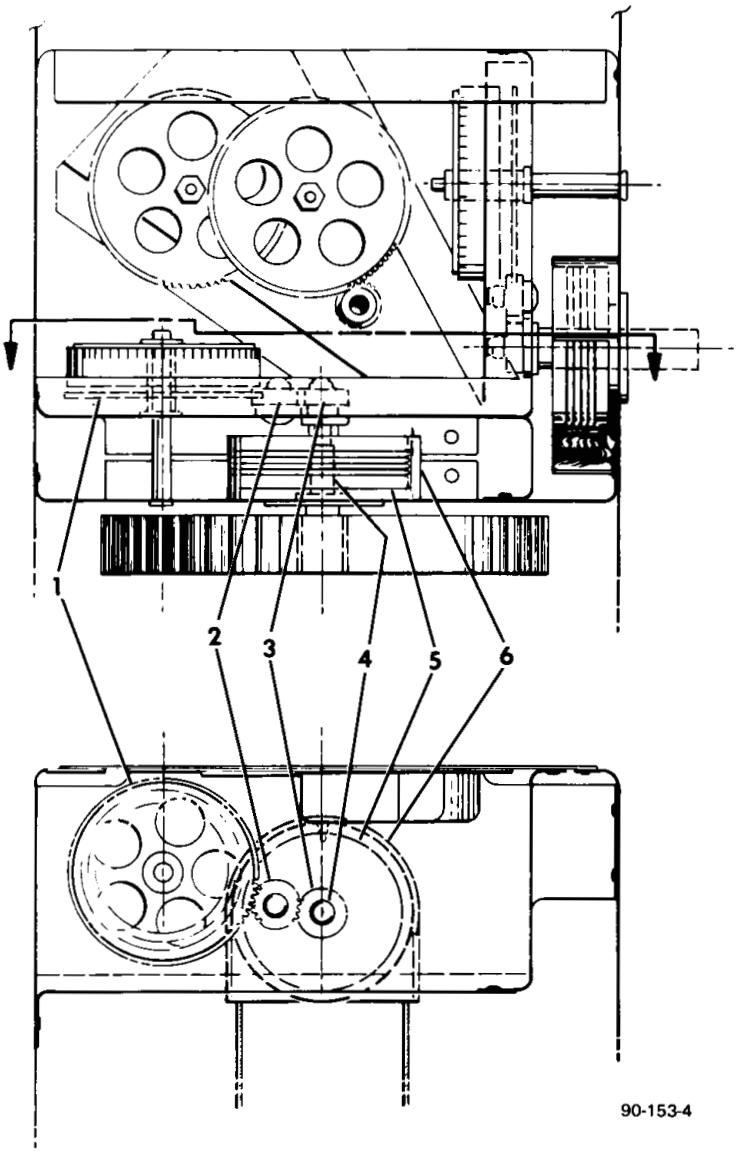
(6) Connect the master brake cylinders.

(7) Install the nonadjustable push-pull tubes from the pedals to the bellcranks.

(8) Connect the nose steering push-pull rod.

(9) Connect the cables to the bellcrank.

(10) Rig the rudder control system.



1. Indicator Gear
2. Idler Gear
3. Drive Gear
4. Drive Shaft
5. Cable Drum
6. Drum Guard

Figure 202
Rudder Trim Tab Indicator Adjustment

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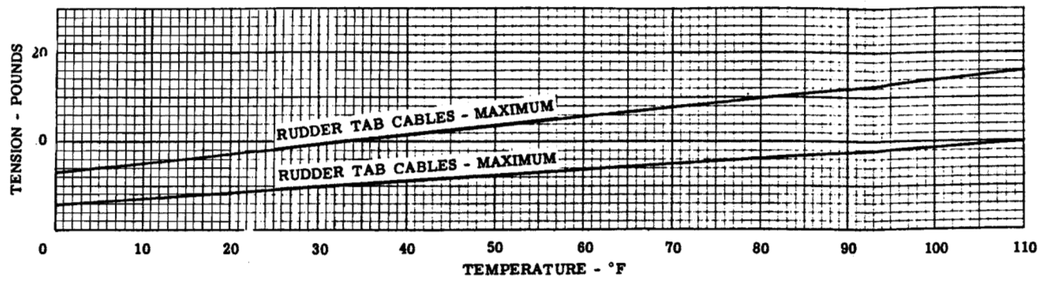

RUDDER TRIM TAB TRAVEL
30° ± 1½° LEFT
30° ± 1½° RIGHT

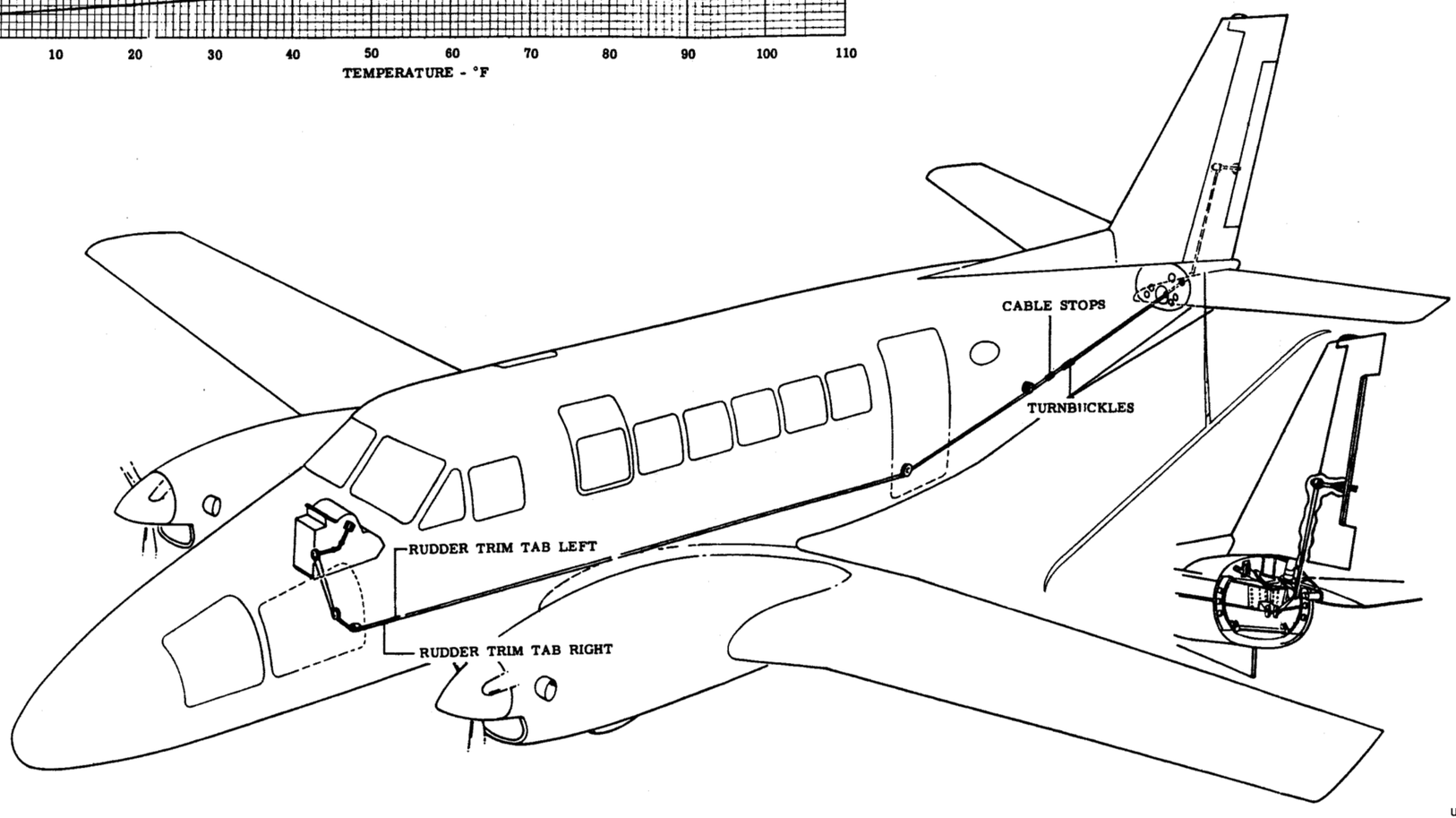

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Figure 203
Rudder Trim Tab Control Rigging and Cable Tension Graph

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4. RUDDER TRIM TAB CABLE

A. Removal

(1) Remove all seats and floorboards from the left side of the cabin.

(2) Remove the toilet (if equipped) and the baggage compartment floorboards.

(3) Remove the tail section access door and the belly access door below the pedestal.

(4) Remove the cable retaining pins and fairleads.

(5) Disconnect the cables at the turnbuckles and connect lead lines to the forward cable ends. Identify the lines for tab left and right movement to assure proper rerouting of the cables. Pull the cables through the access opening below the pedestal.

(6) Remove the front and right side access panels from the pedestal.

(7) Remove the nut from the rudder tab control shaft, then remove the bolt securing the control wheel to the shaft. Remove the shaft and the indicator drive gear. The cable drum and guide can then be removed from the pedestal.

(8) Remove the cable retaining pins from the pedestal pulley brackets and pull the drum and cable from the pedestal.

(9) Unwind the cable from the drum and remove the cable lock pin.

B. Installation

(1) Place the exact center of the cable in the drum slot and install the cable lock pin. The cable must be installed as shown in chapter 27-10-00, Figure 203. Wrap each end of the cable around the drum 2-1/4 turns.

(2) Set the indicator at "0", position the drum and cable guide, and install the control wheel, shaft, and indicator drive gear.

(3) Route and connect the cables, install the cable retaining pins and fairleads, then rig the cables.

(4) Replace all access panels, floor covering and seats.

5. RUDDER TRIM TAB ACTUATOR AND CABLE

A. Removal

(1) Remove the tail cone, the fuselage access doors under the horizontal stabilizer, and the actuator access plate on the rudder.

(2) Disconnect the rudder tab cables at the turnbuckles in the tail section and attach lead lines to the actuator cable ends.

(3) Remove the cable retaining pins and the fairlead on the rudder bellcrank bracket.

(4) Disconnect the trim tab actuator shaft at the trim tab.

(5) Remove the bolt attaching the actuator and remove the actuator and cables through the access opening.

(6) Remove the actuator housing from the actuator and remove the cable chain from the sprocket.

B. Installation

(1) Center the actuator travel,. place the center of the chain on the sprocket and reassemble the actuator.

NOTE: The cable with RH threads moves the trim tab to the right. It must come off the left side of the actuator.

(2) Attach the lead lines to the cable ends and install the actuator and cables through the actuator access hole.

(3) Install the trim tab actuator and access cover, then connect the actuator shaft to the trim tab.

(4) Connect the cables at the turnbuckles and install the cable retaining pins and the fairlead.

(5) Rig the cables and replace the tail cone and access doors.

6. RUDDER TRIM TAB INDICATOR

A. Adjustment

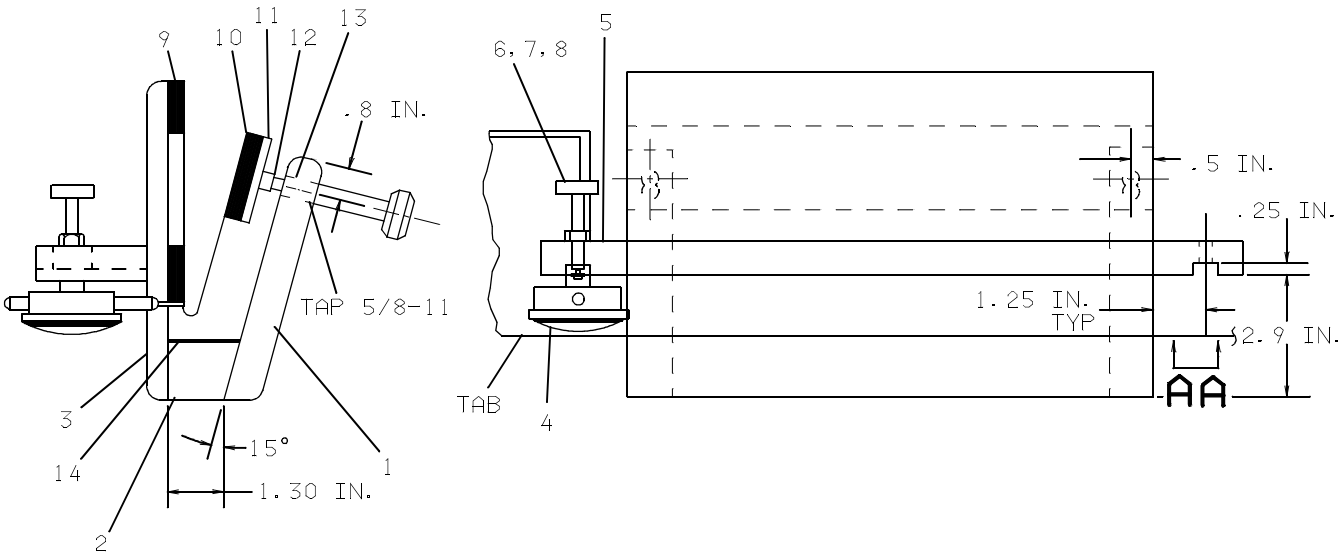
(1) Loosen the setscrew In the side of the trim adjusting knob.

(2) Remove the knob from the shaft.

(3) Raise the indicator dial until it turns freely.

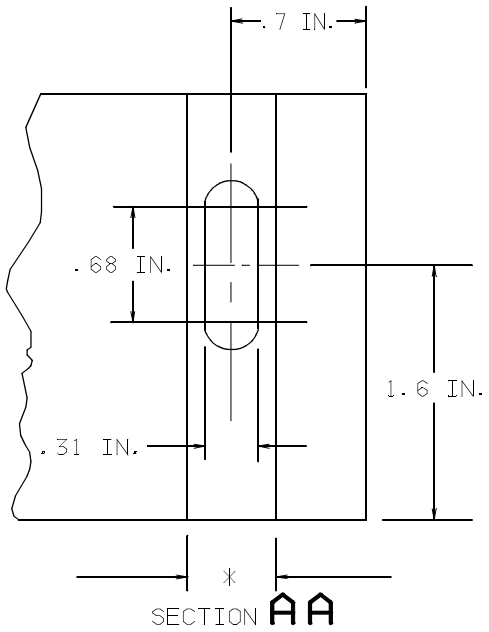
(4) Turn the dial to the desired position and re-engage the reduction gears.

(5) Replace the knob and tighten the setscrew.



ITEM NO.	QUANT.	DESCRIPTION
1	2	3/4 X 1 X 6 ALUMINUM OR EQUIV.
2	2	1 X 1 3/8 X 1 3/4 ALUMINUM OR EQUIV.
3	1	1/2 X 7 1/2 X 10 ALUMINUM OR EQUIV.
4	1	C81Q INDICATOR **
5	1	3/4 X 2 1/2 X 14 ALUMINUM EQUIV.
6	1	1/4 DIA. X 2 CORROSION RES. STL.
7	1	1/4 DIA. X 1 CORROSION RES. STL.
8	1	1/4-28 NUT
9	1	3/8 X 5 X 10 RUBBER
10	1	3/8 X 2 X 10 RUBBER
11	1	1/4 X 2 X 10 CORROSION RES. STL.
12	2	1/2 X 13 X 3 VLIER TORQUE SCREW
13	2	KN813 KEENSERT OR TAP 1/2-13
14	2	1/8 X 1 X 3/4 RUBBER

**P/N OF FEDERAL PRODUCTS CORP., PROVIDENCE, R.I.



*THIS GROOVE TO BE A SNUG FIT TO THE SCREW BRACKET ON THE DIAL INDICATOR

100-135-8

Figure 204
Fabricating Check Fixture for Trim Tab Deflection

7. RUDDER TRIM TAB

A. Rigging

(1) Disconnect the trim tab from the actuator.

(2) Set the control wheel on the pedestal so the tab indicator reads "0".

(3) Rig the cables to the tension shown in Figure 203.

(4) Position the actuator screw in the midpoint of its travel.

(5) Adjust the actuator linkage until the trim tab is in neutral and connect the trim tab to the actuator.

(6) Adjust the tab travel to 30° ± 1-1/2° right and 30° ± 1-1/2° left (2.46 to 2.71 inches from the rudder to the tab trailing edges) with the cable stops in the fuselage tail section. Tighten the stops 40 to 50 inch-pounds torque and safety as shown in Chapter 27-10-00, Figure 205.

(7) Visually check that rudder tab movement corresponds to the movements indicated on the tab indicator: "Left, tab moves right, "Right", tab moves left.

B. Free Play Check

The following check of the rudder trim tab should be performed at 100-hour intervals to detect the presence of wear.

A check fixture (P/N 45-135030-9/810), or the equivalent as shown in Figure 204, is required for making the inspection for free play of the tab. This check fixture may be fabricated (from aluminum or wood) as shown in Figure 204.

(1) Securely lock the control surfaces to prevent movement of the rudder. Set the rudder tab in the neutral position.

(2) Tape the dial indicator check fixture to the rudder so that the dial indicator point is positioned 4.00 inches aft of the tab hinge line and at the top edge of the tab.

(3) Apply a small piece of masking tape (for paint protection) 4.87 inches left of the tab hinge line and along the centerline of the tab actuator. This will be the point of pressure against the tab by the push-pull scale.

Table 201
RUDDER TRIM TAB FREE PLAY LIMITS

1.5 POUND READING	3 POUND READING	
B_____		
2B_____	-A_____	= X_____
D_____		
2D_____	-C _____	= Y_____
X _____	+ _____	= E _____

(E= 0.021 inch maximum)

(4) Apply another piece of masking tape in the corresponding position on the right surface of the tab for the same purpose.

(5) Zero the dial indicator at no load initially. Do not reset during the checking procedure.

(6) With the push-pull scale at the point of the masking tape, apply a full 3-pound left side load. Record this reading as "A".

(7) Release half of the load until a 1.5-pound left side load is obtained. Record the dial reading as "B".

(8) Apply a full 3-pound right side load at the masking tape on the right surface. Record the dial reading as "C".

(9) Release half of the load until a 1.5-pound right side load is obtained. Record the dial reading as "D".

(10) Enter the recorded values on a copy of Table 201 and proceed as follows:

(a) Multiply "B" by 2 and record as "2B".

(b) Subtract "A" from "2B" and record as "X".

(c) Multiply "D" by 2 and record as "2D".

(d) Subtract "C" from "2D" and record as "Y".

NOTE: The results of "X" and "Y" can be negative numbers. If the sum of "X" and "Y" is negative, that number must still be within the limits designated by "E".

(e) Add "X" and Y" and record as "E".

(11) If deflection of the tab is within the allowable limit, the tab and its linkage are in good condition.

(12) If the free play is excessive, disconnect the tab actuator rod and visually inspect the bolts and bushings for indications of excessive wear. Replace excessively worn parts.

FLIGHT CONTROLS
ELEVATOR AND TAB
MAINTENANCE PRACTICES

1. ELEVATOR

A. Removal

(1) Remove the three bolts which secure the elevator torque tube collar to the elevator horn.

(2) Remove the hinge bolts which attach the elevator to the horizontal stabilizer.

(3) Disconnect the elevator static ground wires.

NOTE: Any time an elevator is replaced or repaired, the stabilizer trim system should be flight tested as outlined in this section.

(4) Remove the elevator.

B. Installation

WARNING: On airplanes U-132, U-146, U-148 and After, and prior serials with Kit No. 99-5011 installed, elevators which have the boundary separation wedge on the trailing edge must be installed in pairs. Installing an elevator with the wedge and an elevator without the wedge on the same empennage can alter the flight characteristics of the airplane.

(1) Attach the elevator horn to the hinge assembly and torque the bolt 50 to 70 inch-pounds. If additional torque is required to align the cotter pin hole, it is permissible to torque the bolt to 100 inch-pounds maximum.

NOTE: If the torque tube collar has been removed from the elevator torque tube, it may be installed as follows:

1. Place the torque tube collar on the torque tube and insert the taper pins wet with zinc chromate primer.

2. Torque the nuts 30 to 40 inch-pounds. A maximum torque of 60 inch-pounds may be used to align the cotter pin holes. The small end of the pin body (tapered shank) should extend 0.000 to 0.060 inch beyond the surface of the support.

(2) Compare the length of the torque tube which extends past the torque tube collar and the depth of the recess in the control horn. If the torque tube extends far enough to bottom out in the control horn before the torque tube collar and control horn flanges butt together, material should be removed from the torque tube as necessary to allow the flanges to butt together.

(3) With the end of the torque tube inserted into the recess of the control horn, attach the elevator to the horizontal stabilizer and push the elevator inboard until the torque tube collar butts firmly against the control horn.

(4) With the elevator positioned thus, check the hinges to determine if shims are required between the elevator horn and elevator hinge assembly to center the elevator attaching brackets on the hinges.

NOTE: If either elevator hinge is installed or replaced, torque the elevator hinge attachment bolts 50 to 70 inch-pounds.

Whenever the elevator hinge bearings are replaced the elevator hinge should be dye checked for cracking.

(5) Align the three bolt holes in the flanges and install three bolts. Torque the nuts 30 to 40 inch-pounds before inserting the cotter pin. As each nut is being torqued, ascertain that the respective elevator hinge bracket is gripping the bushing so that the bushing, bolt and hinge bracket will be held tightly together. The bolt must not rotate in the bushing. If the slot in the nut does not align with the hole in the bolt, tighten the nut until the cotter pin can be inserted. Do not exceed 60 inch-pounds torque. Check the hinges and the horn through full travel after completing the installation.

2. ELEVATOR CONTROL HORN ASSEMBLY

A. Removal

(1) Disconnect the elevator pushrod assembly from the control horn assembly.

(2) Remove the pivot bolt from the control horn assembly and remove the control horn assembly.

B. Installation (New Elevator Control Horn Assembly)

(1) Insert an AN174-16/M/ bolt through the horn assembly pivot point and the hinge support assembly, with an AN960-416 washer between the horn assembly and the hinge support assembly.

(2) Place an AN960-416L washer and an AN310-4 nut on the inboard end of the AN174-16 bolt. Tighten the nut until snug, but do not apply the specified torque or install the cotter pin at this time.

(3) Compare the length of the torque tube portion extending past the torque tube assembly collar with the depth of the recess before the flanges on the torque tube assembly collar and horn assembly butt together. Remove material from the torque tube as necessary to allow the flanges to butt together.

(4) With the end of the torque tube inserted into the recess of the horn assembly, temporarily attach the elevator to the horizontal stabilizer with the hinge bolts, and push the elevator inboard until the flange on the torque tube assembly collar butts firmly against the flange on the horn assembly.

(5) While the elevator is positioned as described in Step (4), check the elevator hinges to determine if shims are required between the horn assembly and the hinge support assembly to center the elevator attaching brackets on the hinges. If shims are necessary, install AN960-416 and AN960-416L washers (maximum of two in any combination) between the horn assembly and the hinge support assembly. It washers are added, an AN174-17/M/ bolt may be required instead of the AN174-16 /M/ bolt.

(6) When the elevator hinges and the horn assembly are operating satisfactorily through full travel, remove the elevator. Note the installation of all washers, particularly the number and type of washers used between the horn assembly and the hinge support assembly, to ensure installation of the required washers.

(7) Prime the inside diameter of the horn assembly bushing and the shank of the AN174-16/M/ or AN174-17/M/ bolt with primer (120, Table 6, 91-00-00).

(8) Apply retaining compound (121, Table 6, 91-00-00) evenly but sparingly to the bolt shank. Install the horn assembly with all washers positioned as previously, inserting the bolt shank into the horn assembly bushing with a twisting motion to ensure that both surfaces will be coated with retaining compound.

CAUTION: *Do not loosen the AN310-4 nut to align the cotter pin holes. If the torque requirements described in Step (9) cannot be obtained, add one AN960-416L washer on the inboard end of the AN174-16 or AN174-17 bolt and repeat Step (9) until the cotter pin holes will align within the required torque.*

(9) Torque the AN310-4 nut 50 to 70 inch-pounds and install an MS24665-134 cotter pin through the nut and the AN174-16 or AN174-17 bolt. A maximum torque of 100 inch-pounds may be applied to the AN310-4 nut to align the cotter pin holes.

(10) Allow 2 to 4 hours for the retaining compound (121, Table 6, 91-00-00) to dry.

3. ELEVATOR PUSHROD

A. Inspection

The elevator pushrods must be inspected every 500 hours as follows:

(1) Remove the tail cone and access panels required to gain access to the elevator pushrods.

(2) Mark each pushrod tube and rod end for later installation.

(3) Remove the elevator pushrods from the airplane.

(4) Measure the distance from center to center of the rod end bolt holes and note for later installation.

(5) Remove the rod ends from the pushrod tubes.

(6) Using low power magnification, inspect the pushrod tube for corrosion and cracks. Replace any pushrod tubes that show evidence of corrosion or cracks.

(7) Inspect the pushrod tubes for straightness using a flat surface that is as long as the pushrod tube. Rotate the pushrod tube and check for gap between the flat surface and the pushrod tube using a 0.060 inch feeler gage. Replace any pushrod with a gap of more than 0.060 inch gap.

(8) Install the rod ends in the pushrod tube to the measurements noted in Step (4).

CAUTION: *The threads of the rod ends should be visible through the inspection holes in the ends of the pushrod tubes.*

(9) Install the elevator pushrods in the airplane.

(10) Install the tail cone and access panels.

4. ELEVATOR CONTROL CABLE

A. Removal

(1) Remove the pilot's seat, floor covering and floor panels on the left side of the crew compartment.

(2) Remove all of the access plates on the left passenger seat deck.

(3) Remove the small access plate in the center of the entrance way.

(4) Remove the baggage compartment floor panel and the toilet (if installed).

(5) Remove the tail section access panel.

(6) Label both forward elevator cable ends in relation to their attaching point on the quadrant.

(7) Remove the cable retaining pins from the pulley brackets.

(8) Disconnect the elevator cables from the quadrant and at the turnbuckles in the fuselage tail section.

(9) Connect lead lines to the cables and remove the cables through the fuselage tail section.

NOTE: Label the lead lines for cable connection at the forward or aft points of the quadrant to assure proper installation of the cables.

(10) Disconnect the two aft elevator cables from the forward and aft end of the bell crank assembly. Label the cable ends in relation to their respective attaching points.

B. Installation

(1) Connect the longer of the two aft cables to the aft end of the bell crank assembly and the shorter cable to the forward end.

(2) Attach the forward cables to the quadrant, route them through the pulleys and connect the turnbuckles (Ref. Figure 201).

NOTE: The shorter (elevator down) cable attaches to the forward position on the quadrant.

(3) Replace the cable retaining pins in the pulley brackets.

(4) Rig the cables as outlined under the heading ELEVATOR CONTROL SYSTEM RIGGING.

(5) Replace all access panels, floor coverings, seats, and toilet (if equipped).

5. ELEVATOR CONTROL SYSTEM

A. Rigging

NOTE: Set the horizontal stabilizer in the neutral (0°) position and place the elevator travel gage at station 55.15 on the horizontal stabilizer to determine elevator positions in this procedure.

(1) Remove the tail cone, two round access plates on each side of the aft fuselage, the pilot's seat and the floorboard immediately below the pilot's seat.

(2) Install the five 3/16 inch rig pins (Ref. Figure 201). The three aft rig pins may not align if the adjustments in Steps (8) and (9) have been accomplished previously.

NOTE: If necessary, the 3/16 inch diameter rig pin used in the elevator control horn and hinge may be tapered at one end to facilitate installation. Starting at a point 0.5 to 0.75 inch from the end, taper the pin down to a diameter of approximately 0.06 inch at the end.

(3) With the elevators in neutral and the rig pin installed in the forward bell crank, position the control wheel in neutral by adjusting the forward pushrod to a length of 16.5 inches from pin center to pin center. On airplanes U-132, U-146, U-148 and After, and prior serials with Kit No. 99-5011 installed, the forward pushrod should be adjusted to a length of 16.8 inches.

NOTE: The inspection holes on the elevator control pushrods (U-165 and After) have been increased from 0.062 inch diameter to 0.093 inch diameter, and the rods have had an anti-corrosion treatment. For inspection and anti-corrosion treatment of the control rods on those airplanes prior to U-165, refer to Service Instructions No. 0334-152, Rev II or subsequent.

(4) Rig the elevator control cables to the tension shown on the graph (Ref. Figure 201).

(5) Check each elevator for neutral (0°) position with the elevator travel gage.

(6) If the elevators are not in the neutral (0°) position, disconnect the aft pushrods and insert the two aft rig pins. Check the position of each elevator with the elevator travel gage.

(7) If the trailing edge of one or both elevators deviates MORE THAN ± 2° off the neutral (0°) position, the elevator(s) should be replaced.

(8) If the elevator trailing edge deviates NOT MORE than ± 1° off the neutral (0°) position, remove the two aft rig pins from the elevator horns and adjust the two aft pushrods which connect the torque shaft to the elevator horns until the elevator is in the neutral (0°) position.

(9) If the trailing edge is MORE THAN ± 1° but NOT MORE than ± 2° off neutral, remove the two aft rig pins from the elevator horns and the torque shaft rig pin. Then adjust the elevator to ± 1° with the adjustable pushrod which connects the bell crank to the torque shaft. Repeat the procedure noted in Step (8) to bring the elevator to the neutral (0°) position.

(10) Remove the rig pin.

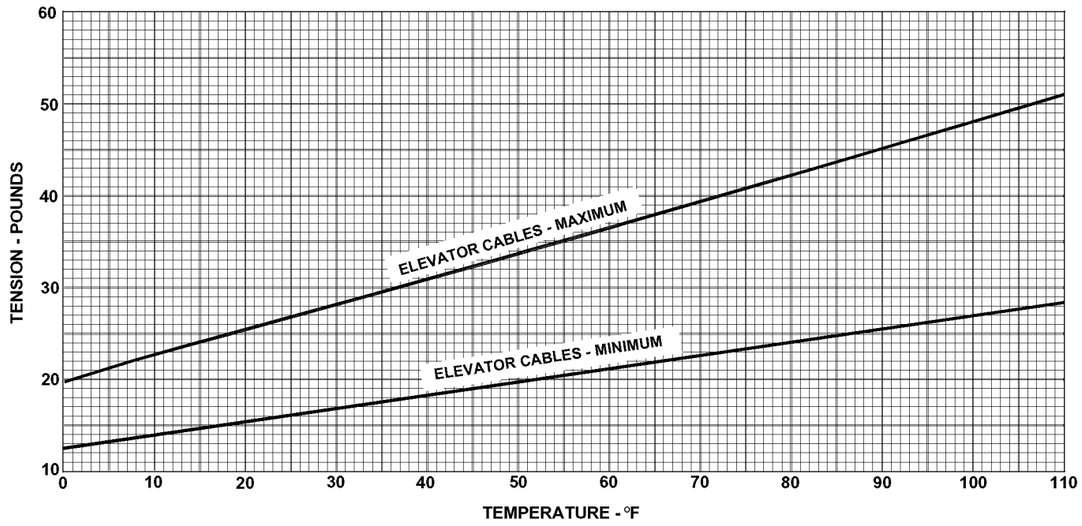
WARNING: Check to see that the threads extend beyond the inspection holes provided in the push-pull rod ends.

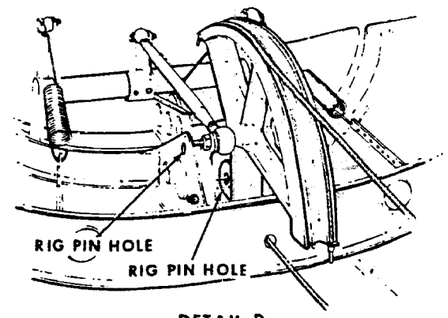
(11) Adjust the elevator surface travel with the adjustable stops provided at the elevator horns for 12° +1°/ -0° trailing edge UP (with the horizontal stabilizer at 0 incidence) and 15° +1°/ -0° trailing edge DOWN (with the horizontal stabilizer at 0 incidence). After the travel has been set, adjust the quadrant stops on the forward bell crank to maintain 0.37 ± 0.06 inch clearance.

NOTE: Visually check that the elevator movement corresponds to the movement of the control column. Check for proper downspring force and elevator system friction. Use a hand force gage, or equivalent tool such as a fish scale, to measure the force required to move the control column from the full forward position through neutral to full aft. Note the gage reading while the control column passes through neutral. Measure the force required to move the control column from the full aft position forward through neutral. Note the gage reading as the control column passes through neutral going forward. The average of the two preceding readings should indicate a downspring force of 24 to 26 pounds. If this reading is not obtained, readjust the downspring to obtain this tension. If the existing strap will not provide sufficient adjustment, there is a longer strap available which can be utilized. The difference between the two readings should indicate an elevator system friction of 8 to 12 pounds. On airplanes U-132, U-146, U-148 and After, and prior serials with Kit No. 99-5011 installed, the elevator downspring force should be 18 to 20 pounds and the elevator system friction should be 0 to 10 pounds.

(12) Remove the elevator travel gage.

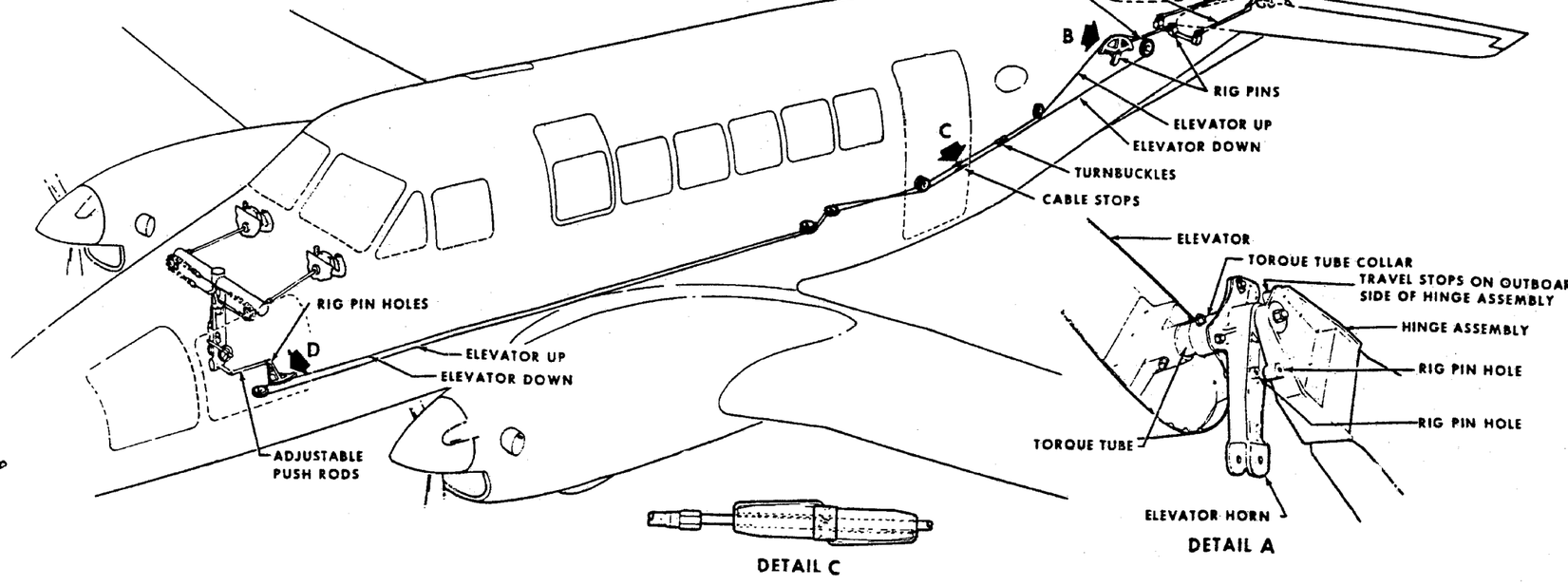
(13) Install the pilot's floorboard, seat, the tail cone and fuselage access plates.



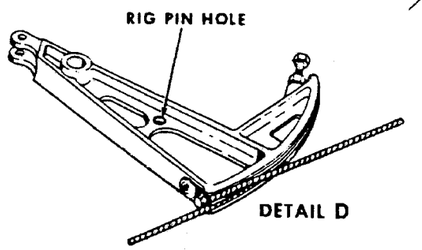


DETAIL B

ELEVATOR TRAVEL
12° + 1° -0° UP
15° + 1° -0° DOWN



DETAIL A



DETAIL D

DETAIL C

U#27B
034097AA.AI

Figure 201
Elevator Rigging and Cable Tension Graph

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6. ELEVATOR AFT BELLCRANK AND TORQUE TUBE

A. Inspection

Inspect the bell cranks for loose taper pins and cracks near the taper pin holes at 100-hour intervals (Ref. Figure 202). If the taper pins and bell cranks are loose or require replacement, observe the following precautions:

NOTE: Any time the aft bell crank has been removed for replacement or repairs, torque the bellcrank pivot bolt 56 to 78 inch-pounds upon installation.

(1) The small end of the tapered shank of the taper pin should be flush with, or extend no more than 0.060 inch through the retaining washer and nut. If the pin does not extend within these limits, the hole should be reamed out as necessary to obtain a proper fit.

(2) If the pin extends beyond 0.060 inch, the taper pin is worn or the hole is too large. If the taper pin is worn, install a new pin and check the pin for proper extension.

(3) Set the pin with a light tap from a rawhide or nylon mallet. When securing the pin in place, install the washer with the cupped side toward the bell crank arm. Torque the retaining nut 30 to 35 inch-pounds.

CAUTION: *To avoid damaging the tube or bell crank casting, do not drive the taper pins or over torque them.*

(4) If the hole in the bell crank is too large, the assembly must be replaced or the hole reamed out as required for installation of the next larger size taper pin.

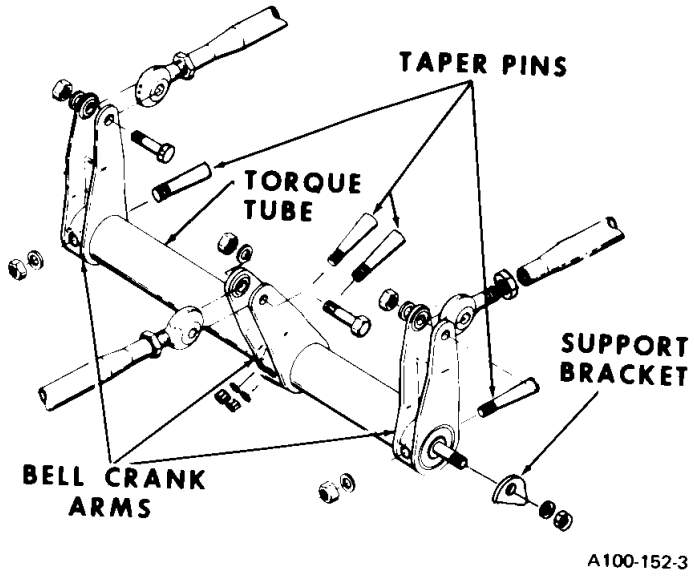

Figure 202
Elevator Bellcrank Torque Tube

7. ELEVATOR TORQUE TUBE

A. Inspection

The elevator torque tube should be periodically inspected for cracks, particularly around the taper pin holes (Ref. Figure 203). The first inspection should be accomplished at 5,000 hours of operation with subsequent inspections at each 1,000 hours thereafter.

B. Torque Tubes Assembled With EA9309NA

(1) Remove all power from the airplane and disconnect the battery. Display warning notices prohibiting connection of airplane electrical power.

(2) Gain access to the torque tube assemblies by removing the tailcone.

NOTE: In the following Step, have another technician present to verify that there is no movement in the elevators.

(3) Verify that the both torque tubes have no movement at the supports:

(a) Lift both elevators to approximately the center position.

(b) Alternately move the left side up and the right down and vice versa.

(4) If movement is noted at the supports, proceed to Step (3) of the following procedure.

(5) If no movement is noted, perform a visual inspection of both torque tube assemblies using a 10X or stronger magnifying glass:

(a) Pay particular attention to the taper pin locations.

(b) Verify that there are no cracks, corrosion, and/or damage on the entire exposed portion of the torque tube assembly.

(6) If both torque tube assemblies are found to be airworthy, proceed to Step (14) of the following procedure.

(7) If cracks, corrosion or damage is noted, replace the existing torque tube assemblies with new P/N 115-610010-325 or P/N 115-610010-383 assemblies (Ref. 27-30-00). Proceed to Step (13)(b) of the following procedure.

C. Torque Tubes Assembled Without EA9309NA

(1) Remove all power from the airplane and disconnect the battery. Display warning notices prohibiting connection of airplane electrical power.

(2) Gain access to the torque tube assemblies by removing the tailcone.

(3) Disconnect the left and/or right control horn from the inboard torque tube supports (Ref. Figure 203).

(4) Remove the elevator(s) from the airplane (Ref. ELEVATOR REMOVAL).

(5) Remove the attaching hardware from the center torque tube supports.

(6) Remove the torque tube assemblies from the elevator(s).

(7) Mark for installation and remove the taper pins from both the inboard and center torque tube supports.

(8) Remove both the inboard and center supports from the torque tubes (Ref. Figure 203).

WARNING: Solvents, primers, and paints are flammable and toxic to skin, eyes, and respiratory tract. Skin and eye protection is required. Avoid repeated or prolonged contact. Keep away from flames or sources of heat. Use in a well ventilated area or respiratory protection equipment may be required.

(9) Thoroughly clean the torque tubes and both supports with aliphatic naphtha.

(10) Using a 10X or stronger magnifying glass, perform a visual inspection of both entire torque tubes and support brackets for cracks, corrosion, or damage. Pay particular attention to the taper pin locations. Verify that there are no cracks, corrosion, and/or damage on the entire exposed portion of the torque tube assembly.

NOTE: All personnel performing a fluorescent liquid penetrant inspection must be certified Nondestructive Test Personnel who have obtained a Level II or Level III certification in fluorescent liquid penetrant inspection methods in accordance with NAS 410 (Ref. SIRM, Chapter 20-00-00).

(11) Perform a fluorescent liquid penetrant inspection of the taper pin bores on the torque tube and both supports for cracks.

NOTE: A magnetic particle inspection on the torque tube only is allowed in lieu of the fluorescent liquid penetrant inspection method (Ref. SIRM, Chapter 20-00-00).

(12) If cracks, corrosion or damage is noted, replace the existing torque tube assembly(s) with new P/N 115-610010-325 or P/N 115-610010-383 assembly(s) (Ref. 27-30-00). Proceed to Step (13)(b).

(13) If no cracks, corrosion or damage is found, assemble the center and inboard torque tube supports to the torque tubes with structural adhesive EA9309NA. Using the taper pins removed in Step (7), install “wet” using structural adhesive EA9309NA. Avoid application of adhesive on taper pin threads.

CAUTION: *DO NOT TAP TAPER PINS. Install the taper pins by hand. The taper pins must be installed into their original holes as previously marked in Step (7). The small end of the pin body (tapered shank) should extend 0.000 to 0.060-inch beyond the surface of the support. If the pin does not extend within these dimensions, the hole is considered oversized and a new torque tube assembly will be required. Torque the nuts on the taper pins 35 to 40 inch-pounds. If required, a maximum torque of 60 inch-pounds is permissible if a cotter key is required for alignment.*

NOTE: Apply epoxy primer to all exposed surfaces of the torque tube assemblies per the applicable SIRM, Chapter 20-30-00.

(a) Install new P/N MS24665-132 cotter pins in the aligned taper pin/nut holes or install new self locking nuts as required per installation.

(b) Install the existing or new torque tube assemblies and connect the control horn.

(c) Install the elevator(s), using the existing attaching hardware (Ref. ELEVATOR INSTALLATION).

(14) Install the tailcone.

(15) Connect the airplane battery, remove warning notices, and restore power.

(16) Return airplane to service.

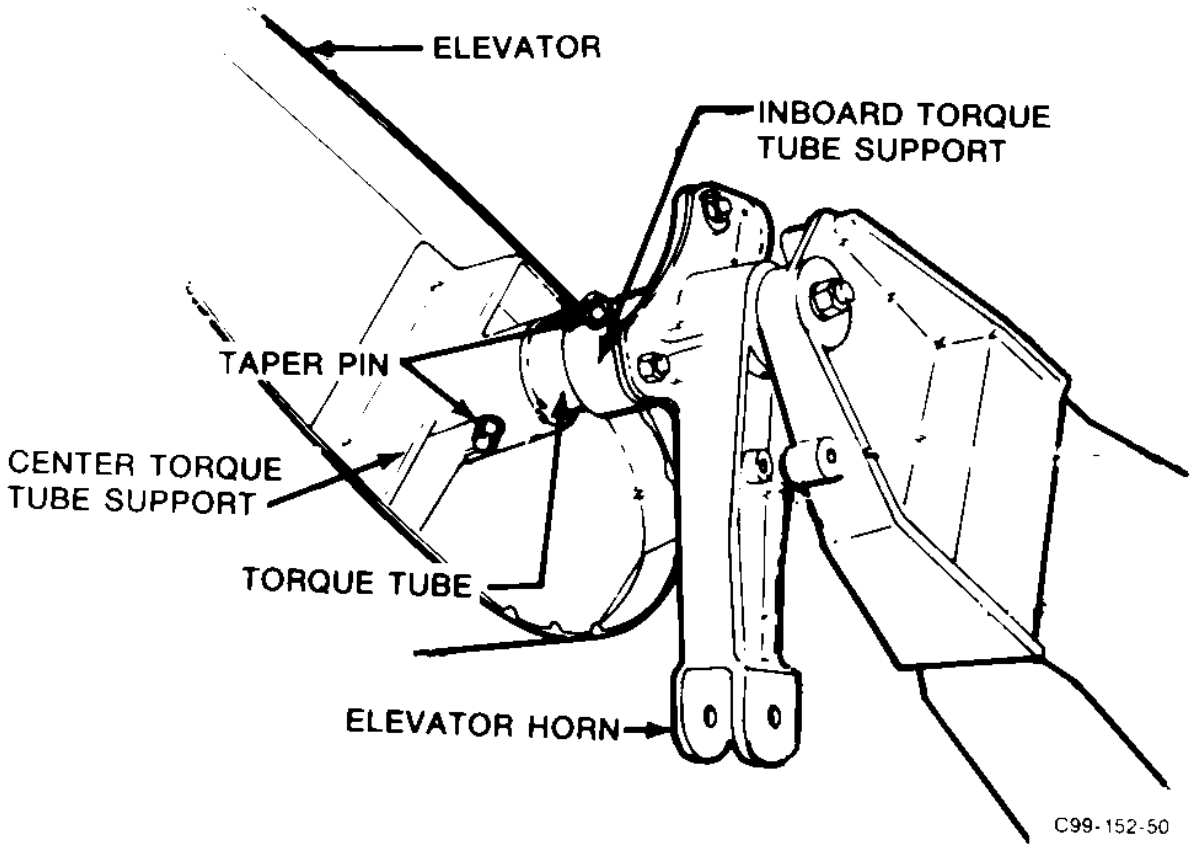

Figure 203
Elevator Torque Tube Inspection

FLIGHT CONTROLS
STALL WARNING SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

A. (U-1 Thru U-164, except U-50)

The stall warning system consists of a circuit breaker on the side panel by the copilot, a transistor switch mounted at the RH end of the defroster duct behind the instrument panel, and a lift transducer on the leading edge of the left wing. A stall warning horn is mounted aft of the light control panel in the overhead. When aerodynamic pressure on the lift transducer vane indicates that a stall is imminent, the transistor switch is actuated to complete the circuit to the stall warning a horn. Airplane serials U-137 and After, and prior airplanes complying with Service Instruction No. 0378-355, are equipped with a heated mounting plate and vane of the lift transducer to provide ice protection effectively at temperatures to -22°F at speeds up to 235 knots.

CAUTION: *Stalling airspeeds should be expected to increase when ice has accumulated on the airplane due to the distortion of the wing airfoil. For the same reason, stall warning devices are not accurate and should not be relied upon.*

B. U-50, U-165 and After and Earlier Airplanes Incorporating Kit 99-3027)

The lift transducer provides the information necessary to the transistorized switching device which transmits current to the stall horn, thereby sounding the stall warning at a point just above that at which a stall would occur.

The lift transducer contains a reed switch and a permanent magnet. The magnet is mounted on the transducer vane which responds to aerodynamic forces applied during flight. When the magnet is moved into close proximity with the reed switch, the reed switch will close to ground. This ground signal is sensed by the transistor switch which begins to conduct current to the horn.

Since ice on the transducer vane would render the system inoperative, the transducer vane and mounting plate are electrically heated. Current for the transducer vane heater is taken from the 5-ampere (2 ampere on airplanes with Kit 99-3027) circuit breaker that also provides power to the transistor switch module. A separate 15-ampere (5 ampere on airplanes with kit 99-3027) circuit breaker provides current for the mounting plate heater.

CAUTION: *Due to distortion of the airfoil when ice has accumulated on the wing of the airplane, stalling airspeeds should be expected to increase. Consequently, the stall warning system may tend to lose it's accuracy during flight in icing conditions.*

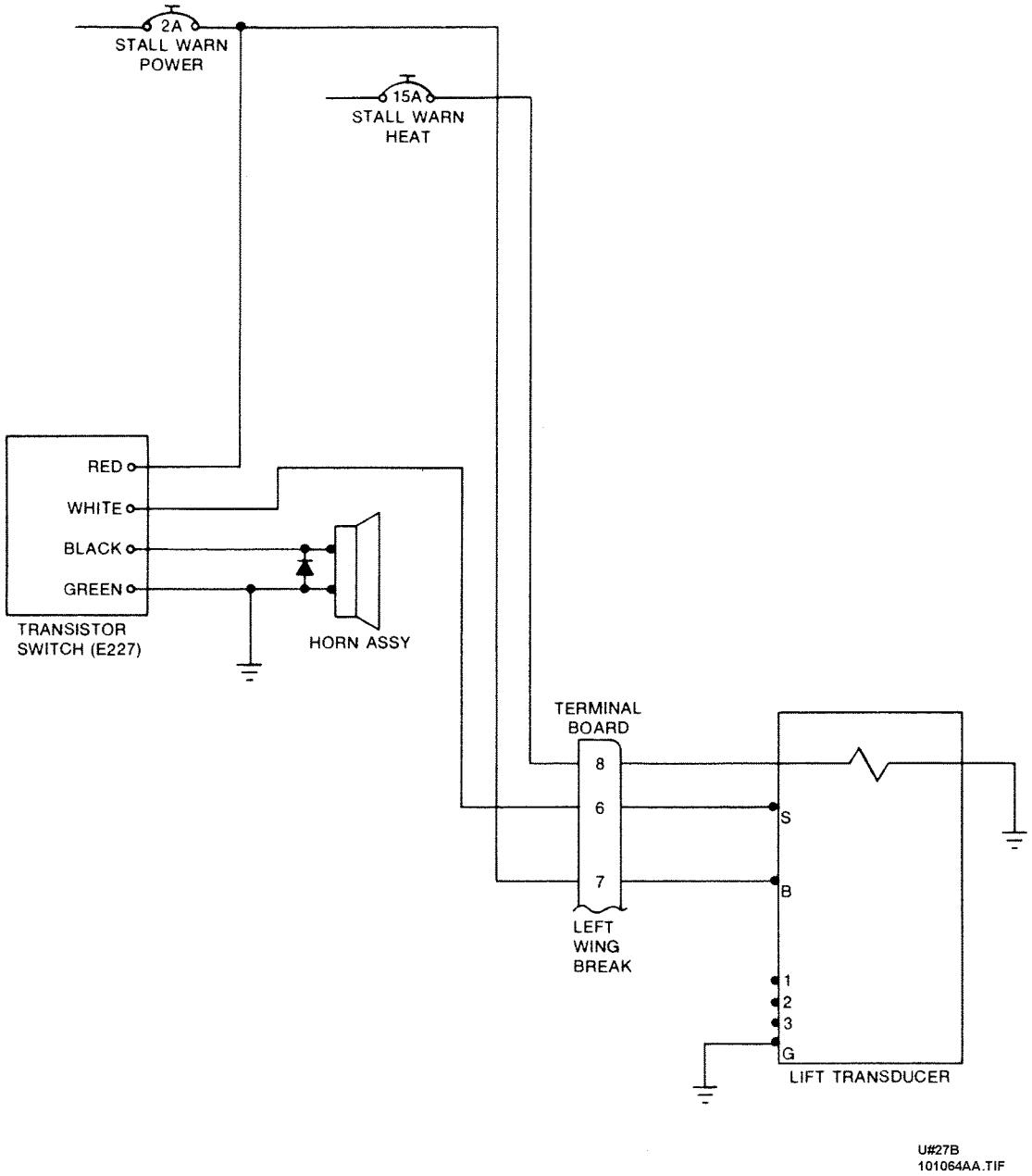

Figure 1
Stall Warning System U-1 thru U-136 (except U-50) with Kit 99-3020 installed, and U-137 thru U-164 (all serials without Kit 99-3020)

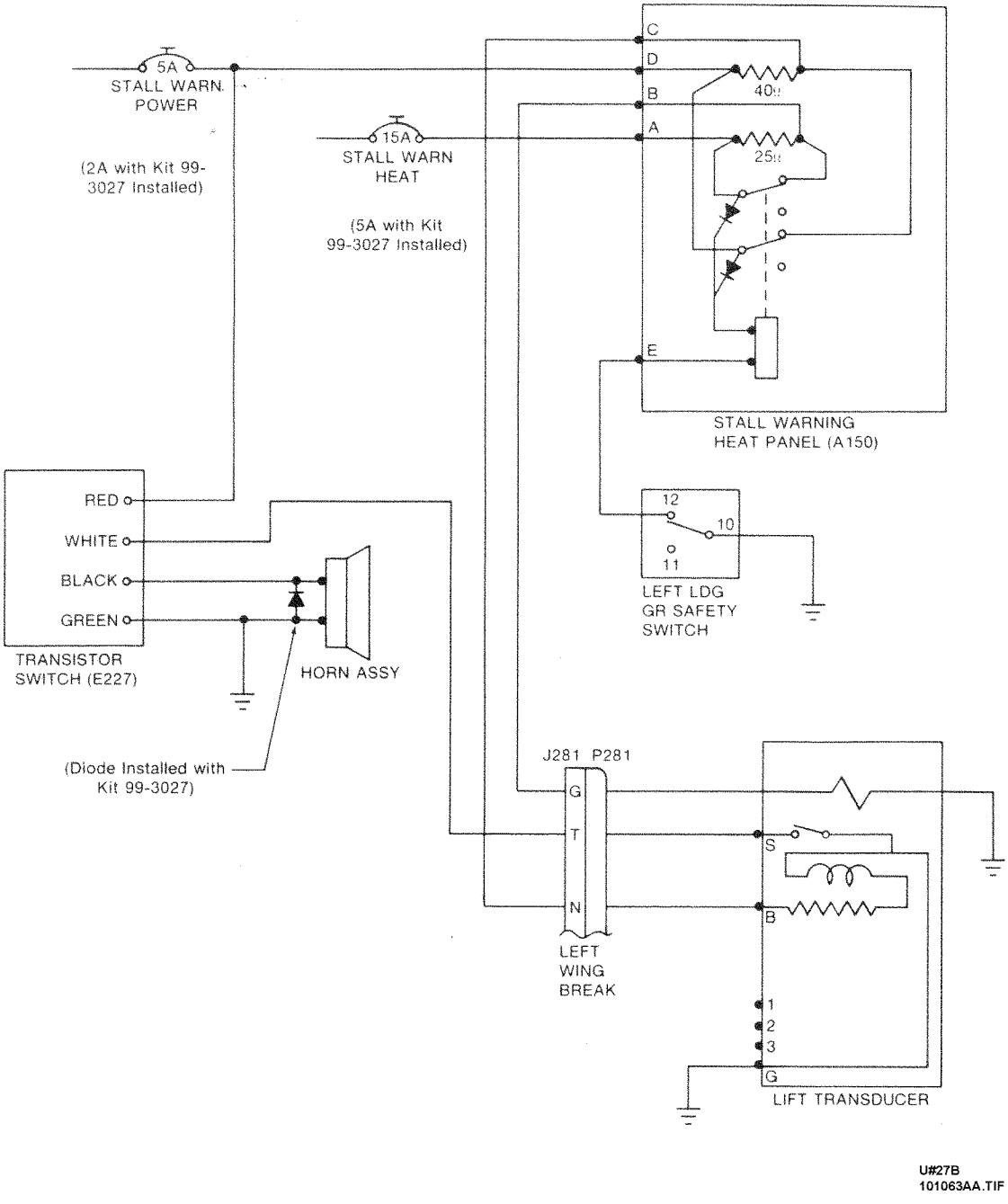

Figure 2
Stall Warning System
(U-50, U-165 and After. and earlier airplanes with Kit 99-3027 Installed)

FLIGHT CONTROLS
STALL WARNING SYSTEM
TROUBLESHOOTING

1. STALL WARNING

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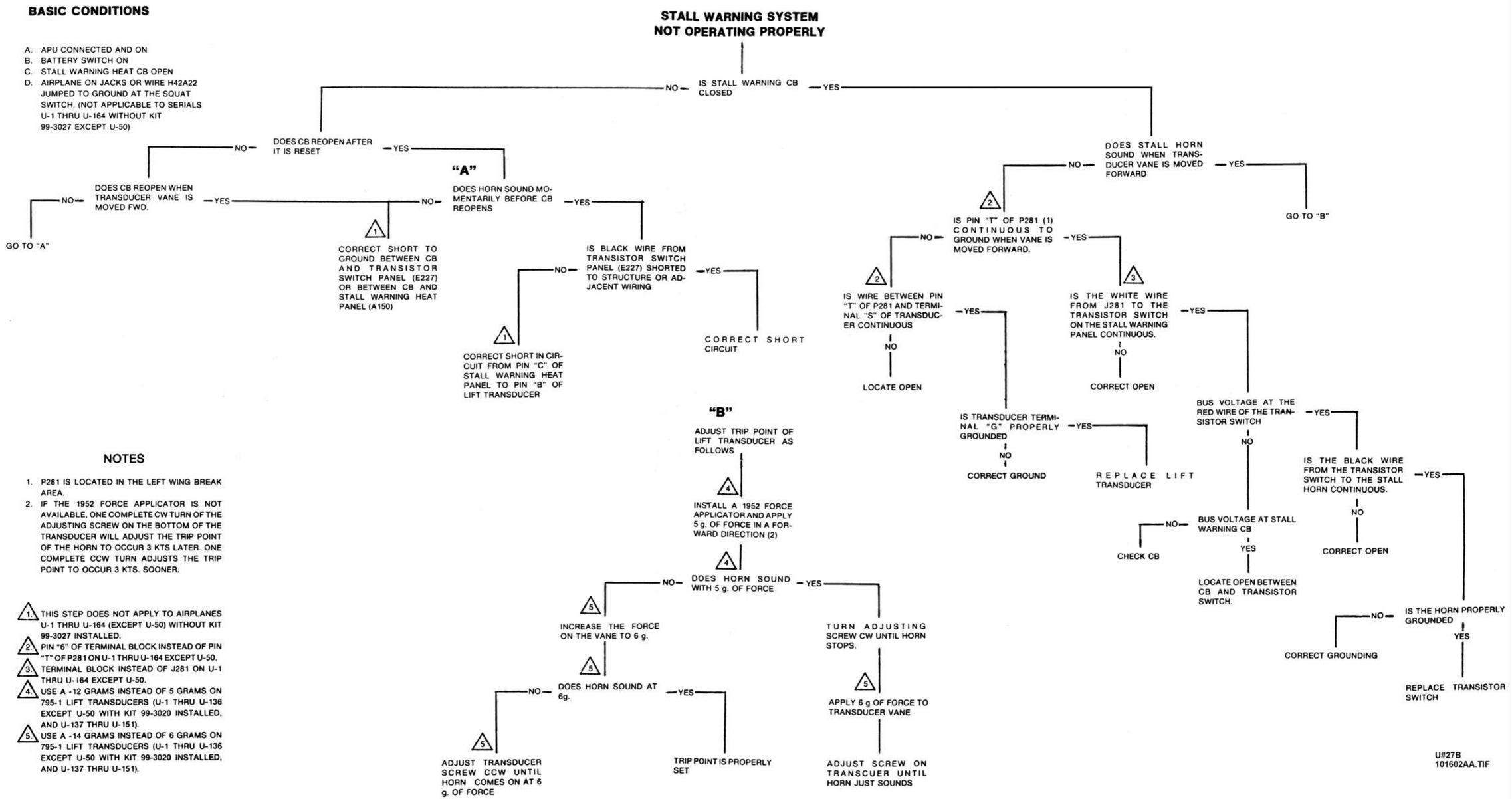

Figure 101
Troubleshooting - Stall Warning System

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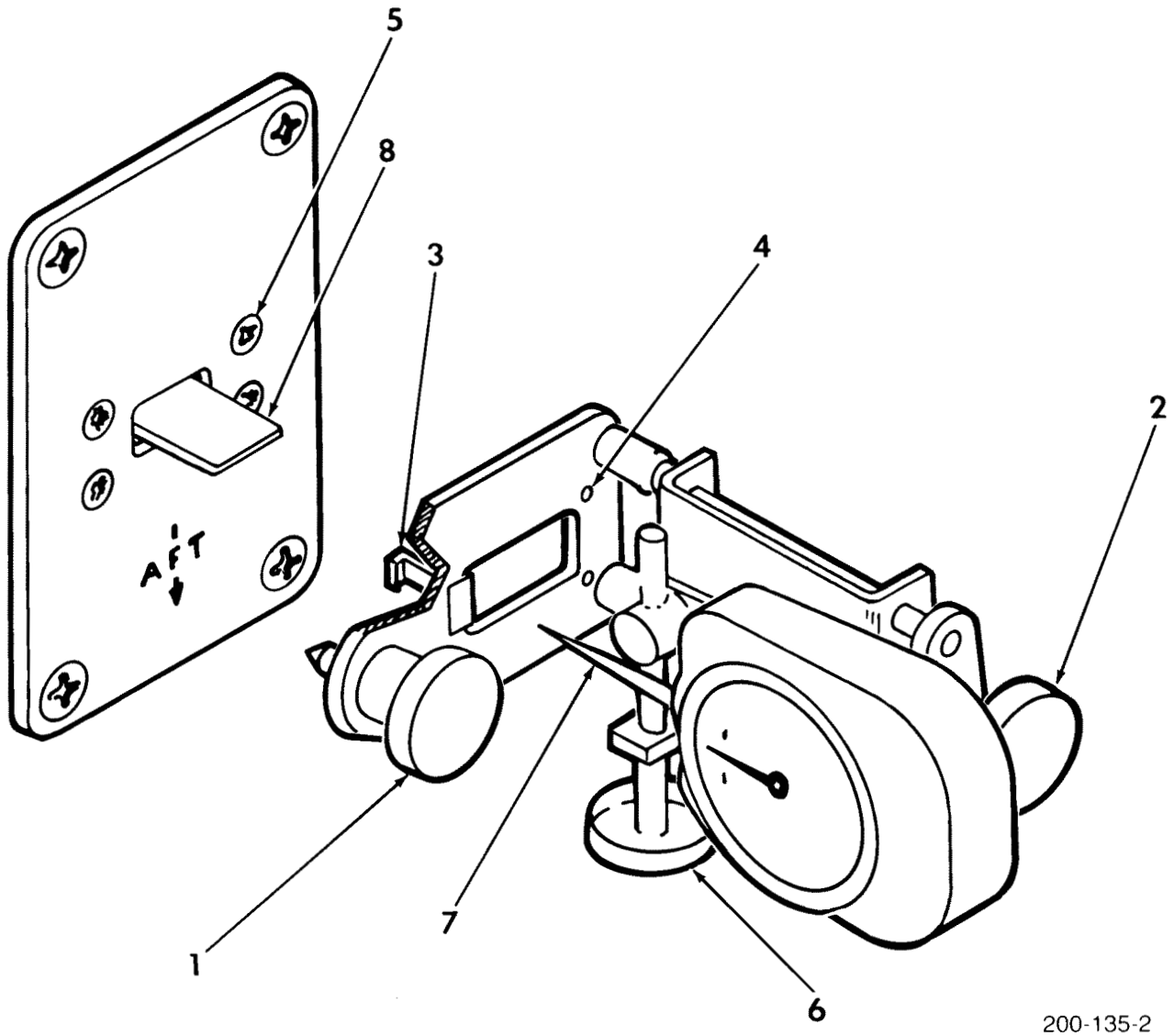


1. Insert positioner anchor tab (3) into vane opening.
2. Hook anchor tab (3) onto the mounting plate.
3. Keeping anchor tab engaged, shift positioner until cone points (4) enter recesses in the screw heads (5).
4. Turn screw (1) counterclockwise until knob clutch slips.
5. Turn screws (2) and (6) as required to bring gage pressure arm (7) to bear on surface of vane (8) within 1/32" of tip.
6. Apply pressure to aft surface of vane (8) for negative (−) tip grams and forward surface for positive (+) tip grams.
7. Turn screw (6) to adjust force (tip grams) as required.

Figure 102
1952 Force Applicator Usage Instructions

FLIGHT CONTROLS
STALL WARNING SYSTEM
MAINTENANCE PRACTICES

1. FLIGHT

A. Calibration and Adjustment

Perform the following flight check in smooth air at a safe altitude under stabilized conditions:

(1) With the flaps and landing gear down and power off (idle power), slowly reduce speed (at a rate not higher than one knot per second) until the stall warning horn sounds. Record the indicated airspeed at the time the warning horn was actuated. Continue to slowly reduce speed until the airplane stalls, then record the stall speed. The stall warning horn should actuate at an airspeed 5 to 10 knots faster than that at which the stall occurred.

NOTE: If the stall warning horn did not actuate at the proper speed, land the airplane and remove the screws securing the mounting plate of the lift transducer to the left wing leading edge. Turn the adjusting screw on the bottom of the unit counterclockwise if the warning horn actuated at a speed of more than 10 kts above stall speed, or clockwise if the warning horn actuated at a speed of less than 5 kts above stall speed. One complete rotation of the adjusting screw changes the transducer by 3 knots.

(2) With the flaps and gear up and power off (idle power) the stall warning should actuate within 5 to 10 knots above a stall.

FLIGHT CONTROLS
HORIZONTAL STABILIZER
MAINTENANCE PRACTICES

1. HORIZONTAL STABILIZER CONTROL SYSTEM

The horizontal stabilizer is an electrically operated flight control surface with a total leading edge up travel of 3 1/4° to 3 1/2° on the 99 and 99A (except airplanes with Kit No. 99-5011 or 99-5014 and 99-5015 installed) and 4° to 4 1/4° on the A99A, B99 and C99, and prior airplanes with Kit No. 99-5011 or 99-5014 and 99-5015 installed. The leading edge down travel is 3 1/4° to 3 1/2°. The neutral position is indicated by a pointer on the leading edge of the left stabilizer with respect to a fuselage reference rivet (Ref. Figure 201). Travel of the stabilizer is controlled by a switch in the flight compartment, which is marked to indicate full up, neutral and full down. The up and down travel should be checked periodically with the travel board.

NOTE: If the stabilizer fails to indicate the full up and down requirements when its travel is checked, refer to HORIZONTAL STABILIZER ACTUATOR ADJUSTMENT. However, if the actual travel of the stabilizer is determined to be correct, and the indicator does not correspond to the travel of the stabilizer, the indicator should be adjusted as noted under the heading HORIZONTAL STABILIZER TRIM POSITION INDICATOR CHECKS AND ADJUSTMENT.

2. HORIZONTAL STABILIZER ACTUATOR

A. Removal

CAUTION: *Prior to removing the actuator, brace the horizontal stabilizer to prevent it from damaging the fuselage fairing pan when it swings free as the actuator mounting bolts are removed.*

(1) Remove the aft fuselage access panel inside the cabin area of the aft bulkhead.

(2) Having gained access to the actuator, disconnect the stabilizer connecting rods from the actuator attach points.

(3) Disconnect the electrical leads from the actuator.

(4) Disconnect the actuator from the fuselage crossmember to which it is attached and remove the actuator from the airplane.

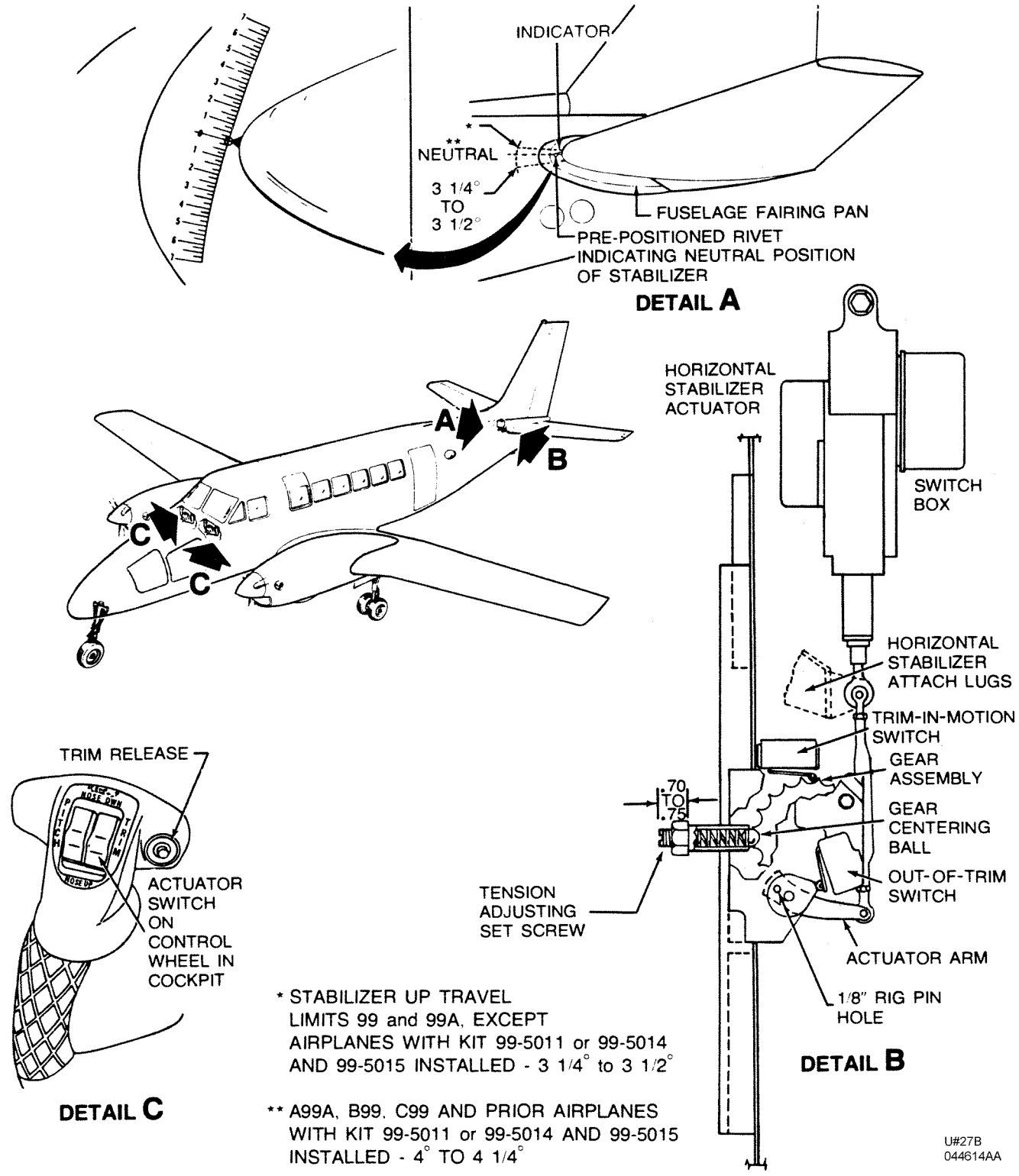


Figure 201
Horizontal Stabilizer

B. Installation

(1) Hang the actuator in the upper mounting brackets (rams pointed down and switch box forward).

(2) Check the alignment of the actuator rod ends with the horizontal stabilizer lugs in three positions of stabilizer travel (full up, neutral and full down).

WARNING: Ensure that the actuator Upper Attach Lugs have been correctly installed to the fuselage cross member. Alignment of the lugs with the bolt holes is critical to prevent inadvertent attachment without retaining the lugs with the bolts. Verify actual alignment and attachment (Step (3)) prior to proceeding.

NOTE: It should be possible to install the rod ends between the lugs in full up, neutral and down positions without forcing the actuator rams.

(3) Secure the actuator to the fuselage crossmember with two upper bolts and one of the lower bolts. Connect one of the actuator lower rod ends to its respective fitting on the stabilizing structure. Leave the nuts loose on the three mounting bolts to facilitate alignment of the fourth mounting bolt.

CAUTION: *Install the two washers on each of the two upper bolts outside of the fuselage brackets (Ref. Figure 202).*

(4) Install the mounting bolt that connects the remaining actuator rod end to the other stabilizer fitting if the connection can be made without forcing the bolt. If the fitting and rod end are so misaligned that the attaching bolt cannot be installed, proceed as follows:

(a) Break the safety wire and loosen the jam nut on the actuator rod end.

(b) Rotate the rod end until properly aligned with the stabilizer fitting.

(c) Insert the attaching bolt through the rod end and fitting, then retighten and safety the jam nut while holding the rod end parallel to the fitting. Tighten the actuator mounting nuts in place.

NOTE: If necessary, apply force by hand on the leading edge of the horizontal stabilizer in the direction required to align the bolt.

(5) Connect the electrical leads of the actuator and install the access panel.

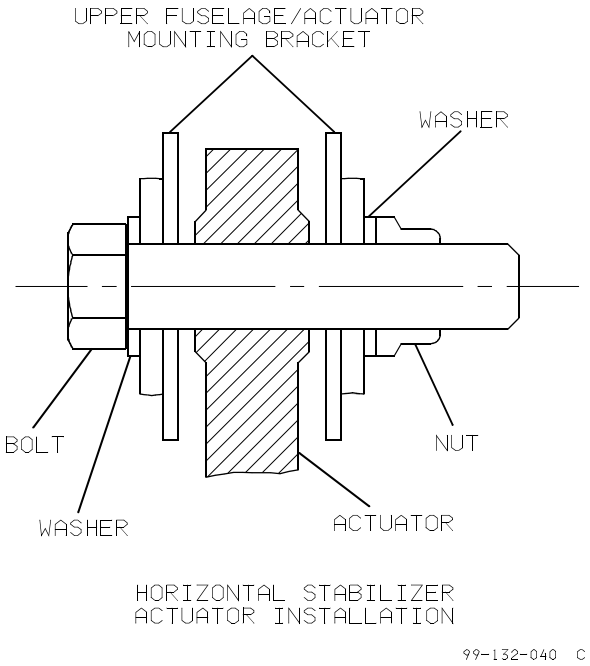


Figure 202
Horizontal Stabilizer Actuator Installation

C. Adjustment

(1) Place the horizontal stabilizer travel radius template over the reference rivet along the stabilizer arc of travel (Ref. Figures 201 and 204).

(2) Operate the stabilizer by either the main or standby trim system and observe the stabilizer leading edge travel limits as follows:

(a) On the 99 and 99A, airplanes without Kit No. 99-5011 or 99-5014 and 99-5015 installed, the leading edge UP (nose-down trim) travel is 3 1/4° to 3 1/2°.

(b) On the A99A, B99, C99 and prior airplanes with Kit No. 99-5011 or 99-5014 and 99-5015 installed, the leading edge UP travel is 4° to 4 1/4°.

(c) Leading edge DOWN (nose-up trim) travel for all airplanes is 3 1/4° to 3 1/2°.

(3) If the stabilizer leading edge DOWN travel requires adjustment, adjust the stabilizer actuator rod ends only. Make adjustments in 1/2 turn increments, checking down travel after each adjustment.

(4) If the stabilizer leading edge UP travel is not within the required limits, adjust the horizontal stabilizer actuator as follows:

(a) Operate the trim system to place the stabilizer in the full leading edge DOWN (nose-up trim) position.

(b) Remove the cover on the horizontal stabilizer actuator to expose the traveling nuts.

(c) Mark the traveling nuts with a marking pen, or other suitable means, to indicate their position in relation to the anti-rotation bar (each nut has three notches).

(d) Remove the safety wire and screws securing the anti-rotation bar.

CAUTION: *Exercise extreme care when removing the anti-rotation bar to prevent the traveling nuts from rotating. The extend traveling nut in the horizontal stabilizer actuator is preset and should not be adjusted (Ref. Figure 203).*

(e) Remove the anti-rotation bar.

(f) To reduce the actuator travel in the stabilizer leading edge UP position, adjust the left retract limit traveling nut (as seen facing the actuator) to the left, away from the right extend limit traveling nut.

NOTE: Turning the nut 1/3 turn changes the stabilizer travel approximately 0.074°.

(g) To increase the actuator travel in the stabilizer leading edge UP position, adjust the left retract limit traveling nut to the right, toward the right extend traveling nut.

CAUTION: *Do not attempt to cycle the actuator with the anti-rotation bar removed.*

(5) Replace the anti-rotation bar, making sure the bar is in the slots of the traveling nuts.

(6) Operate the trim system and observe the stabilizer UP travel limits. If further adjustment is necessary, repeat Steps (a) thru (g).

(7) Safety wire the screws securing the anti-rotation bar.

(8) Replace the actuator cover and safety wire the attaching screws.

(9) Verify that the trim position indicator reads 0° with the pointer on the stabilizer on the 0° (reference rivet) mark.

(10) Verify that the trim indicator does not exceed the red line when the stabilizer is operated to the full leading edge UP (nose-down trim) position.

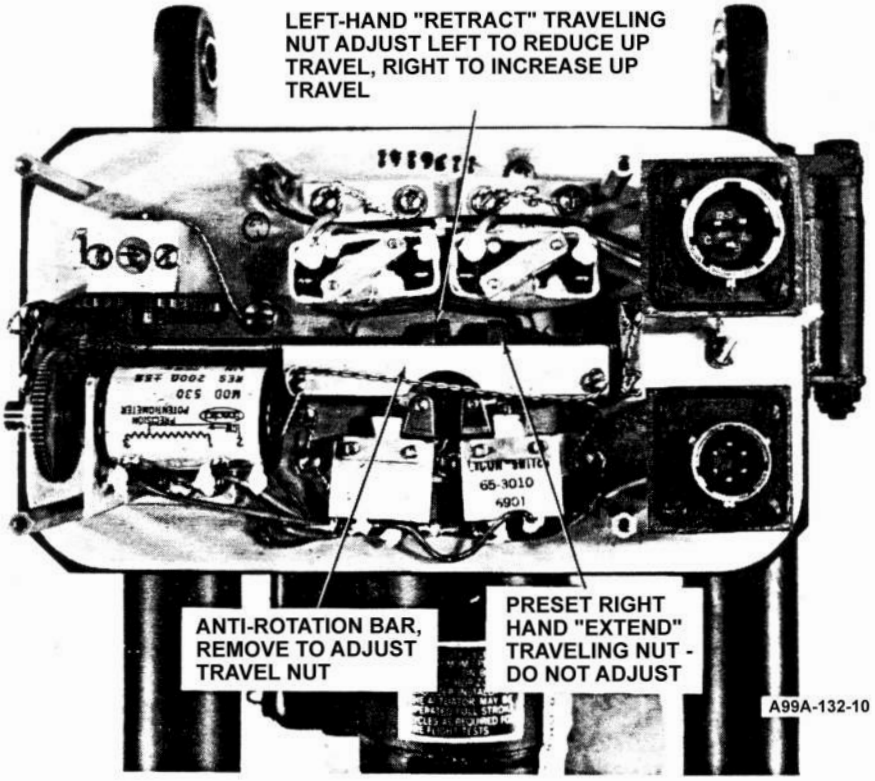
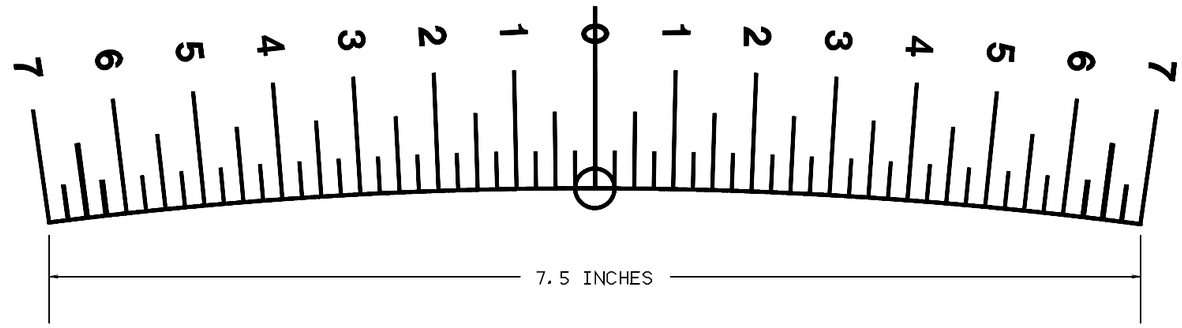


Figure 203
Horizontal Stabilizer Actuator Traveling Nuts



IF PRINTING FROM IML SOFTWARE ENSURE THE PROPER PAPER SIZE IS ADJUSTED TO PRINT ON 11 X 17 PAPER. VERIFY BY CONFIRMING THAT THE DISTANCE FROM NOTED ARROW POSITIONS IS 7.5 INCHES.

AN EXACT DUPLICATE OF THIS FIGURE MAY BE CUT OUT AND RETAINED AS A PERMANENT TRAVEL GAGE. ATTACH THE DUPLICATE TO A THIN SHEET OF ALUMINUM OR CLEAR PLASTIC TO PRESERVE IT FOR FUTURE USE.

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Figure 204
Horizontal Stabilizer Travel Radius Template

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3. HORIZONTAL STABILIZER TRIM POSITION INDICATOR

A. Checks and Adjustments

As required or whenever the horizontal trim actuator has been replaced, the pitch trim indicator should be checked and adjusted in the following manner:

(1) Connect an auxiliary power unit to the airplane and adjust the bus voltage to 28 volts.

(2) Turn the main trim ON and operate to full NOSE-UP limit.

(3) Locate the Constant Voltage Regulator Rheostat. On airplane serials U-1 through U-164 except U-50, the rheostat is located on the Upper Electrical Equipment Panel under the center aisle floor, immediately forward of the main spar. On airplane serials U-50, U-165 and After, the rheostat is located in the pedestal. Adjust this rheostat to obtain an indication of full NOSE-UP trim on the indicator. Tap the indicator lightly to assure a correct reading.

(4) Run the trim to a full NOSE-DOWN limit. Tap the indicator lightly to assure a correct reading. Verify that the indicator pointer is within 1/8 inch of full nose-down position.

(5) Run the trim to an indication of NEUTRAL position. Tap the indicator lightly to assure a correct reading. Verify that the indicator pointer, located on the leading edge of the left horizontal stabilizer, is within 1/16 inch of the center of the fuselage reference rivet (Ref. Figure 201).

(6) Disconnect the auxiliary power unit and repeat the NOSE-UP and NEUTRAL checks at battery voltage (24 volts) to determine that the Constant Voltage Regulator is functioning correctly. (Identical indication should be observed at the 28 volt check and at the 24 volt check).

4. HORIZONTAL STABILIZER TRIM ACTUATOR

A. Load Testing

After the first 2000 hours of actuator service and every 2000 service hours thereafter, the horizontal stabilizer trim actuator must be load tested. This consists of both a tension and compression load test. The load test is to be accomplished by obtaining the required Instructions Kit No. 99-5012-1S and may be done in the following manner:

NOTE: As an alternate to the load test, the actuator may be replaced with a new or overhauled actuator.

(1) Fabricate two clamp fixtures as described in Instructions Kit No. 99-5012-1S.

(2) Install a tail support stand similar to the one shown, and block the landing gear wheels to prevent movement of the airplane (Ref. Figure 207).

(3) Remove the two round access plates, located just below the leading edge of the horizontal stabilizer.

(4) Connect a 28 vdc regulated ground power supply to the airplane.

(5) Position the horizontal stabilizer travel template and check the stabilizer travel limits. Reset the limit switches if required and recheck the travel limits.

(6) After it is determined that the travel limit is correct, install the two clamp assemblies fabricated from the Instructions Kit on the horizontal stabilizer as shown in the kit (Ref. Figure 205).

(7) To accomplish the actuator tension load test, load the LH and RH clamps at the forward shackle with 430 lbs. of weight on each shackle. DO NOT EXCEED THIS WEIGHT. Figure 206 illustrates an example of weight loading at the aft shackle. (The actuator may be in either the extended or retracted position).

CAUTION: *Weights must be added simultaneously and equally to both the LH and RH sides of the stabilizer.*

Operate the actuator through a full extend and retract cycle by using the pilot's control of the primary motor. Check the time required for travel between full retract and full extend, then recheck. Refer to Table 201 for the travel times.

CAUTION: *Allow a 5 minute cooling period between the retract and extend cycles.*

Table 201
FULL TRAVEL TIME CHECK

MODEL	NORMAL TRAVEL TIME	MAXIMUM TRAVEL TIME
99 and 99A (except airplanes with Kits No. 99-5011 or 99-5014 and 99-5015 installed)	18 seconds	25 seconds
A99, B99, C99 (and prior airplanes with Kits No. 99-5011 or 99-5014 and 99-5015 installed)	20 seconds	27 seconds

If the trim system does not operate within the maximum time limits indicated in Table 201, replace the primary motor and repeat the travel time check. If the system still does not operate with the limits, the actuator would be replaced or overhauled. The actuator will also have to be replaced or overhauled if any clutch slippage or abnormal noise or movement is noted during the travel time check.

NOTE: If the clutch is slipping, there will be considerable noise and chatter or there will be no movement of the horizontal stabilizer.

(8) To accomplish the actuator compression test, remove the weights from the forward shackles of the clamp assemblies and attach them to the aft shackles of the clamps and repeat Step (7).

CAUTION: *Weights must be added simultaneously and equally in each side of the stabilizer (430 lbs per side). DO NOT EXCEED THIS WEIGHT.*

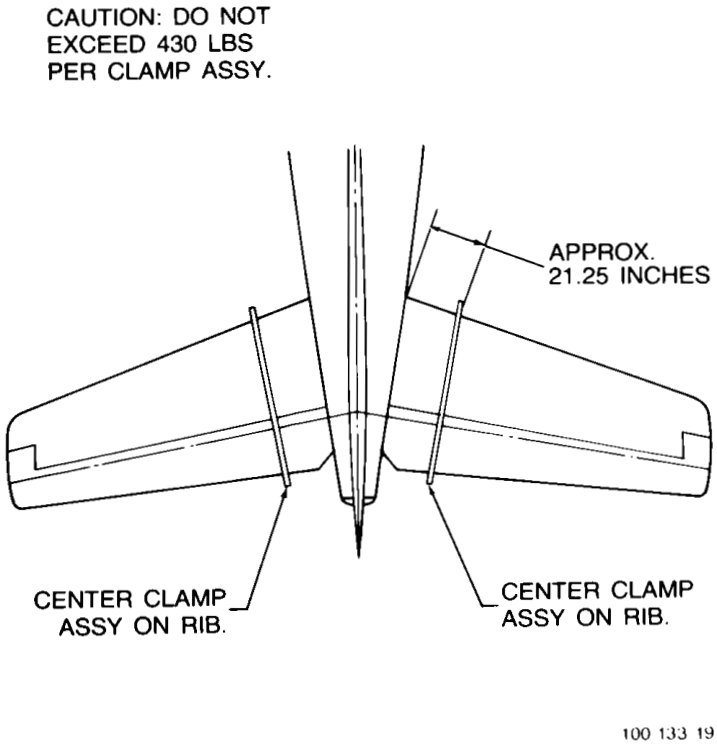

Figure 205
Positioning of Clamp Assemblies

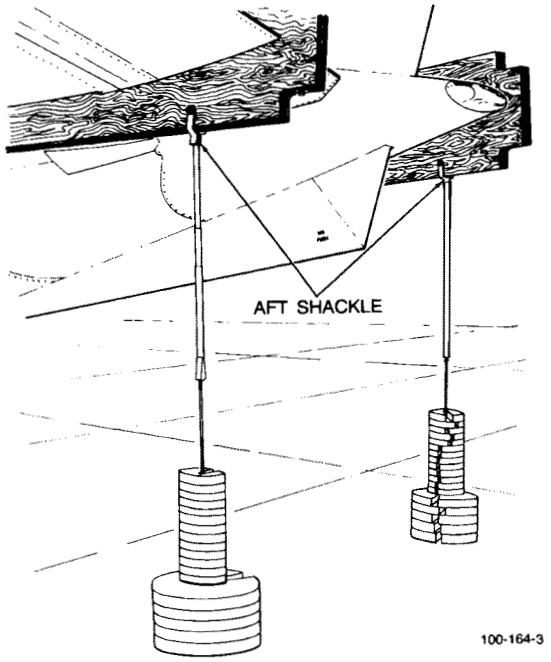

Figure 206
Loading of Shackles

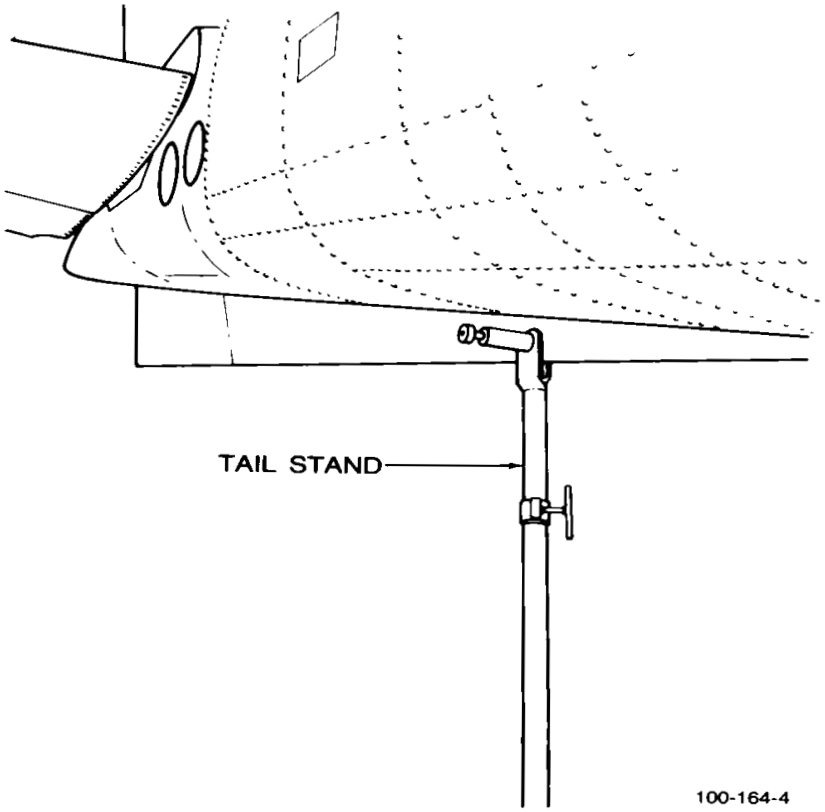

Figure 207
Positioning of Tail Stand

5. PITCH TRIM SWITCH

A. Replacement Criterion

A cracked detent spring or springs is an indication that a switch has exceeded its life expectancy or has been mishandled in some way. This type of failure is considered a safe failure since the pilot still can move the switch to the OFF position to prevent trim operation and the failure results in a mushy feel rather than the crisp detent action which is normal.

In general, the switch should be replaced whenever the crisp detent action becomes mushy. Since there are four springs for each direction on each switch, this will allow an operator adequate use after the first indication of failure until switch replacement during scheduled maintenance.

6. HORIZONTAL STABILIZER TRIM POTENTIOMETER

A. Adjustment

If the indicator cannot be calibrated within the proper limits, the horizontal stabilizer position transmitter potentiometer must be adjusted in the following manner:

(1) Operate the trim to a full NOSE-DOWN position.

(2) Disconnect the STANDBY TRIM connector at the actuator.

(3) Remove the cover from the trim actuator unit.

(4) Adjust the rheostat to obtain a reading of 0 to 5 ohms as measured between pins D and E of the STANDBY TRIM actuator connector.

(5) Replace the cover and reconnect the STANDBY TRIM connector.

(6) Repeat the HORIZONTAL STABILIZER POSITION INDICATOR ADJUSTMENTS.

7. AURAL TRIM WARNING SYSTEM

The trim warning system has two separate functions which provide an aural warning to the pilot and copilot that the horizontal stabilizer is: (1) in motion or (2) out of the takeoff trim position. Each system is controlled by a circuit breaker on the copilot's circuit breaker panel and is mechanically actuated by horizontal stabilizer movement through a ratchet-driven gear assembly.

A. Trim-in-Motion System

The trim-in-motion system provides the pilot and copilot with an aural warning tone that is interrupted at a rate directly proportional to the horizontal stabilizer rate of travel. The tone has a rate of approximately 2 beats per second for main trim operation. Tone interruption is produced by the actuating arm of the trim-in-motion switch riding up and down on the teeth of a rotating gear in the gear assembly. The electrical pulses from the switch are routed to the trim indicator module mounted on the right side of the fuselage immediately aft of the rear cabin bulkhead. Within the module, a delay timer will modulate the signal to an oscillator which produces a tone of 900 cycles per second. This tone is directed to a speaker mounted overhead in the cockpit and to the pilot and copilot headphone jacks. The volume level may be adjusted separately for cockpit speaker and headphone, but the warning tone will always be heard over any received avionics signals.

B. Takeoff Out-of-Trim System

The takeoff out-of-trim system provides the pilot and copilot with a continuous aural warning whenever the following conditions are fulfilled:

- The power control levers are at 90% N_1 or greater, the airplane is on the ground, and the horizontal stabilizer position is out of the takeoff position.

When these conditions exist, the electrical circuit will function as follows:

- Electrical power from the 5 amp circuit breaker is directed through the pedestal mounted power lever switch whenever the left power control lever is advanced to 90% N_1 or greater. Power is then routed to the cam actuated out-of-trim switch mounted on the gear assembly in the aft fuselage below the horizontal stabilizer.
- When the stabilizer is out of the takeoff range, the out-of-trim switch will actuate, routing power to the takeoff out-of-trim relays mounted on the forward bulkhead adjacent to the left defroster duct. A ground to energize the relays is provided by the RH landing gear safety switch. The takeoff and out-of-trim relays select the contacts for continuous power and ground to activate the landing gear warning horn.

8. POWER LEVER SWITCH

A. Adjustment

(1) Remove the covering on the left side of the pedestal to gain access to the power control lever switch.

(2) Run the left engine to determine the position of the left power control lever at 90 ± 2% N_1. Mark the position of the lever on the pedestal with a piece of tape and shut down the engine.

(3) Adjust the power lever switch to actuate in this position.

(4) Remove the tape and install the pedestal side covering.

9. TRIM-IN-MOTION SWITCH

A. Adjustment

(1) Remove the aft fuselage access panel in the rear baggage compartment.

(2) Position the switch to actuate on the upslope and deactuate on the downslope of each tooth. The free motion designed into the trim-in-motion gear will allow a finger, placed against the gear, to deflect it approximately one tooth in either direction while the gear assembly is stationary. The actuator roller of the switch should rest between the gear teeth simultaneously with the gear centering ball (Ref. Figure 201). The setscrew and locknut may be adjusted to regulate tension of the centering ball against the gear. Adjustment of the centering ball tension will vary the actuation of the switch and beat frequency heard in the cockpit. To assure correct tension, loosen the locknut and turn the setscrew all the way in. Operate the trim system and listen for a minimum beat frequency of approximately 3 beats per second. Back off the setscrew until each beat doubles (approximately 6 beats per second) when moving in either direction. Secure the setscrew with the locking nut. The setscrew should extend between 0.70 to 0.75 inch from the end of the threaded tube.

10. TRIM INDICATOR MODULE

A. Adjustment

(1) Remove the aft fuselage access panel in the rear baggage compartment.

(2) Access holes on the top and bottom of the trim indicator module will allow adjustment of the potentiometers that control the volume level of the cockpit speaker and the pilot and copilot headphones.

11. TAKEOFF OUT-OF-TRIM SWITCH AND GEAR ASSEMBLY

A. Adjustment

(1) Remove the aft fuselage access panel in the rear baggage compartment.

(2) Tape the horizontal stabilizer travel template onto the fuselage so that the 0° mark is directly over the fuselage reference rivet and the template aligns with the stabilizer travel radius (Ref. Figure 201).

(3) Move the stabilizer until the pointer is in the 0° position.

(4) If the linkage between the horizontal stabilizer and gear assembly has been disconnected or its adjustment disturbed, the gear assembly must be rigged before the out-of-trim switch can be adjusted. If the linkage adjustments have not been disturbed, proceed to Step (6).

(5) To rig the gear assembly, disconnect the linkage between the stabilizer and the gear assembly. Rotate the arm of the gear assembly until a 1/8 inch diameter rig pin can be inserted in the rigging hole in the cam, bracket and arm of the assembly. Adjust and attach the linkage to this position, which corresponds to 0° stabilizer position. Remove the rigging pin.

(6) Determine that the left power control lever is in the IDLE position and then attach the leads of the battery test lamp to the common and normally open terminal of the out-of-trim switch.

(7) Move the stabilizer up from the neutral position until the test lamp lights; the stabilizer pointer should indicate 1/4° ± 1/4°. If the light does not come on, adjust the switch to actuate within this tolerance.

NOTE: Loosen the switch bracket bolts slightly to allow adjustment of the switch. The tip of a screwdriver blade placed between the edge of the switch bracket and the adjacent plate allows adjustments to be made in small increments.

(8) Move the stabilizer down from the neutral position until the test lamp lights: the stabilizer pointer should indicate 2 1/4° ± 1/4°. If the light does not come on, adjust the switch to actuate within this tolerance. If necessary, repeat Steps (7) and (8) until the lamp will light in both stabilizer positions.

(9) Remove the travel template and the test lamp and replace the aft fuselage panel.

(10) Check to see that the trim indicator on the pedestal corresponds to the correct position of the stabilizer; if not, refer to the procedure outlined under HORIZONTAL TRIM POTENTIOMETER ADJUSTMENT.

12. TRIM WARNING SYSTEM (100 HOUR INSPECTION)

The 100-hour inspection of the trim warning system consists of performing an operational check of the trim-in-motion and the takeoff out-of-trim functions. This inspection may be accomplished with the airplane on the ground, engines off and external power applied to the airplane.

A. Trim-in-Motion Warning

(1) Turn the battery switch ON.

(2) Insert the headphone leads into the pilot's headphone jacks.

(3) Operate the main and standby trim systems and listen for a distinct difference in tone frequency (approximately 6 beats per second for the main and 3 beats per second for standby). This should be heard over the cockpit speaker and the pilot's headphones.

(4) Install the headphone leads in the copilot's headphone jacks and repeat Step (3).

B. Out-of-Trim Warning

(1) Install the stabilizer travel template and turn the battery switch ON.

(2) Be sure the right landing gear safety switch is compressed.

(3) Advance the left power control lever to a position which corresponds to 90% N_1 or greater to actuate the power lever switch in the pedestal.

(4) Move the horizontal stabilizer up from the neutral position (airplane nose-down) until the warning horn sounds; the stabilizer pointer should indicate a 1/4° ± 1/4°. If the stabilizer is not in the correct position, refer to the adjustment procedure for the out-of-trim switch.

(5) Move the horizontal stabilizer down from the neutral position (airplane nose-up) until the warning horn sounds; the stabilizer pointer should indicate 2 1/4° ± 1/4°. If the stabilizer position is not correct, refer to the adjustment procedure for the out-of-trim switch.

13. AURAL TRIM WARNING SYSTEM

A. Electrical Checks

Refer to the applicable circuit diagrams in the Wiring Diagram Manual. Should the trim-in-motion warning system fail, the following electrical checkout of common voltage and ground sources, signal input and trim indicator module should pinpoint the source of malfunction.

(1) Disconnect the electrical connector from the trim indicator module located in the aft fuselage compartment.

(2) Using a voltmeter, determine that there is 28 vdc at socket insert 1 of the female plug.

(3) Using an ohmmeter, check for ground at socket insert 2 of the female plug.

(4) Operate the trim system, check for interrupted power at socket insert 3 and then insert 4 of the female plug.

If the above system checks are correct, the source of failure is probably in the trim indicator module. Failure of the cockpit speaker or either headphone would indicate individual wiring or component malfunction. If a failure of the takeoff out-of-trim system occurs, check out the system by setting up an out-of-trim condition by advancing the left power lever and running the trim system out of the green range on the pedestal mounted trim indicator. Check for common voltage and ground sources at the takeoff trim-in-motion interlock relay, trim-in-motion relay and the landing gear warning horn.

14. AURAL TRIM WARNING GEAR ASSEMBLY

A. Removal

The aural trim warning gear assembly is located in the RH aft fuselage below and adjacent to the horizontal stabilizer actuator. Access and removal may be accomplished as follows:

(1) Remove the aft fuselage access panel inside of the cabin area at the furthermost aft cabin bulkhead.

(2) Locate the aural trim warning gear assembly. Remove the linkage rod connecting the horizontal stabilizer actuator and the trim warning gear assembly at the gear assembly.

NOTE: Do not disturb the adjustment of the connecting rod linkage.

(3) Remove the electrical leads to the aural trim warning gear assembly.

NOTE: Identify the electrical leads (by tape or other suitable means) as they are removed to facilitate reinstallation.

(4) Remove the bolts attaching the gear assembly to its mounting bracket.

B. Installation

(1) Locate the aural trim warning gear assembly on its mounting bracket and install the attaching bolts.

(2) Install the electrical leads on their respective terminals.

(3) Connect the linkage rod from the actuator to the gear assembly.

NOTE: If linkage rod adjustment is required, refer to Step (5) under TAKEOFF OUT-OF-TRIM SWITCH AND GEAR ASSEMBLY ADJUSTMENT.

15. STABILIZER TRIM SYSTEM (U-1 THRU U-131, U-133 THRU U-145, AND U-147 WITHOUT KIT NO. 99-5011, 99-5014 OR 99-5015 INSTALLED)

A. Flight Test

The following flight test procedures should be accomplished each time a horizontal stabilizer or elevator is repaired or replaced and whenever the elevator control system is rerigged.

(1) Remove the access panel on the aft cabin bulkhead to provide access to the trim actuator.

(2) Attach the travel indicator template to the trim actuator housing with a 1 1/8 inch diameter clamp or tape (Ref. Figure 208). Position the template so that the numbers on the template extend downward, parallel to one of the actuator rams.

(3) Run the trim to the neutral (0°) position. Verify that the indicator pointer, located on the leading edge of the left horizontal stabilizer, is within 1/16 inch of the center of the fuselage reference rivet. Refer to Figure 201 for the location of the reference rivet.

(4) Make a mark on the actuator ram, with a marking pen or other suitable means that corresponds with the zero (0)° mark on the travel indicator template. Run the trim to the full up and full down positions and check to see that the travel indicator template is positioned so that it indicates full up and down travel.

NOTE: The travel indicator template and the mark on the actuator ram must be positioned so that both may be read from the aft cabin bulkhead while in flight.

(5) Load the airplane to obtain a gross weight of 8900 ± 200 pounds with the center of gravity at FS 182.0. This loading includes a pilot and copilot.

(6) With the airplane loaded as specified in Step (5), take off and climb to a cruising altitude where the air is smooth.

(7) Adjust the power to 1000 foot-pounds of torque and an engine speed of 2200 rpm (both engines) and trim to a speed of 200 kts. IAS, (hands off) with the pilot and copilot both in their seats.

(8) Without readjusting the pitch trim, check the position of the horizontal stabilizer. This can be accomplished by one crew member checking the position indicated on the travel indicator template attached to the trim actuator. This horizontal stabilizer should be 1.25° ± 0.5° leading edge up (airplane nose down) position.

(9) Adjust the trim to obtain a speed of 100 kts. IAS (hands off) with the gear and flaps down and the power set to 640 foot-pounds of torque with the pilot and copilot in their seats.

(10) Without readjusting the pitch trim, check the position of the horizontal stabilizer at the travel indicator template on the trim actuator. The horizontal stabilizer should be at 0.5° ± 0.5° leading edge up (airplane nose-down) position.

(11) Adjust the pitch trim to an airplane full nose-up (horizontal stabilizer leading edge down) position with the airplane in a clean configuration (gear and flaps up). The engines should be set at 200 foot-pounds of torque with the pilot and copilot in their seats and hands off the control wheel. The airplane speed should be 123 ± 7 kts.

(12) If the airplane does not perform as specified in Steps (6) thru (11), new elevators shall be installed on the airplane. When these elevators are installed, the elevator system shall be rigged and the flight check accomplished.

NOTE: When P/N 115-610010-151 elevators are used, 3 pounds per elevator is added at FS 473.6. Weight and balance changes should be reflected in the airplane records.

(13) After satisfactory completion of the flight tests, remove the travel indicator template from the trim actuator and the load from the cabin. Replace the access panel on the aft cabin bulkhead.

16. HORIZONTAL STABILIZER ATTACHMENT POINTS

A. Check and Repair

The horizontal stabilizer attachment points should be checked for wear every 300 hours (Ref. Figure 209). With the tail cone removed, check the wear of the hinge brackets at the pivot hole and check the bearings in the pivot support assemblies for looseness. With one man at the stabilizer tip, alternately lifting up and pulling down on the stabilizer, a second man at the tail should check for movement of the bolt in the hinge bracket and for any looseness in the pivot support assembly bearings. If any looseness is found in the bearing, replace the pivot bolt support assembly, complete with bearing. If any movement of the bolt in the hinge brackets (P/N 115-440031-3 and -4) is observed on airplanes U-50, U-165 and After; if the old bearing in the hinge bracket can be removed and the hole in the bearing does not exceed 0.500 + 0.001 - 0.0005 inch, new bearing(s) (P/N 115-440031-7 or -9) may be installed. If any movement of the bolt in the hinge brackets (P/N 115-440031-1 and -2) is observed on airplanes prior to U-165, except U-50, the wear may be repaired as follows:

(1) Support the stabilizer in the airplane and remove the hinge brackets.

(2) Check the wear of the bearings into the face of the hinge brackets by the pivot support and pivot support bearings.

(3) Mill off the hinge brackets to remove the worn portions. Milling should be held to a minimum to provide a smooth, flat bearing surface. The maximum metal removed should not reduce the lug thickness to less than 0.36 inch.

(4) Ream the holes in the hinge point bracket to 0.500 + 0.0010 - 0.0005 inch diameter and install new bushings (P/N 115-440031-7 and -9) in the holes. Install the 115-440031-7 bushings on the outboard side of the hinge point brackets and the 115-440031-9 bushings on the inboard side. Install the bushing flush with the machined surface of the hinge point bracket.

NOTE: Apply adhesive (102, Table 6, Chapter 91-00-00) to the bushings prior to installation.

(5) Attach the hinge point bracket to the airplane with the six bolts (4 places) and torque 50 to 70 inch-pounds.

(6) Install the two bolts (bolt heads inboard) through the pivot support assemblies and the hinge point brackets with a washer inserted under the bolt head and washers (maximum of three) under the nut to prevent the nut from bottoming against the grip of the bolt.

NOTE: Any side clearance between the hinge bracket and the stabilizer pivot point should be taken up with the installation of AN960-616 or AN960-616L washers, depending on the fit. The hinge bracket should fit snugly against the pivot bearing.

(7) Install the tail cone.

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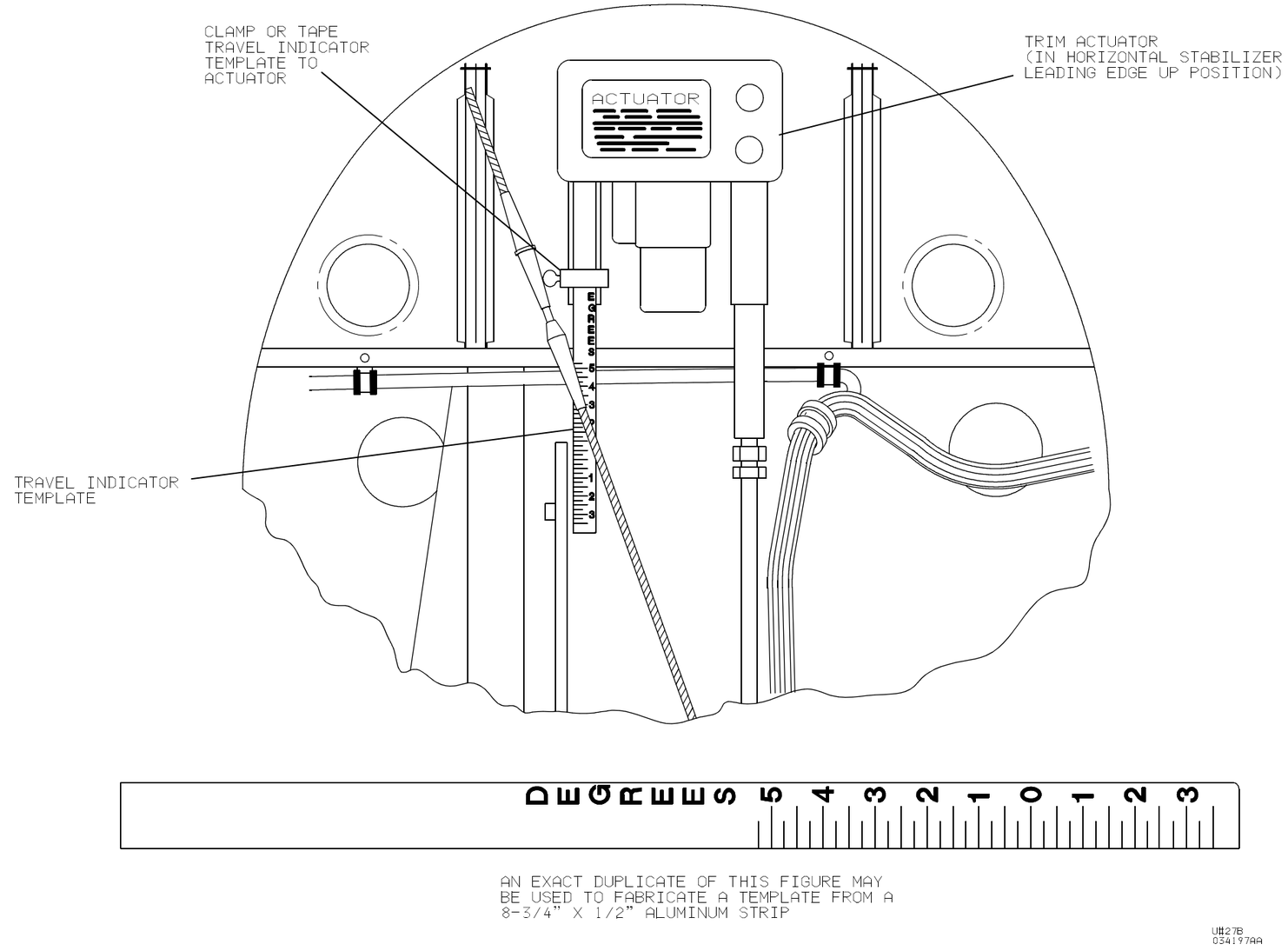

Figure 208
Horizontal Travel Indicator Template and Installation

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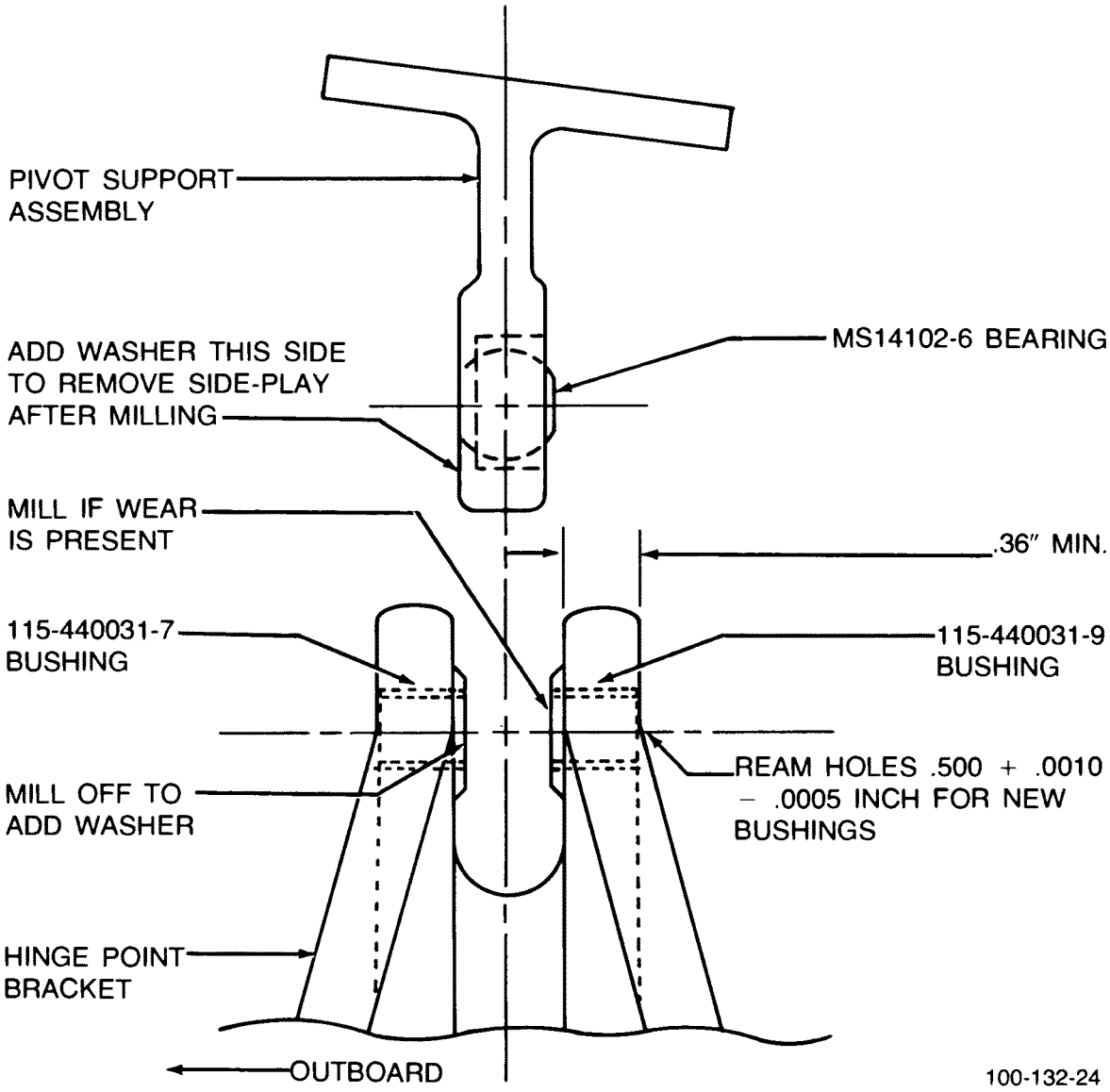

Figure 209
Horizontal Stabilizer Attachment Points

FLIGHT CONTROLS
FLAPS
DESCRIPTION AND OPERATION

1. GENERAL

A. U-1 thru U-164, except U-50

Driven from a single electric motor, the flaps consists of a section on the outer wing panel and one on the center section of each side of the fuselage. A flexible shaft extends from the motor gearbox to a jackscrew actuator for each section. Limit switches installed on the right inboard flap track stop the flap travel at 0° full up and 43° full down. An infinite travel control switch on the pedestal will also stop and set the flaps in any position in between the full up and full down position. The approach position is located by a detent in the switch travel slot at the 13° position. To indicate the position of the flaps, a rheostat operated by the right inboard flap controls an indicator mounted at the top of the pedestal.

B. U-50, U-165 and After

A flap control printed circuit board, located in the PCB card rack, forward of the main spar, mounted on the center aisle subfloor processes information from a potentiometer attached to the flap control handle in the flight compartment and a potentiometer actuated by the left inboard flap to position the flaps at a desired position between full up and full down. Moving the flap control handle changes the resistance of the flap control handle potentiometer and unbalances the circuitry of the flap control PCB. The flap control PCB then attempts to rebalance the circuit by energizing the flap motor to drive the flaps which, in turn, changes the resistance of the potentiometer mounted on the flap. When the PCB is satisfied that the circuit is once again balanced, the PCB interrupts power to the motor and the flaps stop moving. The potentiometers are adjusted so that balancing can never be achieved in either the full up or full down position of the flap control handle, and power will then be interrupted by the up or down limit switch. Discrete outputs from the flap control PCB are responsible for energizing the flap motor relay which transmits power from a 20-amp circuit breaker to the flap motor. A 5-amp circuit breaker provides control current to the flap control PCB.

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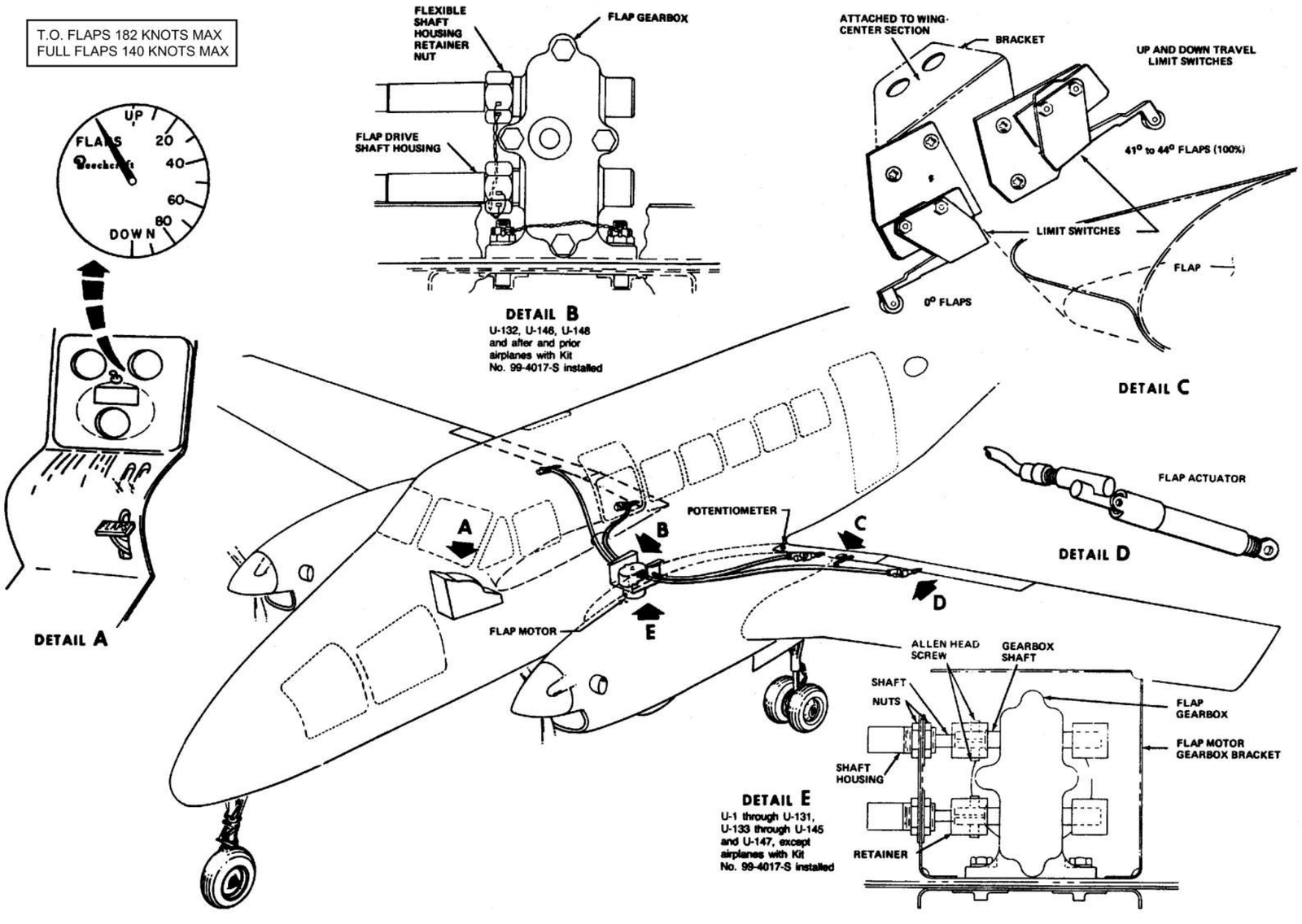

Figure 1
Flap Control System

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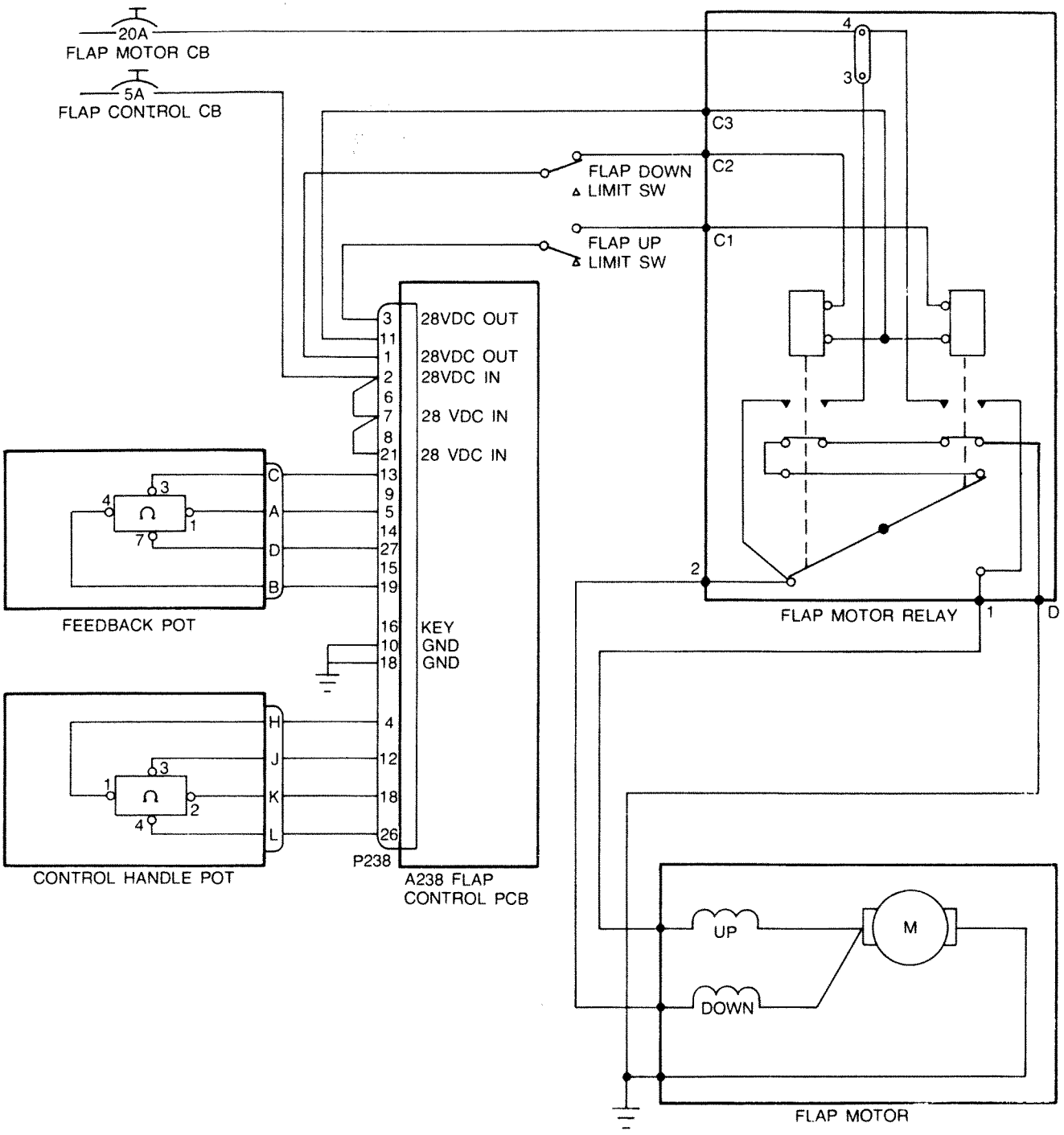

U-50, U-165 AND AFTER

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Figure 2
Flap Control Schematic

FLIGHT CONTROLS
FLAPS
TROUBLESHOOTING

1. FLAP CONTROL SYSTEM

A. U-50, U-165 and After

The following troubleshooting diagram is designed to give a logical sequence of checks which should be made, the results of which will lead the technician to a specific action and should result in fault correction. Use of this flow diagram is intended only as an aid to the technician and is not intended to replace the technicians own logic and reason in approaching the task of troubleshooting a fault in this system.

The conditions necessary to begin this troubleshooting procedure are stated at the top of the diagram and must be closely adhered to if this guide is to accomplish its purpose.

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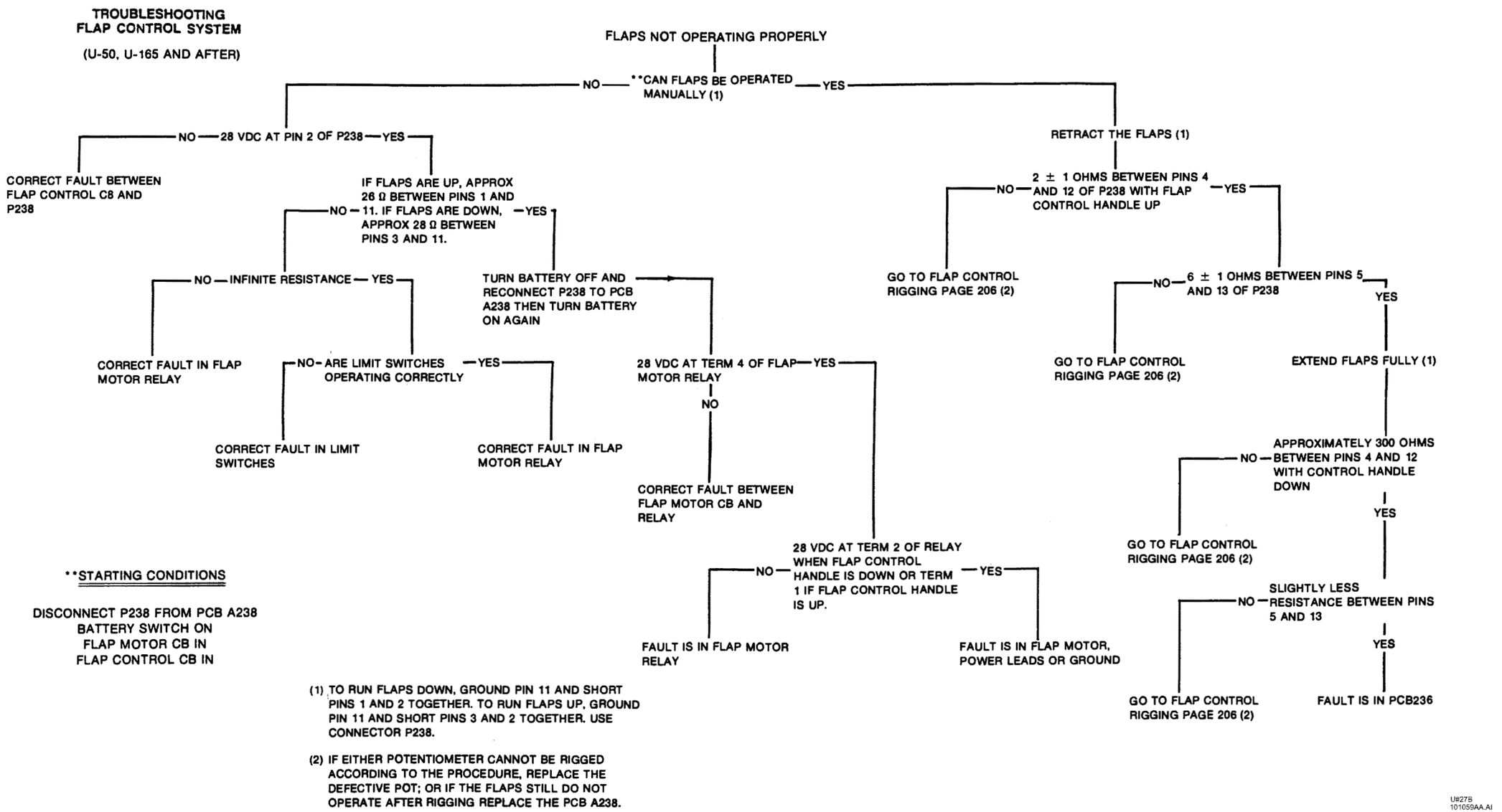

Figure 101
Troubleshooting - Flap Control System
(U-50, U-165 and After)

FLIGHT CONTROLS
FLAPS
MAINTENANCE PRACTICES

1. FLAP

A. Removal

CAUTION: *Never operate the flaps with a flap removed as the limit switches will be inoperative and serious damage to the flaps, wings, and the actuating system can result.*

(1) Place the flap control switch in the DOWN position and operate until the flaps are fully down.

(2) Pull the flap motor control circuit breaker.

NOTE: Place the following identification tags on the flap motor circuit breaker and the flap handle: DO NOT OPERATE WHILE THE FLAPS ARE REMOVED.

(3) Provide a suitable support for the flap being removed.

(4) Remove the screws attaching the bonding jumpers to the flap.

NOTE: Measure the extension of the flap actuator screw so it can be reattached in its original position.

(5) On the LH inboard flap, remove the bolt that attaches the flap drive actuator and linkage for the feedback potentiometer.

(6) On the RH inboard flap, remove the bolt attaching the flap position transmitter linkage and the flap drive actuator to the flap.

(7) Disconnect and remove the flap actuators from the LH and RH outboard flap segments to be removed.

(8) Remove the attaching screws and access plugs located on the upper surface of the flap on either side of the flap tracks.

NOTE: Each inboard flap has four access plugs and each outboard flap has three access plugs.

(9) Remove the nuts, washers, and bolts securing the flap to the aft track of the wing bracket. Remove the flap from the wing.

(10) Note the correct position of the nylon washers with respect to the rollers and the track bearings to facilitate reinstallation.

B. Installation

(1) When installing the flap, be certain the flanges and nylon washers are facing each other, that is, to the center of the flap.

(2) Position the flap and align the holes of the flap to the aft track of the wing bracket. Install the attaching bolts, nuts, and washers.

(3) Install the access plugs on the upper surface of the flap on either side of the flap tracks and secure with the attaching screws.

NOTE: When installing the flap actuators, ensure that the flap actuator screw is extended to the same degree as marked upon removal.

(4) Position and reconnect the flap actuators for the LH and RH outboard flaps.

(5) On the RH inboard flap, install the bolt attaching the flap position transmitter linkage and the flap drive actuator to the flap.

(6) On the LH inboard flap, install the bolt that attaches the flap drive actuator and linkage for the feedback potentiometer.

(7) Install the bonding jumpers to the flap.

(8) Remove the identification tags from the flap motor control circuit breaker and the flap control switch.

(9) Check the flap system for proper rigging.

NOTE: Be certain to adhere to the dimensions given, when installing the flaps (Ref. Figure 201).

2. FLAP TRACK

A. Wear Limits

Inspect the slots in the flap tracks for excessive wear. Maximum wear will occur in the full up, full down and approach positions. The maximum slot width limit permissible is 0.785 inch for the inboard and outboard flap tracks. A maximum allowable tolerance of 0.050 inch side wear into the track, caused by the flange of the roller, will normally occur in the full up or static position. Where the roller flange or insulating washer, P/N 105000-3, has worn into the side of the track, the edge of the wear area should be tapered in the direction of travel to prevent the roller from binding on the track. Additional chafing washers, as necessary, may be installed to prevent excessive side clearance between the roller flange and the track.

Track wear within the above limitations may be dressed smooth with light emery cloth to prevent roller binding. To help reduce wear, it is recommended that the tracks and rollers be kept clean and lubricated, using a dry spray lubricant (122, Table 6, 91-00-00). The teflon chafing washers should also be kept in good condition.

NOTE: The 3NC1010P needle bearings manufactured by Torrington Bearing Co., used in flap bearing assembly P/N's 35-165004, 99-160011 and BC33062 do not have provisions for lubrication. These bearings should be replaced on condition.

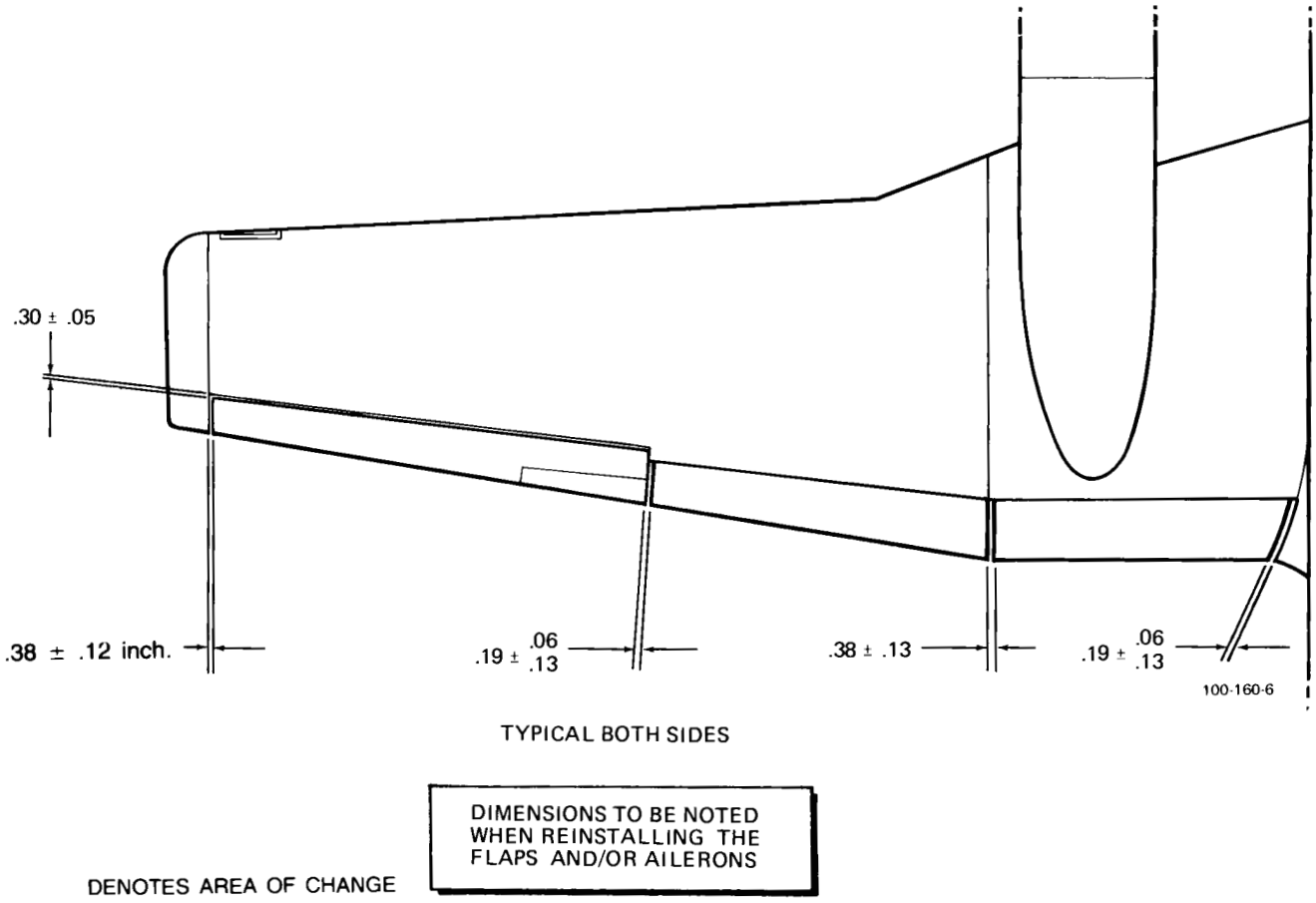

Figure 201
Flap and Aileron Installation

B. Repair

WARNING: If the track is worn beyond the maximum allowable dimension, repair must NOT be attempted. Replace excessively worn flap tracks.

NOTE: New inboard and outboard flap track slot dimensions (dimension A) are 0.750 to 0.785 inch (Ref. Figure 202). The maximum allowable side wear (dimension B) on any flap track is 0.050 inch below the original surface (track plane).

(1) Clean the area of the track that is to be repaired with solvent (34, Table 6, 91-00-00) or equivalent solvent.

(2) Roughen the area to be repaired with 180-grit emery cloth.

(3) Clean the roughened area with soap and water, rinse, and dry with a lint-free cloth or blow dry with filtered shop air.

NOTE: Devcon WR-11220 (123, Table 6, 91-00-00) should be used in repair areas with less than 0.02 inch of wear. Devcon WR-11410 (124, Table 6, 91-00-00) should be used in repair areas with 0.02 inch or more of wear.

(4) Select the type of Devcon as dictated by the amount of wear and mix a sufficient amount to perform the repair.

(5) Apply the Devcon to the worn area and allow it to cure until hard enough to work.

(6) When sufficiently hard, work the Devcon repair into the original track dimension.

(7) Finish and repair with 320-grit emery cloth.

(8) Apply Alodine 1200, 1200S or 1201 (89, Table 6, 91-00-00) to the repair area.

(9) Apply one coat of zinc chromate primer.

(10) Do not operate the flaps until the Devcon repair has fully hardened.

NOTE: The Devcon should be fully cured in approximately 6 to 8 hours in an environment of at least 60°F.

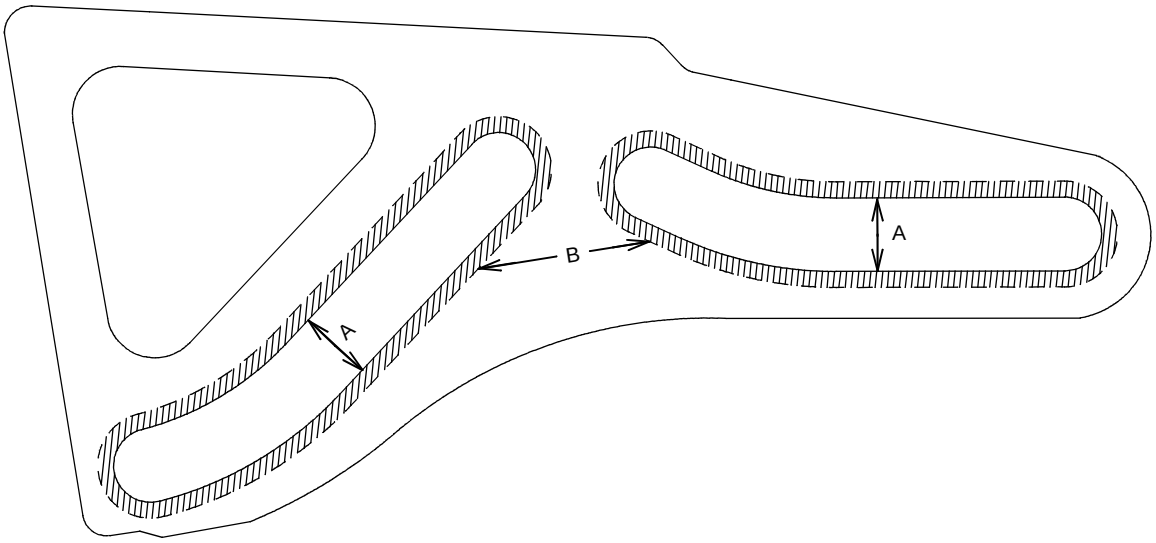

Figure 202
Flap Track Repair

3. FLAP MICROPOSITIONER SHOCK MOUNTS

A. Lubrication

To prevent deterioration of the rubber shock mounts on the flap micropositioner, spraying the rubber portion of the mount with one of the following is recommended. This should be accomplished the first time the floorboards are removed to gain access to the micropositioner and whenever new rubber shock mounts are installed on the micropositioner.

- #1329.... Krylon Silicone Spray, Bordon Co., NY, NY.
- #8042.... Crown Silicone, Crown Industrial Prod. Co., Hebron, IL.

4. FLAP MOTOR AND GEARBOX (U-1 THRU U-147 EXCEPT U-132, U-146 WITHOUT KIT NO. 99-4017-1S INSTALLED)

A. Removal

(1) Remove the center aisle floorboard immediately forward of the rear spar.

(2) With the flap motor control circuit breaker pulled, disconnect the electrical leads from the flap motor and identify with tags to facilitate reinstallation.

(3) Remove the retaining nuts securing the flap drive shaft housings to the gearbox mounting bracket.

(4) Remove the inboard housing locknuts.

(5) Loosen the two allen head screws in each retainer and rotate the retainer to disengage the shafts from the gearbox.

(6) Remove the bolts, nuts, and washers securing the motor and gearbox assembly to the rear spar. Remove the assembly from the airplane.

(7) Remove the two screws and lock washers securing the flap motor to the gearbox.

B. Installation

(1) Secure the flap motor to the gearbox with the two retaining screws and lock washers. Be certain the lock washers are in good condition or new.

(2) Secure the motor and gearbox assembly to the rear spar with the retaining bolts, washers and nuts.

(3) Insert the flexible shafts through the retainers and engage with the gearbox shafts. Tighten the inboard housing locknuts.

(4) Tighten the retaining nuts securing the flap drive shaft housings to the gearbox mounting bracket.

(5) Rotate the shaft retainer 90° and tighten the two allen head screws in each flexible shaft retainer.

(6) Connect the electrical leads to the flap motor and remove the identification tags. Engage the circuit breaker.

(7) Cycle the flap system to check for proper rigging. Ensure that no rollers bottom out at either end of travel and that the flaps are synchronized.

(8) Install the center aisle floorboard over the flap motor and gearbox assembly.

5. FLAP MOTOR AND GEARBOX (U-132, U-146, U-148 AND AFTER, AND PRIOR AIRPLANES WITH KIT NO. 99-4017-1S INSTALLED)

A. Removal

(1) Remove the center aisle floorboard immediately forward of the rear spar.

(2) With the flap motor control circuit breaker pulled, disconnect the electrical leads from the flap motor and identify them with tags to facilitate installation.

(3) Remove the safety wire from the retaining nuts securing the flexible shafts to the gearbox and remove the nuts.

(4) Disengage the shafts from the gearbox. Remove the bolts, washers, and nuts securing the motor and gearbox assembly to the rear spar. Remove the assembly from the airplane.

NOTE: Refer to the 99 Airliner Series Component Maintenance Manual for separation of the flap motor and gearbox.

B. Installation

NOTE: Refer to the 99 Airliner Series Component Maintenance Manual for reassembly of the flap motor and gearbox.

(1) Secure the motor and gearbox assembly to the rear spar with the retaining bolts, washers and nuts.

(2) Connect the flexible shafts to both sides of the gearbox with the retaining nuts and safety wire the nuts.

(3) Connect the electrical leads to the flap motor and remove the identification tags. Engage the circuit breaker.

(4) Cycle the flap system to check for proper rigging. Ascertain that no rollers bottom out at either end of travel and that the flaps are synchronized.

(5) Install the center aisle floorboard over the flap motor and gearbox assembly.

6. FLAP ACTUATOR

A. Removal

(1) Fully extend the flaps and disconnect the actuator from the flap.

(2) Remove the inspection plate on the lower surface of the wing and uncouple the flexible drive shaft.

(3) Remove the mounting bolts and actuator from the bracket. Remove the bushings from the actuator.

B. Installation

(1) Position the actuator with its vent hole up and install the mounting bolts and bushings.

(2) Connect the drive shaft and replace the access cover. The shaft drive key must be aligned with the slot in the actuator.

(3) Extend the actuator until the flap synchronizes with the next flap, then connect it to the flap.

(4) Check the flap rigging.

(5) If a new or overhauled actuator is installed, lift lightly on the trailing edge of the flap while running the flaps through a complete extension-retraction cycle. There should be no roughness or indication of binding in the actuator.

7. FLAP CONTROL SYSTEM (U-1 THRU U-164, EXCEPT U-50)

A. Rigging

NOTE: All flap rigging must be accomplished using an external power source capable of delivering and maintaining 28.25 ± 0.25 volts during the entire rigging operation.

(1) Disconnect the micropositioner from the flap control circuit. Apply power to wire number D46A22 to run the flaps up and wire number D49A22 to run the flaps down. Rig the flap limit switches in the following manner:

(a) Adjust the up-limit switch so that the right inboard flaps fair with the bottom of the fuselage.

(b) Adjust the down-limit switch for 41° to 44° of flap travel.

(c) Synchronize the other three flaps by removing the bolts from the actuator arms and screwing the arms in or out of the actuators.

NOTE: The left inboard flap must fair flush with the bottom of the fuselage with the flaps up. The outboard flaps may vary from 1/2 inch above to 1/4 inch below the inboard flaps at the trailing edges. When the ailerons are properly rigged and in the neutral position, the outboard flaps may be up to 1/2 inch below, but cannot be above the ailerons.

(2) Adjust the wing follow-up potentiometer to read 5 ± 2 ohms in the UP position (terminal 1 to 3), and approximately 295 ohms in the DOWN position (terminal 1 to 3). Lengthen the potentiometer actuator arm to decrease rotation, and shorten to increase rotation in order to arrive at the proper reading.

(3) Position the flap to the approach position. Apply power to the flap control potentiometers.

(4) Put the flap handle in the approach detent. Rotate the flap handle potentiometer until the voltage reading between wire numbers D51A22 and D47A22 at the micropositioner reads zero.

(5) Connect the micropositioner. Run the flap UP and back down to APPROACH. Adjust the dead band resistor to 100 ± 5 ohms.

(6) Check the APPROACH position at the flap when positioned by the flap handle. Adjust the flap handle potentiometer as necessary to obtain 13° + 0° - 1° when going to the APPROACH position from the UP position and 13° + 0° - 1° when going to the APPROACH position from the DOWN position.

(7) Check that the flap goes full UP and full DOWN. Adjust the flap handle stops as necessary.

(8) Adjust the flap position indicator transmitter as required for proper indication.

(9) Check the switching transistors for proper operation by interrupting both the ground or power circuits on the potentiometer (one at a time). The circuit breakers should clear within 2 seconds.

NOTE: If adjustment of the flap position indicator transmitter is required, it may be accomplished in the following manner:

1. Extend the flaps to the full down position.

2. The flap position indicator transmitter is attached to the wing edge midway along the length of the flap adjacent to the flap track. Adjustment is made by increasing or decreasing the length of the rod connecting the transmitter to the flap. A distance of approximately 4 inches from the shoulder of the adjusting nut to the tip of the rod with the flap in the full extended position should indicate a reading of 100% flaps on the indicator in the cockpit.

(10) After rigging has been completed, run the flaps down and check the position indicator. If the indicator does not show 100% FLAPS, loosen the transmitter arm bolt and rotate the transmitter shaft until the reading is correct, then tighten the transmitter arm bolt.

(11) Run the flaps up and check the position indicator for 0% flaps. If the indication is not correct, lower the flaps and lengthen (if greater than 0%) or shorten (if less than 0%) the position transmitter fork to correct approximately one-half of the indication discrepancy.

(12) After the indicator travel has been corrected, adjust the transmitter arm as in Step (10) above.

8. FLAP CONTROL SYSTEM (U-50, U-165 AND AFTER)

A. Rigging

CAUTION: *Never connect or disconnect any PCB when power is on the airplane.*

(1) Locate the PCB card rack in the subfloor of the forward cabin center aisle and remove plug P238 from card number 5 (flap control PCB).

(2) Connect a jumper wire from pin 11 of the plug to ground.

(3) To drive the flaps down or up, the following procedure may be used:

(a) Down: Connect a jumper wire between pins 1 and 2 of plug P238; the flaps will run down when the battery switch is turned on.

(b) Up: Connect a jumper wire between pins 2 and 3 of plug P238; the flaps will run up when the battery switch is turned on.

(4) Rig the flap limit switches as follows:

(a) Adjust the up-limit switch so that the right inboard flap fairs with the bottom of the fuselage to wing root fairing.

(b) Adjust the down-limit switch for 41° to 44° of flap travel.

(c) Synchronize the other three flaps by removing the bolts from the actuator plugs and screwing the plugs in or out of the actuators.

NOTE: The left inboard flap must fair flush with the bottom of the fuselage to wing root fairing with the flaps up. The outboard flaps may vary from 1/2 inch above to 1/4 inch below the inboard flap at the trailing edges. When the ailerons are properly rigged and in the neutral position, the outboard flaps may be up to 1/2 inch below, but cannot be above the ailerons.

(d) Extend and retract the flaps and ensure that the flaps are fully extending and retracting when the travel limit switches are actuated. Check for binding and freedom of travel.

(5) With the flaps fully retracted, measure the resistance between pins 4 and 12 of plug P238; this value should be 2 ± 1 ohms. It not, adjust the control handle potentiometer (R107) for the proper resistance and record this value.

(6) Move the flap control handle to the approach position and again measure the resistance between pins 4 and 12; record this value.

(7) Move the flap control handle to the down position and again measure the resistance between pins 4 and 12; record this value.

(8) Measure the resistance between pins 5 and 13; this value must be 5 ± 1 ohms. If not, the resistance can be increased by adjusting the flap feedback potentiometer (R108) located in the flap well forward of the left inboard flap. Shortening the potentiometer arm will increase the resistance while lengthening the arm will reduce the resistance.

(9) Lower the flaps to between 12° and 13° according to Step (3) and measure the resistance between pins 5 and 13 again; adjust the shaft of the feedback potentiometer until this resistance matches the resistance measured between pins 4 and 12 when the flap control handle was in the approach position. This adjustment can be made by loosening the adapter-clamp of the shaft of the potentiometer until the shaft will rotate inside of the adapter-clamp.

(10) Fully extend the flaps according to Step (3) and once again measure the resistance between pins 5 and 13; this value must be at least 3 ohms less than the resistance measured between pins 4 and 12 when the flap control handle was in the down position. If not, adjust the length of the feedback potentiometer arm and repeat Steps (8), (9) and (10) until all conditions have been met.

(11) After the rigging has been completed, reconnect plug P238 to the flap control PCB (A238), run the flaps down and check the position indicator for a 100% reading. If the indicator does not show 100%, loosen the position transmitter arm bolt and rotate the transmitter shaft until the reading is correct, then tighten the transmitter shaft arm bolt. The flap position transmitter is located in the flap well forward of the right inboard flap.

(12) Retract the flaps and check the position indicator for 0% flaps. If the indication is not correct, lower the flaps and lengthen (if greater than 0%) or shorten (if less than 0%) the position transmitter fork to correct approximately one-half of the indication discrepancy, then repeat Step (11).

FLIGHT CONTROLS
GUST LOCKS AND DAMPER
DESCRIPTION AND OPERATION

1. CONTROL LOCKS

The control lock consists of a U-shaped clamp and three pins connected by a chain. The pins lock the primary flight controls and the U-shaped clamp fits around the engine power control levers and serves to warn the pilot not to start the engine's with the control locks installed. It is important that the locks be installed or removed together due to the possibility of an attempt to taxi or fly the airplane with the power levers released and the pins still installed in the flight controls. Install the control locks in the following sequence:

A. Installation

(1) Position the U-clamp around the engine power controls.

(2) Insert the small pin in the elevator-aileron pilot's control from the upper side of the pilot's control column.

NOTE: On airplane serials U-165 and after, the control wheel must be fully forward and rotated to the left 15 degrees before the control pin can be inserted.

(3) Insert the largest pin in the pilot's right rudder pedal by pushing forward on the left rudder pedal and inserting the pin into the hole, located on the inside of the pilot's right rudder pedal.

(4) Neutralize the rudder pedals and slide the pin into the hole in the left rudder pedal.

B. Removal

(1) Remove the pins from the pilot's left and right rudder pedals.

(2) Remove the elevator-aileron pin from the upper side of the pilot's control column.

(3) Remove the U-shaped clamp from the engine power controls.

NOTE: The control locks should be removed before towing, as serious damage could result in the steering linkage if towed by a tug with the rudder locks installed.

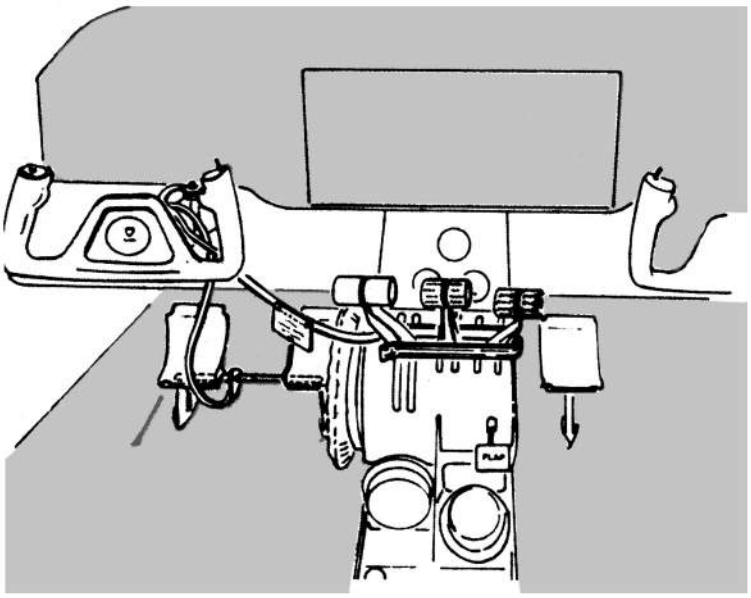

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Figure 1
Control Lock

List of Effective Pages

CH-SE-SU	PAGE	DATE
28-LOEP	1	Nov 1/10
28-CONTENTS	1 thru 4	Nov 1/10
28-00-00	1 thru 8	Nov 1/10
	201 thru 208	Nov 1/10
28-10-00	201 thru 213	Nov 1/10
28-20-00	1 thru 3	Nov 1/10
	201 thru 205	Nov 1/10
28-21-00	201 and 202	Nov 1/10
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	201 thru 203	Nov 1/10
28-41-00	1	Nov 1/10
	101 thru 103	Nov 1/10
	201 thru 204	Nov 1/10
28-42-00	1	Nov 1/10
	201 thru 212	Nov 1/10

Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

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FUEL
FUEL SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

A. Fuel System (99, 99A and B99)

The fuel system consists of two separate systems connected by a crossfeed system. Each fuel cavity is lined with a rubber bladder-type cell. Fuel for each engine is supplied from a 57 gallon nacelle tank, a 441 gallon center section tank, a 38 gallon wing leading edge tank, a 23 gallon wing outboard tank and a 23 gallon wing inboard tank. The tanks in each wing and the nacelle, all interconnected, have a combined total of 185 usable gallons. The wing fuel systems are interconnected through a crossfeed to supply a total of 370 usable gallons to either or both engines as circumstances warrant. Fuel flows by gravity from the wing tanks into the nacelle tank, except for 28 gallons that are trapped in each wing because of gravity feed valve cracking pressure, wing dihedral, and the location of the gravity feed lines in the tank walls. Each wing system has two filler caps, one in the nacelle tank and one in the wing leading edge tank.

NOTE: Aircraft U-146, U-148 and after have a 56 gallon nacelle tank. The total usable fuel in each wing and nacelle is 184 gallons. The wing fuel systems have a total of 368 usable gallons.

To assure that the system is properly filled, and to prevent a void in the system, service the nacelle tank first, then the wing tanks.

The inboard wing tank is cross-vented to the center section tank, the wing leading edge tank, and the outboard wing tank. The wing tank system is vented through a recessed vent coupled to a static vent in the underside of the wing adjacent to the nacelle. One vent is recessed to prevent icing. The second vent is added as a backup should the other vent become plugged. In the event that both vents are plugged, a vent check valve just outboard of the wing leading edge tank and a vent check valve adjacent to the nacelle filler cap will open to allow fuel flow at a vacuum of approximately one inch of water. A flame arrestor, which protects the system from ignition through the vent system should lightning strike the airplane is also incorporated in the system at this point.

A crossfeed line connects the two nacelle tanks. A switch controlled crossfeed valve is connected into the line at the aft inboard corner of the LH nacelle tank.

NOTE: To facilitate draining the fuel system, a tee connection is installed in the crossfeed line inboard of the crossfeed valve. Access to the tee connection is through the cover on the underside of the LH center section adjacent to the inboard side of the nacelle. The tee drains the fuel through the RH side. The LH side only may be drained by gravity through the nacelle drain with the crossfeed valve closed.

When the valve is in its normally closed position, each engine draws fuel from its respective fuel tank system. When the manually operated crossfeed control switch mounted on the pedestal just below the fuel quantity gages is actuated, power is drawn from a 10 ampere circuit breaker on the RH panel beside the copilot to the solenoid of the crossfeed valve. The crossfeed valve then opens to allow either or both engines to draw fuel from the entire fuel system. Power is also drawn from the crossfeed valve to illuminate a white CROSSFEED light in the annunciator panel mounted on the center of the instrument panel.

CAUTION: *On U-40 and after, except U-132 (refer to Fuel Systems A99A), the NAC NOT FULL lights will extinguish when the fuel quantity select switch is placed in the nacelle position. The light will not illuminate again while the switch remains in this position. Therefore, the nacelle tanks can be drained without the operator receiving a warning light.*

A jet transfer pump mounted on the inboard side of the main landing gear wheel well transfers fuel from the wing fuel sump (center section tank) to the nacelle tank. The jet pump receives its motive flow from the nacelle boost pump through a shutoff valve, filter, and check valve mounted on the inboard side of the wheel well immediately below the jet pump. The purpose of the shutoff valve is to permit filter removal without draining the fuel cells. The check valve prevents the engine from sucking air when the boost pump is not operating. An internal bypass valve in the filter permits the continued flow of the fuel should the filter become plugged or blocked with ice. The jet pump draws fuel from the center section tank and transfers it to the nacelle tank. Should the jet pump fail or the center section tank become empty, a float switch installed in the nacelle tank will close and illuminate the LEFT NAC NOT FULL or RIGHT NAC NOT FULL light on the annunciator panel, when the nacelle tank fuel level drops 2 gallons below full. This circuit is protected by a 5 ampere circuit breaker on the RH side panel beside the copilot. Should the jet transfer pump fail, all but 28 gallons of fuel will flow by gravity into the nacelle tank. Since the jet transfer pump depends on boost pump pressure for its motive flow, the jet transfer pump will cease to operate any time both the primary and secondary boost pumps fail. As long as either of the boots pumps operate and there is fuel in the wing tanks, the jet transfer pump will keep the nacelle full.

Fuel to the engine is supplied directly from the nacelle tank. The gravity feed line from the wing tanks is connected to a flanged fitting on the outboard side of the nacelle tank. A check valve in the line aft of the nacelle fitting allows fuel flow only into the nacelle tank. The motive line from the nacelle tank boost pumps to the jet transfer pump and the return line from the transfer pump are connected to the aft side of the nacelle tank. The overflow line that returns to the wing gravity feed line, the excess fuel delivered by the jet transfer pump, is connected into the nacelle tank adjacent to the connections for the transfer pump lines. A check valve at the filler neck with a cracking pressure of one inch of water prevents the fuel cell from collapsing of a partial vacuum when the overflow fuel line becomes empty as fuel is depleted. Fuel to the heater is drawn by suction from the bottom of the LH nacelle tank through a fitting on the inboard side. The supply line connecting the engine to the internal plumbing from the submerged boost pumps is attached to the forward inboard side of the nacelle fuel tank.

The primary and secondary boost pumps are installed in the bottom of the nacelle fuel tank just forward of the cutout for the landing gear. The line plumbed into each boost pump connects to the line that attaches to the fittings for the external lines leading to the jet transfer pump and engine. Either the primary boost pump or its backup unit, the secondary boost pump, is capable of supplying fuel to the engine driven fuel pump under the minimum pressure required by the engine manufacturer.

CAUTION: *Should the boost pumps fail, suction lift operation may be employed, however, suction lift operation is restricted to 10 hours total time between engine overhaul periods. If the pump is operated on suction lift beyond the 10 hour limit, overhaul or replacement of the pump is necessary.*

A check valve in the line leading from each boost pump prevents fuel from backflowing through the inactive pump. Should both boost pumps on one side of the fuel system fail. fuel under pressure can be obtained from the boost pumps and fuel cells on the opposite side by opening the crossfeed valve. The primary and secondary boost pumps are each actuated by a 10 ampere circuit breaker located directly below the master switch on the left subpanel. Fuel pressure from the boost pump provides the motive flow by which the jet transfer pump functions. Fuel under boost pump pressure is routed through a check valve and out the inboard side of the nacelle tank to the engine. The fuel is then routed forward to a normally open firewall shutoff valve installed in the fuel line immediately behind the engine firewall. The firewall shutoff valve for each engine "fuel system is actuated by its respective FIRE PULL handle on the upper center of the instrument panel on either side of the annunciator panel. When the FIRE PULL handle is pulled, its respective firewall shutoff valve closes to shut off the flow of fuel to the engine. This also arms the engine fire extinguisher for action. From the firewall shutoff valve, fuel is routed to the main fuel filter on the lower center of the engine firewall. The 20 micron filter incorporates a bypass valve to permit fuel flow in case of plugging and a drain valve to drain the filter prior to each flight. A pressure switch mounted on the right side of the firewall senses boost pump fuel pressure at the

filter. At 9 to 11 psig of decreasing pressure, the switch closes and actuates the yellow FUEL BOOST light in the annunciator panel. At this time the operating fuel boost pump should be shut off and its backup unit switched on. The yellow FUEL BOOST light will go out with 9 to 11 psig of increasing fuel pressure. From the main filter, fuel is routed through the fuel flow transmitter mounted on the firewall adjacent to the pressure switch. Fuel from the transmitter is routed through the fuel heater that utilizes heat from engine oil to warm the fuel. The fuel is then routed to the fuel control unit that monitors the flow of fuel to the engine fuel nozzles. A heater boot is also installed on the P_y air line of each engine. To minimize battery drain during cold starts and air starts, each airline heater is controlled by a switch within the pedestal that is operated by an actuator on the condition lever. The air line heaters are protected by a 5 ampere circuit breaker in the right side panel beside the copilot.

A solenoid valve in the fuel return line along the RH side of each nacelle immediately aft of the firewall permits purging of the fuel supply line during engine starts. The line is connected to the starter switch and opened only when the switch is in the "START" position.

The fuel gaging system is a capacitance type that is compensated for specific gravity and reads in pounds on a linear scale.

The fuel quantity indicating system includes a selector switch mounted below the fuel quantity gages. The total quantity of fuel in each wing fuel system is indicated by its respective fuel quantity gage when the selector switch is placed in the TOTAL position. If the selector switch is moved to the NACELLE position, the fuel quantity indicating circuitry for all fuel cells except the one in each nacelle is bypassed so that each fuel gage indicates only the quantity of fuel in its respective nacelle fuel cell. Each side of the aircraft has an independent gaging system consisting of a tank unit (capacitance probe) in the nacelle tank, one in the center section tank, one in the inboard wing tank, and two in the leading edge tank. The tank units consist of two concentric metal tubes with the active areas and the separation between the two tubes fixed to allow the dielectric constant of the fuel to be the determining factor of the capacity of the unit. The tank units produce a signal proportionate to the height of the fuel in the fuel tank to compare with a signal in the reference portion of the gage bridge circuit. The algebraic sum of the two signals is amplified and used to drive a servo-motor. When the algebraic sum of the two signals is not equal to zero (bridge balance), the motor will run in either the clockwise or counterclockwise direction, depending on the phase of the resultant signal. The motor drives the wiper of the rebalance potentiometer and the pointer. The wiper of the rebalance potentiometer picks off a voltage proportional to the height of the fuel (provided the empty and full calibration has been performed) and places it across a reference portion of the bridge circuit. This brings the bridge back into balance and the pointer indicates the weight of the fuel contained in the fuel tank. A compensator is also included in the circuitry to provide a correction signal for the reference portion of the bridge to minimize error in the indicator section. The compensator, identical to construction to the tank units, is installed at the lowest point in the nacelle tank so that it will always be submerged in fuel. The compensator senses any variation in the dielectric constant of the fuel caused by temperature changes, variations in manufacturer's fuels. etc.

The fuel system also includes a fuel drain collector system. Fuel from a drain on the fuel control unit is routed to the fuel drain collector tank mounted on the center of the fire seal directly under the accessory section of the engine. This tank contains an internal float for a normally open switch that closes the circuit to a pump mounted on the fireseal just to the right of the tank. When fuel from the fuel control unit is dumped, the tank fills and closes the switch. At the first activation of the fuel line heater from which the power for the pump is derived, fuel is pumped aft to the purge line leading to the nacelle tank. A check valve in the line prevents backflow of fuel during purging of the fuel system at engine start. The drain line is teed into the fuel return line aft of the fuel purge valve to prevent interruption of pressure in the fuel system. A vent line is plumbed from the top of the collector tank downward to the fuel drain manifold on the underside of the nacelle. The vent line is also routed through a flame arrestor.

B. Fuel System (A99A)

The fuel system may be considered the same as that installed on the 99, 99A and B99, except that the 44 gallon center section fuel tank, the jet transfer system, pump and plumbing have been removed. The nacelle tank float switch, which operates the LEFT NAC NOT FULL or RIGHT NAC NOT FULL lights on the annunciator panel, has been removed from both nacelle fuel tanks. The vent/overflow lines from the nacelle tank have been replaced by a single open vent line. With the removal of the nacelle tank filler cap, the system is serviced at a single point in the outboard end of the leading edge tank of each wing because the top of the nacelle tank is approximately 4 inches higher than the leading edge wing filler cap location, the nacelle tank usable fuel is approximately 43 gallons. The fuel capacity is 127 gallons of usable fuel per wing or a total of 254 gallons for the aircraft. A system drain line is teed into the gravity feed line that runs between the outboard fuel cells and the nacelle tank. The drain valve for this line is located on the underside of the wing, outboard of the airplane fuselage.

The fuel quantity selector switch has been removed from the fuel quantity indicating system. Each fuel quantity gage indicates only the total fuel quantity for its respective wing system.

The fuel system also includes a fuel drain collector system. Fuel from a drain on the fuel control unit is routed to the fuel drain collector tank mounted on the center of the fire seal directly under the accessory section of the engine. This tank contains an internal float for a normally open switch that closes the circuit to a pump mounted on the fireseal just to the right of the tank. When fuel from the fuel control unit is dumped, the tank fills and closes the switch. At the first activation of the fuel line heater from which the power for the pump is derived, fuel is pumped aft to the purge line leading to the nacelle tank. A check valve in the line prevents backflow of fuel during purging of the fuel system at engine start. The drain line is teed into the fuel return line aft of the fuel purge valve to prevent interruption of pressure in the fuel system. A vent line is plumbed from the top of the collector tank downward to the fuel drain manifold on the underside of the nacelle. The vent line is also routed through a flame arrestor.

C. Fuel System (C99)

Five interconnected rubber bladder-type fuel cells in each wing provide fuel storage in the C99 airplanes. The total fuel storage capability of each wing system is 184 gallons of usable fuel for engine operation. Under normal circumstances the wing systems operate independently; however in the event of an in-flight emergency, both systems can be used to supply fuel to one engine through a crossfeed interconnect.

Integral fuel cells in each system are vented through a vent network which is common to all the fuel cells. Anti-syphon protection and explosion protection is provided for by the installation of certain check valves and a flame arrestor. A ram air scoop and a heated vent air intake have been installed to insure positive venting of the system under the most adverse flight and climatic conditions.

Two engine-driven fuel pumps deliver fuel to each engine. An electrically driven fuel pump, installed inside the nacelle fuel cell, is used as an emergency backup pump and during crossfeed operations.

A series of drain points on the underneath side of the wing allow for a variety of draining operations ranging from removing fuel contaminates and defueling the airplane to the draining of specific lines and components of the fuel system.

The wing center section tank is the lowest tank in the system and is gravity fed from the tanks in the outboard wing section; therefore, a fuel transfer system was installed to transfer fuel from the center section tank into the nacelle tank. Should a failure in the fuel transfer system be encountered, the last 28 gallons contained in the center section tank would become unusable fuel.

During engine start procedures, a solenoid valve in the vapor return line opens and permits air and any excess fuel purged from the fuel control unit to return to the fuel system.

After engine shutdown, a small amount of fuel present in the fuel nozzle manifold drains into a small collector tank. The tank is mounted to one of the lower fire shields in the aft engine compartment. An electric float switch senses the tank fuel level and activates an electric pump which then transfers the fuel back to the nacelle tank. When the collector tank is emptied, the float switch turns off the pump. The entire operation is automatic and requires no input or additional duties from the crew.

Fuel system operation and performance is monitored through various indictors (gages and annunciator lights) found in the flight compartment. Fuel gages indicate total fuel in all tanks or fuel quantity in the nacelle tanks only. Fuel quantity gaps indicate in pounds of fuel remaining.

Fuel flow gages indicate the actual amount of fuel, in pounds per hour, being metered to each engine at any given time.

In addition to the indicating gages mentioned earlier, caution lights in the annunciator panel alert the flight crew to the following fuel conditions: nacelle tank not full, no motive flow, and low fuel pressure.

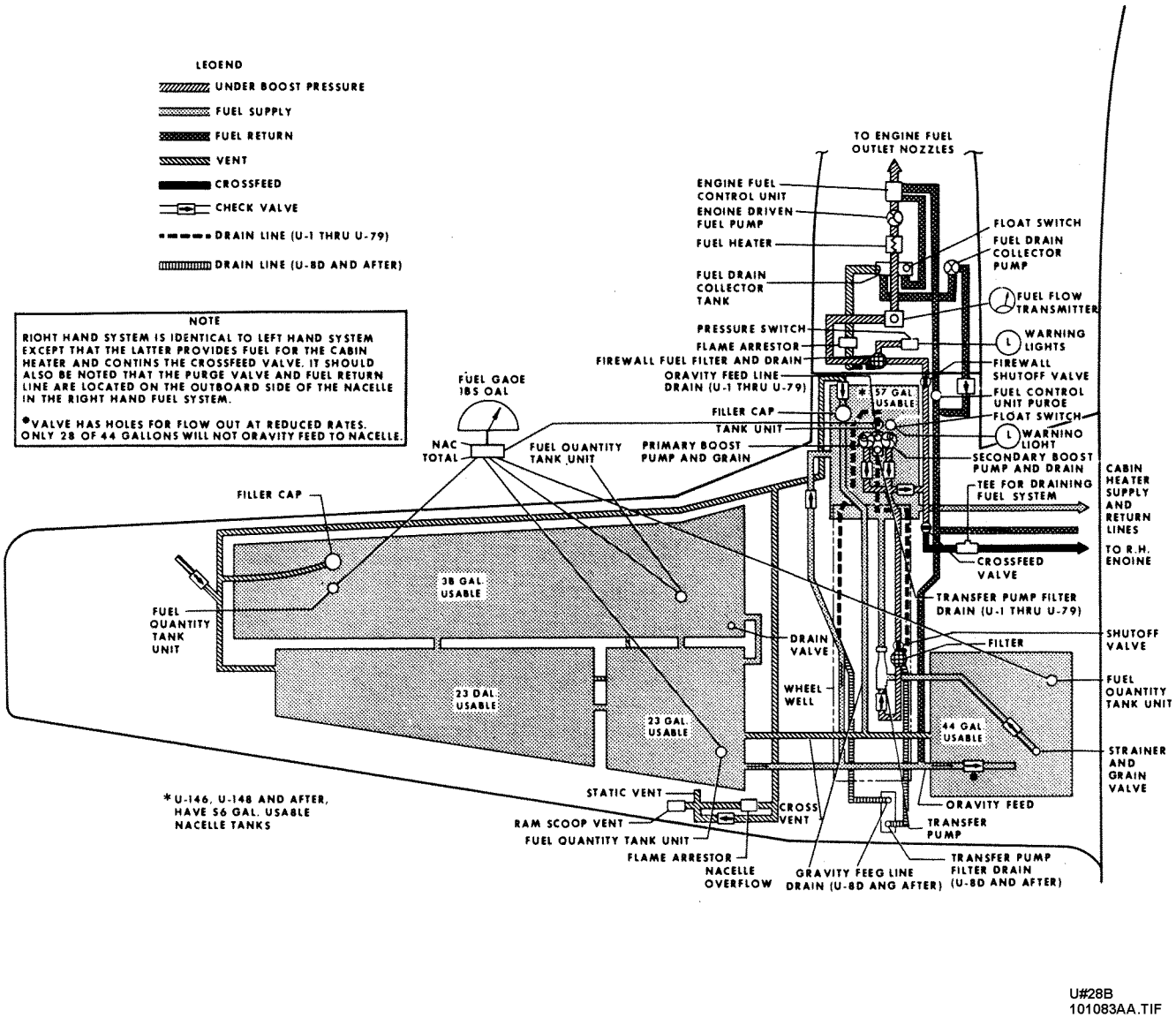


Figure 1
Fuel System Schematic (99, 99A and B99)

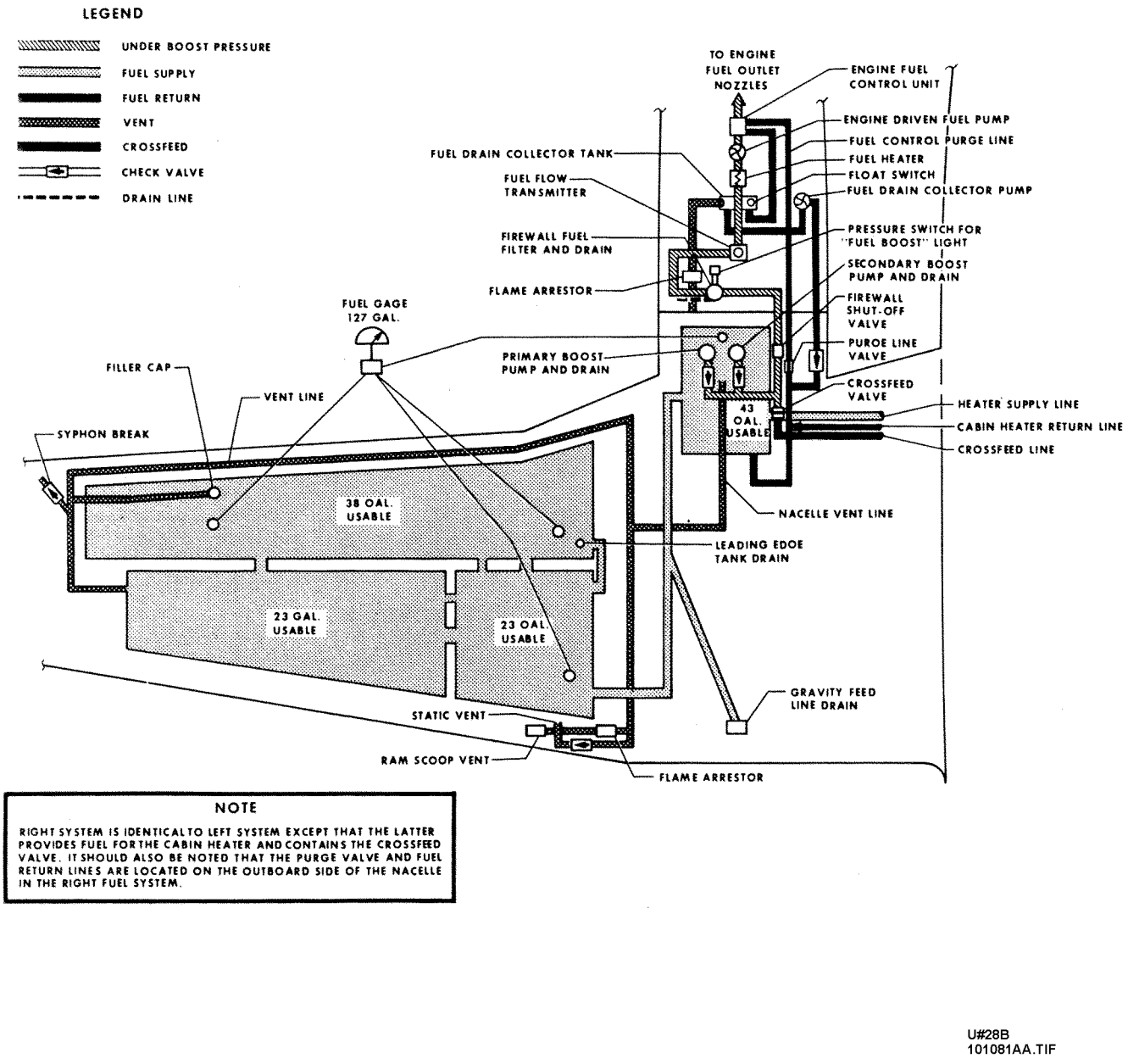


Figure 2
Fuel System Schematic (A99A)

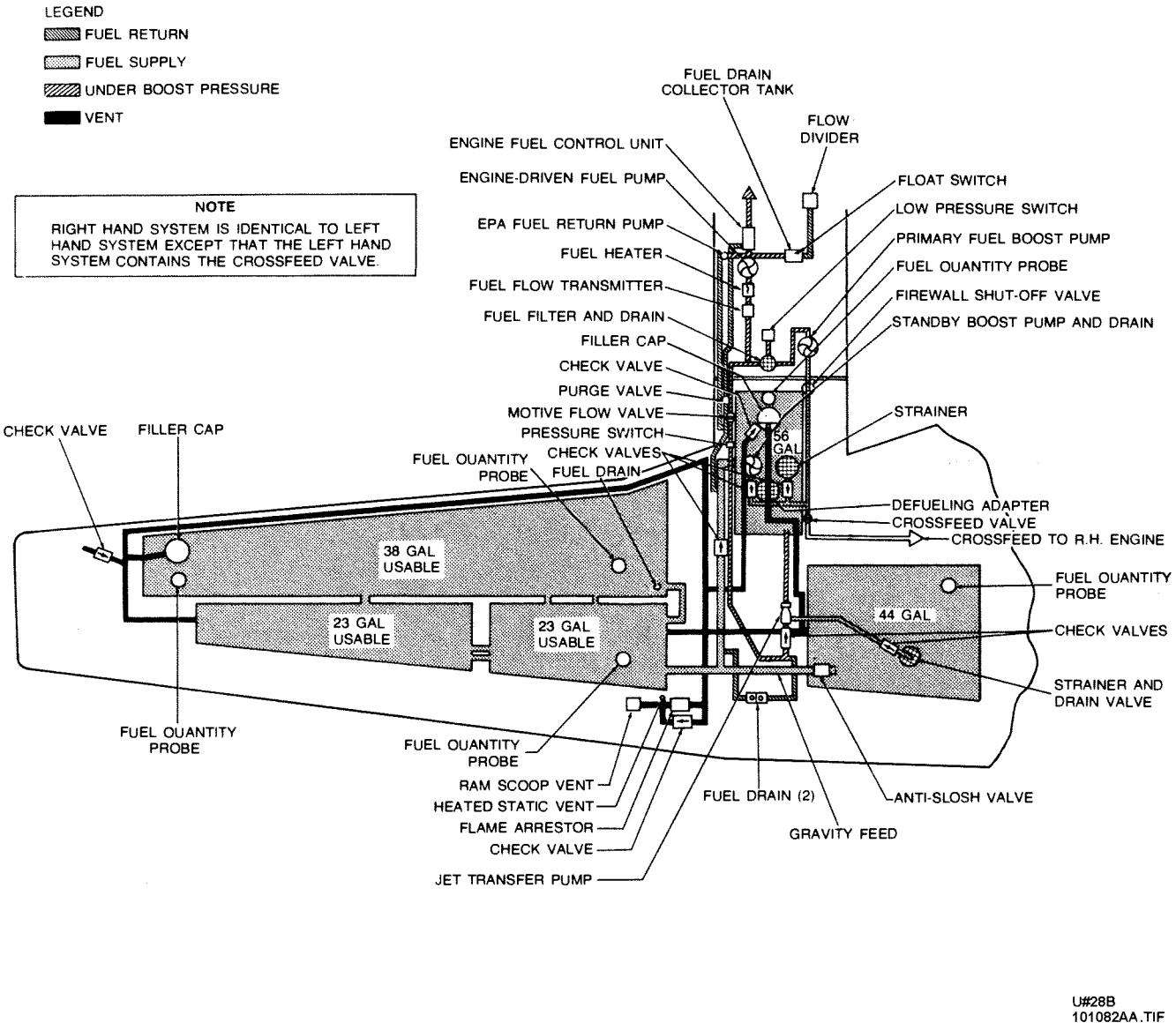


Figure 3
Fuel System Schematic (C99) (U-50, U165 and After)

FUEL
FUEL SYSTEM
MAINTENANCE PRACTICES

1. FUEL

A. Handling

All hydrocarbon fuels contain some dissolved and some suspended water. The quantity of water contained in the fuel depends on temperature and the type of fuel. Kerosene, with its higher aromatic content, tends to absorb and suspend more water than aviation gasoline. Along with the water, it will suspend rust, lint and other foreign materials longer. Given sufficient time, these suspended contaminants will settle to the bottom of the tank. However, the settling time for kerosene is five times that of aviation gasoline. Due to this fact, jet fuels require good fuel handling practices to assure that the Model 99 is serviced with clean fuel. If recommended ground procedures are carefully followed, solid contaminants will settle and free water can be reduced to 30 parts per million (ppm), a value that is currently accepted by the major airlines.

Since most suspended matter can be removed from the fuel by sufficient settling time and proper filtration, they are not a major problem. Dissolved water has been found to be the major fuel contamination problem. Its effects are multiplied in airplane operating primarily in humid regions and warm climates.

Dissolved water cannot be filtered from the fuel by micronic type filters, but can be released by lowering the fuel temperature, such as will occur in flight. For example, a kerosene fuel may contain 65 ppm (8 ounces per 1000 gallons) of dissolved water at 80°F. When the fuel temperature is lowered to 15°F. only about 25 ppm will remain in solution. The difference of 40 ppm will have been released as supercooled water droplets which need only a piece of solid contaminant or an impact shock to convert them to ice crystals. Tests indicate that these water droplets will not settle during flight and are pumped freely through the system. If they become ice crystals in the tank, they will not settle since the specific gravity of ice is approximately equal to that of kerosene. 40 ppm of suspended water seems like a very small quantity but, when added to suspended water in the fuel at the time of delivery, is sufficient to ice a filter. When the critical fuel temperature range is from 0° to -20°F. which produces severe system icing, water droplets can freeze at any temperature below 32°F.

Water in jet fuel also creates an environment favorable to the growth of a microbiological "sludge" in the settlement areas of the fuel cells. This sludge, plus other contaminants in the fuel, can cause corrosion of metal parts in the fuel system as well as clogging the fuel filters.

Although the Model 99 uses bladder type fuel cells and all metal parts, except the main boost pumps and transfer pumps, are mounted above the settlement areas, the possibility of filter clogging and corrosive attacks on fuel pumps exist if contaminated fuels are consistently used. Since fuel temperature and settling time affect total water content and foreign matter suspension, contamination can be minimized by keeping equipment clean, using adequate filtration equipment and careful water drainage procedures, storing the fuel in the coolest areas possible, and allowing adequate settling time. Underground storage is recommended for fuels. Filtering the fuel each time it is transferred will minimize the quantity of suspended contaminants carried by the fuel.

The primary means of fuel contamination control by the owner/operator is "Good Housekeeping." This applies not only to fuel supply. but to keeping the airplane system clean. The following is a list of Steps that may be taken to prevent and recognize contamination problems.

(1) Know your supplier. It is impractical to assume that fuel free from contaminants will always be available, but it is feasible to exercise precaution and be watchful for signs of fuel contamination.

(2) Assure, as much as possible, that the fuel obtained has been properly stored, filtered as it is pumped to the truck, and again as it is pumped from the truck to the airplane.

(3) Perform filter inspections to determine if sludge is present.

(4) Maintain good housekeeping by periodically flushing the fuel tankage system. The frequency of flushing will be determined by the climate and the presence of sludge.

(5) Since aviation gas is an alternate fuel, it should be used occasionally as a means to change fuel tank environment, thus destroying a possible microbiological growth pattern. The 150 hours maximum operation on aviation gas per a "Time Between Overhaul" should be observed.

(6) Use only clean fuel servicing equipment.

(7) After refueling, allow a three hour settle period whenever possible, then drain a small amount of fuel from each drain.

CAUTION: *Remove spilled fuel immediately to prevent the surface thus contaminated from causing tire damage.*

B. Fuel Grades and Types

Jet A, Jet A-1, Jet B, JP4, JP-5 and JP-8 fuels may be in any ratio. Aviation gasoline, grades 80 red (80187), 100 green (100/130) and 100LL blue, are emergency fuels and may be mixed in any ratio with the normal fuels when necessary. However, use of the lowest octane rating available is suggested due to its lower lead content. The use of aviation gasoline shall be limited to 1% hours operation during each Time Between Overhaul (TBO) period.

Table 201 gives fuel refiner's brand name, along with the corresponding designations established by the American Petroleum institute (API) and the American Society of Testing Material (ASTM). The brand names are listed for ready reference and are not specifically recommended by Hawker Beechcraft Corporation. Any product conforming to the recommended specification may be used.

C. Filling the Tanks

When filling the airplane fuel tanks. always observe the following:

(1) Make sure the airplane is statically grounded to the servicing unit and to the ramp.

(2) Service the nacelle fuel tanks before filling the outboard fuel tanks. On the A99A, the nacelle tank filler caps have been removed, thereby allowing single point refueling for each wing at the outboard edge.

(3) Allow a three hour settle period whenever possible, then drain a small amount of fuel from each drain point.

CAUTION: *When inserting or removing the fuel nozzle, use extreme care to prevent the fuel hose from rubbing against the deicer boot. Also, do not allow fuel to contact the deicer boot.*

2. DRAINING FUEL SYSTEM

NOTE: Before beginning the defueling operation, always statically ground the airplane structure.

A. Prior to U-165, Except U-50

The boost pumps may be utilized to drain the fuel system. To accomplish this, remove the cover from the underside of the LH center section adjacent to the inboard side of the nacelle to gain access to the tee connection used for draining. Connect a line to the tee and pump the fuel into a suitable tank or container. The crossfeed valve must be in the open position of the LH fuel system is to he drained.

A faster, alternate method of draining the fuel system with the boost pumps is to disconnect the flexible fuel heater line (under the engine cowling at the engine accessory case), connect a drain line to the flexible line and then pump the fuel overboard into a large, suitable container. By using this method and leaving the crossfeed valve closed, the fuel may be evacuated from only one wing, thereby not requiring complete draining of the entire system.

B. U-50, U-165 and After

A defueling adapter, installed in the nacelle fuel cell and located aft of the standby pump, contains a check valve to prevent fuel drainage when the plug is removed. Each wing fuel system may be drained as follows:

(1) Remove the cover on the bottom of the nacelle to gain access to the adapter plug.

(2) Remove the plug and screw an AN832-12 union into the adapter. The fuel will begin draining as the check valve is unseated by the union.

(3) The fuel may be gravity drained or pumped out with the aid of a fuel truck.

NOTE: Remove the filler cap during draining to prevent damage or collapse of fuel cells.

(4) The quickest way to drain the wing center section tank is to turn the standby electric fuel pump on and use the airplane's transfer system to move the fuel from the center section tank into the nacelle tank. The center section tank can also be drained by using the sump drain in the center section near the fuselage and letting the fuel gravity flow into a suitable container. The wing center section tank need not be defueled in order to replace the standby electric boost pump.

3. ENGINE FUEL FILTERS AND SCREENS

NOTE: All inline fuel filters and engine driven pump filters should be inspected and cleaned every 100 hours as noted in the titled paragraphs following. Additional inspection and cleaning should be accomplished for the following conditions:

(1) At any time a suspected contaminated fuel is used.

(2) At any time the submerged fuel boost pumps are replaced The sump tank area should also be properly cleaned at this time.

If abnormal quantities of contaminants are found, notify the local FAA General Aviation District Office so that corrective action may be taken regarding any other airplanes which may have received the same fuel.

4. CLEANING FILTERS

NOTE: All inline fuel filters and engine driven pump filters should be inspected and cleaned every 100 hours as noted in the titled paragraphs following. Additional inspection and cleaning should be accomplished for the following conditions:

A. Firewall Filter

Clean the firewall filter every 100 hours as follows:

(1) Open the access door on the LH side of the aft lower cowling to gain access to the firewall filter.

(2) Remove the drain hose from the firewall filter.

(3) Cut the lockwire securing the filter housing retaining nut and remove the nut.

(4) Remove the filter housing from the filter body.

(5) Remove the locknut and the filter retaining nut and remove the filter pack assembly (the packs need not be removed from the center tube).

(6) Inspect the filter pack for foreign material and microbiological sludge.

(7) Plug the open ends of the center tube and wash the unit in solvent (38, Table 6, 91-00-00), or its equivalent.

(8) Install the filter pack assembly; torque the retaining nut 10 to 15 inch-pounds.

(9) Install the locknut and torque the locknut 40 to 60 inch-pounds while holding the retaining nut in place.

(10) Install the filter housing and torque the retaining nut 80 to 120 inch-pounds and safety wire.

(11) Attach the drain hose to the firewall filter.

B. Transfer Line Filter

Clean the transfer line filter every 100 hours as follows:

NOTE: The jet transfer system and filter are not installed on the A99A. The jet transfer line filter is not installed on the C99.

(1) Shut off the valve adjacent to the transfer line filter on the inboard side of each main landing gear wheel well.

(2) Disconnect the drain line from the transfer line filter.

(3) Cut the lockwire securing the filter housing retaining bolt and remove the bolt, filter housing, retaining spring and filter pack assembly.

(4) Inspect the filter pack assembly for foreign material and microbiological sludge. Do not remove the packs from the center tube.

(5) Plug the open ends of the center tube and wash the unit in solvent (38, Table 6, 91-00-00), or its equivalent.

(6) Install the filter pack assembly, retaining spring, housing and retaining bolt. Torque the retaining bolt 80 to 100 inch-pounds and safety wire.

(7) Connect the drain line to the transfer line filter. Open the valve in the transfer line after servicing the filter.

C. Fuel Pump Screen (Vickers Fuel Pump)

Clean as follows at intervals of 100 hours:

(1) Cut the lockwire securing the fuel screen cap-nut to the fuel pump.

(2) Unscrew and remove the cap-nut and screen together, lifting the assembly vertically to avoid damaging the screen body.

(3) Disconnect the fuel inlet line at the pump end to drain off the residue of fuel.

(4) Blow all foreign matter from the screen with clean, dry compressed air.

(5) Install a new packing to the groove on the screen cap-nut.

(6) Install the screen and cap-nut assembly and tighten the cap-nut to a torque of 75 to 100 inch-pounds, then safety the cap-nut with lockwire.

(7) Connect the fuel inlet line to the pump. torque the coupling nut in accordance with the engine Maintenance Manual, and safety the nut with lockwire.

(8) Perform an engine run-up, then check the pump body for leaks after engine shut-down.

D. PESCO Fuel Pump Filter (Every 100 Hours)

(1) Cut the lockwire and remove the attaching bolts (1) and washers (2) from the filter cover (3) on the bottom of the fuel pump (Ref. Figure 201).

(2) Insert a No. 10 (0.190) 32UNF3B screw in the filter removal hole in the cover subassembly and pull firmly on the screw to remove the filter (4) from the pump housing.

CAUTION: *Never pry the cover off with a sharp instrument for it may damage the pump housing or cover.*

(3) Separate the spring (5) and filter (4) from the cover (3) by removing the retaining bolt (6) and washer (7).

(4) Remove the preformed packings (8 and 9) from the cover.

(5) Clean the filter with solvent (38, Table 6, 91-00-00) and blow dry with filtered, compressed air.

(6) Install new preformed packings (8 and 9) on the filter cover (3).

(7) Secure the filter (4) and spring (5) to the filter cover (3) with the retaining bolt (6) and washer (7). Torque the retaining bolt to 25 to 30 inch-pounds.

(8) Install the assembled filter and cover in the pump housing and torque the attaching bolts (1) 40 to 46 inch-pounds. Safety the cover bolts with lockwire.

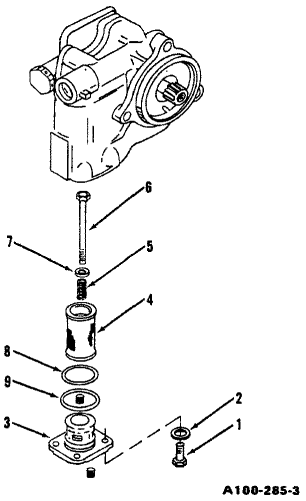


Figure 201
PESCO Fuel Pump Filter

E. PESCO Fuel Pump Micron Filter (Every 100 Hours)

(1) Unscrew and remove filter housing from fuel pump (Ref. Figure 202).

(2) Remove filter element and discard.

(3) Before installing new filter element, unscrew inlet screen cover and remove assembly.

(4) Remove long bolt, spring, washers and separate screen from cover. Wash the screen with solvent and replace in reverse order of disassembly.

(5) Tighten bolt and torque 20 to 23 inch-pounds.

(6) Install two new preformed packings on screen cover and assemble screen and cover assembly to pump body.

(7) Tighten cover to compress the packing and obtain metal-to-metal contact and then lockwire.

(8) After cleaning and installing screen, install a new fuel pump outlet filter. (Check for pressure of preformed packing in internal diameter. The element must be installed so that the preformed packing slides over the spigot in the pump body).

(9) Replace preformed packing on filter housing.

(10) Install filter housing and tighten to compress preformed packing to obtain metal-to-metal contact.

(11) Lockwire square section of housing to hole on nameplate boss. Check for leaks after engine ground checks.

Table 201
Fuel Brands and Type Designations

PRODUCT NAME	DESIG	PRODUCT NAME	DESIG
AMERICAN OIL		PURE OIL COMPANY	
American Jet Fuel Type A	Jet A	Purejet Turbine Fuel Type A	Jet A
American Jet Fuel Type A-1	Jet A-1	Purejet Turbine Fuel Type A-1	Jet A-1
ATLANTIC REFINING COMPANY		RICHFIELD PETROLEUM COMPANY	
Arcojet-A	Jet A	Richfield Turbine Fuel A	Jet A
Arcojet-A-1	Jet A-1	Richfield Turbine Fuel A-1	Jet A-1
Arcojet-B	Jet B		
BP TRADING COMPANY		SHELL OIL COMPANY	
BP A.T.K.	Jet A-1	Aeroshell Turbine Fuel 640	Jet A
BP A.T.G.	Jet B	Aeroshell Turbine Fuel 650	Jet A-1
		Aeroshell Turbine Fuel JP-4	Jet B
CALIFORNIA TEXAS COMPANY		SINCLAIR OIL COMPANY	
Caltex Jet A-1	Jet A-1	Sinclair Superjet Fuel	Jet A
Caltex Jet B	Jet B	Sinclair Superjet Fuel	Jet A-1
CITIES SERVICE COMPANY		STANDARD OIL OF CALIFORNIA	
Turbine Type A	Jet A	Chevron TF-1	Jet A-1
		Chevron JP-4	Jet B
CONTINENTAL OIL COMPANY		STANDARD OIL OF KENTUCKY	
Conoco Jet-40	Jet A	Standard JF A	Jet A
Conoco Jet-50	Jet A	Standard JF A-1	Jet A-1
Conoco Jet-60	Jet A-1	Standard JF B	Jet B
Conoco Jet JP-4	Jet B		
GULF OIL COMPANY		STANDARD OIL OF OHIO	
Gulf Jet A	Jet A	Jet A Kerosene	Jet A
Gulf Jet A-1	Jet A-1	Jet A-1 Kerosene	Jet A-1
Gulf Jet B	Jet B		
HUMBLE OIL COMPANY		TEXACO	
Enco Turbo Fuel A	Jet A	Texaco Avjet K-40	Jet A
Enco Turbo Fuel 1-A	Jet A-1	Texaco Avjet K-58	Jet A-1
Enco Turbo Fuel 4	Jet B	Texaco Avjet JP-4	Jet B
Esso Turbo Fuel A	Jet A		
Esso Turbo Fuel 1-A	Jet 1-A		
Esso Turbo Fuel 4	Jet B		
MOBIL OIL COMPANY		UNION OIL COMPANY	
Mobil Jet A	Jet A	76 Turbine Fuel	Jet A-1
Mobil Jet A-1	Jet A-1	Union JP-4	Jet B
Mobil Jet B	Jet B		
PHILLIPS PETROLEUM COMPANY			
Philjet A-50	Jet A		
Philjet JP-4	Jet B		

NOTE: Jet A - Aviation kerosene-type fuel with -40°F (-40°C) maximum Freeze Point.

Jet A-1 - Aviation kerosene-type fuel with -58°F (-50°C) maximum Freeze Point.

Jet B - Aviation wide-cut gasoline type fuel similar to MIL-T-5624 grade JP-4, but may have Freeze Point of -60°F (-51°C) instead of maximum -76°F (-60°C).

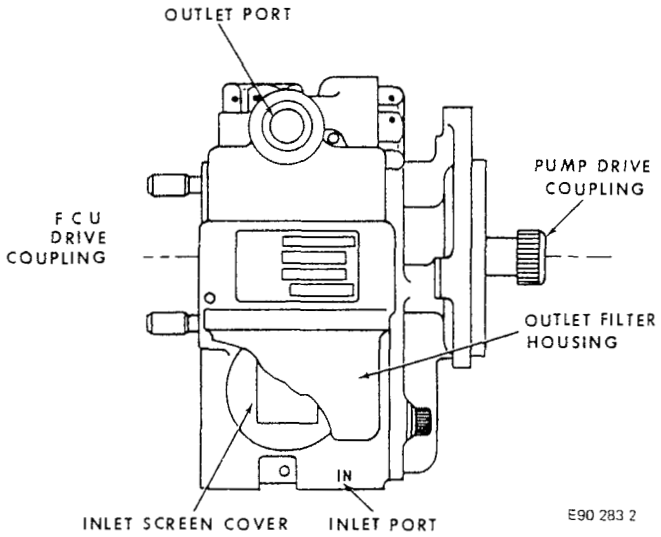


Figure 202
PESCO Fuel Pump Filter

FUEL
STORAGE
MAINTENANCE PRACTICES

1. FUEL CELLS

Rubber bladder-type fuel cells are equipped with nipples for attaching plumbing lines to the fuel cell. In order to avoid damaging the fuel cell or its nipple fittings, the following torque values should be applied to the nipple fitting clamps during installation:

Table 201

NIPPLE FITTING ID (IN.)	TORQUE (IN.-LBS)
0.25 to 0.50	12 to 16
0.75 to 1.00	15 to 20
1.50	25 to 30
2.00	30 to 35
3.00	35 to 40

Goodyear fuel cells having yellow nipples shall be torqued to 25 $\pm$ 5 inch-pounds: all others shall be torqued according to Table 201.

Always position hose clamps on internal nipple fittings so that the screw body travels freely as the screw is tightened. Never allow the screw body to wedge between nipple fitting OD and tank wall. This can result in false torque readings, leaks and/or nipple fitting damage.

NOTE: Goodyear bladder-type fuel cells returned to the manufacturer for repair or replacement will not be repaired if foreign material has been applied to the fittings of the fuel tank wall, either inside or outside. Foreign material is considered to be any material, such as sealants, coatings, cements, etc., not approved by the manufacturer for the installation, operation or repair of a bladder-type fuel cell.

To facilitate installation of bladder-type fuel cells, allow the fuel cell to set at room temperature (above 60°F) for a period of time sufficient to ensure flexibility.

2. NACELLE FUEL CELL

A. Removal

(1) Make sure that all electrical power is disconnected on the airplane.

(2) Drain all fuel from the nacelle and wing panel cells.

NOTE: The filler cap, transfer pump and plumbing are removed on the A99A airplanes.

(3) Remove the screws securing the access cover to the top of the nacelle and fuel filler neck. Remove the access cover.

CAUTION: *Handle the tank units carefully, for dents in the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precaution against dropping the connector insulator into the fuel cell.*

(4) Disconnect the wires leading to the tank unit, float switch and compensator terminal post. Tag the wires to ensure proper installation.

(5) Working through the filler cap opening, remove the bolt securing the vent line to the filler neck.

(6) Remove the bolts securing the fuel cell cover to the fuel cell. Remove the cover from the fuel cell.

(7) Working through the cover opening, disconnect the vent line, boost pump line, and transfer pump return line from the cover on the aft side of the fuel cell. Remove the vent line through the cover opening on top of the tank.

(8) Working through the cover opening, remove the four retaining bolts, washers, and nuts securing the support brackets for the boost pump outlet lines to the cover bracket.

(9) Working in the wheel well, disconnect the line to the transfer pump, the line from the transfer pump, and the vent line from the aft fuel cell cover. Remove the bolts securing the cover to the cell and remove the cover and gaskets.

(10) Remove the primary and secondary boost pumps from the bottom of the fuel cell. Refer to BOOST PUMPS in Chapter 28-20-00 FUEL DISTRIBUTION - MAINTENANCE PRACTICES.

(11) This Step applies to U-50, U-165 and After. Working through the boost pump opening, remove the safety pins securing the fuel strainer to the fuel cell fasteners, then remove the strainer from the fuel cell.

(12) Unsafety and remove the retaining bolts securing the access cover behind the boost pumps in place. Remove the cover and gaskets from the bottom of the nacelle tank.

(13) Working through the cover opening on top of the nacelle tank, unclamp the transfer pump return line from the line leading to the crossfeed tee at the inboard side of the nacelle tank, and remove the transfer pump return line.

NOTE: One electric boost pump and the combustion heater have been removed on the C99.

(14) Disconnect the boost pump line at the fitting directly above the tee connection to the secondary (outboard) boost pump outlet line. Remove the boost pump line with the three check valves through the cover opening in top of the tank.

(15) Disconnect the line from the boost pumps at the crossfeed tee on the inboard side of the nacelle tank, then remove the boost pump lines through the cover opening on top of the tank.

(16) Remove the access panel on the inboard side of the nacelle immediately forward of the center section leading edge.

(17) The following procedure applies only to left hand fuel cell installations:

(a) Working through the access opening on the inboard side of the nacelle, disconnect the heater fuel supply line.

(b) Unsafety and remove the four retaining bolts and remove the connector flange for the heater supply line through the access opening on the inboard side of the nacelle forward of the center section leading edge.

(18) Working through the access opening on the inboard side of the nacelle, disconnect the lines to the crossfeed tee. Unsafety and remove the six bolts and washers securing the crossfeed tee in place. Remove the tee and gaskets from the fuel cell installation.

(19) Remove the access panel on the outboard side of the nacelle just forward of the wing leading edge.

(20) Working through the access opening on the outboard side of the nacelle, disconnect the fuel line at the elbow. Unsafety and remove the six bolts and washers securing the elbow in place, then remove the elbow and its two gaskets.

(21) Unsnap the Carr fasteners securing the top of the nacelle fuel cell to the adjacent structure.

(22) Collapse the fuel cell and remove it through the top opening. Use care to avoid damaging the fuel cell.

B. Installation

NOTE: Expose the fuel cell to temperatures of at least 60°F for a period sufficient to ensure flexibility during installation.

(1) Check the fuel cell cavity for foreign objects or debris. Collapse the cell and install it through the access opening. Use care to prevent damaging the fuel cell.

NOTE: The filler cap, transfer pump and plumbing are removed on the A99A airplane.

(2) Snap the Carr fasteners securing the top of the fuel cell to the holes in the adjacent nacelle structure.

(3) Working from the top of the nacelle, push the fuel cell into position inside the fuel cell bay.

(4) Working inside the wheel well, place a gasket between the fuel cell fitting and the skin and between the skin and cover when positioning the aft cover over the opening.

(5) Install and torque the cover retaining bolts 45 to 50 inch-pounds, then safety the bolts.

(6) Working inside the wheel well, connect the transfer pump line, vent line, and transfer pump return line to the aft side of the cover.

(7) Working through the access opening on the outboard side of the nacelle just forward of the wing leading edge, place a gasket between the fuel cell fitting and the skin and between the skin and elbow when positioning the elbow over the opening. Install and torque the six elbow retaining bolts 45 to 50 inch-pounds, then safety the bolts. Connect the fuel line to the elbow and reinstall the access panel.

(8) Working through the access opening on the inboard side of the nacelle just forward of the center section leading edge, place a gasket between the fuel cell fitting and the skin and between the skin and crossfeed tee when positioning the tee over the opening. Install and torque the six tee retaining bolts 45 to 50 inch-pounds, then safety the bolts. Connect the fuel lines to the crossfeed tee.

(9) The following procedure applies only to left hand fuel cell installations and airplane serials prior to U-165, except U-50.

(a) Working through the access opening on the inboard side of the nacelle, place a gasket between the fuel cell fitting and the skin and between the skin and connector flange for the heater fuel supply line when positioning the flange over the opening.

NOTE: Be sure to install the flange with the countersunk side adjacent to the nacelle in order to provide efficient sealing and prevent possible leakage.

(b) Install and torque the four flange retaining bolts 45 to 50 inch-pounds, then safety the bolts.

(c) Working through the access opening on the inboard side of the nacelle, reconnect the heater fuel supply line.

(10) Install the access panel on the inboard side of the nacelle.

NOTE: One electric boost pump and the combustion heater have been removed on the C99.

(11) Working through the cover opening in top of the nacelle tank, position the boost pump lines with the three check valves in the bottom of the fuel cell so that the boost pump check valves and lines align with the openings for the boost pumps, then connect the remaining check valve and line to the crossfeed tee on the inboard side of the fuel cell.

(12) Place a gasket between the fuel cell fitting and the skin and between the skin and cover plate when positioning the cover plate over the access opening under the nacelle directly behind the boost pump opening. Install and torque the cover plate retaining bolts to 45 to 50 inch-pounds, then safety the bolts.

(13) Working through the cover opening in top of the nacelle fuel tank, install the four retaining bolts, washers, and nuts securing the support brackets for the boost pump outlet lines to the bracket on the cover plate installed in the preceding step.

(14) Working through the cover opening in the top of the nacelle fuel tank, install in the fuel tank the line that connects the outboard boost pump line to the upper inboard fitting of the cover in the aft end of the fuel cell.

(15) Working through the cover opening in the top of the nacelle fuel tank, connect the transfer pump return line to the lower fitting of the cover in the aft end of the nacelle fuel cell. Clamp the lower end of the line to the line from the boost pump at a point adjacent to the crossfeed tee on the inboard side of the fuel cell.

(16) Working through the cover opening in the top of the nacelle fuel tank, connect the lower end of the vent line to the cover in the aft end of the fuel cell.

(17) Place a gasket on each side of the fuel cell cover pan. Align the fuel cell cover with the mounting holes on top of the pan. Install and torque the cover retaining bolts to 45 to 50 inch-pounds, then safety the bolts.

CAUTION: *Handle the tank units carefully for dents in the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(18) Working through the filler cap opening, secure the vent line to the filler neck with the retaining bolts.

(19) Connect the wires to the compensator, float switch and tank unit terminal post. Make sure that the wires are reconnected to the correct post as marked upon removal.

(20) This Step applies to U-50, U-165 and After. Working through the boost pump opening, place the fuel strainer in position over the fuel cell fasteners, then secure the strainer in place with the safety pins.

(21) Apply Permatex No. 2 around the filler neck flange and reinstall the fuel access cover on the nacelle with the retaining screws.

(22) Install the primary (prior to U-165 except U-50) and secondary boost pumps. Refer to BOOST PUMPS in Chapter 28-20-00 FUEL DISTRIBUTION - MAINTENANCE PRACTICES.

3. WING LEADING EDGE FUEL CELL

A. Removal

The leading edge wing cells are located in the outboard wing panel leading edge.

(1) Make sure all electrical power to the airplane is disconnected.

(2) Drain the wing fuel cells.

(3) Remove the access plate on the lower surface of the wing leading edge. Working through this opening, disconnect the vent line at the outboard end of the cell.

NOTE: The clamp securing the nipple cap just forward of the vent line must be removed.

(4) Remove the access cover over the fuel filler neck and outboard tank unit.

(5) Remove the tank unit retaining screws and lift the tank unit up to disconnect the electrical leads and remove the lead strap, then remove the tank unit.

CAUTION: *Handle the tank units carefully for damage to the surface of the probe tube will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(6) Remove the bolt securing the vent line to filler neck and remove the line from the fuel cell.

(7) Remove the fuel cell cover retaining screws and remove the cover.

(8) Remove the bolts securing the filler neck to fuel cell and remove the filler neck.

(9) Working through the filler neck opening, remove the strap securing the tube for the tank unit wiring to the interconnect at the aft side of the fuel cell.

(10) Working through the filler neck opening, remove the nipple clamp on the aft side of the fuel cell. Pull the nipple off the tube.

(11) Remove the access panel just outboard of the nacelle on the upper wing skin.

(12) Remove the inboard tank unit access cover.

(13) Disconnect the ground straps and electrical leads from the fuel cell cover and tank unit.

(14) Remove the tank unit retaining screws and lift the tank unit up to disconnect the electrical leads and the lead strap, then remove the tank unit.

CAUTION: *Handle the tank unit carefully for damage to the surface of the probe tube will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(15) Remove the fuel cell cover.

(16) Working through the opening, remove the strap securing the tube for the tank unit wiring to the interconnect at the aft side of the fuel cell.

(17) Working through the opening, remove the 2 nipple clamps at the aft side of the fuel cell and pull the nipples free of tubes.

(18) Disconnect the vent line at the inboard end of the cell. Access is gained through the access opening adjacent to the nacelle.

(19) Remove the tube with the tank unit wiring from the fuel cell out through the filler neck opening.

(20) Remove the drain valve from the bottom of the cell.

(21) Release the snap fasteners securing the cell to the upper wing skin.

(22) Collapse the cell and remove it through the inboard tank unit access opening.

B. Installation

NOTE: Expose the fuel cell to temperatures of at least 60°F for a period sufficient to ensure flexibility during installation.

(1) Check the fuel cell cavity for foreign objects or debris. Collapse the cell and install it through the inboard tank unit opening.

(2) Secure the snap fasteners that hold the fuel cell against the upper wing skin.

(3) Install the drain valve at the lower inboard end of the cell.

(4) Connect the vent line at the outside inboard end of the fuel cell. Torque the clamps on all cells equipped with type WWD clamps (Ref. Table 201).

(5) Working through the inboard tank unit opening, connect the lines at the aft side of the cell. Torque the clamps on all cells equipped with WWD clamps (Ref. Table 201).

(6) Connect the tank unit wires to the cover plate and install the round cover plate at the inboard end of the fuel cell.

(7) Install the fuel cell cover.

(8) Position the tank unit in the fuel cell, install and torque the retaining screws to 25 ± 5 inch-pounds. Safety the screws.

(9) Connect the electrical lead to the tank unit.

(10) Working through the filler neck opening, connect the line at the aft side of the cell. Torque the nipple clamp on all cells equipped with type WWD clamps.

(11) Insert the tube containing the interconnecting wiring for the tank units through the filler neck opening. Strap the outboard end of the tube to the top of the fuel cell interconnect.

(12) Working through the inboard tank unit opening, strap the end of the tube containing the interconnecting wiring for the tank units to the top of the inboard fuel cell interconnect.

(13) Place the vent tube in the cell and insert it into the aft nipple fitting at outboard end of the cell.

(14) Position the filler neck and gasket on the fuel cell and install the retaining bolts. Safety the bolts.

(15) Connect the vent line to the filler neck.

(16) Install the inner fuel cell access cover.

(17) Apply Permatex No. 2 around the filler neck flange and install the access cover.

(18) Insert the outboard tank unit into it's opening until the electrical leads from the tube can be attached. Connect the leads to the tank unit and secure them with the nylon straps, then completely insert the tank unit into the fuel cell. Install and safety the tank unit retaining screws.

CAUTION: *Handle the tank unit carefully for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(19) Working through the access opening on the lower surface of the wing, connect the vent line and reinstall the clamp securing the nipple cap in place at a point just forward of the vent line. Torque the clamps on all cells equipped with type WWD clamps.

(20) Insert the inboard tank unit into its opening until the electrical leads from the interconnecting tube to the terminals on the cover can be attached. Connect the leads to the tank unit and secure them with the nylon strap, then completely insert the tank unit into the fuel cell. Install and safety the tank unit retaining screws.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tube will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(21) Install all removed access plates.

4. WING OUTBOARD FUEL CELL

NOTE: All inline fuel filters and engine driven pump filters should be inspected and cleaned every 100 hours as noted in the titled paragraphs following. Additional inspection and cleaning should be accomplished for the following conditions:

A. Removal

The outboard wing fuel cell is the outboard cell located aft of the outboard wing panel main spar.

(1) Make sure all electrical power to the airplane is disconnected.

(2) Drain the wing fuel cells.

(3) Remove the two oval access covers located on the lower wing skin, just aft of the spar.

(4) Disconnect the vent line at the outboard end of the cell.

(5) Remove the access cover located under the inboard access opening.

(6) Working through the access opening, remove the three nipple clamps (one forward - two inboard) and pull the nipples off the tubes.

(7) Unsnap the fasteners that secure the fuel cell to the upper wing skin.

(8) Collapse the cell and remove it through the access opening in the lower wing skin.

B. Installation

NOTE: Expose the fuel cell to temperatures of at least 60°F for a period sufficient to ensure flexibility during installation.

On airplane serials U-151 and After, new interconnects have a bead added so that fuel cells aft of the main spar can be installed without reopening the leading edge cavity.

(1) Check the fuel cell cavity for foreign objects or debris. Collapse the cell and insert it through the wing access opening.

(2) Secure the snap fasteners to the top of the wing.

(3) Insert the 3 tubes into the tube nipples. Torque the clamps on the cells equipped with type WWD clamps (Ref. Table 201).

(4) Install the cover on the bottom of the fuel cell. Safety the bolts.

(5) Connect the vent line at the outboard end of the cell. Torque the clamps on all cells equipped with type WWD clamps.

(6) Install all removed access covers.

5. WING INBOARD FUEL CELL

A. Removal

The inboard wing fuel cell is the inboard cell located aft of the outboard wing panel main spar.

(1) Make sure all electrical power is disconnected from the airplane.

(2) Drain the wing fuel cells.

(3) Remove the fuel tank unit access cover.

(4) Unsafety and remove the tank unit retaining screws. Lift the tank unit up to disconnect the electrical leads and the lead strap, then remove the tank unit.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(5) Remove the oval access cover on the bottom of the wing and forward of the outboard flap.

(6) Remove the fuel cell cover and disconnect the tank unit leads.

(7) Working through the fuel cell access opening, remove the nipple clamps at the outboard, inboard, and forward sides of the cell. Pull the nipples free of the tubes.

(8) Unsnap the fasteners that hold the fuel cell to the upper wing skin.

(9) Collapse the cell and remove it through the access opening.

B. Installation

NOTE: Expose the fuel cell to temperatures of at least 60°F for a period sufficient to ensure flexibility during installation.

On airplane serials U-151 and After, new interconnects have a bead added so that fuel cells aft of the main spar can be installed without reopening the leading edge cavity.

(1) Check the fuel cell cavity for foreign objects or debris. Collapse the cell and insert it through the lower access opening.

(2) Snap the fasteners that hold the cell to the upper wing skin.

(3) Working through the cell access opening, insert all tubes into the nipple fittings. Torque the clamps on all cells equipped with type WWD clamps (Ref. Table 201).

(4) Connect the tank unit leads and install the fuel cell cover on the bottom of the cell. Safety the retaining bolts.

(5) Insert the tank unit into its opening until the two electrical leads can be attached. Connect the leads to the tank unit and secure them with the nylon strap, then completely insert the tank unit into the fuel cell. Install and safety wire the tank unit retaining screws.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(6) Install all removed access covers.

6. CENTER SECTION FUEL CELL (IF INSTALLED)

A. Removal

The center section fuel cells are located in the wing center section, just inboard of the nacelles.

(1) Make sure all electrical power to the airplane is disconnected.

(2) Drain the wing cells.

(3) Remove the fuel tank unit access cover located adjacent to the fuselage.

(4) Remove the tank unit retaining screws and lift the tank unit up to disconnect the electrical leads and lead strap, then remove the tank unit.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(5) Remove the round access plate located outboard of the tank unit access cover.

(6) Remove the fuel cell cover and disconnect the tank unit leads.

(7) Working through the cell opening, remove the cotter pin, nut, and bolt securing the valve to the end of the cell gravity feed line at the aft outboard side of the cell. Remove the valve from the line.

(8) Remove the nipple clamps from the gravity feed line and vent line. Pull the nipples off the lines.

(9) Working in the wheel well, remove the WWD clamp from the line connecting the transfer pump to the center section fuel cell.

(10) Remove from the underside of the center section, the small round access cover over the fuel strainer and valve assembly. Remove the large rectangular access cover outboard of the fuel strainer cover.

(11) Unsafety and remove the retaining bolts for the strainer and drain valve assembly.

(12) Unsnap the fasteners securing the inboard end of the fuel cell to the upper wing skin.

(13) With one arm thrust through the strainer assembly opening, reach through the rectangular access opening and work the cell outboard until the nipple clamp at the inboard side of the fuel cell can be reached with the other hand. Remove the clamp and work the fuel line and check valve out through the strainer assembly opening.

(14) Unsnap the fasteners securing the cell to the upper wing skin.

(15) Collapse the fuel cell and remove it through the rectangular access opening in the underside of the center section.

B. Installation

NOTE: Expose the fuel cell to temperatures of at least 60°F for a period sufficient to ensure flexibility during installation.

(1) Check fuel cell cavity for foreign objects or debris. Collapse the cell and install it in the wing.

(2) Snap the fasteners that secure the outboard end of the fuel cell to the upper wing skin.

(3) Insert the fuel line and check valve into the fuel cell through the fuel strainer opening.

(4) Work the end of the fuel line through the fuel cell nipple until the jumper clamps are adjacent to the end of the nipple. Install and torque the nipple clamps on all cells equipped with type WWD clamps (Ref. Table 201).

(5) Reaching through the large rectangular access opening, work the fuel cell inboard until the end of the fuel cell tube can be guided through the grommet in the wheel well wall with the arm thrust through the fuel strainer opening.

(6) Working in the wheel well, install the WWD clamp on the line connecting the fuel cell line to the transfer pump.

(7) Snap the fasteners that secure the outboard end of the fuel cell to the upper wing skin.

(8) Secure the fuel strainer and drain valve assembly in place with the retaining bolts and washers. Torque the bolts to 50 ± 5 inch-pounds and safety them.

(9) Working through the upper access opening, insert the cross vent tube and gravity feed tube in the cell nipples on inboard side of the fuel cell. Install and torque the nipple clamps on all cells equipped with type WWD clamps.

(10) Position the check valve on the cell gravity feed tube and install the bolt and nut. Torque the nut 15 to 20 inch-pounds. Install a cotter pin in the bolt and nut.

(11) Connect the tank unit leads to the cover and position the gasket and cover on the fuel cell. Install and torque the retaining bolts 45 to 50 inch-pounds, then safety the bolts.

(12) Install the access covers on the underside of the center section.

(13) Insert the tank unit into its opening until the electrical leads can be attached. Connect the leads to the tank unit and secure them with a nylon strap, then completely insert the tank unit into the fuel cell. Install and safety wire the tank unit retaining screws.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector into the fuel cell.*

(14) Install the access covers on top of the wing center section.

C. Fuel Cell Storage

CAUTION: *Never store fuel cells in the vicinity of electrical equipment, such as generators and motors. The movement of brushes across the commutators of these units results in minute sparks that cause the formation of ozone gas. Ozone has a highly destructive effect on fuel cells. Although it leaves no visible indication, ozone makes the material of which the cells are formed brittle so that the cells will disintegrate upon the application of stress.*

NOTE: Preservation is not necessary for in service fuel cells when the cell is to be empty for an indefinite period. The fuel cell should not be open to the atmosphere except for the normal vent lines in the airplane.

If AVGAS was in the fuel cell prior to emptying, purge the cell with aviation kerosene prior to leaving empty for the indefinite period.

For information pertaining to the storage of Uniroyal fuel cells, refer to Report FC 1473-73 entitled Recommended Handling and Storage Procedures for Bladder Type Fuel and Oil Cells, which is published by Uniroyal Inc., Engineered Systems Department, Mishawaka, Indiana.

Goodyear fuel cells are constructed of polyethylene: they need not be treated for storage. The only stipulation on storage of Goodyear cells is that they be carefully wrapped for protection against dust, then be stored at room temperatures in a location shielded from sunlight.

D. Fuel Cells Inspection

(1) Examine the lower wing surface and around vent and drain nipple areas for fuel stains.

(2) If any evidence of fuel seepage is found, remove the fuel cell access plates on the lower wing surface and inspect the areas between the fuel cell liners and the wing skins for fuel seepage from the fuel cells.

(3) Inspect the fuel cells for deterioration and possible cracks or seepage, particularly around vent, drain outlet and interconnect nipples.

(4) Interconnect nipples that are not hard or cracked should be checked by scraping with a finger nail. If material can be scraped off, the cell should be replaced.

CAUTION: *Bladder type fuel cells on airplanes which have been in service for extended periods may have become hard and brittle. These fuel cells should be checked very carefully. The inspection at this time could cause cracks in the material which in turn could cause fuel seepage.*

(5) If seepage from a fuel cell is confirmed, the age and general condition of the cell should be considered.

Fuel cells that have become hardened and are not pliable should be replaced. Repair is not recommended since the handling of such cells is likely to result in cracking of the hardened cell material. Refer to the applicable Parts Catalog for the correct replacement part numbers.

Fuel cells that are pliable and are not hardened may be considered repairable in accordance with the applicable Shop/Maintenance Manuals.

NOTE: Goodyear BTC-39 construction type fuel cells removed from the airplane for any reason MUST BE REPLACED and MUST NOT BE REPAIRED or REINSTALLED in the airplane. Specific inspection instructions for BTC-39 construction fuel cells are contained in Hawker Beechcraft Service Instructions No. 0895, or subsequent revision.

(6) Fuel cells that have been repaired should be checked for condition of the repair by draining and purging, removing the access plates and inspecting with a mirror and non-explosive flashlight. If the repair shows any indication of blistering or any edge is loose, the cell should be replaced.

7. MINOR REPAIR OF FUEL CELLS

A. Uniroyal Fuel Cells

No damage area larger than two inches may be repaired. Cells with damage on the radius or in the fitting area must be returned to the manufacturer for repairs. Lesser damage may be repaired. Refer to Chapter 28 of the MODEL 99 AIRLINER SERIES COMPONENT MAINTENANCE MANUAL (P/N 99-590015-5).

B. Goodyear Fuel Cells

Punctures, slits up to 3 inches in length, abraded holes, and loose hangers or glove snaps may be repaired in the field. Any damage more extensive in nature must be returned to the manufacturer for repair. Refer to Chapter 28 of the MODEL 99 AIRLINER SERIES COMPONENT MAINTENANCE MANUAL (P/N 99-590015-5) for repair procedures.

C. Testing for Fuel Cell Leaks

Refer to Chapter 28 of the MODEL 99 AIRLINER SERIES COMPONENT MAINTENANCE MANUAL (P/N 99-590015-5) for the manufacturers recommended test procedures.

FUEL
DISTRIBUTION
MAINTENANCE PRACTICES

1. GENERAL

A. 99, 99A and B99

The fuel system consists of two separate systems connected by a crossfeed system. Each wing system has two filler caps, one in the nacelle tank and one in the wing leading edge tank. Fuel flows by gravity from the leading edge wing tank (through the inter-connects) to the outboard and inboard wing tanks to the center section tank and then to the nacelle tank.

A crossfeed line connects the two nacelle tanks. A switch controlled crossfeed valve is connected into the line at the aft inboard corner of the LH nacelle tank. When the valve is in its normally closed position, each engine draws fuel from its respective fuel tank system. When the manually operated crossfeed control switch mounted on the pedestal just below the fuel quantity gages is actuated, power is drawn from a 10 ampere circuit breaker on the RH panel beside the copilot to the solenoid of the crossfeed valve. The crossfeed valve then opens to allow either or both engines to draw fuel from the entire fuel system.

A jet transfer pump mounted on the inboard side of the main landing gear wheel well transfers fuel from the wing fuel sump (center section tank) to the nacelle tank. The jet pump receives its motive flow from the nacelle boost pump through a shutoff valve, filter, and check valve mounted on the inboard side of the wheel well immediately below the jet pump. The purpose of the shutoff valve is to permit filter removal without draining the fuel cells. The check valve prevents the engine from sucking air when the boost pump is not operating. An internal bypass valve in the filter permits the continued flow of the fuel should the filter become plugged or blocked with ice. The jet pump draws fuel from the center section tank and transfers it to the nacelle tank. Should the jet transfer pump fail, all but 28 gallons of fuel will flow by gravity into the nacelle tank. Since the jet transfer pump depends on boost pump pressure for its motive flow, the jet transfer pump will cease to operate any time both the primary and secondary boost pumps fail. As long as either of the boost pumps operate and there is fuel in the wing tanks, the jet transfer pump will keep the nacelle full.

Fuel to the engine is supplied directly from the nacelle tank. The gravity feed line from the wing tanks is connected to a flanged fitting on the outboard side of the nacelle tank. A check valve in the line aft of the nacelle fitting allows fuel flow only into the nacelle tank. The motive line from the nacelle tank boost pumps to the jet transfer pump and the return line from the transfer pump are connected to the aft side of the nacelle tank. The overflow line that returns to the wing gravity feed line the excess fuel delivered by the jet transfer pump is connected into the nacelle tank adjacent to the connections for the transfer pump lines. A check valve at the filler neck with a cracking pressure of one inch of water prevents the fuel cell from collapsing of a partial vacuum when the overflow fuel line becomes empty as fuel is depleted. Fuel to the heater is drawn by suction from the bottom of the LH nacelle tank through a fitting on the inboard side. The supply line connecting the engine to the internal plumbing from the submerged boost pumps is attached to the forward inboard side of the nacelle fuel tank.

The primary and secondary boost pumps are installed in the bottom of the nacelle fuel tank just forward of the cutout for the landing gear. The line plumbed into each boost pump connects to the line that attaches to the fittings for the external lines leading to the jet transfer pump and engine. Either the primary boost pump or its backup unit, the secondary boost pump. is capable of supplying fuel to the engine driven fuel pump under the minimum pressure required by the engine manufacturer.

CAUTION: *Should the boost pumps fail. suction lift operation may be employed, however. suction lift operation is restricted to 10 hours total time between engine overhaul periods. If the pump is operated on suction lift beyond the 10 hour limit, overhaul or replacement of the pump is necessary.*

A check valve in the line leading from each boost pump prevents fuel from backflowing through the inactive pump. Should both boost pumps on one side of the fuel system fail, fuel under pressure can be obtained from the boost pumps and fuel cells on the opposite side by opening the crossfeed valve. The primary and secondary boost pumps are each actuated by a 10 ampere circuit breaker located directly below the master switch on the left subpanel. Fuel pressure from the boost pump provides the motive flow by which the jet transfer pump functions. Fuel under boost pump pressure is routed through a check valve and out the inboard side of the nacelle tank to the engine. The fuel is then routed forward to a normally open firewall shutoff valve installed in the fuel line immediately behind the engine firewall. The firewall shutoff valve for each engine fuel system is actuated by its respective FIRE PULL handle on the upper center of the instrument panel on either side of the annunciator panel. When the FIRE PULL handle is pulled, its respective firewall shutoff valve closes to shut off the flow of fuel to the engine.

B. A99A

The fuel system may be considered the same as that installed on the 99, 99A and B99, except that the center section fuel tank, the jet transfer system, pump and plumbing have been removed. The nacelle tank float switch has been removed from both nacelle fuel tanks. The nacelle tank filler cap has been removed. The system is serviced at a single point in the outboard end of the leading edge tank of each wing.

C. A99A

Two filler openings are provided for fuel servicing of the C99 Airliners. One opening is located at the high point of the nacelle tank and another opening is located at the high point of the wing leading edge tank. Both openings are closed by tab locking filler caps. To assure that both tanks are properly serviced with fuel and to preclude the possibility of air pockets being trapped, the nacelle tank should be serviced first.

The fuel in the wing leading edge tank and the fuel in the outboard tank flows by gravity into the inboard wing tank through interconnecting plumbing lines. A gravity feed line from the inboard tank low point connects, through a tee in the gravity feed line, to the fuel inlet of the wing center section tank. An anti-slosh valve, installed at the wing center section tank inlet, prevents fuel from flowing back into the inboard tank during various flight attitudes. The third outlet of the tee in the gravity feed line, carries fuel, through a check valve to the nacelle tank. This check valve in the nacelle gravity feed line prevents fuel from flowing from the nacelle tank back into the inboard wing tank and into the wing center section tank; thus, all five fuel cells are interconnected and share a common gravity feed system.

A fuel transfer interconnect, between the wing center section tank and the nacelle tank, permits fuel flow from the center section tank into the nacelle tank. A portion of the fuel in the center section tank will not gravity feed into the nacelle tank; therefore, a jet transfer pump installed in the fuel transfer interconnect line pulls the fuel from the center section tank and transfers it to the nacelle tank by way of the suction produced when a motive flow passes through a venturi in the jet pump.

Motive flow for the jet pump is provided by the engine driven primary boost pump. The motive flow is controlled by a solenoid valve installed in the motive flow line. Energizing current for the motive flow valve is taken from the standby pump circuit breaker through the crossfeed switch. During crossfeed operations, the motive flow valve on the side receiving fuel will be closed. This action is necessary in order to prevent pressurizing the nacelle tank during crossfeeding.

Should the motive flow to the jet pump be interrupted for any reason, a normally closed switch in the motive flow line immediately downstream of the motive flow valve will illuminate a warning light on the annunciator panel alerting the pilot to the no-fuel-transfer condition.

A crossfeed line between the two wing systems makes it possible to supply fuel to either engine from the opposite wing system. Selecting either crossfeed switch position automatically closes the opposite wing system electric boost pump relay which simultaneously energizes the boost pump and the crossfeed relay. Power to the boost pump control switches is supplied through the normally closed contacts of the crossfeed switch; therefore, manual control of the boost pumps can only be accomplished when the crossfeed switch is in the OFF position.

FUEL
DISTRIBUTION
MAINTENANCE PRACTICES

1. FUEL TRANSFER PUMP

A jet transfer pump is installed in the fuel line on the inboard side of each main landing gear wheel well.

A. Removal

(1) Drain the fuel cells.

(2) Disconnect the motive flow, inlet, and outlet lines from the pump at the inboard side of the wheel well.

B. Installation

(1) Connect the inlet port of the pump to the line from the fuel cell in the wing center section.

(2) Connect the line from the motive flow line to the motive flow port (large end) of the pump.

(3) Connect the remaining line to the outlet port of the pump.

2. BOOST PUMPS

The boost pumps are installed side by side in the bottom of each nacelle tank just forward of the cutout for the main landing gear wheel well. The primary boost pump is outboard and secondary boost pump is inboard. U-50, U-165 and After made use of only one electric boost pump. The primary boost pump on these airplanes is engine-driven.

NOTE: Airplanes U-136 and After have flex lines for the pump outlet and do not require the bayonet type connector.

When a boost pump is determined inoperative (Ref. Fuel Boost Pump Test Procedure) and is to be replaced, the remaining pump may be utilized to drain the applicable nacelle and wing fuel cells.

This may be accomplished by disconnecting the flexible fuel heater line inside the cowl on the engine accessory case and pumping the fuel into a large, suitable container.

3. PRIMARY BOOST PUMP (PRIOR TO U-165, EXCEPT U-50)

A. Removal

(1) Make sure all electrical power to the airplane is disconnected.

(2) Drain the nacelle and wing fuel cells.

(3) Open the access door below the pumps.

(4) On airplanes prior to U-80, disconnect the line at the drain manifold and at the elbow in the wheel well then remove the drain manifold from its mounting bracket.

(5) Disconnect the electrical lead to the inboard pump.

(6) Cut the safety wire from around the retaining bolts and remove the bolts.

(7) Pull the pump free at the quick-disconnect fitting to the pump outlet line, and remove the pump from the cell.

B. Installation

(1) Insert two headless 1/4 UNF bolts, 1-1/2 inches long, into the pump mounting ring in the cell to aid in the installation of the pump.

(2) Position one gasket between the cell and the tank liner and one gasket between the tank liner and the pump.

(3) Engage the male and female fittings of the pump and outlet line.

(4) Position the pump in the mounting opening with the large plug aft, and work the cell over the pump. Install and torque the retaining bolts to 25 ± 5 inch-pounds. Safety the bolts.

NOTE: Prior to inserting the boost pump in place, lubricate the packing on the tube extending from the pump with JP-4 fuel.

(5) Connect the electrical leads to the pump.

(6) On airplanes prior to U-80, reattach the drain manifold to its mounting bracket, and install the line between the manifold and the elbow in the wheel well. Close the access cover.

4. SECONDARY BOOST PUMP

A. Removal

(1) Make sure all electrical power to the airplane is disconnected.

(2) Drain the nacelle and wing fuel cells.

(3) Open the access door below the pumps.

(4) Disconnect the line at the drain manifold, then remove the drain manifold from its mounting bracket.

(5) Disconnect the electrical lead to the pump.

(6) Cut the safety wire from around the retaining bolts and remove the bolts.

(7) Pull the pump free at the quick-disconnect fitting to the pump outlet line, and remove the pump from the cell.

B. Installation

(1) Screw two headless 1/4 UNF bolts, 1-1/2 inches long, into the pump mounting ring to aid in installation of the pump.

(2) Position one gasket between the cell and the tank liner and one gasket between the tank liner and the pump.

(3) Engage the male and female fittings of the pump and outlet line.

(4) Position the pump in the mounting opening with the large plug inboard, and work the cell over the pump. Install and torque the retaining bolts to 24 ± 5 inch-pounds. Safety the bolts.

NOTE: Prior to inserting the boost pump in place, lubricate the packing on the tube extending from the pump with JP-4 fuel.

(5) Connect the electrical leads to the pumps.

(6) Reattach the drain manifold to its mounting bracket, and install the line to the drain manifold. Close the access cover.

5. ENGINE-DRIVEN FUEL BOOST PUMP (U-50, U-165 AND AFTER)

A. Removal

(1) Disconnect the inlet, outlet and drain lines from the boost pump.

(2) Remove the four bolts which secure the pump to the vacuum pump pad of the engine accessory case, and remove the pump from the airplane.

B. Installation

(1) Apply lubricant (111, Table 6, 91-00-00) sparingly to the drive spline of the boost pump.

(2) Engage the pump drive spline with the pump positioned on the vacuum pump pad of the engine accessory case with the seal drain port down.

(3) Install and torque the four mounting bolts 65 to 85 inch-pounds.

(4) Reconnect the inlet, outlet and drain lines to the ports of the pump housing.

6. FUEL BOOST PUMP

The following test procedures may be used to isolate a fuel boost pump problem.

Fuel Boost Pump Test Procedure

SYMPTON	TEST PROCEDURE
1. Pump circuit breaker trips.	a. Turn off pump. b. Mark the pump lead wires and airplane wires to show the connection pairing c. Disconnect the pump leads from the airplane wiring. d. With the airplane wire ends isolated from each other and the airframe, turn the pump switch to the ON position with power supplied to the pump circuits. If the circuit breaker still trips, the airplane wiring is defective.

Fuel Boost Pump Test Procedure (Continued)

SYMPTON	TEST PROCEDURE
1. Pump circuit breaker trips. (Continued)	e. Turn off pump. Using ohmmeter continuity checks, determine which pump lead is grounded to the pump case and which airplane lead is grounded to the airframe. If these two wires are not previously connected, the pump connection was reversed. If this is the case, reverse the connections and recheck the pump operation. f. If the problem is not resolved by the above checks, the pump is defective.
2. Pump will not run - circuit breaker does not trip.	a. Check power source and airplane switch(s) to ensure that power should be available to the pump. b. Turn off the pump. Mark the pump lead wires and airplane wires to show connection pairing. c. Disconnect pump leads from airplane wiring. d. With appropriate switch(s) in the PUMP OPERATING position, check the voltage from both airplane wires to airframe. One of the airplane wires should have bus voltage. If bus voltage is not present on one of the wires, the defect is in the airplane wiring system. e. If bus voltage is present on one of the wires, check continuity of the other airplane wire to ground with an ohmmeter. f. If the continuity check shows connection to airframe ground, temporarily reconnect the pump leads as before. g. Recheck pump operation. If the pump does not run, it is defective. h. If the pump runs, reconnect the leads in a permanent installation configuration and again test the pump.
3. Insufficient pressure (fuel pressure warning light is on) but pump runs. (It is assumed that only one of the boost pumps will not shut off the pressure warning light.) If none of the pumps will shut off the light, the pumps are probably at fault.	a. With jet pump (99 99A and B99 only) inoperative (turn shut off valve off), visually inspect for turbulence (ripples) in the nacelle fuel tank that would indicate fuel leaks from the pump or plumbing into the tank. Press down on the pump discharge fuel lines while monitoring the pressure warning light. A leaking fitting is indicated by changes in pump pressure.

Fuel Boost Pump Test Procedure (Continued)

SYMPTON	TEST PROCEDURE
3. Insufficient pressure (fuel pressure warning light is on) but pump runs. (It is assumed that only one of the boost pumps will not shut off the pressure warning light.) If none of the pumps will shut off the light, the pumps are probably at fault. (Continued)	b. If no leakage can be detected in the nacelle tank, turn shut off valve on and perform check procedures 2. b. through 3. c., making sure the bus voltage measured at the pump connection is not abnormally low compared to other properly functioning pumps in the airplane. c. If no defect is found in the above checks, the pump is defective.

FUEL
FUEL SYSTEM VALVES
MAINTENANCE PRACTICES

1. FIREWALL SHUT-OFF VALVE

Electrically operated, gate-type shut-off valves are mounted behind the firewall on the inboard side of each nacelle. Relief valves are incorporated in the valves to relieve thermal expansion downstream of the valve.

CAUTION: *The fuel shut-off valves are electrically operated only and should not be operated manually. The lever on the side of the valve is an indicator only, not a control.*

A. Removal

(1) Make sure all electrical power to the airplane is off.

(2) Drain the fuel system to a level below the shut-off valve.

(3) Remove the access door on the inboard side of the nacelle just forward of the wing leading edge.

(4) Disconnect the electrical lead and fuel lines from the valve.

(5) Disconnect the inlet and outlet lines from the valve.

(6) Remove the safety wire and four mounting bolts and remove the valve from the mounting bracket.

B. Installation

(1) Position the valve in its mounting bracket, then install the mounting bolts and safety wire.

(2) Connect the inlet and outlet lines.

(3) Connect the electrical leads and fuel lines to the valve.

(4) Replace the access cover.

2. FUEL CROSSFEED VALVE

The line-mounted crossfeed valve allows crossfeed of fuel from either system to either or both engines when circumstances warrant. The crossfeed valve is located on the inboard side of the left nacelle just aft of the leading edge. Access is through the top and bottom wing panels adjacent to the nacelle and aft of the leading edge.

A. Removal

(1) Make sure all electrical power to the airplane is off.

(2) Drain the fuel system to a level below the crossfeed valve.

(3) Disconnect the electrical lead and fuel lines from the valve.

(4) Remove the safety wire and mounting bolts and remove the valve from the mounting brackets.

B. Installation

(1) Place the valve in its mounting bracket, then install the mounting bolts and safety wire.

(2) Connect the inlet and outlet lines to the valve.

(3) Connect the electrical lead to the valve.

3. FUEL CONTROL PURGE VALVE

To gain access to the fuel control purge valve, remove the access cover on the outboard side of the RH nacelle just forward of the wing leading edge or the access cover on the inboard side of the LH nacelle just forward of the center section leading edge.

A. Removal

(1) Make sure all electrical power to the airplane is off.

(2) Remove the cover providing access to the valve from the nacelle.

(3) Disconnect the electrical lead and the fuel return lines from the valve.

(4) Remove the retaining bolts and remove the valve from its mounting bracket.

B. Installation

(1) Position the valve on its mounting bracket and install the retaining bolts.

(2) Connect the electrical lead and fuel return lines to the valve.

(3) Install the access cover.

4. FUEL PRESSURE SWITCHES

A fuel system pressure switch is mounted on the RH side of each firewall and is accessible through the upper aft engine cowling.

A. Removal

(1) Make sure all electrical power is off.

(2) Drain fuel in main cell to a level below the bottom of the pressure switch.

(3) Disconnect the electrical leads and fuel line from the switch.

(4) Remove the mounting bolts and remove the switch from the firewall.

B. Installation

(1) Position the switch on the firewall and install the mounting bolts.

(2) Connect the fuel line to the switch.

(3) Connect the electrical lead to the switch.

FUEL
FUEL HEATERS
MAINTENANCE PRACTICES

1. ENGINE OIL/FUEL HEATER

A. Check

In the event of excessive engine oil consumption, make the following pressure check:

(1) Without removing the oil/fuel heater from the engine, cap off the oil outlet and fuel inlet, then attach a length of rubber hose to the fuel outlet.

(2) Apply filtered shop air or dry nitrogen (100 psig maximum) to the oil inlet and place the end of the rubber hose attached to the fuel outlet into a container of water.

(3) Bubbles in the container of water upon the application of pressure indicates leakage and the need for replacement of the oil/fuel heater.

2. ENGINE FUEL CONTROL LINE HEATER

A heating element is wrapped around the engine air pressure sense line immediately before entering the engine fuel control unit. The line between the fuel control unit and primary governor is similarly heated. Each heating element is controlled by an individual circuit breaker switch located on the circuit breaker panel located on the sidewall to the copilot's right.

FUEL
INDICATING
DESCRIPTION AND OPERATION

1. GENERAL

A. Capacitance Fuel Quantity Indicating System

The fuel quantity indicating system is a capacitance type that is compensated for specific gravity and reads in pounds on a linear scale. An electronic circuit in the system processes the signals from the fuel quantity (capacitance) probes in the various fuel cells for an accurate readout by the fuel quantity indicators. A selector switch located between the fuel quantity indicators in the fuel panel beside the pilot may be set in either the TOTAL or NACELLE positions to determine whether the gages indicate the pounds of fuel in the nacelle and wing fuel cells of the fuel system or the pounds of fuel in the nacelle fuel cell only. Each side of the airplane has an independent gaging system consisting of a fuel quantity (capacitance) probe in the nacelle fuel cell, one in the aft inboard fuel cell, two in the leading edge fuel cell, and one in the center section fuel cell.

Fuel density and electrical dielectric constant vary with respect to temperature, fuel type, and fuel batch. The capacitance gaging system is designed to sense and compensate for these variables. The fuel quantity probe is simply a variable capacitor comprised of two concentric tubes. The inner tube is profiled by changing the diameter as a function of height so that the capacitance between the inner and out tube is proportional to the tank volume. The tubes serve as fixed electrodes and the fuel of the tank in the space between the tubes acts as the dielectric of the fuel quantity probe. The capacitance of the fuel quantity probe varies with respect to the change in the dielectric that results from the ratio of fuel to air in the fuel cell. As the fuel level between the inner and outer tubes rises, air with a dielectric constant of one is replaced by fuel with a dielectric constant of approximately two; thus variation in the volume of fuel contained in the fuel cell produces a capacitance of the fuel quantity probe. This variation in the volume of fuel contained in the fuel cell produces a capacitance variation that is a linear function of that volume. This function is converted to a linear current that actuates the fuel quantity indicator. Capacitance fuel probes are designed to produce a capacitance variation that is linear in relation to variation in weight, even through weight is nonlinear with respect to the fuel level. By varying the capacitance per inch, the nonlinear level signal is changed to a linear function. The fuel gage houses a major portion of the system, including the full and empty adjustment screws.

FUEL
INDICATING
MAINTENANCE PRACTICES

1. NACELLE TANK UNIT

A. Removal

(1) Make sure all electrical power to the aircraft is disconnected.

(2) Remove the access cover over the fuel filler neck on top of the nacelle.

(3) Unsafety and remove the bolts securing the tank unit in place. Lift the tank unit up to disconnect the electrical leads and the lead strap, then remove the tank unit.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

B. Installation

(1) Insert the tank unit into its opening until the electrical leads can be attached. Connect the leads to the tank unit and secure them with the nylon strap, then completely insert the unit into the fuel cell.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(2) Install and torque the tank unit retaining bolts to 25 $\pm$ 5 inch-pounds, then safety the bolts.

(3) Install the access cover.

2. LEADING EDGE FUEL CELL TANK UNITS (INBOARD AND OUTBOARD)

A. Removal

(1) Remove the access cover over the tank unit.

(2) Remove the tank unit retaining screws and lift the tank unit up to disconnect the electrical leads and lead strap, then remove the tank unit.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

B. Installation

(1) Insert the tank unit into its opening until the electrical leads can be attached. Connect the leads to the tank unit and secure them with the nylon strap, then completely insert the tank unit into the fuel cell.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(2) Install and safety wire the retaining screws securing the tank unit in place. Install the access cover.

3. WING INBOARD AND CENTER SECTION FUEL CELLS TANK UNIT (IF INSTALLED)

A. Removal

(1) Remove the access cover to the tank unit.

(2) Remove the tank unit retaining screws and lift the tank unit up to disconnect the electrical leads and the lead strap, then remove the tank unit.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

B. Installation

(1) Insert the tank unit into its opening until the electrical leads can be attached. Connect the leads to the tank unit and secure them with the nylon strap, then completely insert the tank unit into the fuel cell.

CAUTION: *Handle the tank unit carefully, for damage to the surface of the probe tubes will destroy the accuracy of the tank unit as a measuring device. Take precautions against dropping the connector insulator into the fuel cell.*

(2) Install and safety wire the retaining screws securing the tank unit in place. Install the access cover.

4. NACELLE TANK FLOAT SWITCH (99, 99A, B99 AND C99)

A. Removal

(1) Remove all electrical power from the airplane.

(2) Remove the nacelle fuel tank access panel.

(3) Disconnect the electrical connectors from the float switch and the capacitance probe.

(4) Disconnect the plumbing as necessary and remove the capacitance probe from the tank, being careful to avoid damage to the tube portion of the probe.

(5) Remove the attaching bolts and bond straps from the fuel tank cover and remove the tank cover.

(6) Remove the float switch from the tank cover.

B. Installation

(1) Install the float switch in the fuel tank cover fitting.

(2) Install the fuel tank cover, bond straps and attaching bolts.

NOTE: The bond straps must be installed at the bolt holes where the primer has been burnished away.

(3) Torque the attaching bolts 45 to 50 inch-pounds and safety wire.

(4) Install the capacitance probe by torquing the probe attaching bolts 45 to 50 inch-pounds and safety wire.

(5) Connect all plumbing which was disconnected during the removal procedure.

(6) Connect the electrical connectors for the probe and float switch.

(7) Install the nacelle fuel tank access panel.

5. FUEL QUANTITY GAGING SYSTEM

A. Calibration

For calibration of the Simmonds fuel quantity gaging system (99, A99 and B99) refer to 28-41-00. For calibration of the Gull system (C99) refer to 28-42-00.

FUEL
INDICATING (99, A99, B99)
DESCRIPTION AND OPERATION

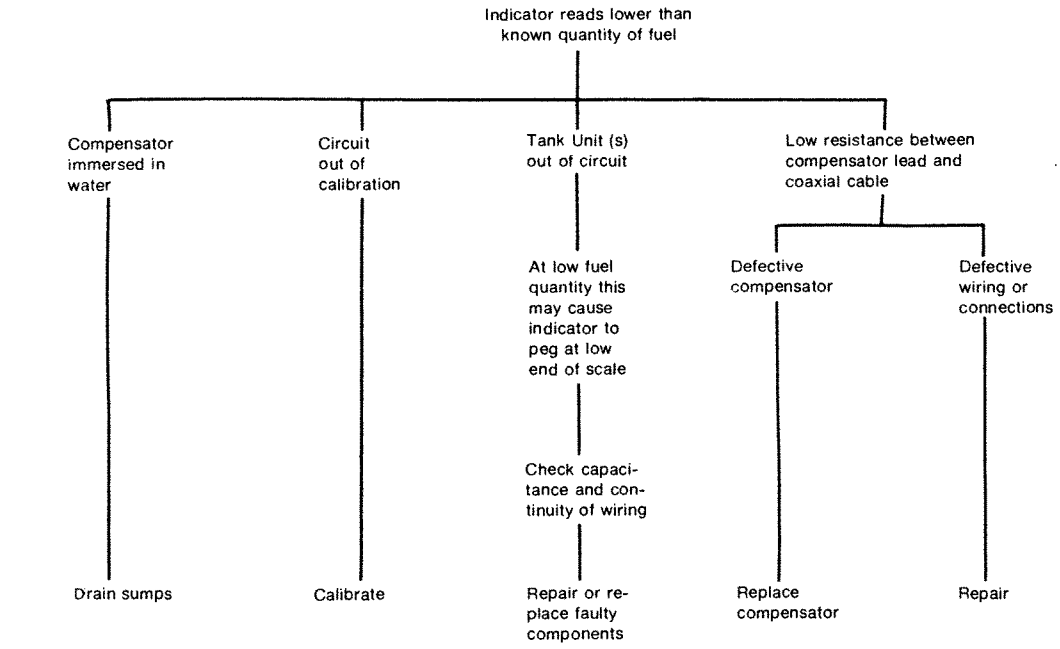
1. GENERAL

A. Simmonds Fuel Quantity Gaging System (Simmonds Installation Number 380787-001)

For the Simmonds Capacitance Fuel Gaging System to function properly, it is required that capacitance signal to the fuel gage be correct. The system operates on 115 VAC 400 Hz. Faults in the system will affect these capacitance values. Some common system wiring faults are High Z to ground shorts, Low Z to ground shorts, Low Z and High Z shorts, shielded wire shorts to ground. Low Z and High Z resistance values not within manufacturer's specifications also give erroneous readings. The High Z wire is shielded (per MIL-C-27500). This shield is continuous from wing leading edge outboard probe to pin "B" of the fuel gage. This shield is insulated from any other grounds by a transparent coating; shorts to ground or openings in this shielding will affect capacitance. All of these values must be maintained within the manufacturer's recommended specifications. For checking probe capacitance, fuel tanks should be drained. This will allow the operator to measure dry probe capacitance (Pf) and check against known values. Total wing probe capacitance can be measured at High Z and Low Z junctions. Nacelle probe capacitance can be measured by removing nacelle High Z wire from wing junction High Z and fuel age pins "C" (compensator) and "E" (Low Z). If these values do not fall within manufacturer's dry capacitance specs, the suspected probe should be isolated and the associated wiring checked for proper megohm resistance values in manufacturer's specifications. It should be mentioned that water or moisture in fuel wiring connectors or in fuel cells will affect system operation. All connectors should be kept clean and dry. Dow Corning #4 can be used on connectors to help displace moisture. Proper test equipment should be obtained to successfully troubleshoot the fuel quantity system. The Model 99 Airliner Series Component Manual contains the Simmonds Precision Technical Information Bulletin No. 408E which contains information regarding test equipment circuit values, troubleshooting, and test procedures. Instruction Manual 2548G Supplement No. 6 of the Barfield Instrument Corporation also contains information and procedures for testing and calibration of the system.

FUEL
INDICATING (99, A99, B99)
TROUBLESHOOTING

1. SIMMONDS FUEL GAGING SYSTEM



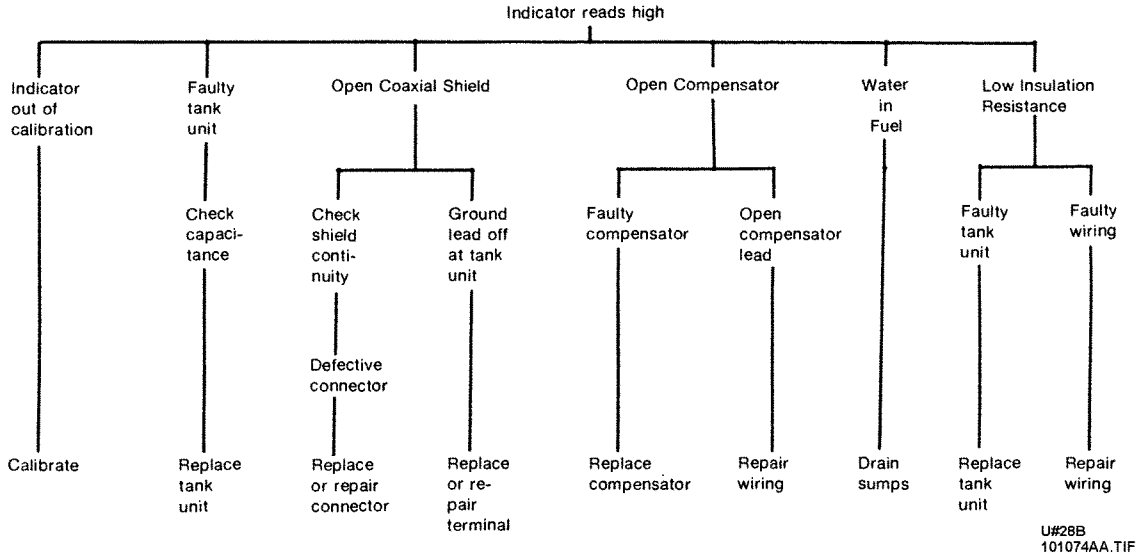


Figure 101 (Sheet 1 of 3)
Troubleshooting - Simmonds Fuel Gaging System

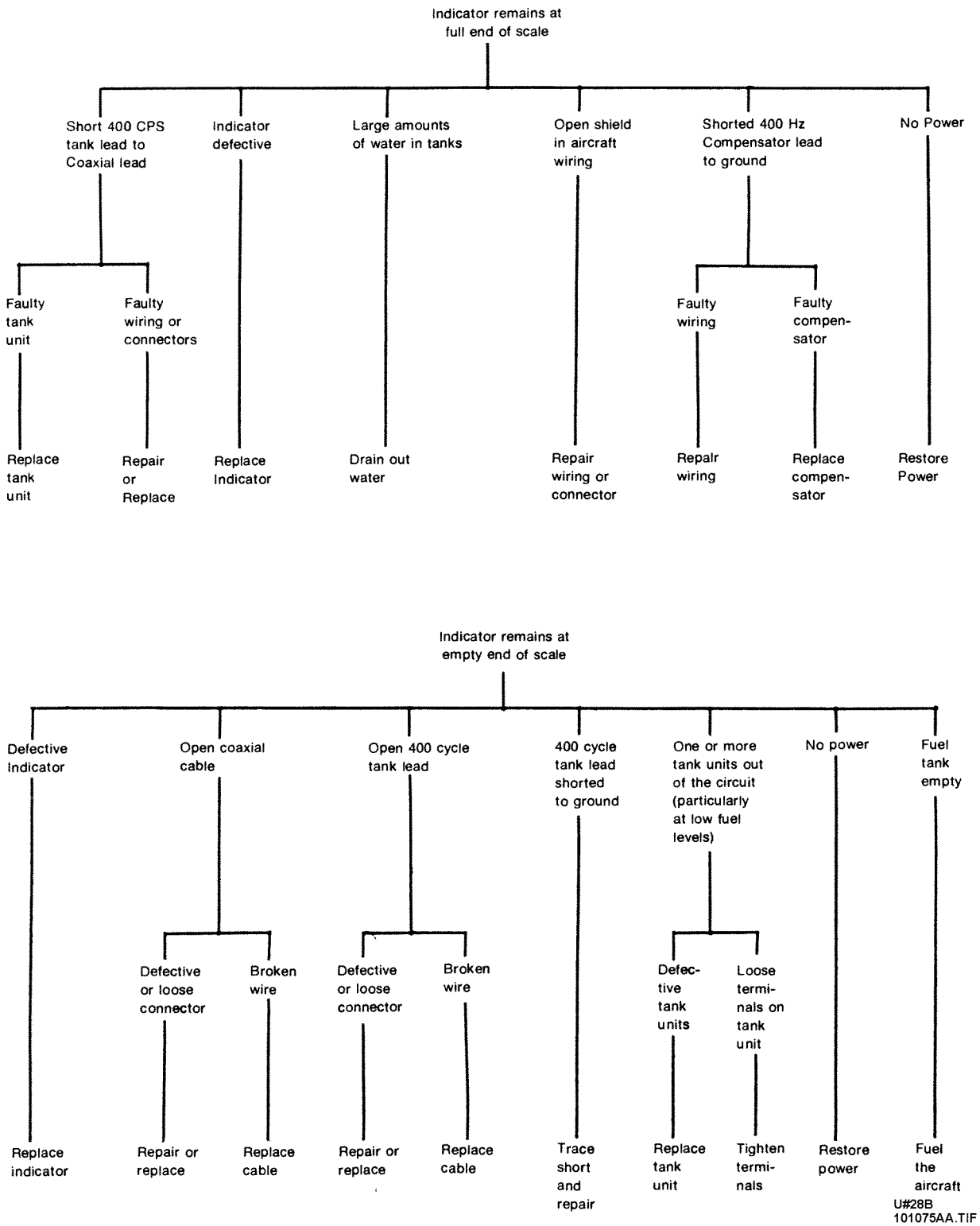


Figure 101 (Sheet 2 of 3)
Troubleshooting - Simmonds Fuel Gaging System

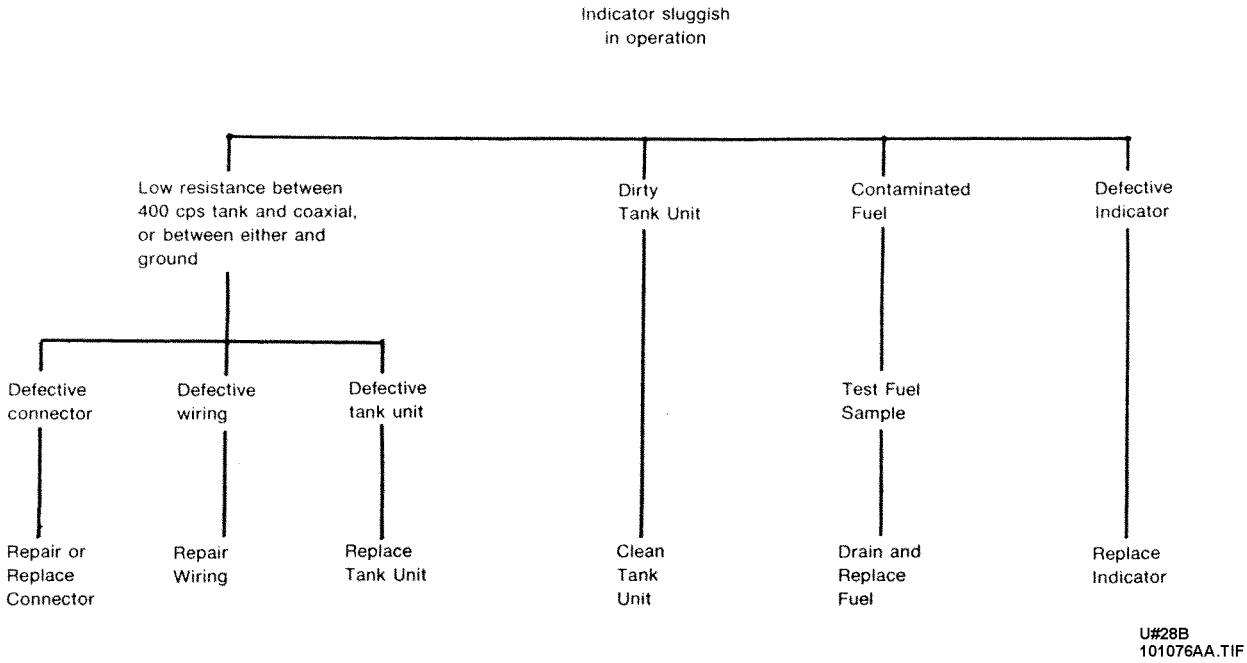


Figure 101 (Sheet 3 of 3)
Troubleshooting - Simmonds Fuel Gaging System

FUEL
INDICATING (99, A99, B99)
MAINTENANCE PRACTICES

1. SIMMONDS GAGING SYSTEM

A. Test

The tank units (probes) are primarily composed of 2 tubes, one inside the other. The outer tube (Low Z) has a load of 400 Hz. 115 volts placed on it. The inner tube is the high impedance (High Z) side of the probe and it is the signal from this High Z that provides the controlling signal to the fuel indicator. Both tubes are insulated from the airplane frame. The capacitance and resistance of some of the components are listed in Table 201. Since it is preferred that the following checks be made with the system drained (dry), let the airplane stand for 1 hour after draining the system so all the fuel will drain from the probes. Both the right and left gaging systems are the same. The following is for one side of the system:

NOTE: These tests are performed with the indicator disconnected. Use test equipment that has current limit circuits ONLY (less that 200 ma).

When measuring tank unit capacitance in the LH and RH cells, short the compensator Low Z (400 Hz) of that side to ground. When measuring capacitance of the compensator, short the tank units Low Z (400 Hz) to ground. This enables a true capacitance reading between the tank unit portion and the compensator.

Table 201
CAPACITANCE (DRY) AND INSULATION RESISTANCE VALUES

Probe Name	Capacitance (pf)	High Z Low Z (Megohms)	High Z Flange (Megohms)	Low Z Flange	T/U to Comp. Low (Megohms)
*Nacelle	32.6 ± 0.5	2500	5	5	10
**Compensator	22.5 ± 0.5	3600	1	1	
Leading Edge Outboard	19.8 ± 0.5	4200	5	5	
Leading Edge Inboard	27.5 ± 0.5	3000	5	5	
Wing Aft	25.4 ± 0.5	3200	5	5	
Center Section	31.0 ± 0.5	2700	5	5	

NOTE

The nacelle tank unit is composed of a sensing probe and compensator.

*Sensing probe's values.

**Compensator’s values.

B. System Capacitance

The LH Fuel Cells or RH Fuel Cells shall be 136.4 51.4 pf (Dry).

The compensator shall be 22.5 pf (Dry) or 46.5 pf (Wet).

C. System Insulation Resistance

High Z to tank units Low Z (400 Hz) - not less than 600 megohms (Dry) and 280 megohms (Wet).

Tank units Low Z (400Hz) to Ground - not less than 1 megohm (Dry or Wet).

High Z to Ground - not less than 1 megohm (Dry or Wet).

High Z to Compensator Low Z (400 Hz) - not less than 3600 megohms (Dry) and 1720 megohms (Wet).

Tank units Low Z to Compensator Low Z - not less than 10 megohms (Dry or Wet).

Coax shield to Ground - not less than 1 megohm.

2. INDICATOR

A. Check

Using a circuit as shown in Figure 201 and Simmonds Field Calibration Unit No. 387991-005 or equivalent, a complete test may be performed on the indicator to determine its status.

(1) Set the compensator decade to a value of 46.5 pf. Set the capacitance decade to an Empty value of 136.4 pf and to a Full value of 286.7 pf. Connect all three leads (Coax and 400 Hz leads) from the test cable to the tester 387991-005 and apply power to the indicator.

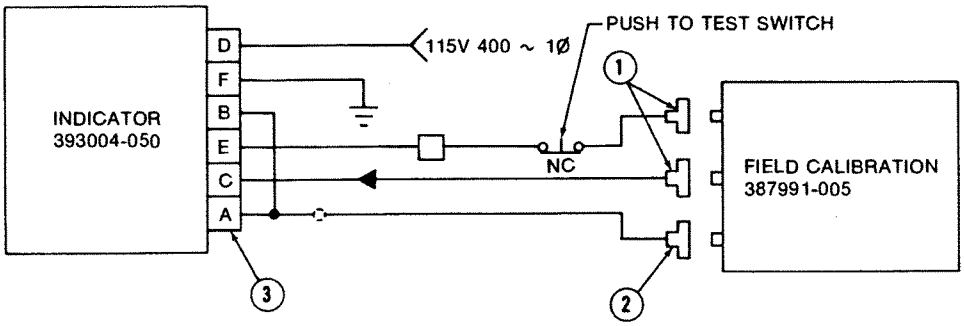
(2) Adjust the indicator for a "Zero" indication, then for a "Full" indication (1300 Ibs).

(3) Using Table 202, a scale check may be performed.

NOTE: When performing Empty and Full adjustments, press and release the "push to test" switch to check the pointer's return to that setting. This switch is also used as a test to see if the indicator is functioning. The switch, when pressed, places an open condition in the Low Z (400 HZ) line to the sensing probe.

Table 202
INDICATOR SCALE CHECK

DIAL SCALE (Fuel Quantity Indicator) (lb)	SCALE ERROR (Pf) Field Calibration Unit Setting
0	136.4
200	158.1 ± 0.7
400	175.9 ± 1.4
600	204.1 ± 1.4
800	235.4 ± 1.6
1000	252.7 ± 0.7
1200	273.8 ± 1.8
1300	286.7



ITEM NO.	PART NUMBER	DESCRIPTION
1	384004	Plug
2	384004	Plug
3	MS3106B-16S-1S	Plug

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Figure 201
Indicator Check

3. FUEL QUANTITY SYSTEM (99, 99A, B99)

A. Calibration

Calibrate the fuel indicators with the fuel system empty. The test equipment used for calibration simulates the total capacitance of the fuel system when full. Field Calibration Test Unit (P/N 99-590001-5) is required for calibration.

NOTE: Equally effective for calibration of the capacitance fuel system is P/N 290F-9 Test Unit and 290F-14 Cable Assembly, both of which may be purchased from Barfield Instrument Corporation, 1478 Central Avenue, Atlanta, GA 30344. It should also be noted that use of crimping tools Type F P/N 90045, Type F P/N 90123-4, and Thermashield P/N 59500 (P/N of AMP. Inc., Harrisburg, PA) will ensure the proper wire connections required for efficient operation of the fuel quantity indicating system.

(1) Disconnect the indicator at the receptacle and unclamp it from the pedestal.

(2) Remove the indicator and attach the test cable as shown in Figure 202, but do not connect the three "tees" (Items 5 and 6) to the Field Calibration Unit.

(3) With airplane power on, turn the RH (when looking at the back of the indicator) adjustment screw on the back of the indicator until the indicator reads “zero".

(4) Set the compensator decade on the Field Calibration Unit to a value of 24 picofarads and the capacitance decade to the value of 153.3 picofarads.

(5) Connect the three "tee" connectors (Items 5 and 6 of Figure 202) to the Field Calibration Unit.

(6) Turn the LH (when looking at the back of the indicator) adjustment screw on the back of the indicator until the indicator reads 1300 pounds.

(7) Disconnect the test cable and install the indicator in the pedestal with the retaining clamp.

(8) Connect the airplane system plug to the indicator receptacle.

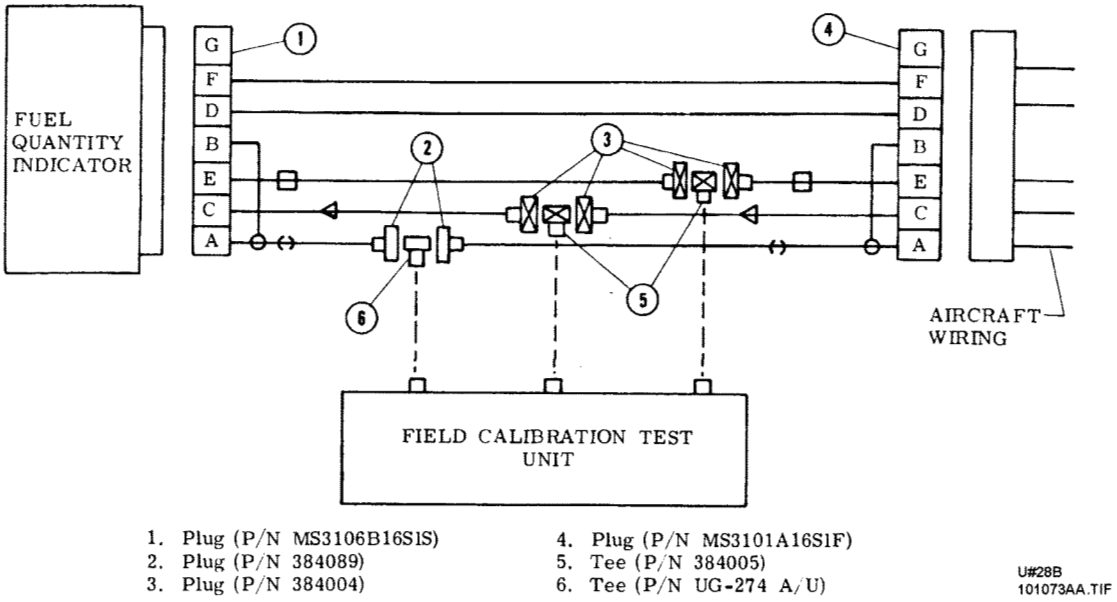


Figure 202
Fuel System Calibration (99, 99A and B99)

FUEL
INDICATING (C99)
DESCRIPTION AND OPERATION

1. GENERAL

A. Gull Fuel Quantity Gaging System

On the Gull system, each fuel quantity probe contains a small circuit network that produces an output current whose average value is directly proportional to fuel level, while automatically compensating for fuel temperature density variations. An AC waveform, whose characteristics are insulated by a regulator from line voltage variations in the incoming +28 VDC, is impressed across the Low Z side (outer tube) of the fuel quantity probe. The ensuing signal is further processed by a DC amplifier, which contains a potentiometer for adjusting the "Full" and "Empty" settings. The DC amplifier controls response time and drives a meter movement.

For detailed maintenance procedures, system troubleshooting, and calibration of the fuel quantity indicators, refer to the "Maintenance Handbook for the Gull Fuel Quantity Gaging System used on the Model 99 Airliner Series, in the Model 99 Air Liner Series Component Maintenance Manual. This manual may also be purchased separately as Publication Number 98-36541 from TMDC, Hawker Beechcraft Corporation, Wichita, KS 67201.

Equipment necessary for testing fuel quantity may also be obtained from the Barfield Instrument Corporation, 1478 Central Avenue, Atlanta, GA 30344, phone (404) 761-4321. Instructions for calibrating and troubleshooting the system are included with the equipment.

B. Fuel Quantity Capacitance Probes

Each capacitance probe consists of two concentric tubes. The tubes serve as fixed electrodes and the fuel of the tank in the space between the tubes acts as the dielectric of the probe. The capacitance of the probe varies with respect to the change in the dielectric that results from the ratio of fuel to air in the tank. The tubular capacitor elements of each capacitance probe are profiled to match the contour of the tank in which it is installed to provide linear capacitance versus volume in the tank at differing flight attitudes. The nacelle capacitance probe contains an integral compensator installed in the lowest point of the fuel cell so that it will always be covered by fuel. It is located thus to sense any variation in the dielectric constant of the fuel caused by temperature changes, variation in the different manufacturer's fuels, etc. The signal output from the compensator is used as a correction signal for the reference portion of measurement circuit to minimize errors in the indicator section.

FUEL
INDICATING (C99)
MAINTENANCE PRACTICES

1. FUEL QUANTITY SYSTEM (U-50, U- 165 AND AFTER)

A. Calibration

Equipment and instructions for accurately calibrating the capacitance fuel gaging system in the C99 Series Airliners can be obtained from Barfield Instrument Corporation, 1478 Central Avenue, Atlanta, GA 30344, or (for the Gull Airborne Test Equipment only) from Hawker Beechcraft Corporation, Commercial Product Support, Wichita, KS 67201.

2. BARFIELD

A. Calibration

The Barfield Instrument Corporation supplies the following equipment to be used in calibrating the fuel quantity indicating system in the C99: Test Box 26436, Tank Unit Simulator 2544-G, Indicator Harness 26436-2, Tank Unit Harness 36436-3 and Pin Adapter 26436-4.

B. Capacitance Check

This check is to be made with no fuel in the tanks.

(1) Connect the test harness to the test box and the airplane wiring as shown in Figure 201.

(2) Set the airplane fuel panel switch to TOTAL.

(3) Set the functions switch of the test box to 225.

(4) Press the TEST button on the test box; test box meter should read 165.5 $\pm$ 15.7 pf.

C. Indicator Calibration

(1) Connect the test harness to the test box and the airplane wiring as shown in Figure 201.

(2) Set the test box function switch to ACFT.

(3) Set the test box range switch to 225.

(4) Turn the airplane battery switch ON.

(5) Place the fuel control panel switch in the TOTAL position.

(6) The indicator needle should be approximately one-half needle width below the ZERO mark (if meter A, Figure 203 is installed), or one needle width below the ZERO mark (if meter B, in Figure 203 is installed). If the needle does not indicate as directed, it may be repositioned by adjusting the EMP 1 potentiometer on the back of the indicator.

(7) Hold the airplane fuel control panel switch in the NAC position.

(8) The indicator needle should be approximately one-half needle width below the ZERO mark (if meter 0, Figure 203 is installed). If the needle does not indicate as directed, it may be repositioned by adjusting the EMP 2 potentiometer on the back of the indicator.

(9) Return the airplane fuel control panel switch to the TOTAL position.

(10) Press the TEST button on the test unit and adjust the simulator to 91.6 pf.

(11) Set the test box function switch to ADD. The indicator should read 1300 lbs.; if not, adjust the FULL potentiometer on the back of the indicator.

(12) Recheck the two empty indications. All three indications must be correct.

NOTE: The above procedure may need to be repeated more than once in order to obtain consistently reproducible readings.

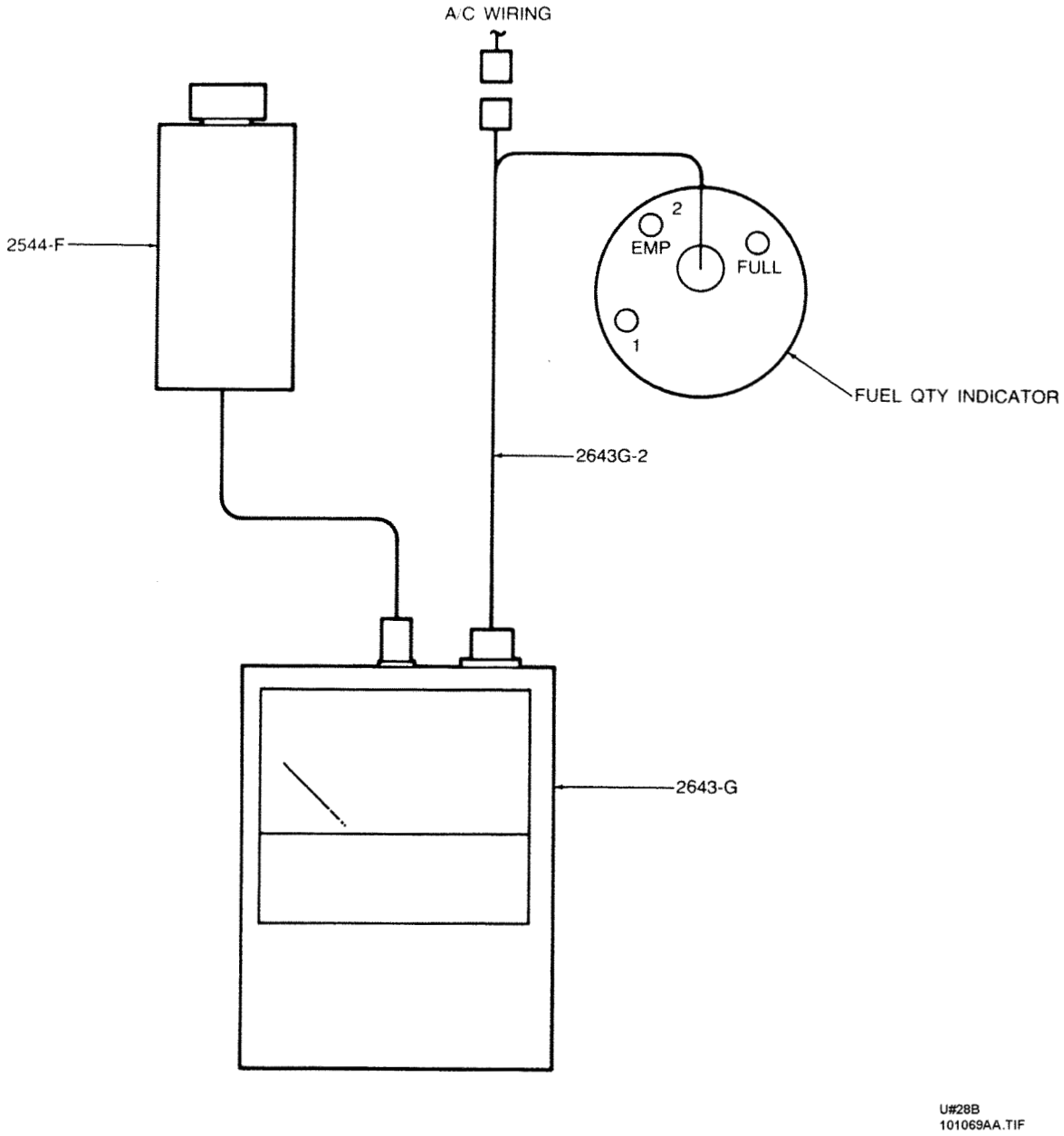


Figure 201
Barfield Test Equipment

3. GULL

A. Calibration

The Gull Airborne Instruments Corporation supplies the following equipment to be used in calibrating the fuel quantity indicating system in the C99; General Aviation Test Box 199-109-001 with adapter harness 081-014-001 and indicator adapter harness 081-013-001.

B. Capacitance Check

This check is to be made with no fuel in the tanks.

(1) Connect the test harness to the test box and the airplane wiring as shown in Figure 202.

(2) Set S1 on the test box to TANK UNITS.

(3) Switch test box power ON.

(4) Place the airplane fuel control panel switch in TOTAL position.

(5) The meter on the test box should read 0.647 ± 0.065.

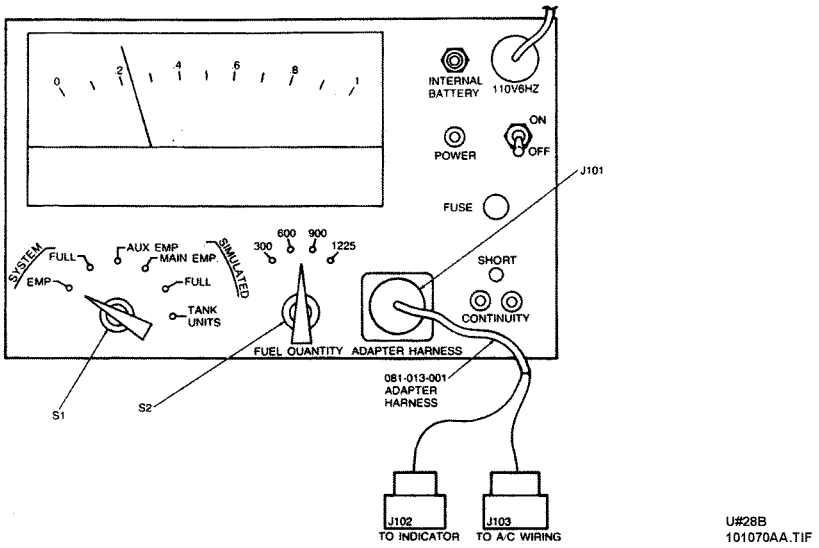


Figure 202
Gull Test Equipment

C. Indicator Calibration

(1) Connect the test harness to the test box and the airplane wiring as shown in Figure 202.

(2) Set S1 on the test box to SYSTEM EMPTY.

(3) Set S2 on the test box to 1225.

(4) Place the fuel control panel switch in the TOTAL position.

(5) Switch the test box power to ON.

(6) The indicator pointer should be approximately one-half needle width below the ZERO mark (if meter A, Figure 203 is installed), or one needle width below the ZERO mark (if meter B, Figure 203 is installed). If the needle does not indicate as directed, it may be repositioned by adjusting the EMP 1 potentiometer on the back of the indicator.

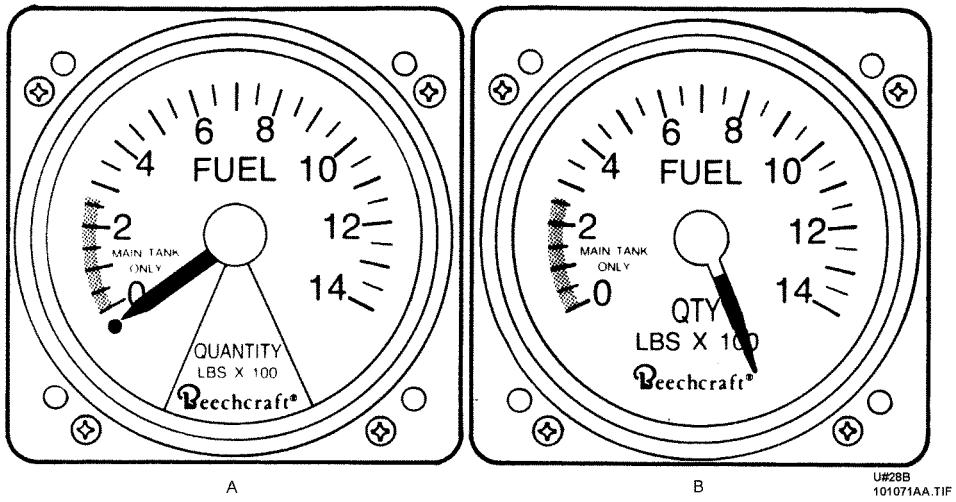


Figure 203
Fuel Quantity Indicators (U-50, U-165 and After)

Table 201

PROBE	CAPACITANCE	GULL METER
100-380006-139 or -143	27.65 ± 2.65	0.1100 ± 0.0110
100-380006-67 or -103	24.75 ± 2.35	0.0982 ± 0.0100
100-380006-73 or -109	32.15 ± 3.05	0.1281 ± 0.0130
100-380006-141 or -145	17.80 ± 1.70	0.0707 ± 0.0080
100-380006-77 or -113	58.90 ± 5.60	0.2344 ± 0.0250

(7) Hold the airplane fuel control panel switch in the NAC position.

(8) The indicator needle should be approximately one-half needle width below the ZERO mark (if meter A, Figure 203 is installed), or one needle width below the ZERO mark (if meter B, Figure 203 is installed. If the needle does not indicate as directed, it may be repositioned by adjusting the EMP 2 potentiometer on the back of the indicator.

(9) Return the airplane fuel control panel switch to the TOTAL position.

(10) Set S1 on the test box to SYSTEM FULL.

(11) The indicator pointer must be at 1225 lbs.; if not, adjust the FULL potentiometer on the back of the indicator.

(12) Recheck the two empty indications. All three indications must be correct.

NOTE: The above procedure may need to be repeated more than once in order to obtain consistently reproducible readings.

4. INDIVIDUAL PROBE CHECKS

Should the capacitance check of the installed system yield results which are unacceptable, the probe values in the accompanying table (Table 201) may be used to detect individual probes which may be out of tolerance. The capacitance values in picofarads are to be used with the Barfield test equipment while the Gull meter values (relative indications found on the Gull test equipment meter) are to be used with the Gull test equipment.

A. Megohm Test for Gulliragen Capacitance Fuel Quantity System

(1) The megohm test of the fuel quantity system will check the resistance between each of the wires going to the fuel probes, and from those wires to airframe ground. Fuel does not have to be drained from the tanks for this test.

This test should be conducted at any time that the tank system capacitance test cannot be satisfactorily completed, (Barfield Fuel Gaging System Test and Calibration Instruction Manual, 2643B-7904, Section 7, pages 15 and 16), or it is suspected that the fuel system is not functioning properly due to shorts or moisture contamination.

CAUTION: *Due to the possibility of damaging sensitive electronics inside each fuel probe or creating an arc inside the fuel cells, this test may only be accomplished by using approved test equipment that is voltage and current regulated. The Barfield fuel quantity system test set 2548 (G, GA, or H) or the Barfield DC-400, are approved test sets. NEVER TEST THE HARNESS WITH THE INDICATOR CONNECTED AS THE INDICATOR MAY BE DAMAGED.*

NOTE: If Barfield DC-400 Test Set is to be used, consult the DC-400 Instruction Manual.

This test will check the resistance between all of the wires that run to the wing of the airplane by checking from pins 1, 2, 5 and 6 of the fuel quantity system connector (airplane harness side).

The wire from pin 1 connects to the yellow wire on all probes. The wire from pin 2 goes to airframe ground. The wire from pin 5 connects to the green wire on all probes. The wire from pin 6 connects to the red wire on the probes of the system that is selected by the tank select switch: main or auxiliary, or total or nacelle.

(2) If a megohm adapter box is not available, set up the 2548 test unit per Steps (2) through (6). Before connecting any leads to the insulation Hi-Z and Lo-Z connections, perform Step (7). Attach leads to the insulation Hi-Z and Lo-Z connectors and use leads as an ohmmeter between pins 1, 2, 5 and 6 of the fuel quantity system connector to monitor the resistance of each possible combination of pins.

Figure 204 is a schematic of the megohm adapter box and Figure 205 is a schematic of the probe test harness.

If a megohm adapter box is available, conduct the megohm test as follows:

(a) Connect the megohm adapter box between airplane harness and the Barfield 2548 fuel quantity test box per Figure 206.

(b) Turn the master OFF-ON switch on the 2548 test box to the ON position.

(c) Place the TEST FUNCTION switch to the CAP BRIDGE position, press the BATTERY test switch and observe that the needle in the 2548 test unit is in the BAT OK range.

(d) Place the TEST FUNCTION switch in the insulation mode.

(e) Place the INS TEST POINTS switch in the Lo-Z/Hi-Z position.

(f) Select the OFF position on the megohm adapter box.

(g) Adjust the METER ADJUST knob for an infinity indication on the 2548 meter.

(h) Rotate the selector knob on the megohm adapter box through each if the nine positions and record all readings.

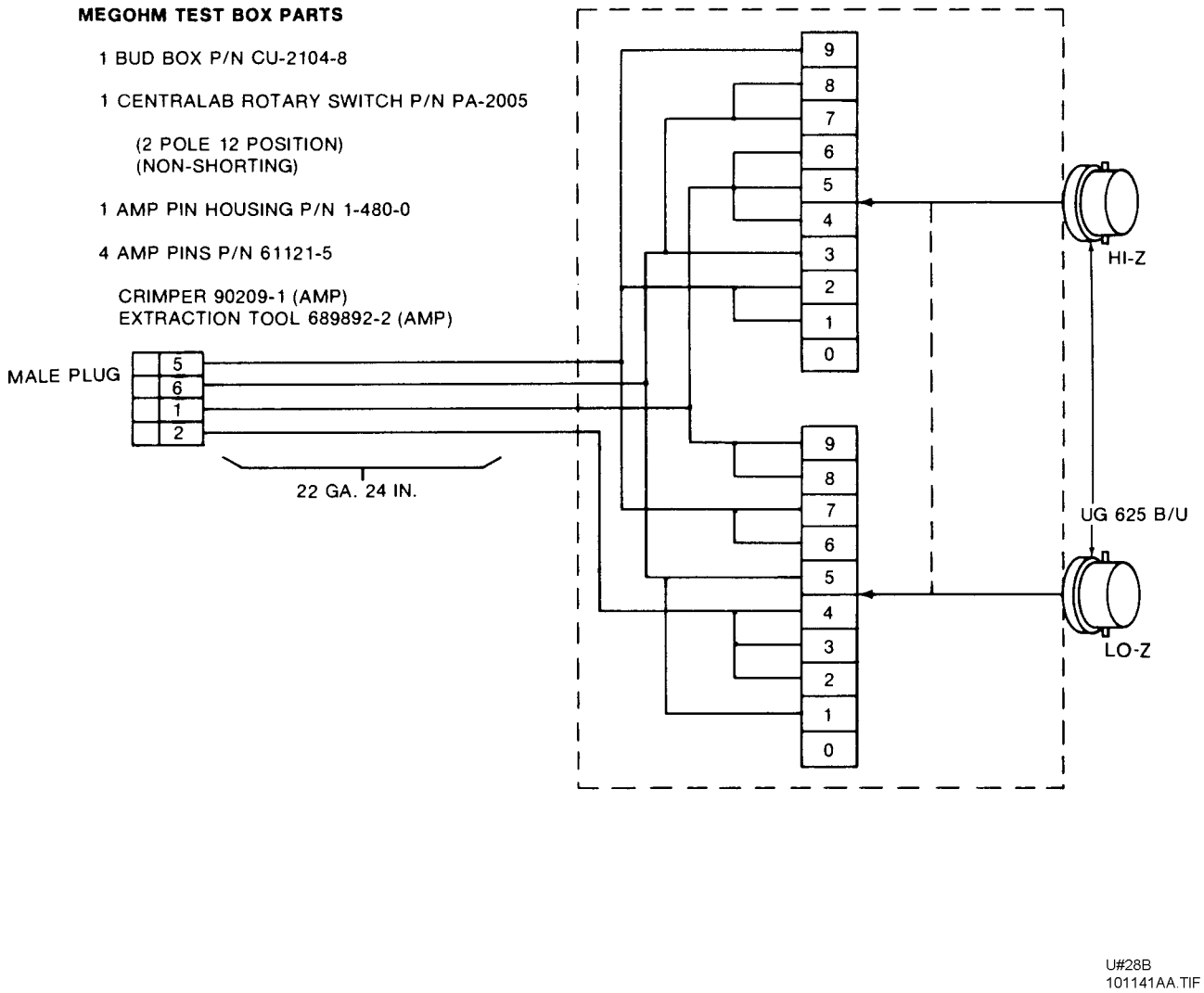
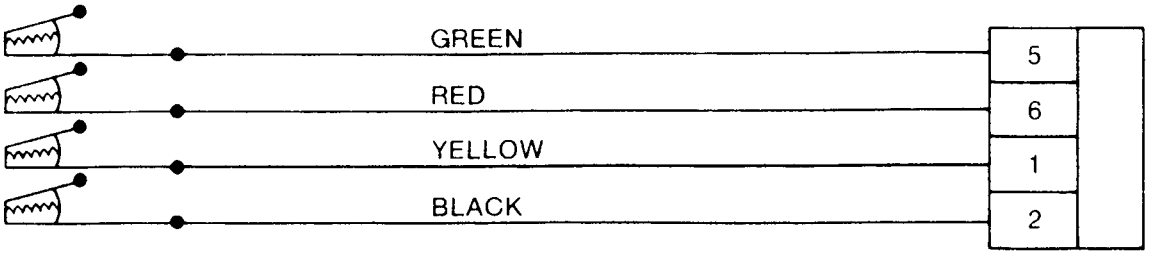


Figure 204
Megohmmeter Test Box



TEST HARNESS PARTS:

1 AMP Socket Housing P/N 1-480409-0
4 AMP Sockets P/N 61123-5
4 Alligator Clips

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Figure 205
Test Harness

1 If the resistance readings are normal (per Step (c)), proceed with Step (i).

2 If the resistance readings are less than normal, then corrective action (per Step (a)) must be taken before proceding with Step (i).

(i) Spray water on any one probe and its associated wiring under its cover plate.

(j) Check the resistance values again of all different combinations per Step (h). Any discrepancies must be corrected before continuing the test.

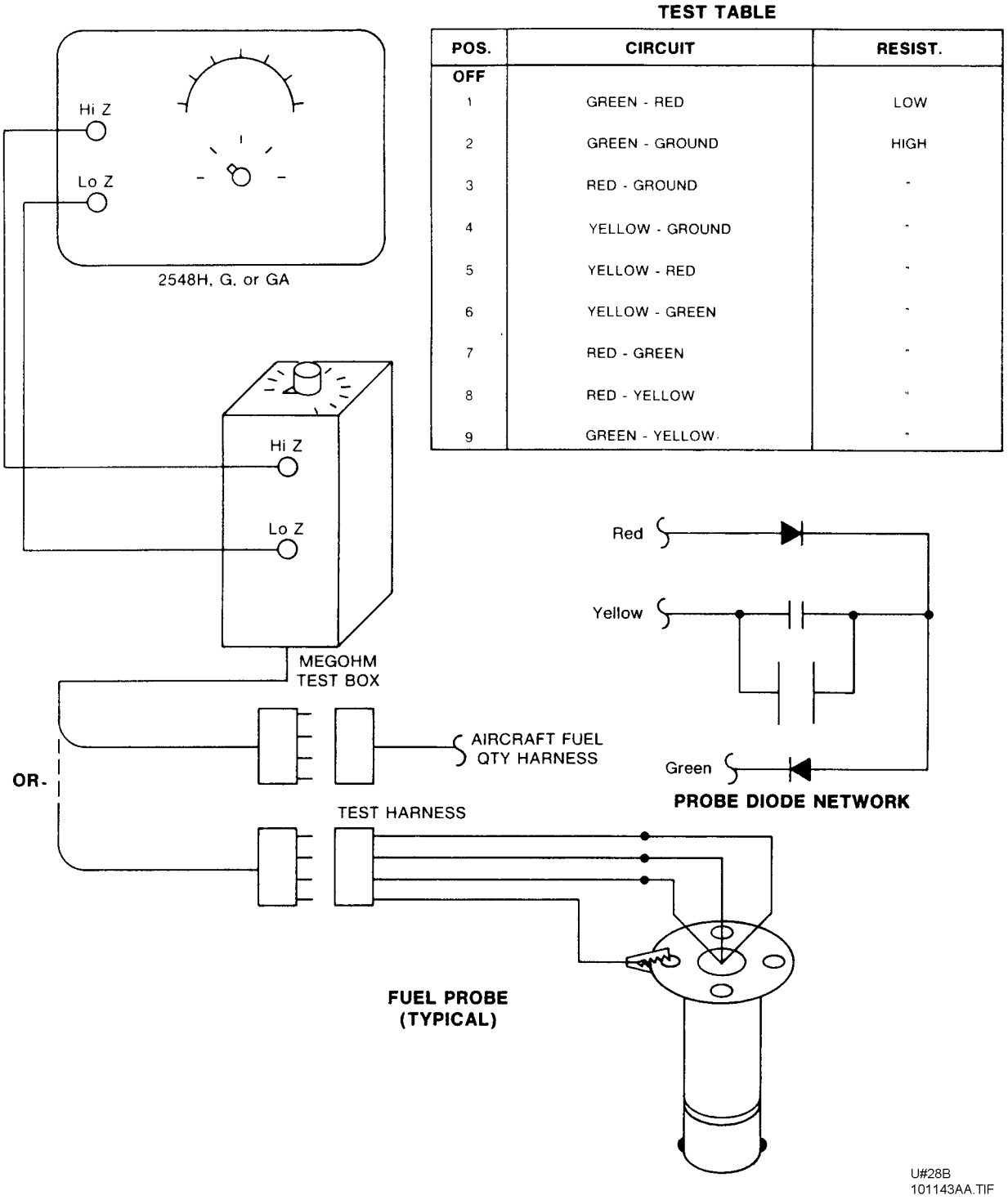
(k) Alternately perform Steps (i) and (j) until all probes and wiring have been checked with water.

(l) Water causes the resistance to decrease from wire to wire and wires to ground. Find the specific cause of resistance change by alternately applying shop air and water spray to a pin point. Cut the in wire or area where water affects the system. Use appropriate material EC1239, RTV, etc. to make an environmental seal.

Because the resistance of the fuel systems between the wires and from the wires to the airframe ground varies from airplane to airplane, there is not a single set of values that can be labeled "normal". There is a value though, that when the resistance drops below it, results in small errors in fuel quantity indication. When the resistance decreases from that value, the error to the system becomes progressively larger until the indication pegs either full or empty.

(3) An indication of 1 to 10 megohms is normal when selecting position one on the megohm adapter box. This position places the 2548 Hi-Z lead with pin 5 and the Lo-Z lead with pin 6 of the fuel quantity system harness, and checks the continuity across the diodes in the system. In all of the other positions, 2 through 9, the indication should be 20 megohms or higher.

Airplanes operated in dry climates may have fuel systems with resistance within the range of 1500 to 5000 megohms.



TEST TABLE

POS.	CIRCUIT	RESIST.
OFF		
1	GREEN - RED	LOW
2	GREEN - GROUND	HIGH
3	RED - GROUND	-
4	YELLOW - GROUND	-
5	YELLOW - RED	-
6	YELLOW - GREEN	-
7	RED - GREEN	-
8	RED - YELLOW	-
9	GREEN - YELLOW	-

Figure 206
Fuel Quantity Megohm Test

Airplanes operated in areas of the country with a high humidity may have resistance a lot lower and in the range of 40 to 500 megohms. Resistances in the range of 5 megohms appear to cause the fuel quantity indicator to display small increments of error. The lower the resistance from that point, the greater the error until the indicator pegs in whichever direction the error is driving the system. The fuel system consists of three wires. Any resistance lower than acceptable from wire to wire or from wire to ground will cause the fuel gage to indicate more or less fuel than is in the tanks.

Fuel systems with resistances slightly above 5 megohms will not display any error in the fuel quantity system, nor can any problems in this system be detected by conducting a system capacitance test. Resistances in this range indicate that there is some insulation breakdown. (Either washing the airplane or the airplane sitting or flying in rain may lower the system resistance below the level that causes indication errors). The source of this low resistance should be determined and corrected to prevent indication error.

The resistance values of the fuel systems are subject to change with any moisture contamination to the wires, connectors, or probes. Some causes for moisture contamination are as follows:

(a) At shield termination points, the insulation of the wires are sometimes cut, then covered up with solder sleeve or shrink teflon.

(b) Any nicks, cracks, or cuts in the wire normally outside of the pressure vessel.

(c) Any unused holes in the fuel probe Matrix connector that does not have a sealing plug in it.

(d) Wire ties or spiral wrap placed too close to a Matrix connector may place a strain on wires going into the connector and elongate the rubber seat at the entrance, allowing moisture to penetrate.

(e) Some fuel probes are susceptible to moisture contamination at the point where the wires enter the top of the probe.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

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ICE AND RAIN PROTECTION
SURFACE DEICER SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

The surface deicer system removes ice accumulation from the leading edges of the wings and the horizontal and vertical stabilizers. Ice removal is performed by alternately inflating and deflating boots. The boots are inflated by air pressure and deflated by vacuum suction. Air pressure and vacuum are obtained by bleeding air from the engine compressors. This bleed air is routed through a regulator valve that is set to maintain the pressure required to inflate the surface deicer boots. To assure operation of the system should one engine fail, a check valve is incorporated in the bleed line from each engine to prevent the escape of air pressure into the chamber of the inoperative compressor. The bleed air from the engine is also routed through an ejector that uses the venturi effect to produce vacuum. The vacuum is used to deflate the deicer boots. The inflation and deflation phases of operation are controlled by a distributor valve. The surface deicer system is actuated by a three position switch on the LH subpanel; the switch is spring loaded to return to the OFF position from either the MANUAL or SINGLE cycle position. When the SINGLE position is selected, the distributor valve opens to inflate the boots.

After an inflation period of approximately 7 seconds, a timer delay switches the distributor valve off to deflate the boots. When deflation is complete, the cycle is complete. When the MANUAL position is selected the boots will inflate and will remain inflated as long as the switch is held in position. When the switch is released, the distributor valve returns to the off position and the boots will deflate. The boots then will remain deflated until again actuated by the switch.

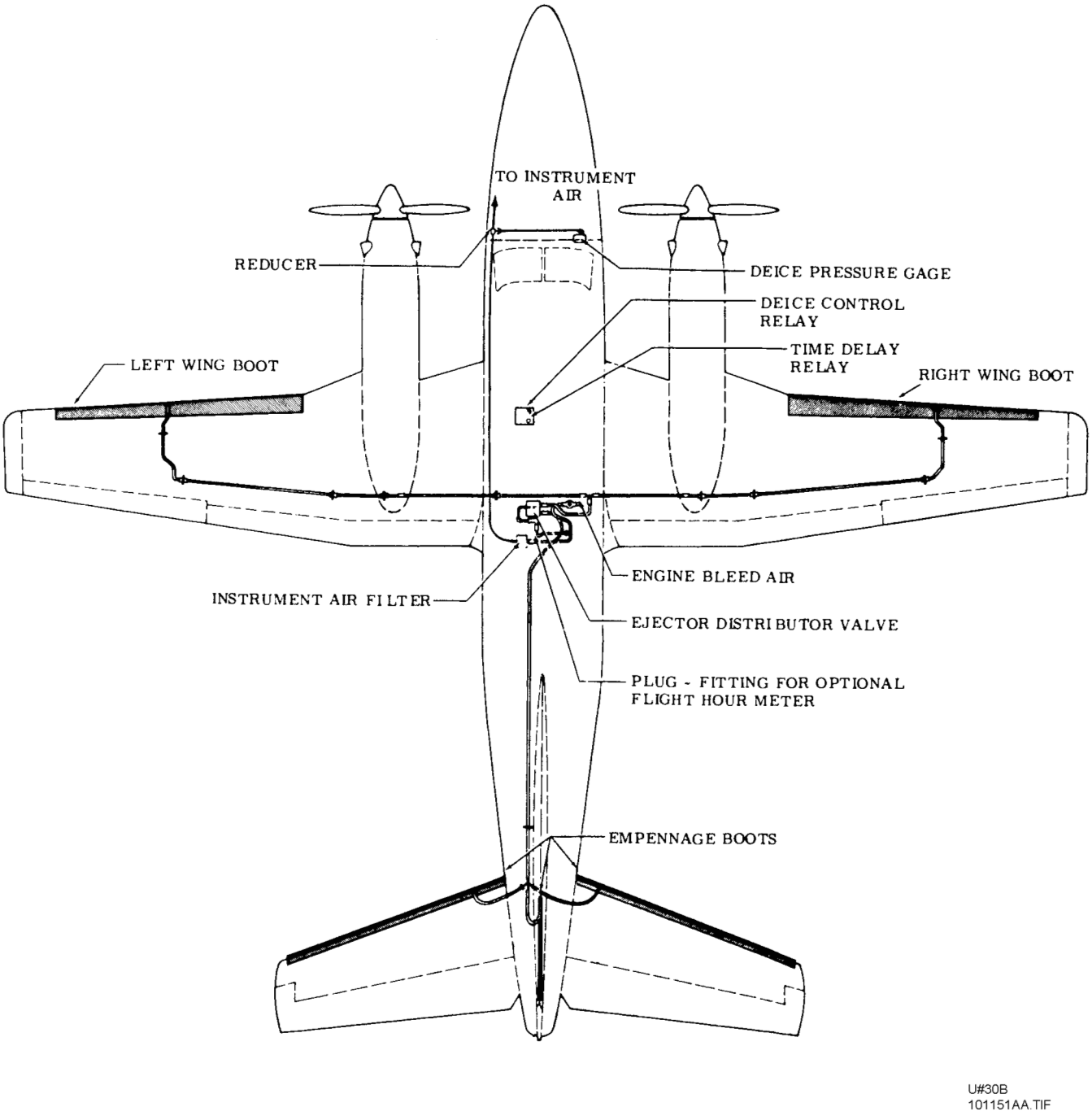

Figure 1
Surface Deice System (U-1 thru U-151, except U-50)

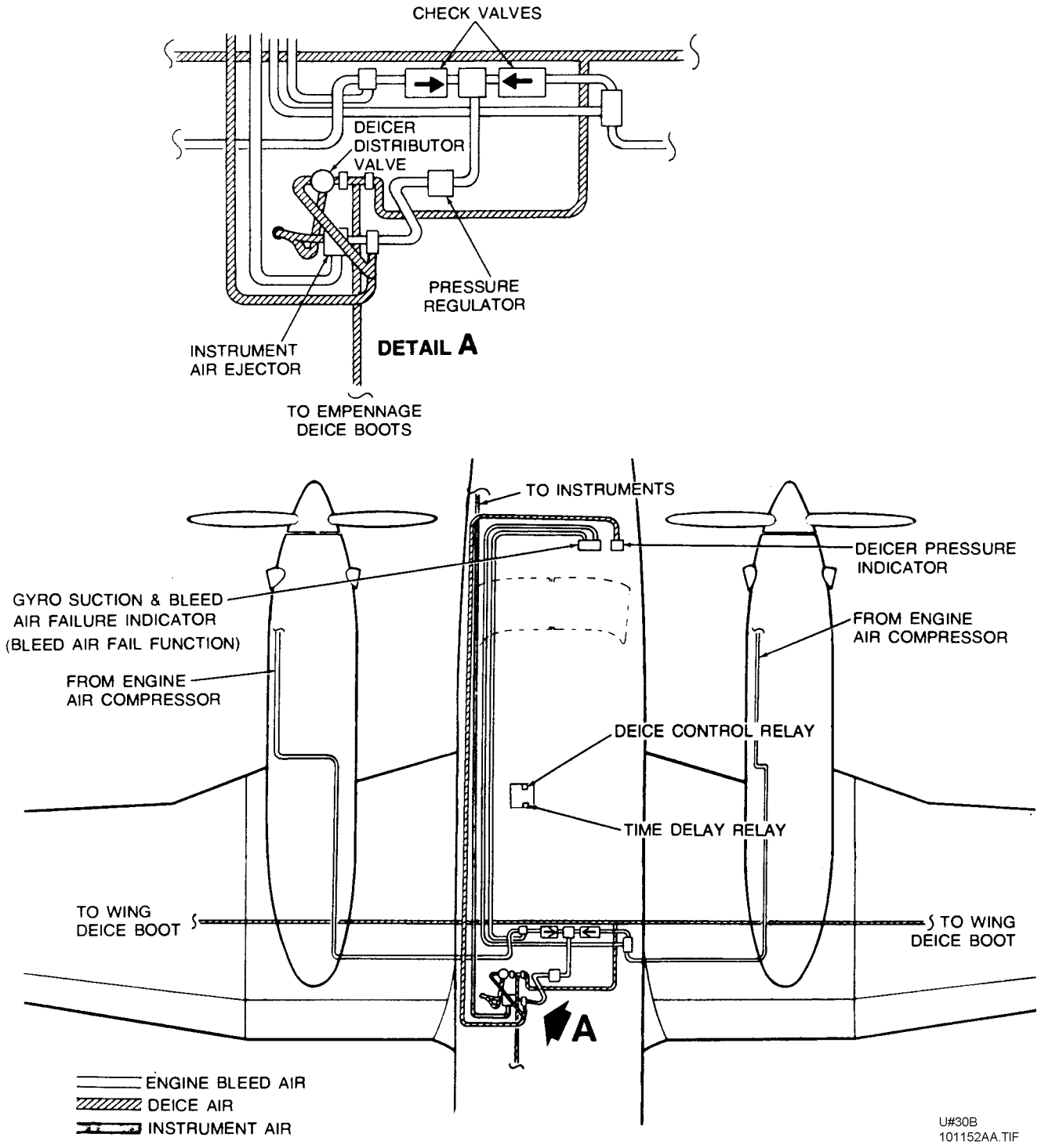

Figure 2
Surface Deice System (U-152 thru U-164)

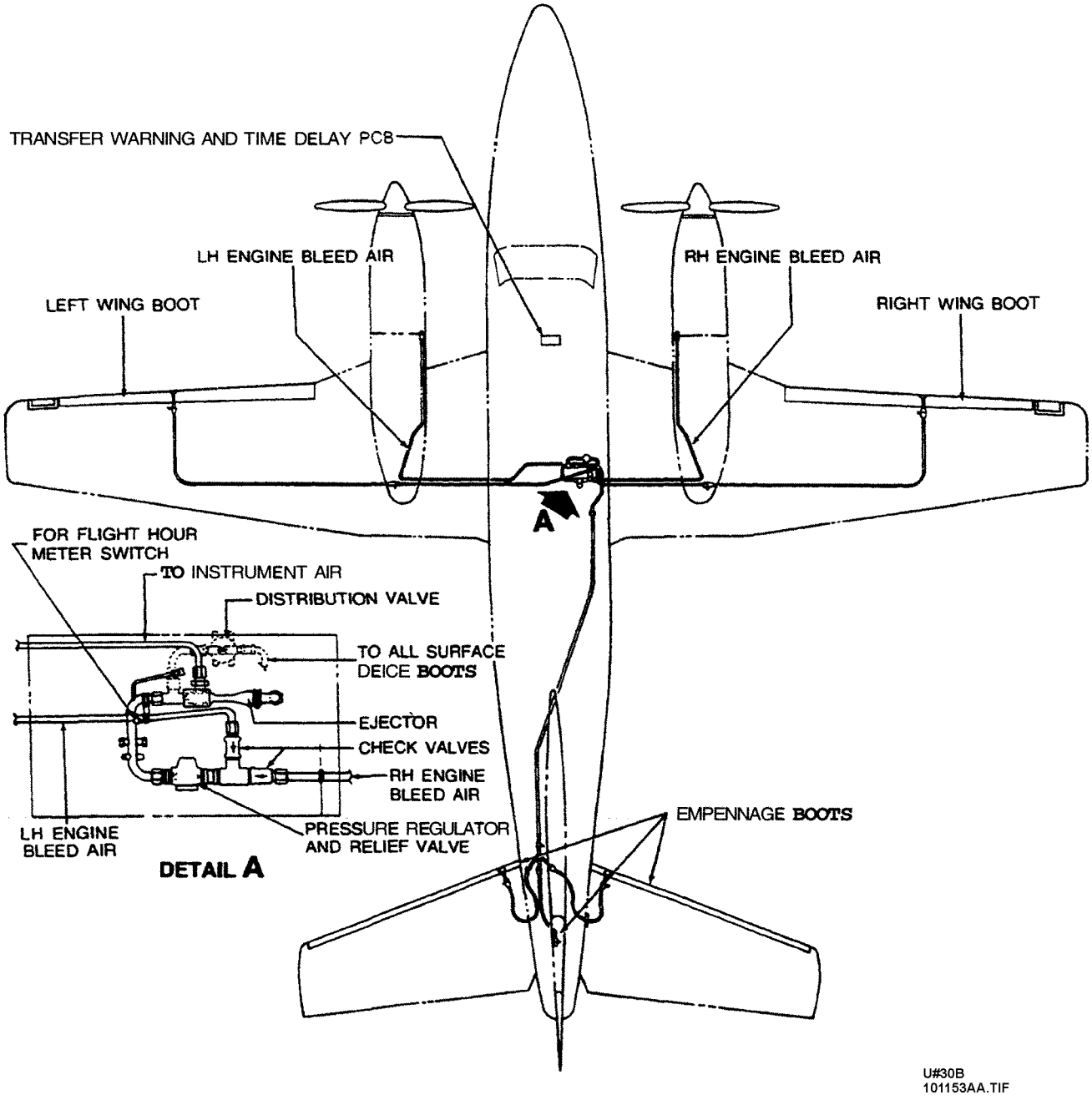

Figure 3
Surface Deice System (U-50, U-165 and After)

ICE AND RAIN PROTECTION
SURFACE DEICER SYSTEM
TROUBLESHOOTING

1. GENERAL

Troubleshooting - Surface Deicer

INDICATION	PROBABLE CAUSE	REMARKS
1. Lack of adequate vacuum to deflate boots completely.	a. Clogged lines.	
	b. Leaking lines.	
	c. Pressure regulator valve needs adjustment.	
	d. Distributor valve is defective.	
2. Lack of adequate air pressure to inflate boots completely.	a. Clogged lines.	
	b. Leaking lines.	
	c. Pressure regulator valve needs adjustment.	
	d. Distributor valve is defective.	
3. Boots fail to inflate with switch in either SINGLE or MANUAL position.	a. Blown or defective fuse.	
4. Boots fail to inflate with switch in SINGLE position.	a. Defective control relay or transfer warning and time delay PCB.	
5. Boots inflate with switch in SINGLE position, but deflate instantaneously when the spring loaded switch returns to off position.	a. Defective timer relay or transfer warning and time delay PCB.	
6. Boots fail to inflate over 12 to 15 seconds with switch in SINGLE position.	a. Defective timer relay or transfer warning and time delay PCB.	
7. Boot inflation time either more or less than 5 to 8 seconds with switch in SINGLE position.	a. Timer relay not properly adjusted or defective transfer warning and time delay PCB.	

ICE AND RAIN PROTECTION
SURFACE DEICER SYSTEM
MAINTENANCE PRACTICES

1. TIMER RELAY (U-1 THRU U-164, EXCEPT U-50)

A. Adjustment

The timer relay that controls the inflation time of the deicer boots is located under the floorboards of the passenger compartment, just forward of the main spar on the left hand side. If the deicer boots do not inflate for 5 to 8 seconds when the switch is in the SINGLE position, the timer relay may need adjusting. This can be accomplished by inserting a screwdriver through the hole in the top of the relay and turning the adjusting screw until a 5 to 8 second inflation time is established. Inflation time is reduced by turning the adjusting screw in a clockwise direction and increased by turning the adjusting screw in counterclockwise direction.

2. TIMER RELAY (U-50, U-165 AND AFTER)

A. Adjustment

The transfer warning and time delay functions of the surface deice system are controlled by number A239 printed wiring board module assembly and are not adjustable. If replacement of the Printed Circuit Board (PCB) is required, it is located in the aft PCB rack, under the center aisle floorboard just forward of FS 158.00.

3. SURFACE DEICER BOOT

A. Cleaning

The surface of the deicer boots should be checked for indications of engine oil after servicing and at the end of each flight. Any oil spots that are found should be removed with a non-detergent soap and water solution. Care should be exercised during cleaning to avoid scrubbing the surface of the boots, as this will tend to remove the special Icex application. It should also be remembered during servicing of the airplane that the deicer boots are made of soft flexible stock, which may be damaged if refueling hoses are dragged over the surface of the boots, or if ladders and platforms are rested against them.

B. Removal

(1) Starting at one corner of the upper trailing edge of the boot, soften the adhesion line with solvent (34, Table 6, Chapter 91-00-00).

(2) Apply a minimum amount of solvent to the seam line while carefully applying tension to peel back the boot.

(3) Using a brush or trigger type oil can to apply solvent, separate the boot from the leading edge for a chordwise distance of four inches across the upper trailing edge.

NOTE: The area between the boot and leading edge which has been separated will act as a reservoir for the solvent; consequently, the boot may be pulled forward with uniform tension.

(4) From the center line of the leading edge, continue to apply solvent while tension is being applied to undercut the adhesive until the boot is removed.

(5) Thoroughly clean all adhesive from the leading edge with clean, grease-free cloths dampened with solvent.

C. Installation

(1) With 3/4 inch masking tape, mask off the area to be covered by the boot, allowing 1/2 to 3/4 inch margin beyond the actual area. Mask off accurately to avoid unnecessary clean up of excess adhesive.

(2) Clean the area to be covered by the boot with a clean, grease-free cloth dampened with solvent (34, Table 6, Chapter 91-00-00). Change cloths frequently, and never dip a used cloth in the clean supply of solvent.

(3) Clean the metal surface with a clean cloth dampened with solvent, then quickly wipe the surface dry with a clean, grease-free dry cloth.

CAUTION: *Use plastic, not metal, containers for Turco Metal-Glo No. 3 (54, Table 6, Chapter 91-00-00). It is also recommended that persons handling the cleaner wear rubber gloves, even though the cleaner is harmless to skin if immediately washed off with soap and water.*

(4) Vigorously scrub the metal surface with a clean, grease-free cloth wetted with Turco Metal-Glo No. 3 (54, Table 6, Chapter 91-00-00). After one minute contact, wipe the surface dry with a clean, grease-free dry cloth. Cover the cleaned surface with kraft paper and wait an hour before checking the surface for cleanliness with a clean, dry, grease-free white cloth. If the clean cloth picks up any dirt, repeat the preceding Steps.

(5) Carefully clean the rough back surface of the boot at least twice with a clean, grease-free cloth moistened with solvent (34, Table 6, Chapter 91-00-00). Change cloths frequently to avoid contaminating the cleaned portions of the boot. If necessary, continue cleaning the boot until completely clean.

NOTE: If the finish on the back of the boot is smooth, mechanically roughen the bonding surface before cleaning.

(6) Prepare test strips as instructed in the following paragraph, TEST CONTROL STRIPS.

(7) Open the container and thoroughly stir the EC1300L adhesive (43, Table 6, Chapter 91-00-00) with a stick. Do not attempt to use adhesive that has jelled too much to drip from the stick.

NOTE: Do not apply adhesive at temperatures below 40°F or when the relative humidity is above 80%. If this requirement can not be met, allow an extended drying time. Do not apply adhesive when the air is contaminated with dust or other elements. The following table will serve as a guide for adhesive application:

Temperature in °F	Drying Time in Minutes
Above 80	30
60 to 80	45
Below 60	60

(8) Either of the two following methods may be used to apply the cement:

(a) Brush one even coat of adhesive (43, Table 6, Chapter 91-00-00) on both the deicer boot back and metal surfaces. After permitting the adhesive to dry for at least 30 minutes, apply a second even coat to both surfaces. Allow the second coat to dry for at least 30 minutes, preferably and hour, before installing the deicer boot.

NOTE: Brushing too long in one area will cause the cement to “roll” or “ball up”.

(b) Using a spray gun (Binks No. 7 with No. 63 PB head, De Vilbiss JGA 75X, or their equivalent), spray one even coat of adhesive on both the deicer boot back and the metal surfaces. After permitting the adhesive to dry for at least 30 minutes, spray a second even cross coat on both surfaces. Allow the second coat to dry for at least 30 minutes, preferably an hour, before installing the deicer boot.

NOTE: For spraying consistency, dilute the adhesive with a solvent blend of 2 parts naphtha (33, Table 6, Chapter 91-00-00) to one part solvent (34, Table 6, Chapter 91-00-00) until a No. 4 Ford cup will empty in 20 seconds.

(9) Snap a chalk line along the center of the leading edge of the airfoil. Snap a corresponding chalk line lengthwise along the center line of the cemented side of the deicer boot. Trace the chalk line with a ball point pen and remove the chalk line with a dry cloth.

NOTE: If the boot has a white center reference line, the additional chalk line is not required; however, the white line should be intensified with a ball point pen after the adhesive is applied.

(10) Attach a hose to the deicer boot connection (nipple) to which a vacuum source is attached. Do not leave fingerprints on the adhesive side of the deicer boot. Have a sufficient number of personnel present to handle the boot effectively.

(11) Using a clean, lint-free cloth heavily moistened (but not dripping) with toluol (36, Table 6, Chapter 91-00-00), reactivate the adhesive on the airfoil leading edge and boot in spanwise strip about 6 inches wide and 24 inches long. Match the boot center line to the leading edge chalk line, apply vacuum air to the boot connection, and roll the boot surface spanwise along the leading edge with a 2 inch rubber roller to remove all air from between the adjoining surfaces. Rubbing the adhesive surfaces excessively will remove some of the adhesive. If the boot is “off course” carefully peel back the boot, using a minimum amount of solvent (34, Table 6, Chapter 91-00-00), and position it properly.

CAUTION: *Do not allow excessive amounts of solvent to contact the boot; the solvent is to undercut the adhesive. If the adhesive is removed from either the boot or leading edge, it must be replaced and allowed to dry for 30 minutes minimum before continuing.*

(12) Continue to activate the surfaces and roll on each top half and lower half of the boot in sequence. Exerting pressure, use the 2 inch rubber roller on the entire surface of the boot parallel with the inflatable tubes and use the narrow steel stitcher roller on the trailing edges. Take care to avoid entrapping air throughout the rolling operation. If an air blister does occur, carefully remove the boot using a brush dipped in solvent, so that the air can escape, then press the surface down until the adhesive surface seals.

(13) Remove the vacuum source and connect the boot nipple to the deicer system tube.

NOTE: On airplane serials U-165 and After, maintain a clearance of 0.02 to 0.06 inch between the stall warning lift transducer and the edge of the deicer boot.

(14) Trim the boot as required with a sharp knife and continue the boot attachment until the total boot area is attached. Reroll the entire surface of the boot, starting at the center line area, then roll the trailing edges with a stitcher roller.

(15) Remove all masking tape and clean excessive adhesive off with a clean white cloth dampened with solvent, being careful not to allow the solvent to run under the edges of the boot.

(16) Perform an adhesion test of the test strips for boot adhesion acceptability as instructed in the following paragraph TEST CONTROL STRIPS.

(17) Install masking tape approximately 1/4 inch around the boot for uniform straight lines, then cover all exposed cement and fair around all cut edges and trailing edges of the boot with sealing compound (46, Table 6, Chapter 91-00-00). Remove the tape before the sealing compound cures.

(18) After the sealing compound has cured approximately 6 to 12 hours, clean the boot surface and the area adjacent to the boot with solvent (34, Table 6, Chapter 91-00-00). After final boot installation, the airplane should not be flown for 24 hours. The boots must not be operated until 48 hours after installation.

4. TEST CONTROL STRIPS

Deicer boot adhesion shall be checked through the use of test strips applied at the time of boot installation. A minimum of two 1 x 8 inch test strips shall be prepared from full thickness pieces of trimmings, scrap or previously rejected boot material. A minimum of two test strips are required for each batch or mixture of adhesive used. The strips should be applied in convenient areas adjacent to the boot. The metal and rubber surfaces shall be prepared in the same manner as prescribed, for the boot installation.

NOTE: The test strips may be run on separate aluminum panels of the same alloy as the leading edge skin. The same surface preparation and bonding requirements as previously stated shall apply.

When applying the test strips, leave approximately an inch of one end free of adhesive for the purpose of attaching a clamp for the adhesion test. Allow the strips to cure for a minimum of four hours or more before testing adhesion.

A. Adhesion Test

Using a spring scale attached to a clamp on the test strip unbonded end, measure the force required to remove the strip at a rate of four inches per minute. The pull shall be applied 180° to the surface (strip doubled back on itself). A minimum requirement of seven pounds tension shall be required to remove the test strip. Should the first pull test fail to meet the adhesion requirements, allow the second test strip

to cure a minimum of eight or more additional hours and perform the same pull test. In the event of failure, the acceptance of the boot installation shall be determined by the BOOT ACCEPTABILITY requirements in the following paragraph.

B. Boot Acceptability

The boots shall be accepted if the test strips meet the adhesion requirements of the preceding paragraph. In the event of failure of both test strips, the acceptability shall be based on the following:

(1) Using solvent (34, Table 6, Chapter 91-00-00) to loosen the adhesion line, carefully lift one corner of the newly installed deicer boot sufficiently to attach a clamp.

(2) Attach a spring scale to the clamp and pull with a force 180° to the surface, in such a direction that the boot tends to be removed on a diagonal.

(3) If a force of seven pounds per inch of width can be exerted under these conditions, the installation shall be considered as satisfactory.

(4) Attach the boot corner as instructed in the paragraph SURFACE DEICER BOOT INSTALLATION.

(5) Failure of the boot to meet the above requirement shall result in complete removal and reinstallation of the boot.

5. SURFACE DEICER BOOT - ICEX APPLICATION

Icex compound (91, Table 6, Chapter 91-00-00), a product of B. F. Goodrich Company, is a silicone base material specifically compounded to lower the strength of ice adhesion on the surfaces of the deicer boots. Icex will not harm the boots and offers added ozone protection. When properly applied and renewed at periodic intervals, Icex provides a smooth, polished film that evens out microscopic irregularities on the rubber surface; therefore, ice formations have less chance to cling and are removed faster and cleaner when the boots are operated. Apply Icex as follow:

(1) Clean the boot thoroughly with a clean white cloth dampened with solvent (34, Table 6, Chapter 91-00-00) or unleaded gasoline.

(2) Wash the boot with mild soap and water. Thoroughly rinse and allow to dry.

(3) Thin the Icex compound, 10% to 25%, with trichloroethane (92, Table 6, Chapter 91-00-00) to obtain a thin coat.

(4) Use a clean cloth saturated with the thinned Icex compand for application. Hand buff with a clean soft cloth until a smooth glossy surface is obtained.

NOTE: The frequency of application will depend upon number of flight hours and type of weather encountered. Reapply as required to maintain a thin polished film on the boot surfaces.

6. SURFACE DEICER BOOT REPAIRS

A. Limitations

NOTE: Patch sizes can be trimmed to accommodate small areas, as long as the patch extends a minimum of 1/2 inch beyond the damaged area on all sides.

To maintain the maximum functional efficiency of the surface deicer boots, the following limits are allowed:

- For pin hole repair, use the pin hole repair kit (127, Table 6, Chapter 91-00-00). Refer to instructions provided with kit for repair procedure.
- Pin hole repairs 1/16 inch or smaller - it is recommended that no more that 20 pin hole repairs are made per 12 inches square **or**
- Small patches 1 1/4 inches x 2 1/2 inches - 3 patches per 12 inches square **or**
- Medium patches 2 1/2 inches x 5 inches - 2 patches per 12 inches square **or**
- 2 small patches and 1 medium patch per 12 inches square.
- Replace any deicer boot that has a cut, tear or rupture greater then 3/4 inch in length.

B. Resurfacing Deicer Boots

The principal factors involved when resurfacing seems advisable are:

- If the surfacing material has abraded off.
- If the surfacing has developed cracks.

(1) Clean the deicer boot thoroughly with toluol (36, Table 6, Chapter 91-00-00) or uncontaminated unleaded aviation gasoline.

(2) Roughen the entire surface of the boot with fine sandpaper.

(3) Clean the surface again with a clean lint-free cloth moistened with toluol or uncontaminated unleaded aviation gasoline.

(4) Apply masking tape beyond the upper and lower trailing edges, leaving a 1/4 inch gap of bare metal.

(5) Brush one coat of cement (40, Table 6, 91-00-00) on the boot and allow it to dry at least one hour. Then apply a second coat and allow it to dry at least four hours before operating the deicers. The airplane may be flown as soon as the cement is dry.

NOTE: If cement has aged 3 months or more, it may be necessary to dilute it with toluol to obtain the proper brushing consistency. Mix thoroughly approximately 5 parts cement to one part toluol.

C. Scuff Damage

This type of damage is most commonly encountered and generally will require no repair; however, close visual inspection when the boots are cycled may expose deep scuffs and cuts. On those occasions, if the entire ply surface (0.010 inch) has been removed, or if the boot is cut, it is necessary to patch the damage. B. F. Goodrich Kit No. 74-451-C contains cold patches suitable for repairing the damage area. Cold patch repairs can be made as follows:

(1) Clean the area around the damage with a clean, grease-free cloth dampened with toluol (36, Table 6, Chapter 91-00-00) or unleaded gasoline.

(2) Using steel wool, buff the area around the damage so that it is moderately, but completely roughened.

NOTE: A locally manufactured buffing shield, fabricated from any thin sheet material, will assure a neater repair

(3) Wipe the buffed area with a clean grease-free cloth dampened with toluol to remove loose particles.

(4) Select a repair patch which will extend at least 1/2 inch beyond the damaged area in all directions.

(5) Apply one even coat of No. 4 cement, provided in the repair kit, to the patch and the damaged area. Allow the cement to dry until tacky.

(6) Apply the patch to the boot with an edge, or the center, adhering first. Work the patch down carefully to avoid air pockets. Thoroughly roll the patch down with a stitcher roller.

NOTE: The patches are manufactured to stretch in one direction only. Be sure to cut and apply the selected patch so that stretch is in the widthwise (spanwise) direction of the inflatable tubes. The length of the patch shall run along the length of the tubes.

(7) Allow the patch to set for 10 to 15 minutes, then clean the patch and surrounding area with a clean cloth dampened with solvent (34, Table 6, Chapter 91-00-00) or toluol (36, Table 6, Chapter 91-00-00).

(8) Satisfactory adhesion of the patch will be reached in about four hours; however, the boot may be inflated to check the repair after a minimum of 30 minutes.

(9) Apply a smooth light coat of EC801A2 sealing compound (46, Table 6, Chapter 91-00-00) to seal and feather the edges of the patch to the boot.

(10) After the cement and sealing compound has dried and cured, apply a thin coating of Icex compound (91, Table 6, Chapter 91-00-00) to the patch and surrounding area.

D. Tube Area Damage

Repair cuts, tears or ruptures to the tube area as follows:

(1) Clean the area around the damage with a clean, grease-free cloth dampened with toluol (36, Table 6, Chapter 91-00-00) or unleaded gasoline.

(2) Buff the area around the damage with a buffing stick, provided in the repair kit.

(3) Wipe the buffed area with a clean, grease-free cloth dampened with toluol to remove loose particles.

(4) Select a patch, from the repair kit of ample size to cover the damaged area and extend at least 5/8 inch beyond the ends and edges of the cut or tear.

NOTE: If none of the patches are the size required, cut a patch of the desired size from one of the larger patches. If this is done, bevel the edges by cutting with shears at an angle.

(5) Complete the installation of the patch as in the paragraph SCUFF DAMAGE, Steps (4) through (10).

E. Loose Surface Ply in Non-Inflatable Area

Repair the damaged area as follows:

(1) Peel and trim the loose surface ply to a point where adhesion of the surface ply is good.

(2) Scrub (roughen) the area in which the surface ply is removed with steel wool. The scrubbing motion must be parallel to the cut edge of the ply to prevent the ply from loosening.

(3) Scrub with steel wool and toluol directly over all edges, but parallel to the edges of the ply, to taper edges of the ply.

(4) Cut a piece of surface ply material, provided in the repair kit, large enough to cover the damaged area and extend at least one inch beyond the damaged area in all directions.

(5) Mask off an area 1 1/2 inches larger in length and width than the size of the damaged area.

(6) Apply one even coat of No. 4 cement, provided in the repair kit, to the damaged area and the surface ply material. Allow the cement to dry until tacky.

(7) Roll the surface ply to the deicer boot with a two inch rubber roller, applying enough tension to prevent wrinkling. If air blisters appear, remove with a hypodermic needle and roll the area.

(8) Remove the masking tape and complete the installation as in the paragraph SCUFF DAMAGE, Steps (7) through (10).

F. Loose Surface Ply in Tube Area

Loose surface ply in the tube area is usually an indication of the boot starting to flex fail. This type of failure is easily detected, as a blister under the surface ply when the boot is pressurized. If detected while a small blister (1/4 to 3/8 inch diameter) and immediately patched, service life of the boot will be appreciably extended. Repair of this type damage is the same as SCUFF DAMAGE.

G. Fabric Back Ply Damage During Removal

If the adhesive from the leading edge shin has adhered to the back surface of the boot, remove the adhesive with steel wool and solvent (34, Table 6, Chapter 91-00-00). In spots where the coating has pulled off the fabric, leaving bare fabric exposed, apply at least two additional coats of adhesive. Allow each coat to dry thoroughly.

7. STALL STRIP

A. Installation

(1) A stall strip is installed near the inboard end of each wing boot. Dimension A is 19.62 ± 0.06 inches at the outboard end of the stall strip which is located at WS 131.98. Dimension A is 20.20 ± 0.06 inches at the inboard end of the stall strip (Ref. Figure 201).

(2) Clean the boot surface thoroughly, remove all old cement. Mask off area where new strip is to be installed and wipe with solvent (34, Table 6, Chapter 91-00-00).

(3) Bostic 1008, a two part cement, (65, Table 6, Chapter 91-00-00) is used to join the stall strip to the deicer boot. Mix the Bostic 1008 in the following manner: 30 parts (by weight) of the base material (in the “A” package) with 1 part (by weight) of the accelerator (in the “B” package).

(4) Apply a coat to both the stall strip and the area to which it will be bonded. Allow to dry 10 to 15 minutes, then install the strip as described in Figure 201. Cement will be set in about 6 hours.

(5) When dry, coat the area with A56B cement (40, Table 6, Chapter 91-00-00) to replace the conductivity of the boot.

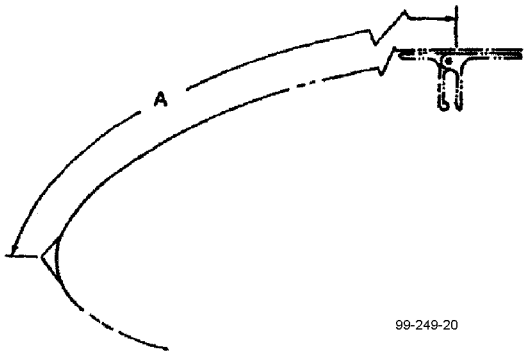

Figure 201
Indicator Check

8. SURFACE DEICER

A. Operational Check

(1) With both engines operating at 70 to 80% gas generator speed, momentarily place the deicer switch in the AUTOMATIC or SINGLE position; the deicer boots should inflate for five or eight seconds, then deflate and reach a vacuum hold-down condition. During inflation, the system pressure gage should register approximately 16 to 17 psi.

(2) Place the deicer switch in the MANUAL position and hold for a few seconds; the deicer boots should inflate and remain inflated while the switch is retained in the MANUAL position. Check for correct system pressure (Ref. PRESSURE REGULATOR VALVE ADJUSTMENT in this chapter).

(3) Release the deicer switch, permitting it to return to the OFF position; the deicer boots should deflate and reach a vacuum hold-down condition.

(4) Repeat Steps (1) through (3) with each engine operating individually at 70 to 80% gas generator speed.

9. PRESSURE REGULATOR RELIEF VALVE

A. Adjustment (U-1 thru U-151, except U-50)

(1) Gain access to the pressure regulator valves by removing the oil cooler air intake scoop at the underneath side of the nacelles.

(2) Remove the center aisle floorboard panel behind the main spar and on the airplanes which are equipped with an hour meter, remove the hour meter pressure switch from the pneumatic pressure intake line and connect a calibrated test gage to the line. On airplanes which are not equipped with an hour meter, remove the cap from the hour meter pressure switch connecting point and attach a test gage. Taps have been provided in the LH and RH bleed air lines to actuate L and R pressure failure indicators on the system pressure gage.

(3) Run up the engine until a reading of between 70 and 80% is registered on the tachometer.

(4) Check pressure for 18 ± 1 psi. Adjust the regulator valve by unsafetying the lock retaining washer and loosening the adjusting screw locknut located at the top of the regulator until a reading of 18 ± 1 psi on the test gage in the fuselage is obtained.

(5) Repeat the above sequence for the other engine.

(6) Run up both engines until a reading of between 70 and 80% is registered on the tachometers and check the “instrument air” system pressure gage on the instrument panel for a reading of 5.5 in. Hg. If the 5.5 in. Hg is not obtained, the instrument air pressure regulator (located forward of the instrument panel) will need to be adjusted.

CAUTION: *Do not exceed the recommended 18 ±1 psi regulator setting, because a higher setting will result in excessive wear on the relief valve.*

NOTE: Step (6) pertains to airplanes equipped with surface deicer equipment. On airplanes not equipped with surface deicer equipment, the system pressure may be read directly from the test gage.

B. Adjustment (U-50, U-152 and After)

(1) Gain access to the pressure regulator valve, just forward or just aft of the rear spar, depending upon airplane serialization.

(2) Run up the engine until a reading between 70 and 80% is registered by the gas generator tachometer.

(3) Check the deicer pressure gage for a reading of 18 ± 1 psi. If necessary, unsafety the lock retaining washer and loosen the adjusting screw locknut located on top of the valve.

(4) Turn the adjusting screw clockwise to increase and counterclockwise to decrease the pressure setting until the pressure gage indicates 18 ± 1 psi.

(5) Tighten and safety the locknut, then shut down the engine.

NOTE: Check the system vacuum gage for a reading of 5.9 in. Hg, for adjustment of the pressure regulator valve may affect system suction enough to necessitate readjusting the vacuum regulator valve.

10. VACUUM REGULATOR VALVE

A. Adjustment

On U-152 thru U-164 the vacuum regulator valve is located on the aft side of the bulkhead at FS 84.00, approximately 20 inches forward of the instrument panel, on the pilot's side. On U-50, U-165 and After, the vacuum regulator valve is located just forward of the bulkhead at FS 94.00 near the left side of the fuselage. Adjust the regulator as follows:

(1) Start one engine and run it up until a reading of 70 to 80% is registered by the gas generator tachometer.

(2) Check the vacuum gage for a reading of approximately 5.9 inches Hg. If the indication is incorrect, adjust the vacuum regulator valve by unsafetying the lock retaining washer and loosening the adjusting screw locknut.

(3) Turn the adjusting screw IN (clockwise) to increase or OUT (counterclockwise) to decrease the vacuum setting.

(4) When the vacuum gage registers a reading of 5.9 inches Hg. with the engine operating within the range designated in Step (1), tighten the locknut and safety it, then shut down the engine.

ICE AND RAIN PROTECTION
ENGINE AIR INLET DEICER
DESCRIPTION AND OPERATION

1. GENERAL

Each engine air inlet is equipped with an electrically heated deicer boot to prevent ice formation over the intakes. The system is controlled by individual right and left inlet switches on the subpanel. Both switches have a TEST position which will indicate system amperage on the prop/inlet ammeter. If both switches are held in the TEST position at the same time, only the amperage from the right lip deice boot will be shown on the prop/inlet ammeter.

Each inlet heater has a power circuit breaker and a relay circuit breaker located on the circuit breaker panel to the copilot's right. On U-50, U-165 and After, the two control circuit breakers for the engine air inlet deice are located on the right side circuit breaker panel. The two power circuit breakers are located on the main power distribution panel assembly, located under the center aisle floorboard just aft of FS 158.00. When the control switch is turned ON, a relay completes the circuit to each boot. Since the boots have a high heating capacity and would burn up without in-flight air to cool them, the relays are grounded through their respective landing gear safety switch. When the landing gears are compressed, the relay circuits are open, de-energizing the relays and opening the circuits to the inlet heaters.

CAUTION: *Do not jack the airplane without first pulling out the circuit breakers for both inlet deicers.*

ICE AND RAIN PROTECTION
ENGINE AIR INLET DEICER
TROUBLESHOOTING

1. GENERAL

Troubleshooting - Engine Air Inlet Deicer

INDICATION	PROBABLE CAUSE	REMARKS
1. Circuit breaker trips repeatedly.	a. Short in circuit.	
2. Boot on one engine will not heat.	a. Open circuit in the boot.	
	b. Lip heater control relay stuck open.	
	c. Open circuit between switch and heater control relay or between relay and power source.	
	d. Open circuit between heater relay and landing gear safety switch.	
3. Entire system will not operate.	a. Control switch will not close (press-to-test indicator lights illuminate).	
	b. Open in power lead.	
	c. Circuit breaker open (press-to-test indicator lights will not illuminate).	

ICE AND RAIN PROTECTION
ENGINE AIR INLET DEICER
MAINTENANCE PRACTICES

1. ENGINE AIR INLET DEICER

A. Operation Check

(1) Raise the airplane on jacks until the main landing gear shock struts are fully extended.

(2) Place one man in the cockpit to operate one of the inlet heater switches while another man places his hand on the corresponding heating element to determine if it is operating correctly.

CAUTION: *Due to the elements high heating capacity, do not operate the inlet heater more than 4 seconds during ground checkout. Allow the inlet heater to cool before operating again.*

(3) Repeat the operation with the other inlet heater.

B. Removal

(1) Disconnect the electrical leads inside the cowl.

(2) Using solvent (34, Table 6, 91-00-00) or toluol (36, Table 6, 91-00-00) to soften the adhesion line between the boot and the cowl, loosen the right hand comer of the boot sufficiently to grasp it with vise grip pliers or a similar tool.

CAUTION: *Unless the boot being removed is to be scrapped, cushion the jaws of any pulling tool to prevent damaging the boot surface.*

(3) Apply a slow, steady pull on the boot to pull it off the cowl surface while continuing to use the solvent to soften the adhesion.

(4) Remove the remaining adhesive from the boot and cowl with methyl propyl ketone or toluol.

C. Installation

(1) Clean the cowl in the area of the boot thoroughly with solvent (34, Table 6, 91-00- 00) or acetone (35, Table 6, 91-00-00). For final cleaning, wipe the solvent off quickly with a clean. dry, lint-free cloth to avoid leaving a film.

CAUTION: *The fiberglass and rubber parts must be thoroughly clean to assure maximum adhesion.*

(2) Moisten a clean cloth with methyl propyl ketone or acetone and clean the inner surface of the deicer boot, changing the cloth frequently to avoid contaminating the clean area. Apply primer (44, Table 6, 91-00-00). Allow to dry 60 minutes.

(3) Thoroughly mix cement (41, Table 6, 91-00-00) or (43, Table 6, 91-00-00) and apply one even brush coat to the cowl. Allow the cement to dry for at least one hour at 40°F or above when the relative humidity is less than 75%, or two hours if the humidity is between 75% and 90%. Do not apply the cement if the relative humidity is higher than 90%.

(4) After allowing sufficient drying time, apply a second brush coat of cement to the cowl and one coat of cement to the inner surface of the boot. Allow the cement to dry.

(5) Insert the electrical leads through the hole in the cowl and position the boot on the cowl inlet. Moisten the cement lightly with toluol and tack the boot to the cowl. being careful to prevent air pockets. Start with the center area of the boot to assure that it will reach but not ride up on the inlet baffles. If the boot is not properly positioned after making contact with the cowl. carefully pull up and replace it properly. Roll firmly along the leading edge with a rubber roller.

CAUTION: *Never use a metal or wooden roller for this purpose, for they would damage the heating elements in the deicer boot.*

(6) Gradually tilting the roller. work the boot carefully over each side of the inlet contour. Avoid trapping air packets under the boot (Ref. Figure 201, Detail B).

(7) Roll outwardly from the leading edge to the trailing edge of the boot. If excess material at the edge tends to form puckers, work them out smoothly and carefully with the fingers. Pull the trailing edge up as necessary and reposition (Ref. Details C and D).

(8) Roll the edges of the boot with a narrow steel stitcher roller.

(9) Clean the cowl with a clean cloth moistened with toluol or methyl propyl ketone. Be careful not to let the solvent run into the edge of the boot.

(10) Mask the cowl about 1/2 inch from all edges of the boot and apply one even coat of cement (40, Table 6, 91-00-00) around the edges of the boot. Allow 1/16 to 1/8 inch overlap on the boot and extend the cement to the masking tape. Remove the masking tape after applying the cement to obtain a neat border.

2. ENGINE AIR INLET DEICER BOOT

A. Repair

For detailed instructions on patching and repairing the air inlet boots, refer to the paragraph in Chapter 30-60-00 that describes the proper method for repairing the propeller deicer boots.

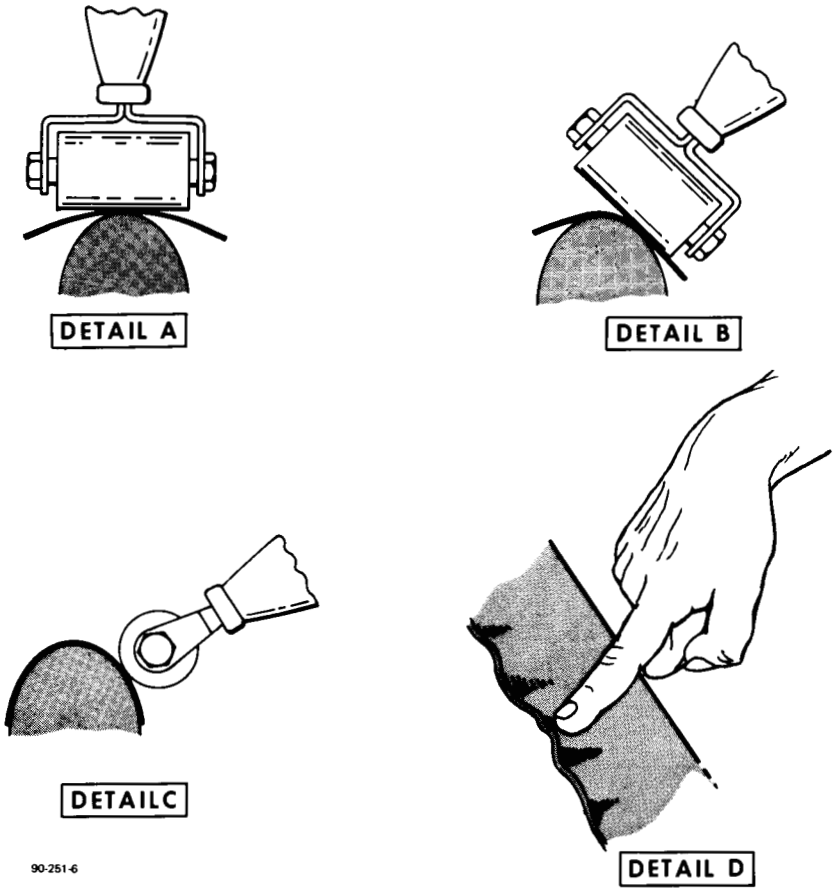

Figure 201
Engine Air Inlet Deicer Boot Installation

ICE AND RAIN PROTECTION
ENGINE INERTIAL ANTI-ICER SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

The system for each engine is controlled by a push-pull handle mounted on a bracket attached to the bottom of the subpanel directly under the pilot's control column. A cable is routed from the control handle. along the leading edge of the wing, and up the RH side of the nacelle to the linkage that moves the vane. When the control handle is pushed in, the linkage attached to the vane is retracted against a spring stop to hold the vane in the up position for flight under normal conditions. When icing is encountered and the control handle is pulled out, the interconnecting cable rotates a torque tube to extend the movable vane, which is then locked in the down position when the attaching linkage passes overcenter. The extended vane causes the velocity of the incoming air to increase. While the lighter portion of the incoming air turns abruptly to enter the engine plenum around the trailing edge of the extended vane, that air which is laden with ice and snow rushes on past the entrance to the plenum and is discharged overboard through the bypass duct over the oil cooler.

ICE AND RAIN PROTECTION
ENGINE INERTIAL ANTI-ICER SYSTEM
MAINTENANCE PRACTICES

1. INERTIAL ANTI-ICER VANE

A. Travel Adjustment

(1) Set the inertial vane in the down position, locked overcenter. If both overcenter stop pins do not contact the stop links, loosen the retaining bolts in the vane brackets and relocate the brackets in their slotted mounting holes until both overcenter stop pins contact the stop links simultaneously. Make sure that the tee control handle and the accompanying nut are run up on the shaft. using all the threads.

(2) Extend the tee handle 4.00 inches ± 0.03 inch, as measured at the instrument panel. Adjust the rod end to align with the attachment in the linkage.

(3) Connect the rod end to the vane and operate the control. The control handle must not move through more than 4 inches of travel, and the vane must go over the center in the down position and fully engage the spring stop.

(4) If the detent is not fully engaged, adjust the nut on the tee handle in the cockpit to obtain additional "up travel" on the vane. Travel will now exceed 4.00 inches. DO NOT READJUST THE ROD END. (If the rod end were readjusted instead of the nut on the tee handle, the pilot might hit the panel with the top of his hand when he is pulling on the push-pull cable).

(5) To complete the rigging, set the tee handle in a vertical position on the rod, using the jam nut on the back of the tee handle. If necessary, lubricate the linkage with lubricating oil (3, Table 6, 91-00-00) to make the movable vane operate freely.

2. BYPASS DUCT

A. Repair

The inertial anti-icer will function effectively only if the inner surface of the bypass duct is aerodynamically clean. Rough spots, dents, or projecting rivets, screws, or other protrusions may allow ice to begin building up and block the bypass duct. To prevent this, make sure there are no protrusions or flaws in the duct. Fill dents or rough spots with smoother (55, Table 6, 91-00-00) and sand the repaired area smooth to blend into the inner contour of the duct. Coat the repaired area with Epoxy Primer (48, Table 6, 91-00-00). Apply a smooth topcoat of Epoxy Black, Gloss Finish No. 54-515, (product of Enmar Incorporated. Wichita, Kansas), or its equivalent to the repaired area.

For repair of the fiber duct over the oil cooler on the underside of the cowling, apply a flush patch in accordance with the procedures under FIBERGLASS REPAIR, Chapter 51. After the repair, apply three coats of white elastomeric polyurethane coating No. CD875-40A P/N of Hughson Chem. Co., Erie, Penn.) to the duct. Minimum thickness of the three coats shall be no less than 0.005. Allow to dry for one hour between coats and allow 48 hours drying time before applying the top color coat.

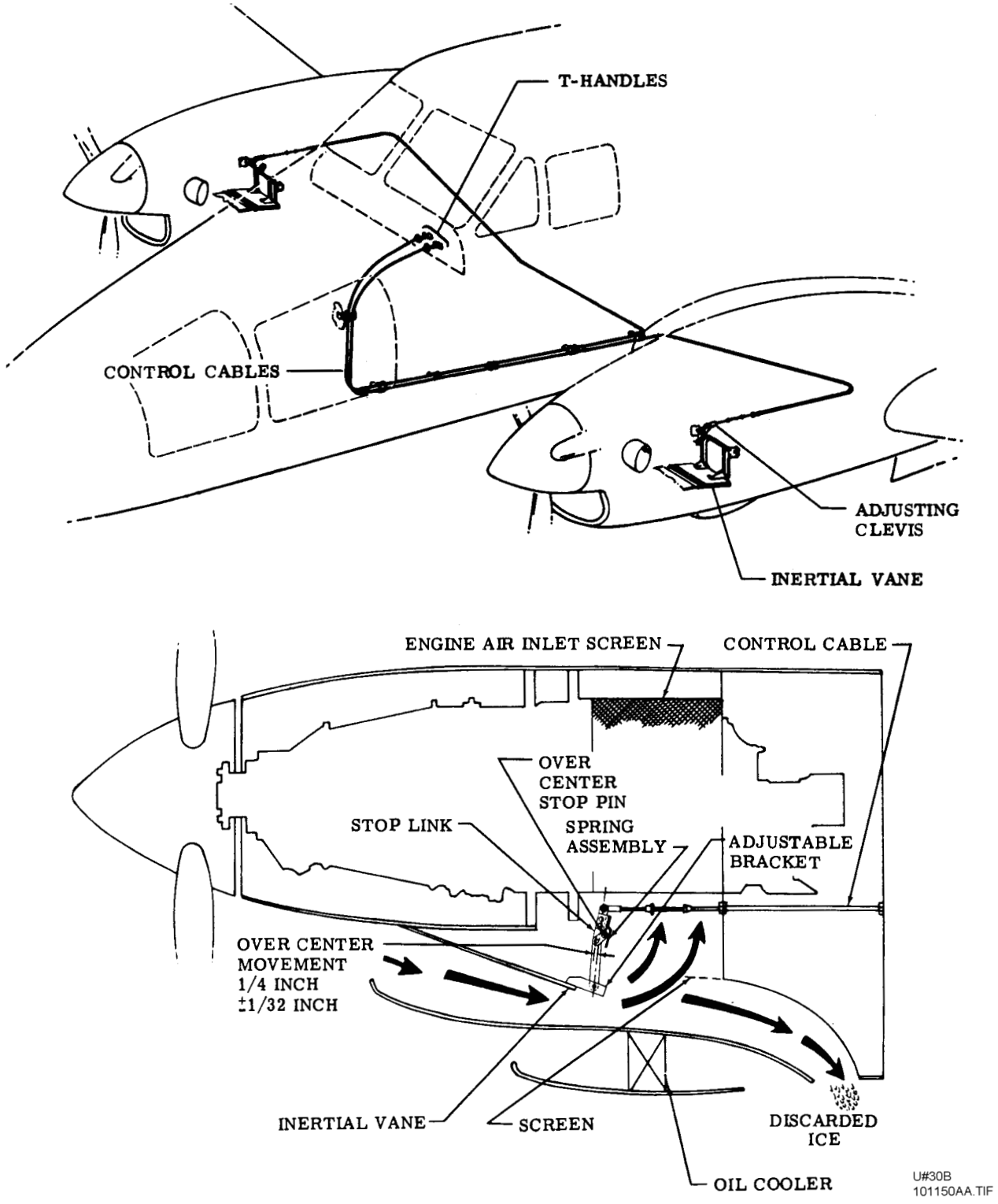

Figure 201
Inertial Anti-Icer System

ICE AND RAIN PROTECTION
BRAKE DEICE SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

A. U-50, U-165 and After

The engine bleed air is routed from the firewall shutoff valve on inboard side of each nacelle, to a combination bleed air shutoff valve and annunciator light trigger switch located beneath an access panel at the aft upper side of each nacelle, aft of nacelle station 136.50. From the switch and valve assembly. the brake deice assembly consists of a tube, two connected hoses and a deice distributor manifold assembly attached at the base of the piston and axle assembly.

NOTE: Notice that the flexible hoses for the brake deicer are the same color and vendor on both main wheels. The Flexfab (Gray) hoses and the Bemco (Red) hoses must not be mixed on any one airplane when maintenance is performed.

The brake deice system is controlled by a BRAKE DEICE - OFF toggle switch located on the pilot's subpanel in the ICE PROTECTION section. When the switch is in the BRAKE DEICE position, power from the airplane electrical system is supplied through a circuit breaker in the right circuit breaker panel assembly to a printed circuit board module assembly (A241), located in the aft PCB rack under the center aisle floorboard just forward of FS 158.0. This module supplies current to the combination bleed air shutoff valve and annunciator light trigger switch at the aft upper side of each nacelle. Each shutoff valve opens and permits the hot bleed air to travel through the lines to the distributor manifold assembly for brake deicing. The annunciator light trigger switch located on the bleed air shutoff valve simultaneously triggers the annunciator light system when the valve opens. The L BRAKE DEICE and R BRAKE DEICE lights in the annunciator panel will illuminate.

If the airplane takes off without the control switch for the brake deice system having been switched OFF. a circuit is completed through the uplock switch to a timing circuit in the control module (PCB A241) when the main landing gears reach the up and locked position. This timing circuit will cycle the deice system off after 10 minutes of operation, thereby closing the bleed air shutoff valve in each nacelle to shut off the flow of bleed air to the brakes so that adjacent components in the wheel well will incur no damage through overheating. The system cannot again be activated until the landing gear is cycled out of the up and locked position.

CAUTION: *The brake deice system should not be activated at a true outside ambient temperature above 50°F.*

Refer to Chapter 32-41-00 for removal and installation procedures of the brake deicer combination bleed air shutoff valve and annunciator light trigger switch, and the distributor manifold.

ICE AND RAIN PROTECTION
PITOT AND STATIC SYSTEM DEICER
DESCRIPTION AND OPERATION

1. GENERAL

A heating element in the pitot mast prevents the pitot opening from becoming clogged with ice. The heating element is connected into the aircraft electrical system through a 5-ampere circuit breaker. On U-50, U-165 and After, the pitot heat is controlled by two 7.5 amp circuit breaker switches located on the left inboard subpanel.

ICE AND RAIN PROTECTION
PITOT AND STATIC SYSTEM DEICER
TROUBLESHOOTING

1. GENERAL

Troubleshooting - Pitot and Static System Deicer

INDICATION	PROBABLE CAUSE	REMARKS
1. Heating element does not heat.	a. Open switch	a. Check for continuity.
	b. Grounded wire.	b. Check for continuity.
	c. Grounded or open circuit.	c. Check for continuity.
2. Circuit breaker keeps tripping.	a. Grounded wire.	a. Check for continuity.

ICE AND RAIN PROTECTION
WINDSHIELD DEICER SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

A. U-1 thru U-164, except U-50

The pilot's windshield is protected against icing by electrical heating elements. The copilot's windshield may also be protected as an optional installation. The system is controlled by a switch located on the pilot's subpanel. An ON-OFF type switch in the standard installation and a three position switch (BOTH-OFF-LEFT) for the optional installation controls the windshield deice system. The controller is mounted on the upper left comer of the forward bulkhead behind the pilot's instrument panel. Controller power is supplied by a 1/2 amp circuit breaker located on the circuit breaker panel to the copilot's right. A 30 amp current limiter (60 amp current limiter with optional installation) on the power distribution panel provides power to the system through the windshield deice relay on the lower equipment panel. The power distribution panels are located under the center aisle floor.

B. U-50, U-165 and After

On U-50, U-165 and After, an optional dual level heated windshield is provided. This installation includes both the pilot and copilot windows and is controlled by the pilot's and copilot's switches on the left inboard subpanel placarded NORMAL-OFF-HI. Each windshield utilizes 882 watts of power in the low heat mode and 1196 watts of power in the high heat mode. In the low heat mode the total heated area of each windshield is 361 square inches, and in the high power mode, the total heated area is 266 square inches.

Windshield heat is protected by two 50 amp circuit breakers: one for the pilot's windshield and one for the copilot's windshield. These two circuit breakers are located on the Main DC Power Distribution Panel located under the center aisle floor. Control for each dual level heated windshield is provided by a windshield anti-ice controller, a low heat relay, a high heat relay, the subpanel switch, and a five amp circuit breaker. The two control circuit breakers are located on the right hand circuit breaker panel. The two windshield anti-ice controllers and the four anti-ice relays are located on the Windshield Deice Control Panel which is located on the left cabin side of the forward cabin bulkhead.

Each windshield contains two terminal blocks: one at the lower inboard comer and one at the lower outboard corner of each windshield. The lower inboard corner terminal block contains the low power electrical lead. The lower outboard corner terminal block contains the high power lead, the two sensing leads, and the ground lead respectively, from outboard to inboard.

ICE AND RAIN PROTECTION
WINDSHIELD WIPER
DESCRIPTION AND OPERATION

1. GENERAL

The windshield wiper installation consists of a motor, arm assemblies, drive shafts and converters located forward of the instrument panel. The system includes a control switch, located in the overhead panel, and the system circuit breaker in the right hand subpanel.

ICE AND RAIN PROTECTION
WINDSHIELD WIPER
MAINTENANCE PRACTICES

1. WINDSHIELD WIPER MOTOR AND CONVERTER

A. Removal

For access to the windshield wiper motor and converters on U-1 through U-164, except U-50, it is necessary to remove two radio sets from the radio panel or work from a prone position up and behind the instrument panel. On U-50, U-165 and After, a large access door is located in the aft bulkhead of the nose baggage compartment. Access to the windshield wiper motor and converters may be gained through this access hole. The motor and converters must be removed as a unit, then disassembled.

(1) Relax tension on the wiper arm assembly by rotating the wiper arm adjustment bolt (1) counterclockwise (Ref. Figure 201).

(2) Remove the wiper arm assembly.

(3) Remove the drive shaft doubler (2) by removing the four attaching screws from each doubler.

(4) Disconnect the electrical wiring from the motor by cutting the lead wires at the existing permanent splices.

(5) Remove the attaching bolts which secure the motor and converters to the airplane structure and remove the motor and converters as a unit.

B. Installation

(1) With the motor and converters assembled as a unit, place the assembly in position and install the attaching bolts. Do not tighten the bolts at this time (Ref. Figure 201).

NOTE: When the converters are attached to the motor, the wiper arm assembly drive shaft of the RH converter must be rotated to the extreme counterclockwise position and the LH drive shaft must be in the extreme clockwise position.

(2) Position the drive shaft doublers and install the attaching screws.

(3) Connect the motor lead wires to the airplane wiring with permanent splice connectors.

(4) Tighten the motor and converter assembly attach bolts.

(5) Install the wiper arms and adjust the arm spring tension as explained in windshield wiper adjustment.

2. WINDSHIELD WIPER

A. Adjustment

NOTE: Do not operate the windshield wiper on a dry windshield.

Run the windshield wiper to the park position. If the wiper blades do not come to rest approximately two inches from the center post of the windshield or if the tension of the wiper arm at the wiper attaching point is not within 4.5 to 5.5 pounds, adjust the windshield wiper in the following manner:

(1) Remove the spring tension on the hub end of the wiper arm by inserting a 1/8 inch diameter pin through the hole in each side of the wiper arm.

(2) Loosen the allen screw on the hub end of the wiper arm until the hub can be spread enough for removal.

(3) Remove the wiper arm and reposition it two inches from the center post by rotating the bronze adjusting sleeve on the drive shaft in the same direction that the wiper arm must be moved to obtain the two inch setting.

(4) Install the arm assembly on the drive shaft, making sure that the arm is pulled up until the washer is in the counterbore of the hub.

(5) Tighten the allen screw until the wiper hub is anchored securely to the drive shaft.

(6) Remove the pin from the holes in the wiper arm.

(7) Set the wiper arm to a tension load of 4.5 to 5.5 pounds at the wiper attaching point. Rotate the spring adjusting screw on top of the wiper arm with the wrench (P/N XW20509), refer to SPECIAL TOOLS in Chapter 12 designed for this purpose. Turn the screws clockwise to decrease or counterclockwise to increase the adjusting spring tension.

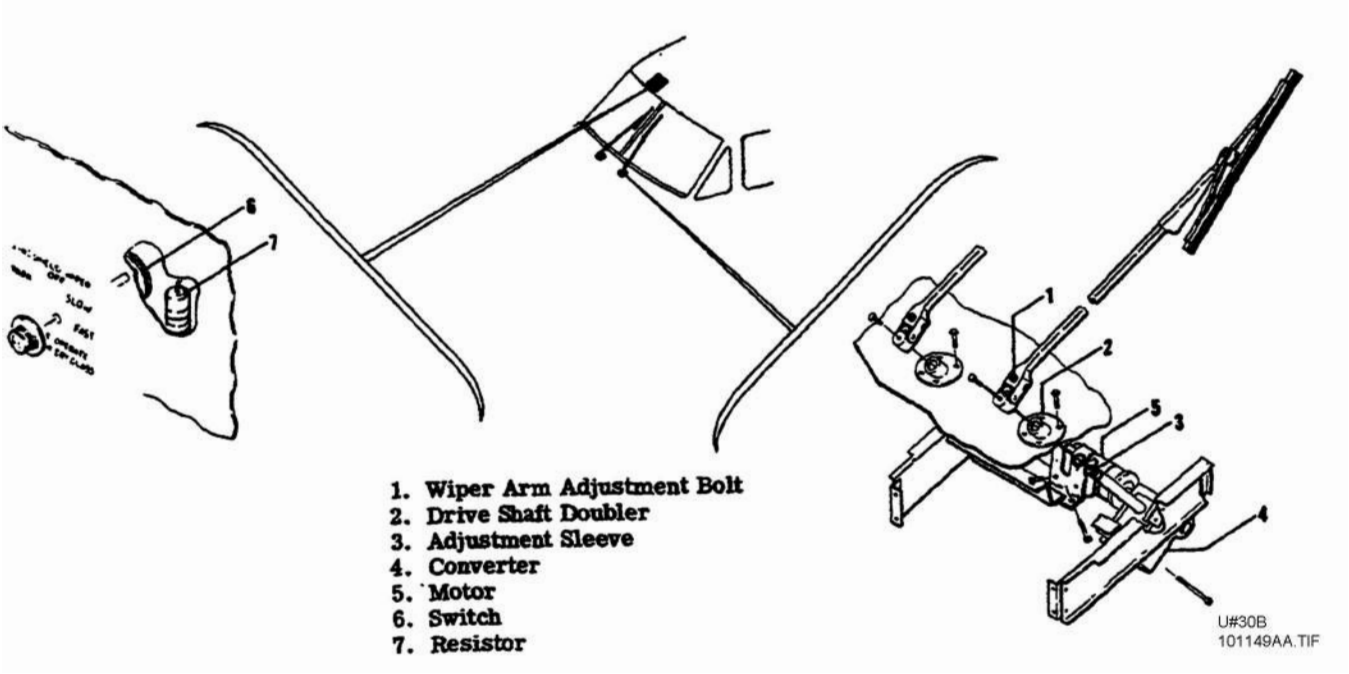

Figure 201
Windshield Wiper

ICE AND RAIN PROTECTION
PROPELLER ELECTRIC DEICER SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

A. U-1 thru U-164, except U-50

The propeller electric deicer system includes an electrically heated boot for each propeller blade, brush assemblies, slip rings, an ammeter and an ON-OFF switch on the subpanel. When the switch is turned ON, the ammeter on the subpanel registers the amount of current (14 to 18 amperes) passing through the system. If the current rises beyond the switch limit. an integral circuit breaker will cut off the power to the timer. The current flows from the timer, located under the center aisle floor aft of the main spar, to the brush assembly mounted on the front of the 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 deicer boots on the propeller blades. Heat from the boots reduces the grip of the ice. which is then removed by the centrifugal effect of propeller rotating and the blast of air stream. Power to the two heating elements 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, LH inboard. Since each of these phases is 30 seconds in duration, the timer makes a complete cycle every two minutes. Whenever the timer switches to the next phase of operation, the ammeter on the LH subpanel will register a momentary deflection.

A manual bypass control switch located immediately to the right of the propeller deicer switch, U-152 and After, is incorporated in the deicer system as a back-up to the normal system. The switch, spring-loaded to the OFF position, must be held in the position selected. Placing the switch in either the left or right position causes the current to by-pass the timer and follow the circuitry of the manual system directly to all the heating elements of the position selected. Since all the heating elements of the position selected will be in operation at the same time, the current load will be approximately twice that of the normal system.

NOTE: Do not hold the switch in the position selected any longer than it takes to melt the accumulated ice.

B. U-50, U-165 and After

Electrothermal propeller deice boots, mounted on the propeller blades, are heated by electrical resistance. Direct current for deice boot heating is supplied through a system of controls by two brushes which run on dual slip rings mounted on the propeller assembly.

The brushes used on this installation are of the modular block type. The dual slip rings are supplied as an assembly. More information concerning maintenance of the modular brush blocks and slip ring assemblies will be found later in this chapter under specific headings.

Current for automatic operation of the deice timing control and all the deice boots is taken from a single 25-ampere switch/circuit breaker located in the ICE PROTECTION section of the pilot's subpanel. When the switch/circuit breaker is placed in the ON position, the deice timer begins to run and automatic cycling of the deice boots is initiated. At intervals of approximately 90-seconds in duration, the timer alternately cycles power to the RH and LH propeller deice boots through the normally closed contacts of the RH and LH manual override relays.

Current for manual operation of the deice boots is supplied by a pair of 20-ampere circuit breakers through the normally open contacts of the RH and LH manual override relays, which are energized through a control switch with a spring return to the OFF position. When the manual override switch is held in the ON position, the propeller deice timer is bypassed. and both RH and LH propeller deice boots operate continuously until the switch is released to the OFF position.

Current for automatic operation is indicated by an ammeter located on the upper portion of the pedestal. The ammeter shunt, both manual override relays and deice timer are located on the propeller deice equipment assembly rack.

NOTE: During manual operation of the deice system, the deice ammeter will remain at zero. A generator loadmeter indication of 12% (equivalent to approximately 30 amperes) will confirm deice operation in the manual mode.

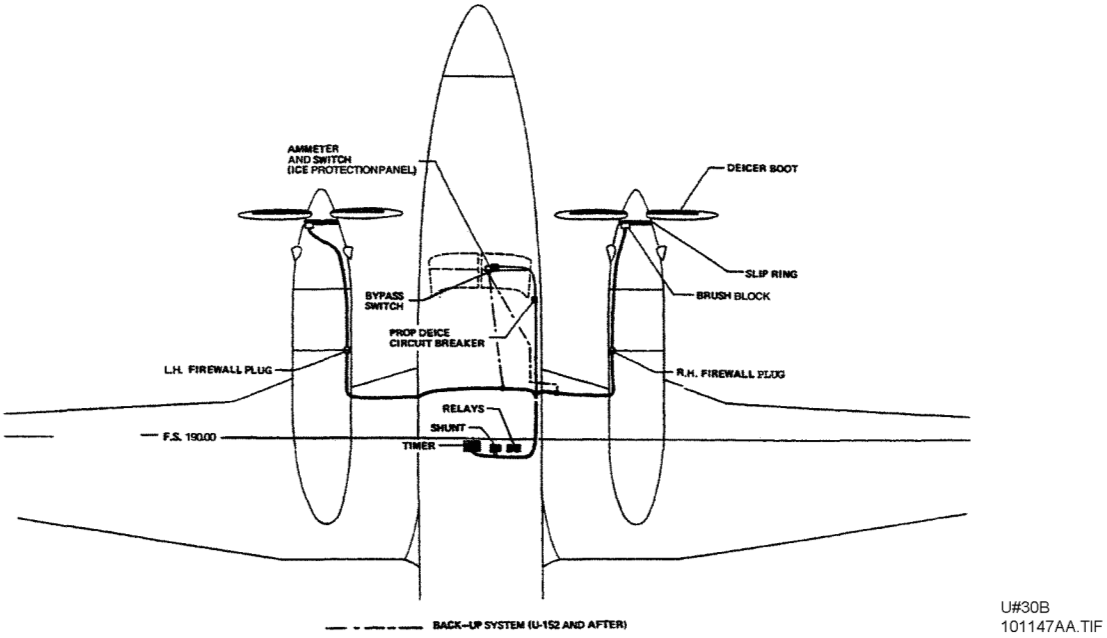

Figure 1
Propeller Electrical Deice System (U-1 thru U-164, except U-50)

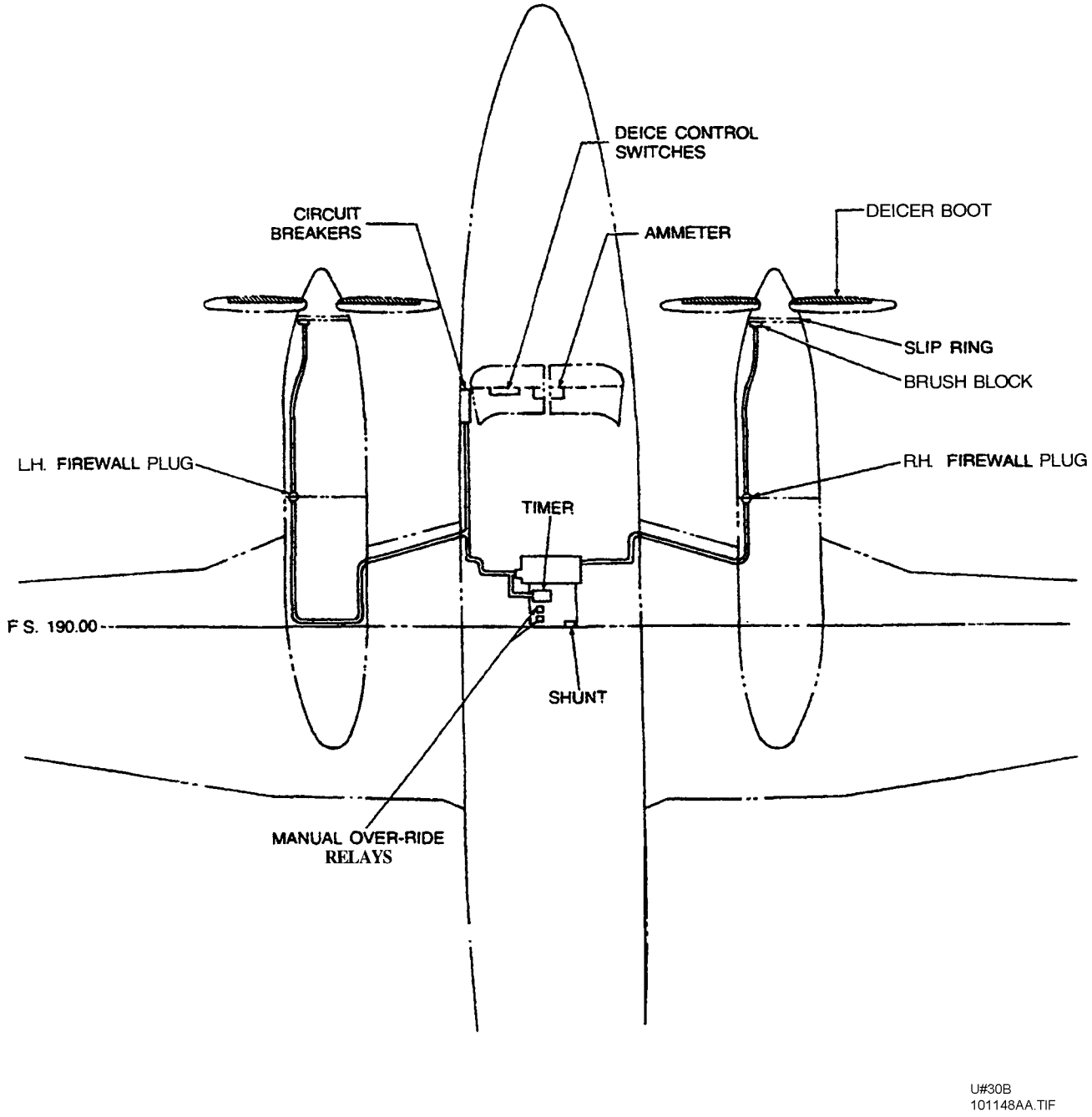

Figure 2
Propeller Electrical Deice System (U-50, U-165 and After)

ICE AND RAIN PROTECTION
PROPELLER ELECTRIC DEICER SYSTEM
TROUBLESHOOTING

1. PROPELLER ELECTRIC DEICER BOOT

For troubleshooting of the propeller electric deicer systems, refer to the appropriate manual for B.F. Goodrich or McCauley deicer systems in the Model 99 Airliner Series Component Maintenance Manual, P/N 99-590015-5 or subsequent.

ICE AND RAIN PROTECTION
PROPELLER ELECTRIC DEICER SYSTEM
MAINTENANCE PRACTICES

1. PROPELLER ELECTRIC DEICER BOOT

For maintenance of the propeller electric deicer systems, refer to the appropriate manual for B.F. Goodrich or McCauley deicer systems in the Model 99 Airliner Series Component Maintenance Manual, P/N 99-590015-5 or subsequent.

List of Effective Pages

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31-CONTENTS	1	Nov 1/10
31-10-00	201 thru 202	Nov 1/10

CHAPTER 31 - INDICATING/RECORDING SYSTEMS

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INSTRUMENTS

31-10-00

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Instruments 201
Access 201
Flight Instruments 202
Radio Equipment 202
Engine Instruments 202

INDICATING/RECORDING SYSTEMS
INSTRUMENTS
MAINTENANCE PRACTICES

1. INSTRUMENTS

NOTE: Refer to Chapter 39-10-00 for removal and installation of instruments and the instrument panel for U-50, U-165 and After.

A. Access

The instrument panel on the Model 99 Airliner is divided into three sections: the center section contains the radio equipment and the engine instruments while the two side sections contain the dual flight instruments (Ref. Figure 201).

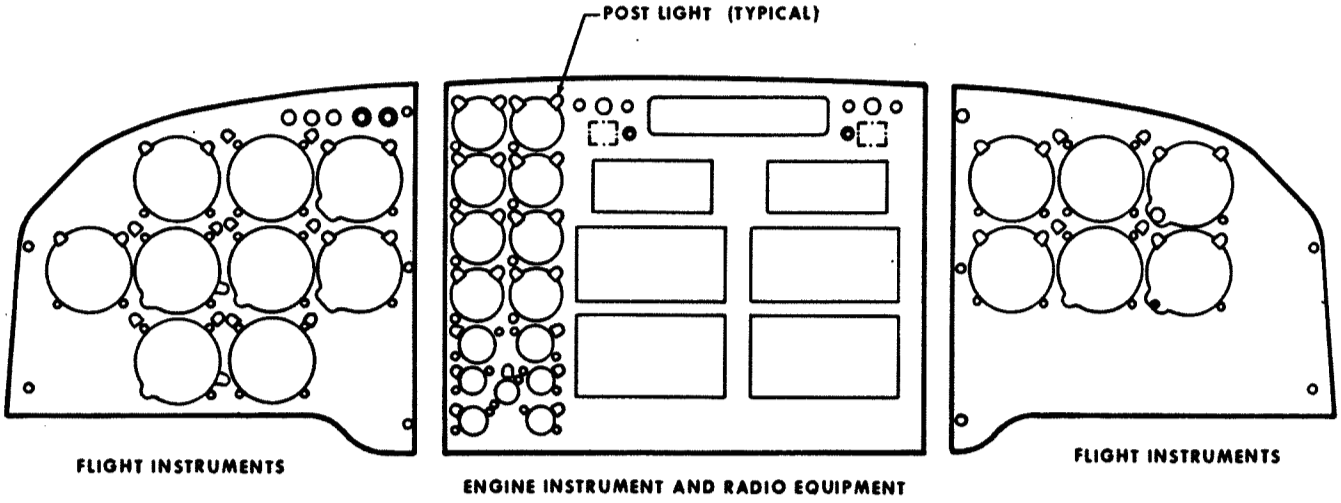

Figure 201
Instrument Access

B. Flight Instruments

Access to the flight instruments is provided by five mounting screws in the panel section. Remove the five mounting screws and tilt the panel section aft; the plumbing connected to the instruments is of sufficient length to provide easy access to the desired flight instrument for maintenance. Disconnect the plumbing connection at the base of the desired instrument and remove the instrument mounting screws at the base of the instrument.

Installation of a flight instrument to the side sections of the instrument panel is accomplished by placing the instrument in position in the panel and securing with the mounting screws. Connect the plumbing at the base of the instrument. Move the panel back in place and secure with the five mounting screws.

C. Radio Equipment

The radio equipment, mounted in the center section of the panel, is the quick removable type. Each component is individually mounted in the panel. Removal of the radio components is accomplished by removing the two attaching screws from each side and sliding the entire component out of the panel. The wiring connections are of sufficient length to permit the component to slide free of the panel. Disconnect the quick disconnect plug from the rear of the unit and remove from the panel.

Each component of the radio equipment is individually mounted in the center section of the panel. Installation may be accomplished by first connecting the appropriate quick disconnect plug to the rear of the unit and then sliding the unit into position in the panel. Secure with the attaching screws at each side of the unit.

NOTE: To avoid marring the finish, wrap padding around the control columns just below the instrument panel.

D. Engine Instruments

Engine instruments may be removed from the center section in the following manner: pull the post light shield off to provide clearance for the instruments. Loosen the retaining screw, located below and to the left of the instrument. and the instrument retaining clamp. Sufficient wiring is provided to allow the instrument to be pulled straight out.

CAUTION: *When installing engine instruments, do not overtighten engine instrument clamps.*

To install an engine instrument in the center section panel, attach the appropriate wiring to the back of the instrument to be installed. Place the instrument in place on the panel and secure with the instrument retaining clamp and screw. Install the post light shield.

NOTE: On U-152 through U-164, all instruments are front mounted to a single instrument panel.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

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LANDING GEAR
GENERAL
DESCRIPTION AND OPERATION

1. GENERAL

A. Mechanical Landing Gear

The landing gear is operated by a split-field series-wound motor located on the forward side of the main center section spar. One field is used to drive the motor in each direction. To prevent over travel of the gear, a relay simultaneously breaks the power circuit to the motor and makes a complete circuit through the armature and the unused field winding. The motor then acts as a generator and the resulting electrical load on the armature stops the gear almost instantly.

The landing gear motor is controlled by the landing gear extension switch located in the center of the subpanel. The main gear actuators are driven by the torque shafts from the motor gearbox and the nose gear actuator is driven by a Duplex chain from a sprocket on the gearbox torque shaft. A spring-loaded friction clutch within the gearbox protects the system in the event of mechanical malfunction. A 200 ampere, remote circuit breaker located on the power distribution panel protects the system from electrical overload. Notched hook and plate attachments fitted to each main gear upper drag leg, and an over-center action of the nose gear drag leg assembly provide positive mechanical down-locks. A jack-screw in each actuator holds the gear in the retracted position.

A solenoid operated down-lock latch prevents the landing gear control handle from being raised while the plane is on the ground. The landing gear safety switch releases the latch when the plane leaves the ground. If necessary, the latch can be manually overridden by pressing down on the red button placarded DOWN LOCK RELEASE.

To prevent accidental gear retraction on the ground, a safety switch on the right main strut breaks the control circuit whenever the strut is compressed.

CAUTION: *Never rely on the safety switch to keep the gear down while taxiing or on landing or take-off roll. Always check the position of the landing gear switch.*

The Model 99 Airliner landing gear incorporates Beech air-oil type shock struts that are filled with both compressed air and hydraulic fluid. Their correct inflation should be assured before each flight.

CAUTION: *Never tow or taxi with a flat strut. Even brief towing or taxiing with a deflated strut can cause severe damage.*

LANDING GEAR
GENERAL
TROUBLESHOOTING

1. MECHANICAL LANDING GEAR CONTROL SYSTEM

Troubleshooting - Mechanical Landing Gear Control System

INDICATION	PROBABLE CAUSE	REMARKS
1. Landing gear will not retract or lower.	a. Circuit breaker tripped in control or motor circuit.	
	b. Loose motor ground connection.	
	c. Open circuit.	c. Test continuity: (1) Control circuit breaker to landing gear switch. (2) Ground fault relay to landing gear power relay. (3) Power relay to motor. Check operations at power relay.
2. Landing gear will lower but not retract.	a. Open circuit between landing gear switch and control relay.	a. Test continuity from up terminal of landing gear switch and energizing wire terminal for the up limit switch.
	b. Up solenoid of control relay inoperative.	
	c. Open circuit between control relay and up winding in motor.	c. Test continuity from up solenoid power terminal to up winding of motor.
3. Landing gear will retract but not lower.	a. Open circuit.	a. Test continuity from down terminal of landing gear switch to down limit switch and energizing wire terminal of the down solenoid.
	b. Down limit switch inoperative.	
	c. Open circuit between control relay and down winding of motor.	c. Test continuity from down solenoid power terminal to down winding of motor.
4. Circuit breaker tripping in circuit.	a. Grounded control circuit.	a. Test for ground: (1) Between circuit breaker and landing gear switch. (2) Between landing gear switch and control relay on side (up or down) of control circuit that is tripping circuit breaker.

Troubleshooting - Mechanical Landing Gear Control System

INDICATION	PROBABLE CAUSE	REMARKS
4. Circuit breaker tripping in circuit. (Cont'd)	b. Grounded motor circuit.	b. Test for ground: (1) Between bus and control relay. (2) In motor leads (with leads disconnected from motor).
	c. Mechanical defect in gear causing overload.	

LANDING GEAR
GENERAL
DESCRIPTION AND OPERATION

1. LANDING GEAR

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from the jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

WARNING: When performing an inspection or maintenance on a Hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all landing gear are down and locked prior to removing the airplane from jacks.

A. Hydraulic Landing Gear

The nose and main landing gear assemblies are operated by a hydraulic power pack, located forward of the main center section spar (U-1 thru U-164, except U-50) or aft of the main center section spar (U-50, U-165 and After) under the center isle floorboard. The power pack consists primarily of a pump, a 28vdc motor, valve body, two section reservoir, filter, gear sector solenoid and uplock pressure switch. Associated plumbing for a normal extend mode, emergency extend mode and normal retract mode is routed from the power pack to each main actuator and the nose actuator. The plumbing for normal extend mode and emergency extend mode is fitted as separate plumbing to the end cap of each actuator. The retract mode plumbing is fitted to the piston rod end of each actuator. The plumbing is color coded (U-50, U-165 and After) (Ref. Table 1).

Table 1

Color	System
Gray	Vent Line
Purple	Fill Line
Yellow	Hand Pump Suction
Brown	Hand Pump Pressure
White	Normal Extension
Blue	Normal Retraction

Nose gear downlock is accomplished by an internal mechanical lock in the nose gear actuator and the overcenter action of the nose drag leg assembly. Notched hook, lock link and lock link guide attachments fitted to each main gear upper drag leg provide a positive mechanical downlock action for the main gear. As the landing gear moves to the down and locked position, the downlock switches are actuated and interrupt current to the pump motor.

The landing gear is held in the retracted position by positive hydraulic pressure. When the system pressure reaches the high pressure limit, a pressure switch mounted on the pump motor assembly will interrupt current to the pump motor. The same pressure switch will activate the pump motor should system pressure drop to the low pressure limit on the hydraulic power pack assembly.

NOTE: On the (U-1 thru U-164, except U-50) Hydraulic power pack (P/N 99-388002-1) was originally installed and has a high pressure limit of 1,600 to 1,650 psig and a low pressure limit of 1,300 ± 100 psig. If the -1 has been replaced with (Kit No. 99-8019-1), contact Hawker Beechcraft Technical Support for information on the high and low pressure limitations.

On the (U-50 and U-165 thru U-253) Hydraulic Power pack (P/N 99-388002-5) was originally installed and has a high pressure limit of 1,800 to 1,850 psig and a low pressure limit of 1,400 psig ± 100 psig. If the -5 has been replaced with (Kit No. 99-8019-3), contact Hawker Beechcraft Technical Support for information on the high and low pressure limitations.

An accumulator charged to 600 ± 30 psig is located under the RH seat deck forward of the pump on U-1 thru U-164, except U-50. On U-50, U-165 and After, the accumulator mounted under the center isle floorboard aft of the power pack is charged to 750 ± 38 psig. The accumulator is designed to absorb line surges during the retract mode, thereby reducing wear on the motor and pump assembly.

A landing gear control handle, placarded UP DN, is located on the pilot's subpanel. A solenoid operated downlock latch prevents the landing gear control handle from being raised while the airplane is on the ground. The landing gear safety switch releases the latch when the airplane leaves the ground. If necessary, the latch can be manually overridden by pressing down on the red button placarded DOWN LOCK RELEASE.

CAUTION: *Never rely on the safety switch to keep the gear down while taxiing, landing or on takeoff roll. Always check the position of the landing gear switch.*

To prevent accidental gear retraction on the ground, a safety switch on the RH main strut breaks the control circuit whenever the strut is compressed.

WARNING: Never use oxygen to inflate the struts. Use only compressed air or nitrogen to inflate the strut.

CAUTION: *Never tow or taxi with a flat strut. Even brief towing or taxiing with a deflated strut can cause severe damage to the strut.*

The Model 99 Airliner landing gear incorporates air-oil type shock struts that are filled with both compressed air and hydraulic fluid. Their correct inflation should be assured before each flight.

The landing gear control circuit is protected by a two ampere circuit breaker located on the circuit breaker panel. The landing gear remote relay receives power from the landing gear remote limiter or circuit breaker; both are located in the power distribution panel. Power is then supplied from the relay to the motor. A time delay relay, located on the lower electrical equipment panel, will interrupt current to the motor and trip the landing gear control circuit breaker if continuous motor operation exceeds the maximum time limit. A gear selector solenoid, located on the valve body end of the pump, is energized when the landing gear control handle is in the UP position; this actuates the gear select valve, allowing system fluid to flow to the up side of the system. The gear select valve is spring loaded in the down position and will only move to the up position when energized.

A HYD FLUID LOW light located on the instrument panel will illuminate in the event the hydraulic fluid level in the landing gear power pack becomes critically low. A sensing unit mounted on the motor end of the power pack provides the necessary switching circuitry to illuminate the light. Three types of sensing units are used in the power pack: (P/N 99-388002-1) and (Kit No. 99-8019-1), has a thermistor type sensing unit. (P/N 99-388002-5) has an optically operated sensing unit with an integrated self test circuit. The internal self test circuit is energized by a switch on the instrument panel and functionally tests the sensing unit's internal circuitry. (Kit No. 99-8019-3), has a liquid level sensor which incorporates a low pressure thermal expansion valve which eliminates a pressure buildup in the "up" side of the system when the landing gear is down and locked.

WARNING: After an emergency landing gear extension has been made, do not move any landing gear controls or reset any switches or circuit breakers until cause of the malfunction has been determined and corrected.

A hand pump, placarded LANDING GEAR EMERGENCY EXTENSION, is located on the floor between the pilot's and copilot's seats. The pump is used when emergency extension of the gear is required.

CAUTION: *If the red knob on the service valve is pushed down while the landing gear is retracted, the electrical power on, and the landing gear control handle is in the DN position, the landing gear will extend immediately.*

A service valve, located just left of the pump and motor assembly (U-1 thru U-164, except U-50) or aft of the power pack (U-50, U-165 and After), may be used in conjunction with the hand pump to raise the gear for maintenance practices. With the airplane on jacks and an external power source attached, pull up on the red knob located on the top of the service valve. On airplanes (U-50, U-165 and After), it will be necessary to pull the detent pin and move the safety device before the knob can be pulled up; to prevent the knob from being pushed down while the airplane is being serviced, slide the block attached to the service valve under the knob. The hand pump can then be pumped to raise the gear to the desired position. After the required maintenance has been performed, push the red knob down, and use either the hand pump or the motor to lower the landing gear. On airplanes (U-50, U-165 and After), remove the block from under the knob, and after the knob is pushed down, install the safety device to prevent the knob from being pulled up.

A fill can, located just inboard of the LH nacelle and forward of the main spar, contains a cap and dipstick assembly, marked FULL and ADD, for convenience of maintaining system fluid level. A line, plumbed to the upper portion of the fill can, is routed overboard to act as a vent.

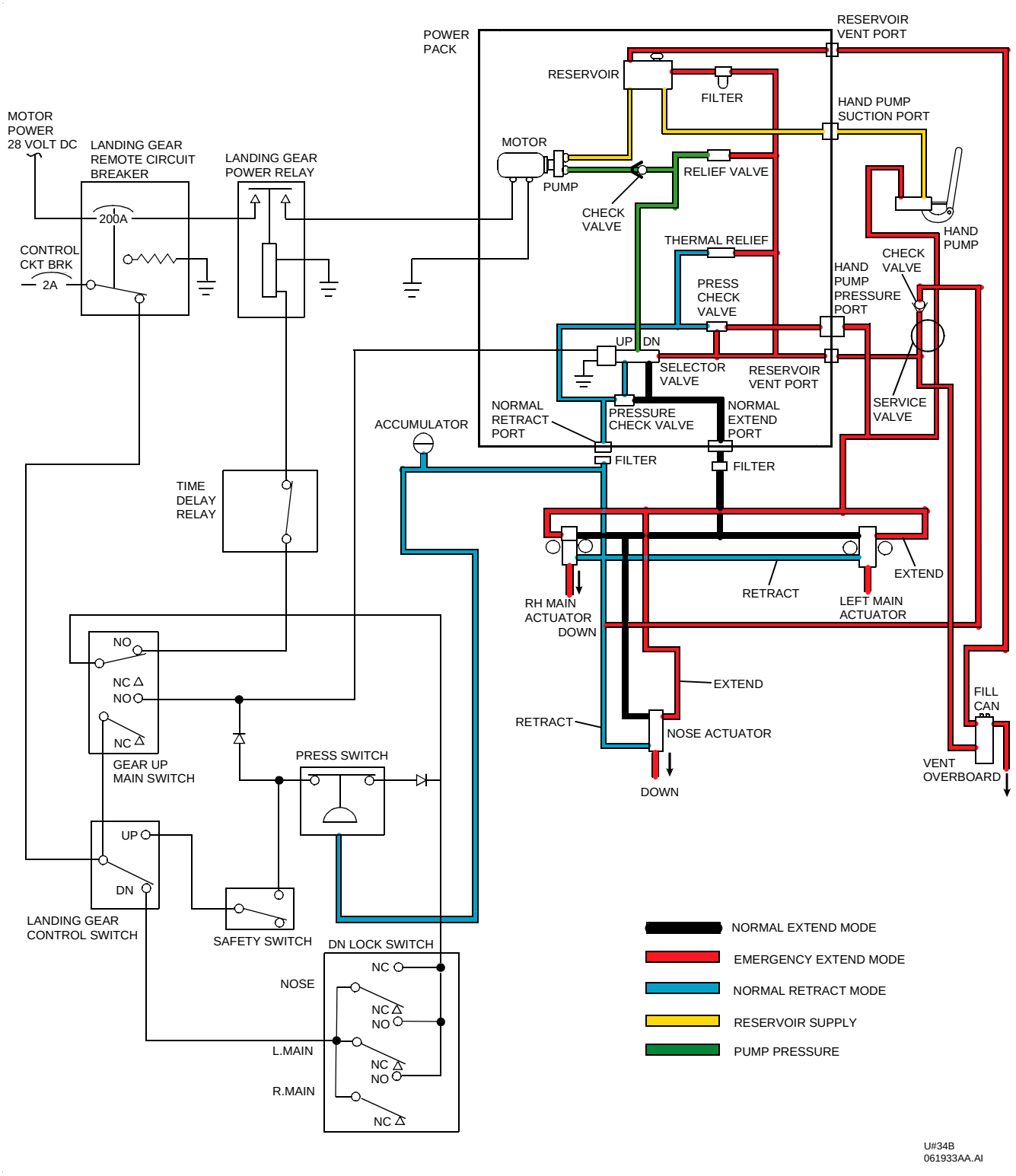

Figure 1
Hydraulic Landing Gear System Schematic (U-1 thru U-164, Except U-50)

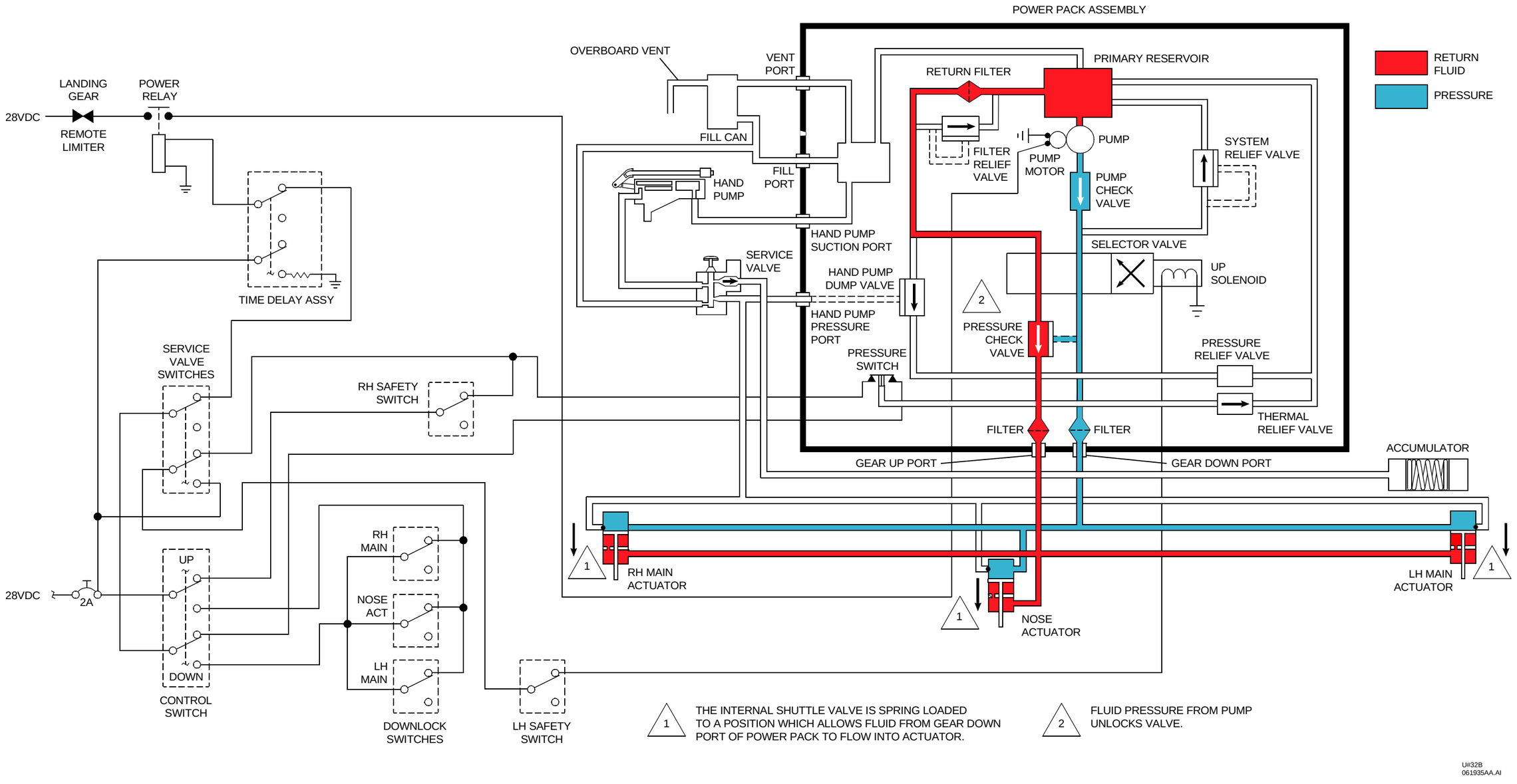

Figure 2
Hydraulic Landing Gear Extending Schematic
(U-50, U-165 and After)

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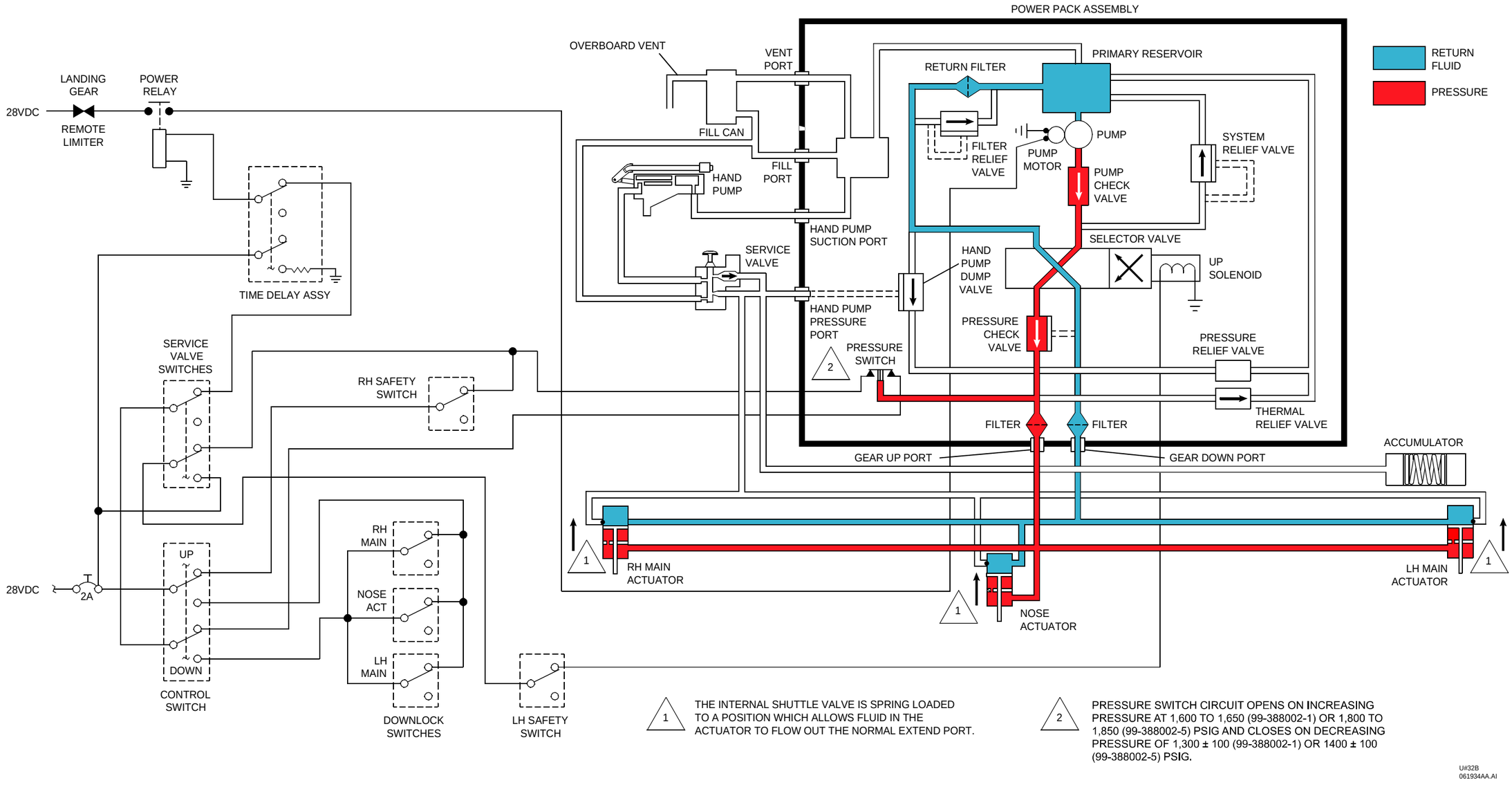

1 THE INTERNAL SHUTTLE VALVE IS SPRING LOADED TO A POSITION WHICH ALLOWS FLUID IN THE ACTUATOR TO FLOW OUT THE NORMAL EXTEND PORT.

2 PRESSURE SWITCH CIRCUIT OPENS ON INCREASING PRESSURE AT 1,600 TO 1,650 (99-388002-1) OR 1,800 TO 1,850 (99-388002-5) PSIG AND CLOSES ON DECREASING PRESSURE OF 1,300 ± 100 (99-388002-1) OR 1400 ± 100 (99-388002-5) PSIG.

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Figure 3
Hydraulic Landing Gear Retraction Schematic
(U-50, U-165 and After)

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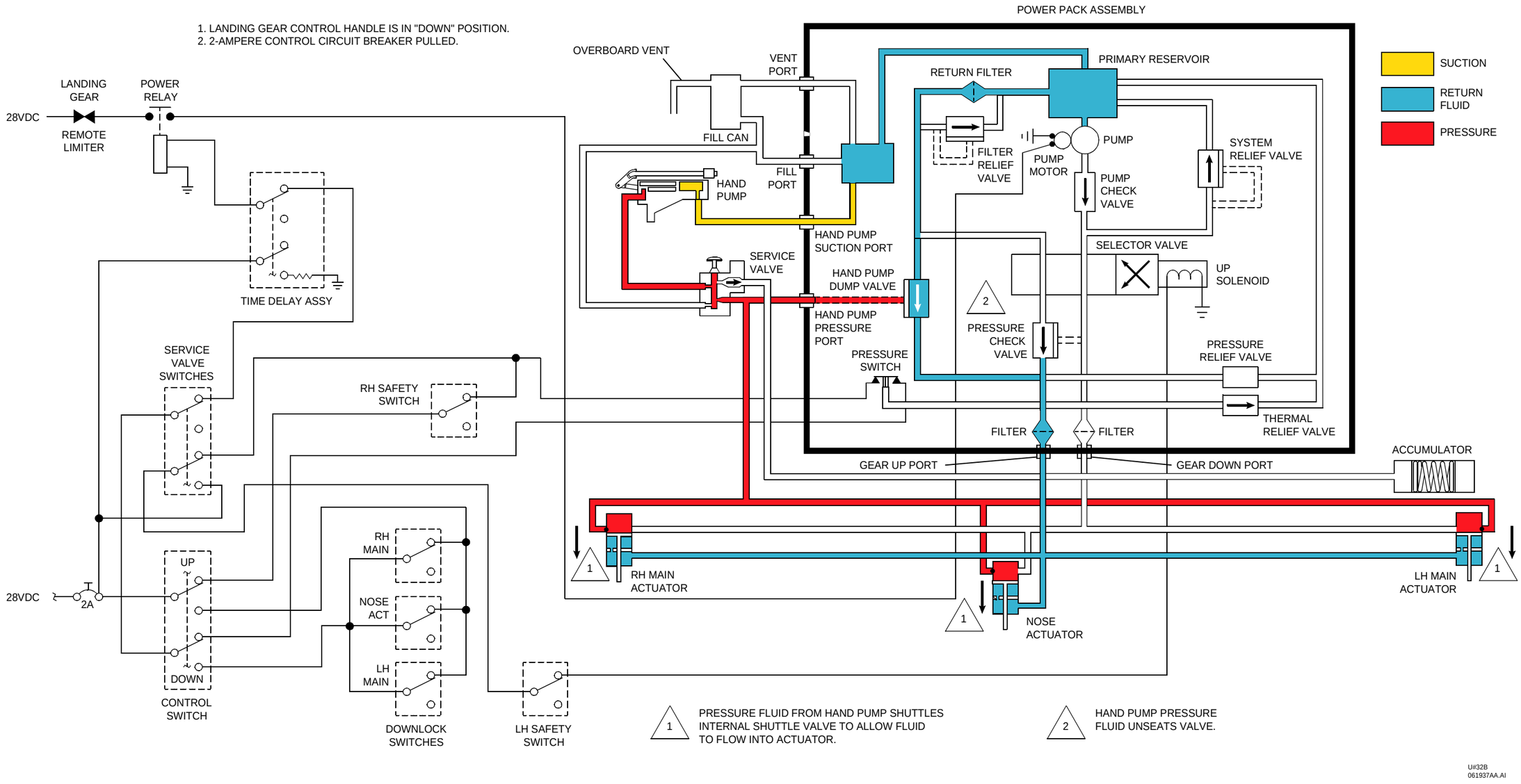

Figure 4
Hydraulic Landing Gear Emergency Extension Schematic
(U-50, U-165 and After)

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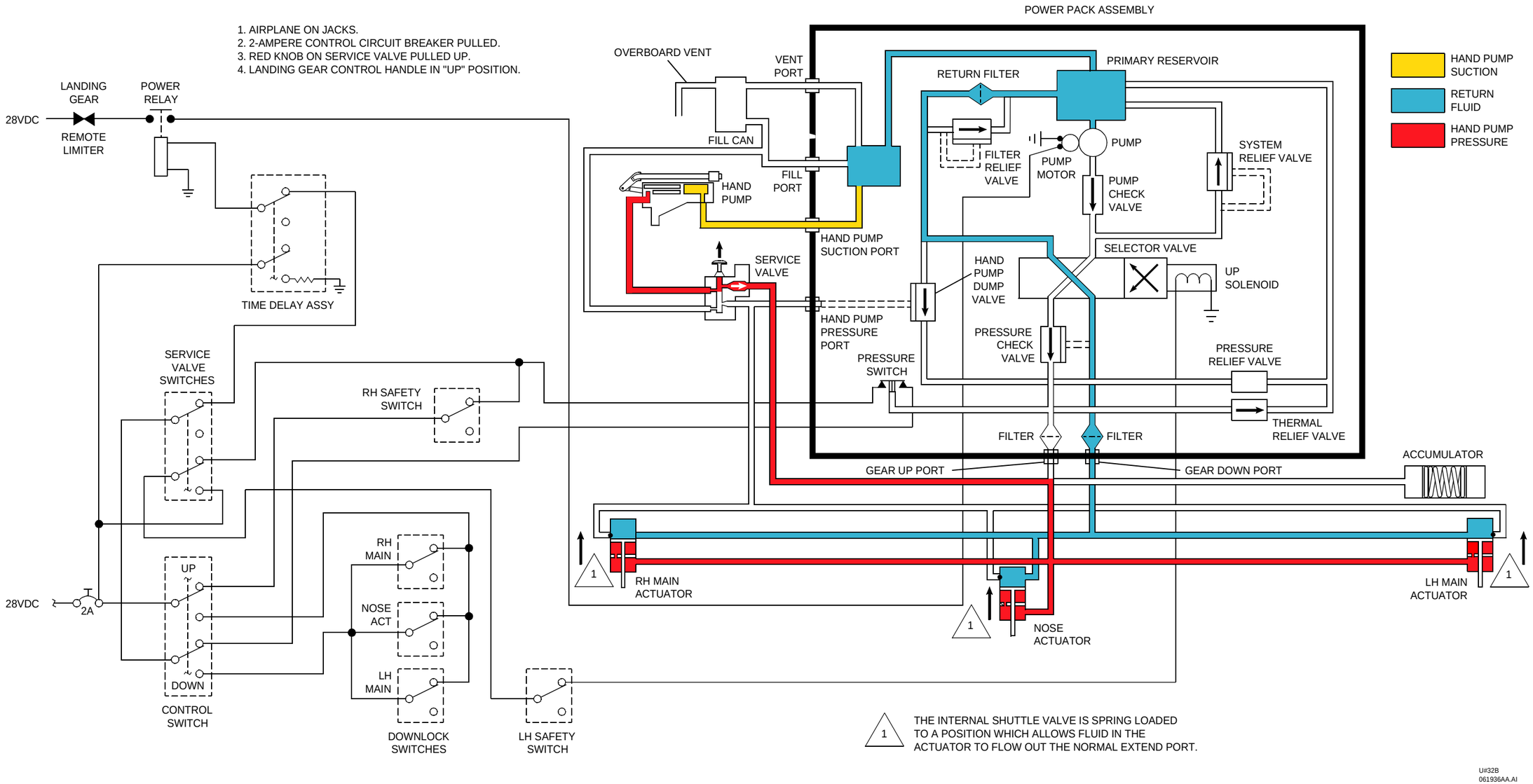

Figure 5
Hydraulic Landing Gear Maintenance Retraction Schematic
(U-50, U-165 and After)

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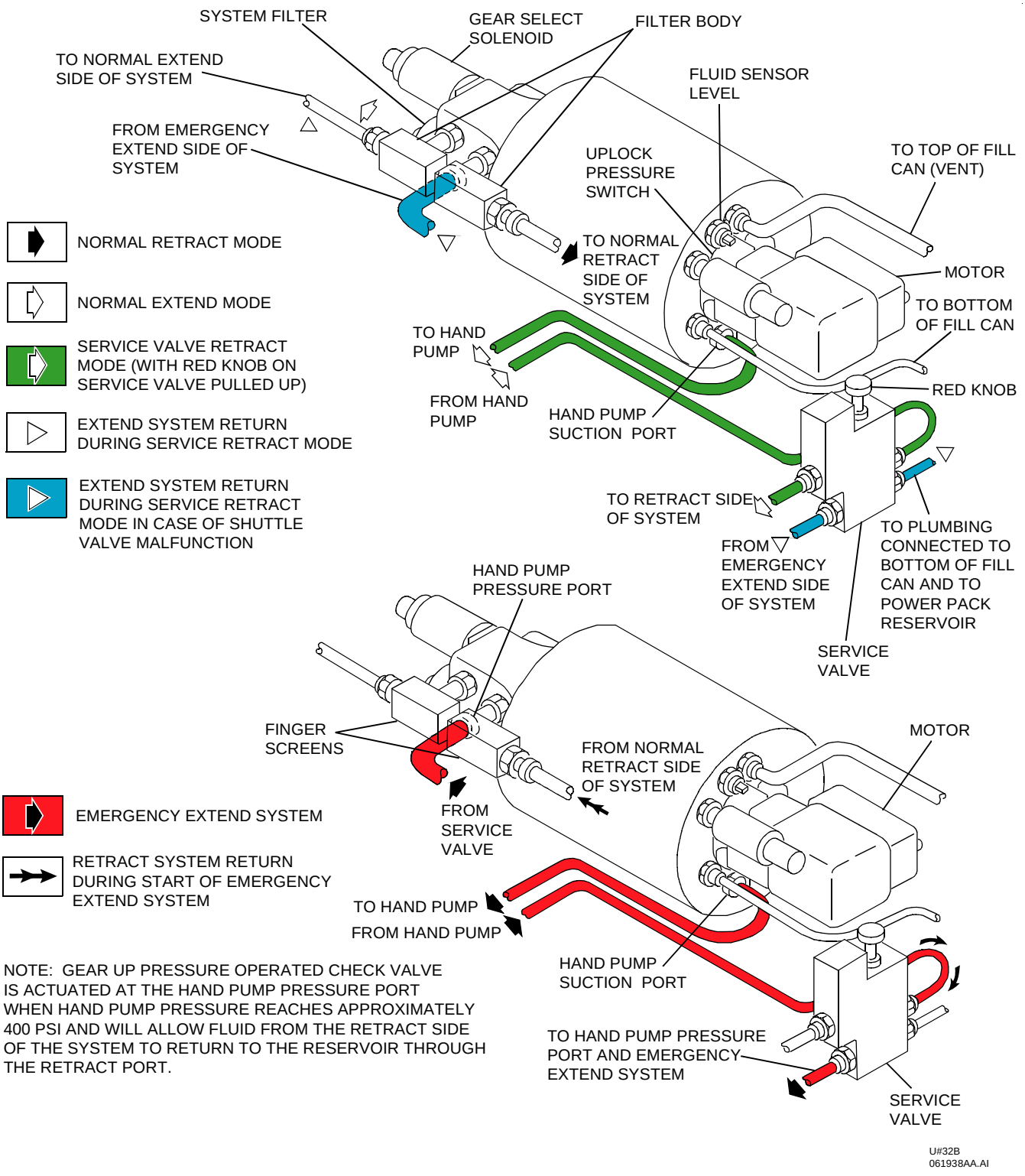


Figure 6
Hydraulic Landing Gear Fluid Flow Schematic

LANDING GEAR
GENERAL
TROUBLESHOOTING

1. HYDRAULIC LANDING GEAR SYSTEM

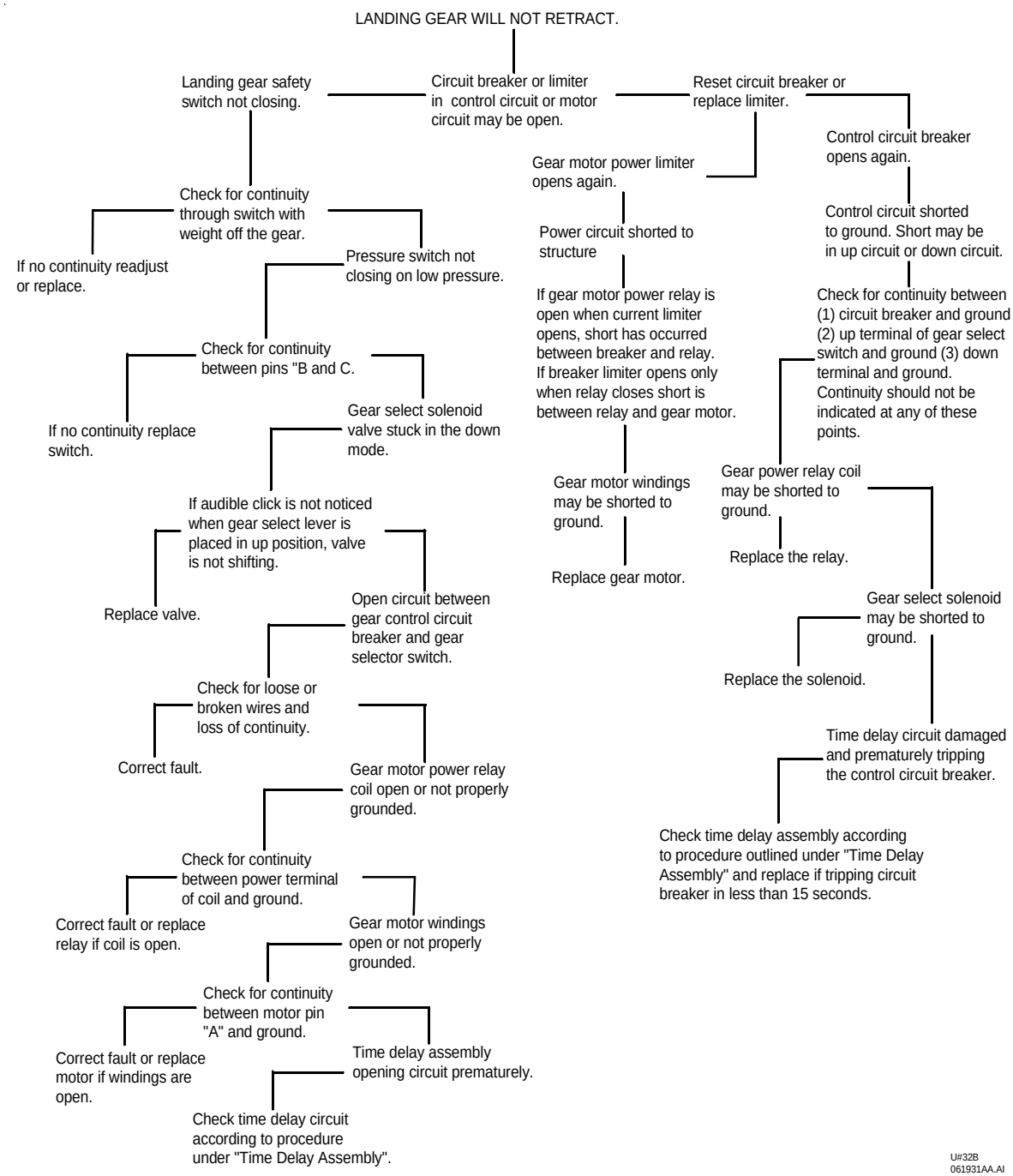


Figure 101 (Sheet 1 of 2)
Hydraulic Landing Gear System - Troubleshooting

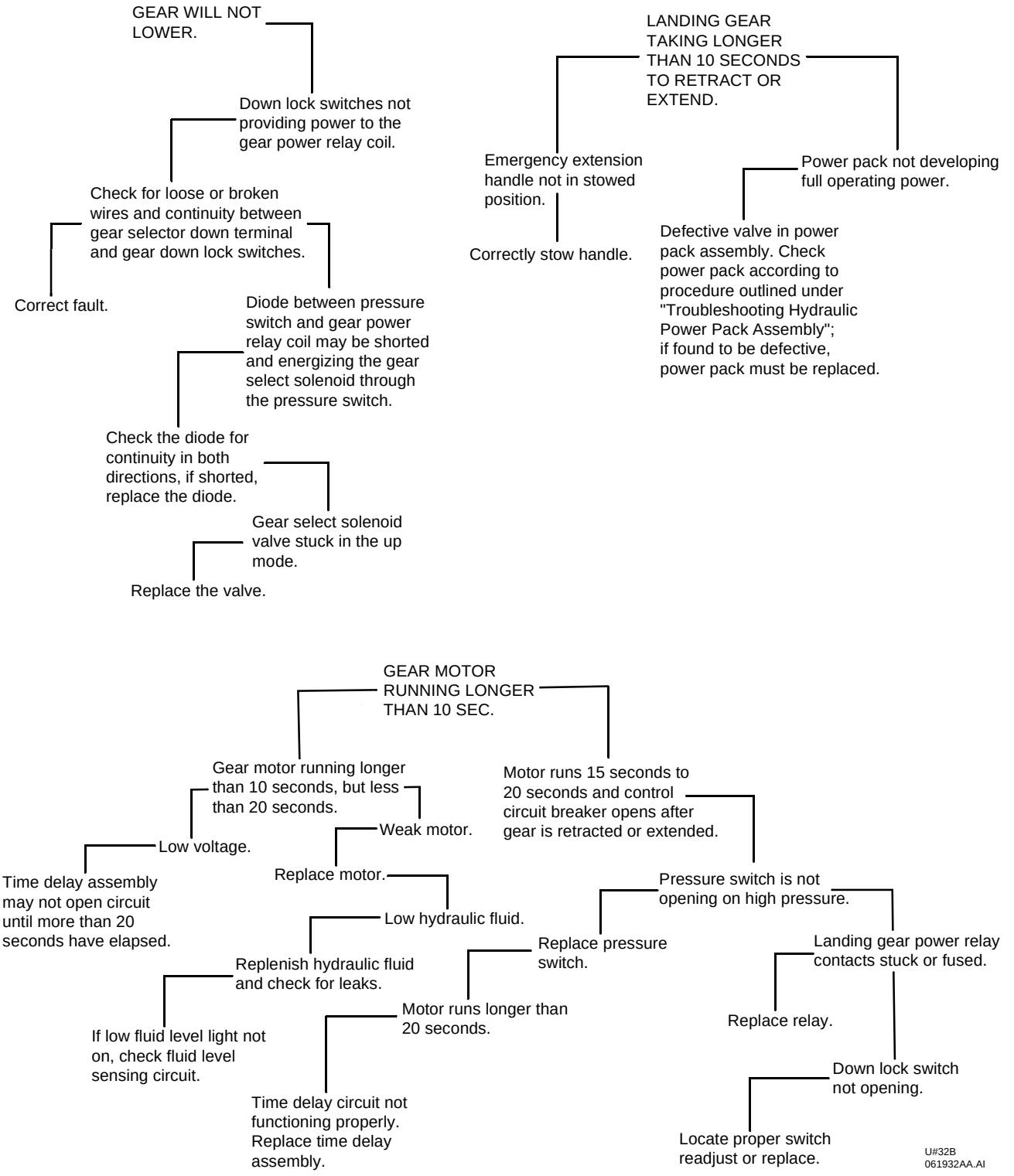


Figure 101 (Sheet 2 of 2)
Hydraulic Landing Gear System - Troubleshooting

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

WARNING: When performing an inspection or maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all three landing gear are down and locked prior to removing the airplane from jacks.

2. TIME DELAY ASSEMBLY

In the normally operating landing gear control circuit, the time delay assembly would apply ground, through a current limiting resistor, to the landing gear control power circuit, causing the 2 ampere gear control circuit breaker to open, preventing further damage to the system. Should the time delay circuit for the landing gear malfunction for any reason, and should the uplock pressure switch fail to interrupt power to the landing gear power pack motor, the motor would continue to run and serious damage to the power pack assembly could result.

A test unit was designed for use in functionally checking the time delay assembly for proper operation (Ref. Figure 101). The following procedure outlines the method for positively checking the operation of the time delay assembly:

(1) Assemble the test unit as shown in Figure 102.

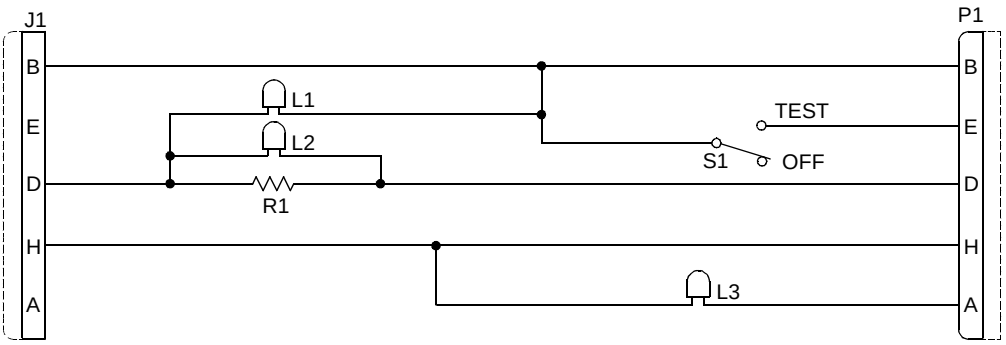
(2) Disconnect plug (P207) from the time delay assembly (A207) and connect the test unit between the plug and the time delay assembly.

(3) Apply power to the circuit by turning on the battery switch. Lamp L1 will illuminate.

(4) Place the switch on the test unit in the test position and begin timing the procedure when lamp L3 illuminates.

(5) After an appropriate time delay, 15 to 20 seconds, lamp L3 should extinguish and lamp L2 will illuminate. Lamp L2 must be dimmer than lamp L1; if not, the shunting resistor is shorted and the time delay assembly must be replaced.

(6) Remove the test unit and connect the plug to the time delay assembly connector.



TEST BOX SCHEMATIC

L1, L2, L3	327 BULBS IN APPROPRIATE LAMP HOLDERS
S1	MST-105 OR EQUIVALENT PART NUMBER ALCOSWITCH
R1	10 OHM, 25 WATT RESISTOR
J1	126-219 OR EQUIVALENT PART NUMBER AMPHENOL
P1	126-222 OR EQUIVALENT PART NUMBER AMPHENOL

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Figure 102
Time Delay Assembly Test Unit

3. HYDRAULIC POWER PACK ASSEMBLY

The following test equipment will permit troubleshooting of the hydraulic power pack assembly:

(1) Hand pump capable of delivering 2,500 psig.

(2) Pressure gage with a range of 3,000 psi.

(3) Continuity checker.

The hand pump should have a reservoir, or be equipped with fittings which will permit connection to a reservoir. The test equipment needs valves capable of controlling the application and release of hydraulic fluid to the unit under test. The test equipment should be capable of holding a pressure of at least 2,500 psig without leaks.

A. Pressure Switch

(1) Perform THREE POINT JACKING procedure (Ref. Chapter 07-00-00). All tires must be clear of the floor.

(2) Connect the hand pump and gage to the gear up port of the unit. Connect the continuity checker to the pins “B” and “C” of the electrical connector of the pressure switch. The continuity checker should indicate that there is continuity between the pins.

(3) Slowly apply pressure with the hand pump while monitoring the continuity checker and the pressure gage. The pressure switch should open, as indicated by a loss of continuity, at a pressure of 1,600 to 1,650 psig (P/N 99-388002-1) or 1,800 to 1,850 psig (P/N 99-388002-5).

(4) Decrease the pressure slowly while monitoring the continuity checker and the pressure gage. The pressure switch should close, as indicated by regaining continuity at a pressure of 1,300 ± 100 psig (P/N 99-388002-1) or 1,400 ± 100 psig (P/N 99-388002-5).

(5) Release the pressure. Disconnect the test equipment. Connect plumbing to power pack.

(6) Cycle the landing gear using the aircraft hydraulic power pack from fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(7) Ensure all three landing gear are down and locked.

(8) Perform LOWERING THE AIRPLANE AFTER THREE POINT JACKING procedure (Ref. Chapter 07-00-00).

B. Gear Uplock Check Valve and Thermal Relief Valve

(1) Perform THREE POINT JACKING procedure (Ref. Chapter 07-00-00). All tires must be clear of the floor.

(2) Connect the hand pump and the pressure gage to the gear up port.

(3) Apply pressure of 1,600 psig. Allow the pressure gage to stabilize. The gage should not indicate a loss of pressure due to an internal leakage.

(4) Slowly increase the pressure while monitoring the pressure gage. The thermal relief valve should relieve at a pressure of 2,200 ± 100 psig (P/N 99-388002-1) or 2,300 ± 100 psig (P/N 99-388002-5). Stop pumping after the thermal relief valve has cracked, and monitor the pressure gage as the pressure decreases. The pressure indication should stabilize at 1,950 ± 100 psig (P/N 99-388002-1) or 2,100 ± 100 psig (P/N 99-388002-5), indicating the pressure at which the thermal relief valve has closed.

(5) Release the pressure. Disconnect the test equipment. Connect plumbing to power pack.

(6) Cycle the landing gear using the airplane hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(7) Ensure all three landing gear are down and locked.

(8) Perform LOWERING THE AIRPLANE AFTER THREE POINT JACKING procedure (Ref. Chapter 07-00-00).

C. Pressure Relief Valve

(1) Perform THREE POINT JACKING procedure (Ref. Chapter 07-00-00). All tires must be clear of the floor.

(2) Connect the hand pump and the pressure gage to the gear down port.

(3) Apply pressure. Some leakage through the gear select valve is normal. Increase the pressure while monitoring the pressure gage. The pressure relief valve should relieve at a pressure of 1,750 ± 50 psig (P/N 99-388002-1) or 1,950 ± 50 psig (P/N 99-388002-5).

(4) Release the pressure. Disconnect the test equipment. Connect plumbing to power pack.

(5) Cycle the landing gear using the airplane hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(6) Ensure all three landing gear are down and locked.

(7) Perform LOWERING THE AIRPLANE AFTER THREE POINT JACKING procedure (Ref. Chapter 07-00-00).

LANDING GEAR
MAIN GEAR AND DOORS
MAINTENANCE PRACTICES

1. GENERAL

WARNING: When performing maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance. Cycle the landing gear and ensure that all three landing gears are down and locked prior to removing the aircraft from jacks.

2. MAIN GEAR SHOCK ABSORBERS

A. Servicing

To check the fluid level in the landing gear shock absorbers, deflate the strut by releasing the air through the valve, then remove the filler valve adapter. The fluid level should be at the bottom of the valve standpipe with the struts fully compressed. If the level is low, add hydraulic fluid (9, Table 6, 91-00-00) to reach the standpipe, work the strut slightly to eliminate any trapped air, then add more fluid as necessary.

WARNING: Release the air pressure entirely before removing the valve adapter.

With the airplane empty except for fuel and oil, inflate the main strut until the piston is extended 4 1/4 to 4 1/2 inches (4 1/2 to 4 3/4 inches with flotation landing gear).

CAUTION: *Never tow or taxi with a flat strut. Even brief towing or taxiing with a deflated strut can cause damage.*

B. Lubrication

Lubricate the main gear wheel bearings and grease fittings as detailed in the Lubrication Schedule in Chapter 12.

3. MAIN LANDING GEAR

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00).

(2) Retract the gear slightly to remove the downlock tension on the drag leg.

(3) Remove the down-position switch and wiring clamps from the drag leg (Ref. Figure 201).

NOTE: The nut next to the switch must be maintained in the same position to provide the correct adjustment when reinstalling the switch.

(4) Disconnect the drag leg from the main gear by removing the cotter pin, nut, washers, bolt and bushings. After disconnecting the drag leg, install the bolt, bushing, washers and nut into the drag leg to facilitate installation. Disconnect the hydraulic brake line plumbing at the flex hose above the gear and cap the hydraulic lines.

(5) On installations equipped with brake deicing, remove the clamp securing the brake deicer hose to the upper end of the strut, then disconnect the hose at the union in the bracket just above the torque knees.

(6) Unsnap the bottom side of the canvas mud guard if installed. Disconnect the wiring (safety switch and down position switch) from the clamps and receptacle at the upper rear of the wheel well.

(7) Remove the access panels covering the landing gear hinge bolts. Remove the cotter pin, nut, washers, hinge bolt and bushing and lower the gear away from the airplane.

NOTE: After removing the bolt, position the washers on the bolt in the same sequence as when installed.

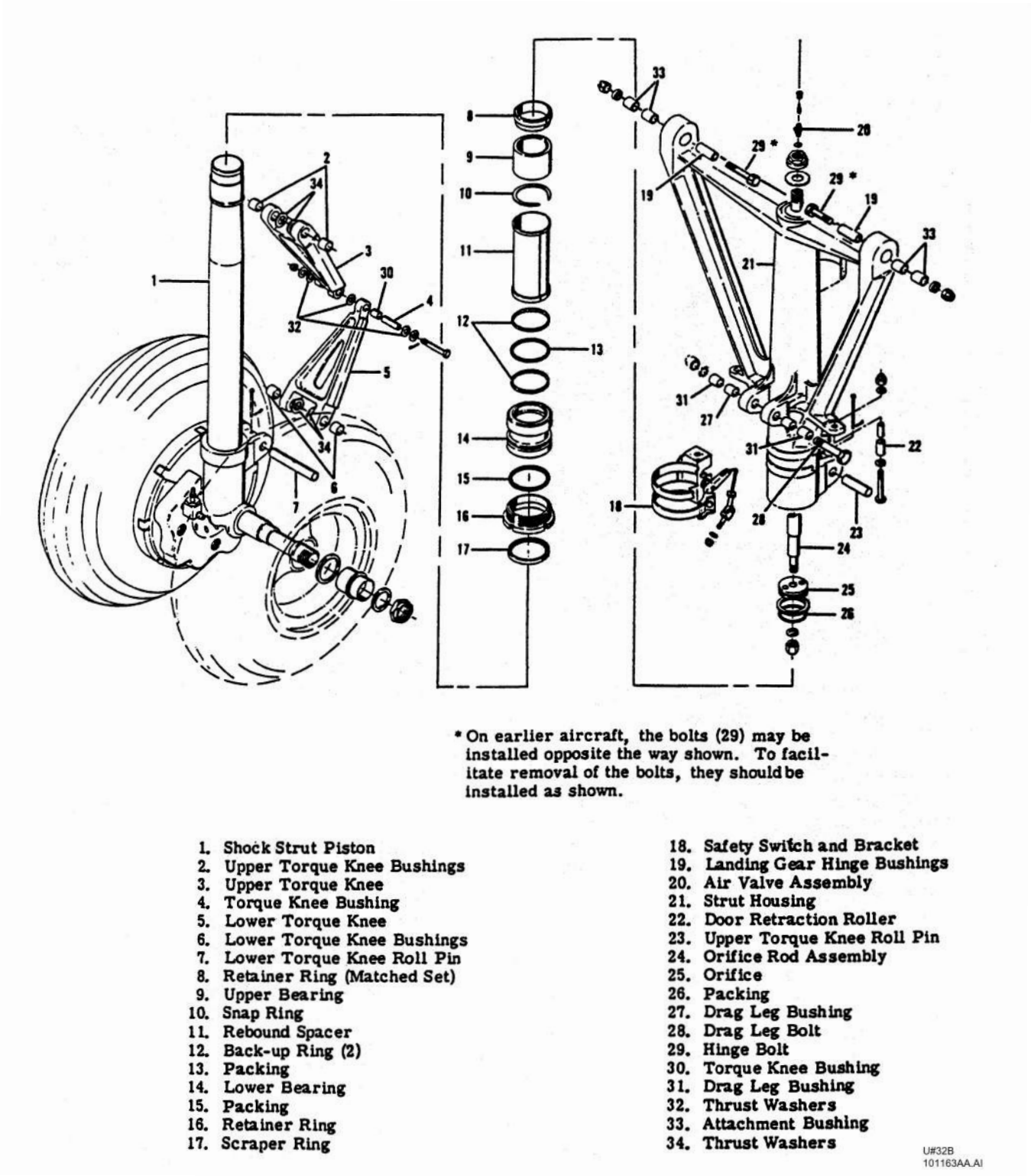


Figure 201
Main Landing Gear Assembly

B. Upper Main Landing Gear Brace Inspection

Inspect the upper main landing gear brace after each 7,500 cycles or 5 years, whichever occurs first. The upper main landing gear brace may be inspected as follows:

(1) With the airplane on jacks, disconnect the drag brace, clean the forging, and visually inspect, with the aid of a magnifying glass, the following areas:

(a) The top area of the cylinder or barrel and the arms leading to the pivot points.

(b) The vertical arms leading to the pivot points.

(c) The lower portion of the barrel, including the outer faces of the torque knee lugs.

(d) The inner and outer faces of the drag brace lugs.

NOTE: Any suspected cracks should be verified by a dye penetrant inspection in accordance with AC43.13-1A. Cracks are cause for replacement.

C. Installation

(1) On airplanes prior to U-152, except U-50, U-149 and those in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0156-200, Rev. II, coat the bushings (19 and 33) and the main gear hinge bolts (29) with metal protector (74, Table 6, 91-00-00). Do not lubricate the bolt threads.

(2) Position the gear and install the hinge bolts through the access openings. Install the 105090CR025-12J washers (minimum of one each side of the strut) as required to provide a 0.005 to 0.030 inch of total clearance between the keel and strut surfaces. Apply a minimum torque of 800 inch-pounds to the hinge bolt nuts; tighten the nuts to the next castellation (but do not exceed 1,200 inch-pounds) and install the cotter key. One AN960-1216L washer may be used under the nut the obtain the required torque value.

NOTE: Install the washers on the main gear hinge bolts in the same sequence as when removed.

(3) Connect the hydraulic brake line to the flex line above the gear.

(4) On installations equipped with brake deicing, connect the brake deicer hose at the union in the bracket just above the torque knees. Install the clamp securing the hose to the upper end of the strut.

(5) Connect the lower drag leg to the landing gear with the bolt, bushing, washer and cotter pin.

NOTE: On airplanes prior to U-152, except U-150, U-149 and those in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, coat the bushings (27 and 31) and drag leg attach bolt (28) with metal protector (74, Table 6, 91-00-00). Do not coat the bolt threads.

(6) Install the down-position indicator switch and wiring clamps to the drag leg.

NOTE: The nut next to the switch must be maintained in the same position to provide the correct adjustment as when previously installed.

(7) Plug the down-position switch and safety switch wiring into the receptacle at the upper rear of the wheel well and snap the canvas mud guard in place if installed.

(8) Install the landing gear hinge bolt access panels.

(9) Inflate the landing gear strut to a strut extension of 4 to 4 1/2 inches (4 1/2 to 4 3/4 inches with flotation landing gear) as indicated on the strut placard with the weight of the airplane on the gears.

(10) Bleed the brake and make a retraction check as outlined in the applicable landing gear rigging procedure.

(11) On airplanes U-50, U-149 and after, and prior airplanes in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, lubricate the main gear hinge points and drag brace joints with grease (17, Table 6, 91-00-00).

4. MAIN LANDING GEAR TORQUE KNEE

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00).

(2) Remove the wheel and brake (Ref. 32-40-00).

NOTE: Cap the hydraulic brake line to prevent contamination of the hydraulic fluid.

(3) Remove the cotter pin, nut, washer and bolt that attach the safety switch arm to the upper torque knee (Ref. Figure 201).

(4) Remove the cotter pin, nut, washers and bolt that connect the upper and lower torque knee. Note the position of the washers to facilitate installation of the torque knees.

(5) Cut the safety wire from the clevis pin and remove the clevis pin from the upper torque knee pin. To provide clearance for removal of the clevis pin on some airplanes, it will be necessary to loosen the two nuts and bolts that secure the safety switch bracket to the strut and then rotate the bracket. Note the position of the washers on the upper torque knee pin to facilitate installation of the torque knee. Tap the upper torque knee pin out and remove the upper half of the torque knee.

(6) Remove the cotter pin, washers and clevis pin from the lower torque knee pin. Note the position of the washers on the lower torque knee pin to facilitate installation, then tap the lower torque knee pin out and remove the lower half of the torque knee.

B. Installation

(1) On airplanes prior to U-152, except U-50, U-149 and those in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, coat the bushings (2, 4, 6 and 30), pins (7 and 23) and the center torque knee bolt with metal protector (74, Table 6, 91-00-00). Do not coat the bolt threads (Ref. Figure 201).

(2) Position the lower half of the torque knee in place with the washers installed in the same position as when previously installed. Tap the lower torque knee pin in place and install the clevis pin, washer, and cotter pin.

(3) Position the upper torque knee in place with the washers installed in the same position as when previously installed. Tap the upper torque knee pin in place and install the clevis pin. Safety wire between the clevis pin and the safety wire hole on the forward side of the retainer ring (16).

(4) Install the bolt, washer, and nut that connect the upper and lower torque knee. Torque the nut to 70 to 90 inch-pounds and install the cotter pin.

(5) Rotate the safety switch bracket, if installed, to its original position and tighten the bolts and nuts. Install the bolt, washer, nut and cotter pin that attach the safety switch arm to the upper torque knee.

(6) Install the wheel and brake (Ref. 32-40-00).

(7) On airplanes U-50, U-149, U-152 and After, and prior airplanes in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200 Rev. II or subsequent, lubricate the torque knee joints with grease (17, Table 6, 91-00-00).

(8) Remove the airplane from the jacks (Ref. Chapter 07-10-00).

5. MAIN LANDING GEAR DRAG LEG

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00).

(2) Retract the gear slightly to relieve the downlock tension on the drag leg.

(3) Remove the wiring clamps and down-position indicator switch (9) (Ref. Figure 202).

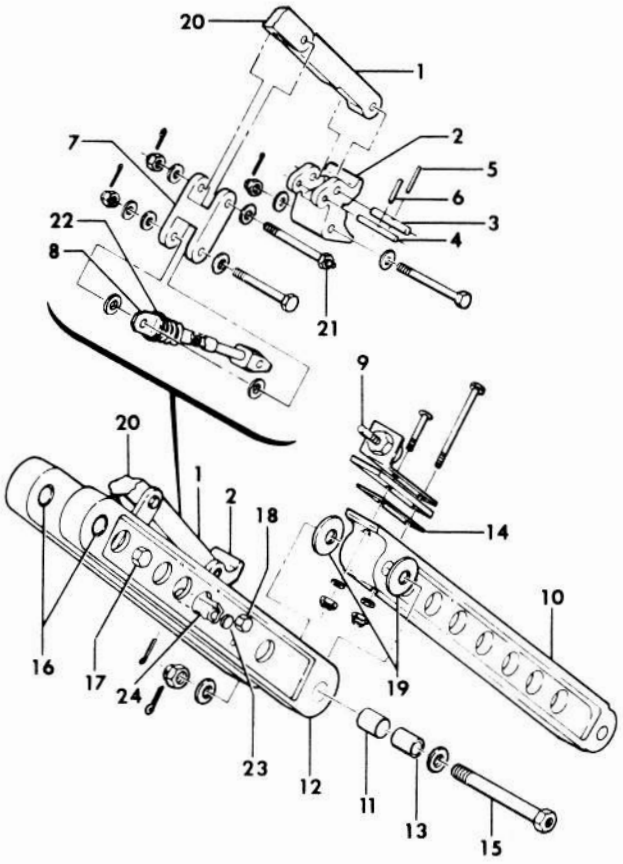
NOTE: To maintain switch adjustment, do not permit the nut on the switch side of the mounting bracket to turn.

(4) Remove the cotter pin, nut and bolt attaching the lower drag leg (10) the landing gear lug. Remove the bushings and attach the bolt, bushing and nut to the drag leg.

(5) Remove the cotter pin, nut, washers and bolt attaching the release and lock link (1) to the actuator clevis (20). Attach the bolt, washers and nut to the release-and-lock link for installation.

(6) Remove the retaining bolt securing the upper drag leg pin to the wheel well attach fitting. Remove the 90° grease fitting from one end of the upper drag leg pin and remove the pin.

NOTE: Note the number of washers on each side of the upper drag leg to facilitate installation of the drag brace in alignment with the attachment fitting on the landing gear strut.



1.	Release and Lock Link	12.	Upper Drag Leg
2.	Down Lock Hook	13.	Inner Bushing
3.	Hook Retainer Pin	14.	Laminated Shim
4.	Hook Retainer Pin	*15.	Bolt
5.	Pin	16.	Bushing
6.	Pin	17.	Bolt
7.	"H" Link	18.	Bolt
8.	Release and Link Guide	19.	Washer
9.	Down Position Indicator Switch	20.	Actuator Clevis
10.	Lower Drag Leg	**21.	Bolt
11.	Bushing	***22.	Spring Retention Washer
		23.	Pin
		24.	Bushing

* Grease fitting installed U-50, U-149, U152 and after and prior airplanes in compliance with Hawker Beechcraft Service instructions No. 0516-200, Rev. II.
** Grease fitting installed U-50, U-154 and after.
*** Installed U-50, U-165 and after.

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Figure 202
Main Gear Drag Leg Assembly

B. Installation

CAUTION: *On airplanes prior to U-152, except U-50, U-149 and those conforming to Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. 11, coat the drag leg bolts (28 of Figure 201 and 15 of Figure 202) and bushings (27 of Figure 201 and 13 of Figure 202) with metal protector (74, Table 6, 91-00-00) prior to installation and at the interval specified in the Lubrication Schedule in Chapter 12 to prevent a galvanic reaction between these parts and the teflon bushings installed with them.*

NOTE: If the upper and lower drag legs (12 and 10) were separated during overhaul, make sure one washer (19) is placed on each side of the lower drag leg (10) upon reassembly (Ref. Figure 202).

(1) Insert the upper drag leg pin through the wheel well fitting and upper drag leg (12). Install the washers between the upper drag legs and the wheel well fitting in the same positions as when removed. Align the hole in the upper drag leg pin with the hole in the wheel well fitting and install the retaining bolt. Install the grease fitting in the end of the pin (Ref. Figure 202).

NOTE: AN960-1016L washers may be shifted from side to side of the wheel well fitting and upper drag legs to align the lower drag leg (10) with the attachment point on the landing gear strut. A minimum of one washer should be maintained between the upper drag legs and the wheel well attachment fitting.

(2) Attach the lower drag leg (10) to the landing gear lug with the attach bolt, bushings, washers, nut and cotter pin.

(3) Attach the down lock release link (1) to the actuator clevis (20) and "H" link (7) with the bolt (21), bushing, washer, nut and cotter pin. Place the washers in the same position as when removed.

NOTE: Tighten the nut snug then back off the nearest castellation and install the cotter pin. The bolt (21) should be free to rotate without binding.

CAUTION: *Do not run the gear with the actuator disconnected.*

(4) Install the down-position indicator switch and attach the wiring clamps to the drag leg.

NOTE: To maintain switch adjustment, do not permit the nut on the switch side of the mounting bracket to turn.

(5) On U-50, U-149, U-152 and After, and prior airplanes in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, lubricate the drag brace joints with grease (17, Table 6, 91-00-00).

(6) Perform a rigging check as outlined in the applicable landing gear rigging procedure.

LANDING GEAR
NOSE GEAR AND DOORS
DESCRIPTION AND OPERATION

1. GENERAL

The nose landing gear incorporates a forward and aft set of landing gear doors. The forward set of doors are connected in such a manner as to open during the extend cycle and remain open until the gear is retracted into the wheel well. The aft set of doors are closed to reduce wind side drag when the landing gear is in the extended position, and are connected in such a manner as to open only when the landing gear is going through the extend or retract cycle.

LANDING GEAR
NOSE GEAR AND DOORS
MAINTENANCE PRACTICES

1. GENERAL

WARNING: When performing maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance. Cycle the landing gear and ensure that all three landing gears are down and locked prior to removing the aircraft from jacks.

2. NOSE GEAR SHOCK ABSORBERS

A. Servicing

To check the fluid level in the landing gear shock absorbers, deflate the strut by releasing the air through the valve, then remove the filler valve adapter. The fluid level should be at the bottom of the valve standpipe with the struts fully compressed. If the level is low, add hydraulic fluid (9, Table 6, 91-00-00) to reach the standpipe, work the strut slightly to eliminate any trapped air, then add more fluid as necessary.

WARNING: Release the air pressure entirely before removing the valve adapter.

With the airplane empty except for fuel and oil, inflate the main strut until the piston is extended 3 to 3 1/2 inches.

CAUTION: *Never tow or taxi with a flat strut. Even brief towing or taxiing with a deflated strut can cause damage.*

3. LANDING GEAR SHIMMY DAMPER

A. Servicing

To check the fluid level in the shimmy damper, insert a wire, approximately 1/16 inch in diameter, through the hole in the washer at the aft end of the piston rod until the wire touches the bottom of the hole in the floating piston. Mark the wire, remove it and measure the depth of insertion. Inserting the wire in the hole of the floating piston, rather than letting it rest against the face of the piston, will give a more accurate check. The wire can be inserted approximately 5-1/4 inches when the reservoir is empty.

NOTE: To determine if the wire is inserted in the hole of the floating piston, insert the wire several times, noting each insertion depth. When the wire is in the hole, the depth will be approximately 1/2 inch greater.

When the shimmy damper is full, the distance from the end of the piston rod to the bottom of the hole in the aft floating piston is 2 5/8 to 2 7/8 inches. When the distance is 0.50 inches from full, (i.e. 3 1/8 to 3 3/8 inches), the shimmy damper should be serviced with hydraulic fluid (9, Table 6, 91-00-00) as follows:

(1) Remove the shimmy damper.

(2) Remove the piston rod end fitting and secure the shimmy damper in a fixed position with the fitting end of the piston rod down.

(3) Remove the snap ring, disc and spring from the aft end of the piston rod.

CAUTION: *The spring in the piston rod is under compression. Remove the snap ring and washer carefully.*

(4) Remove the aft floating piston from the piston rod by screwing a long 10-32 bolt into the piston and pulling it out.

(5) Fill the piston rod with fluid.

(6) Insert the 10-32 bolt through the fitting end of the piston rod and engage the forward floating piston. Pull the piston toward the fitting end of the piston; at the same time, insert the aft floating piston and spring and install the disc and snap ring.

(7) Remove the bolt from the forward floating piston and install the piston rod end fitting.

4. NOSE LANDING GEAR

A. Lubrication

Lubricate the nose gear wheel bearings and grease fittings as detailed in the Lubrication Schedule in Chapter 12.

B. Removal

(1) Jack the airplane so the wheels are clear of the ground (Ref. Chapter 07-10-00). Strut limiters (special tools) will reduce strut extension.

(2) Retract the gear slightly to take the load off the drag leg.

(3) Remove the screws (one in each door) to free the aft set of doors from the center hinge brackets.

(4) Disconnect the push-pull tubes, that operate the forward set of doors, from the nose gear strut by removing the nuts, washers, bushings and bolts.

(5) Remove the nose cone and the two access plates on the avionics floorboard to gain access to the landing gear hinge bolts.

NOTE: Some airplanes have small access openings on the nose lower skin, just outboard of the landing gear hinge bolts. On these airplanes, it is not necessary to accomplish Step (5).

(6) Disconnect the nose gear actuator as follows:

(a) U-1 thru U-164, except U-50 (Ref. Figure 202). Due to the length of the actuator stop bolt (2), it will be necessary to retract the nose gear approximately 1/2 of the way up until the bolt aligns with the small hole in the wheel well RH keel. Remove the cotter pin and nut and remove stop bolt (2) through the hole, retain the washers and bushings for reinstallation.

(b) U-50, U-165 and After (Ref. Figure 203). Remove the bolt (10), nut and washers to disconnect the actuator clevis from the yoke. Remove the actuator stop bolts (2) from the upper drag leg (4).

(7) Disconnect the lower drag leg at the shock strut fitting as follows:

(a) U-1 thru U-164, except U-50 (Ref. Figure 202). Remove the cotter pin, nut and spacers from one side of the lower hinge stud (12) and drive the stud out. Be careful to retain the spacers for installation in their same relative position. U-149, U-152 thru U-164 and prior airplanes which are in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, have the lower hinge stud (12) replaced with a bolt. Remove the cotter pin, nut and washer and drive the bolt out. Be careful to retain the washers and bushings for installation in their same relative positions.

(b) U-50, U-165 and after (Ref. Figure 203). Remove the cotter pin and nut from the lower hinge bolt (8) and drive the bolt out. Note the location of washers and bushings for proper installation. The spacer (11) is used on U-50, U-165 thru U-170 only.

(8) Disconnect the nose gear steering linkage from the shock strut by removing the nut, washer and bolt that attaches the steering linkage yoke to the top left side of the shock strut.

(9) Remove the clamps securing the down-position indicator switch wiring to the upper drag leg and remove the downlock switch.

NOTE: To maintain downlock switch adjustment, do not allow the nut on the aft side of the drag leg to turn.

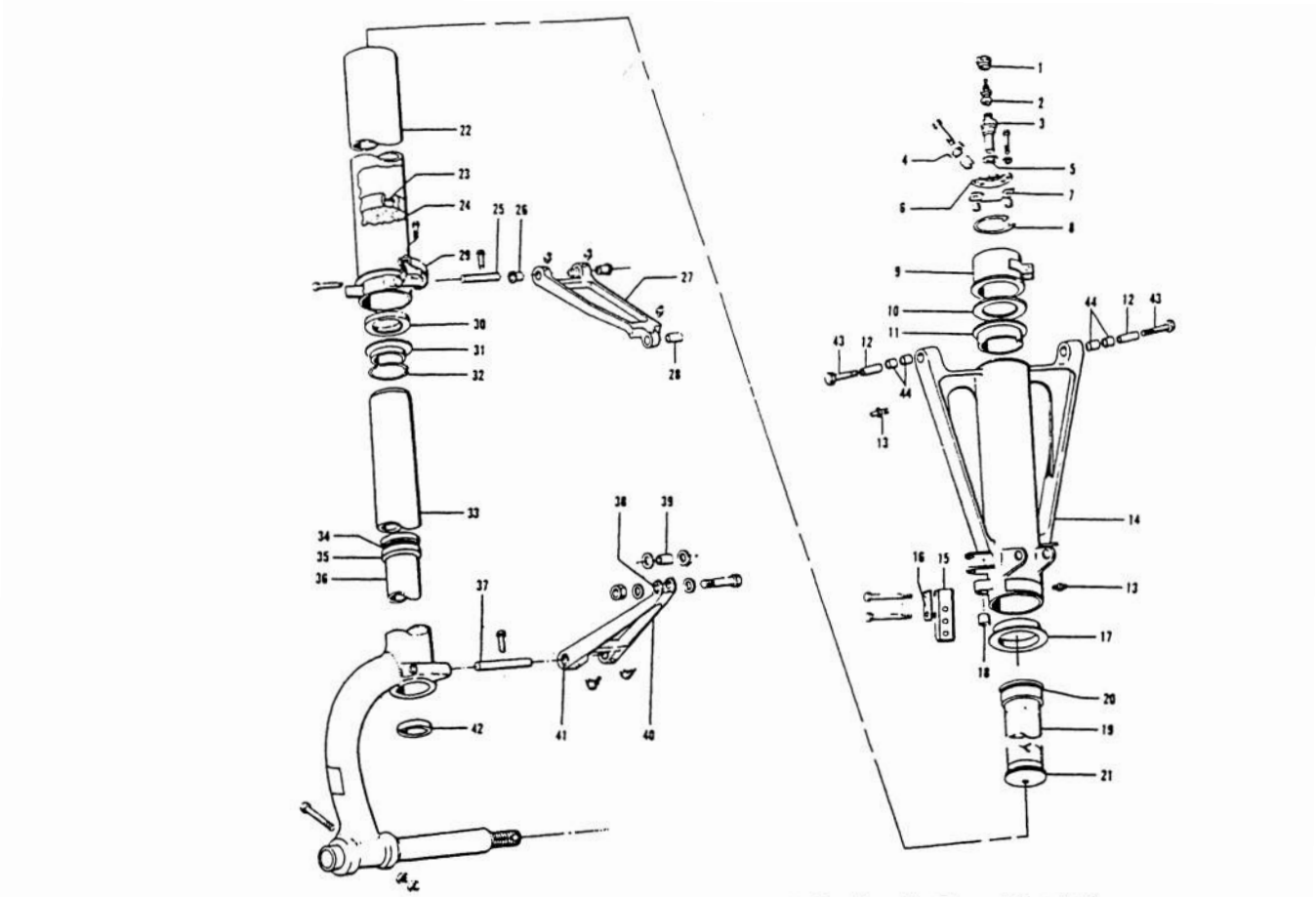
(10) Disconnect electrical wiring from taxi light.

(11) Pull the nose gear to its fully forward position and turn the wheel full right to clear the gear-straightening roller, allowing the lower end of the drag leg to be pulled back through the shock strut web.

(12) Place a block under the wheel to remove part of the load on the attaching bolts and to support the strut when the bolts are removed.

(13) Remove the cotter pins, nuts, washers and nose gear attach bolts and lower the strut.

NOTE: Note the position of the washers to facilitate installation.



1. Cap
2. Valve Core
3. Filler Valve Body
4. Nose Gear Roller
5. Packing
6. Nose Gear Straightner Bracket
7. Nose Gear Sleeve Retainer Bar
8. Snap Ring
9. Sleeve Assembly
10. Laminated Shim
11. Upper Bearing
12. Hinge Point Bushing
13. Bearing Lubricators
14. Strut Housing
15. Nose Gear Travel Stop Fitting
16. Shim
17. Lower Bearing
18. Shimmy Damper Attach Bushing
19. Orifice Support Assembly
20. Packing
21. Metal Ring
22. Upper Shock Absorber
23. Packing (In Brazed Bearing)
24. Felt Lubricating Pad
25. Upper Torque Knee Roll Pin
26. Bushing
27. Upper Torque Knee
28. Torque Knee Joint Bushing
29. Shimmy Damper Arm
30. Wiper Ring
31. Scraper Ring
32. Snap Ring
33. Lower Shock Absorber
34. Packing
35. Plug
36. Inner Shock Absorber Tube
37. Lower Torque Knee Roll Pin
38. Outer Torque Knee Bushing
39. Inner Torque Knee Bushing
40. Lower Torque Knee
41. Roll Pin Bushing
42. Plug (Bonded in Strut)
43. Bolt
44. Bushing

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Figure 201
Nose Gear Assembly

C. Installation

CAUTION: *On airplanes prior to U-152, except U-50, U-149 and those in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. /I, coat the nose landing gear hinge bolts (43) and bushings (12 and 44), hinge stud (12, Figure 202) and bushings (8 and 9, Figure 202) with metal protector (74, Table 6, 91-00- 00) at installation and at the interval specified in the Lubrication Schedule in Chapter 12 to prevent galvanic reaction between the bolts and the teflon bushings.*

(1) Position the strut in the wheel well and install the landing gear hinge bolts and washers. lnstall the bushings over the bolts and into the strut casting after the strut is in position. lnstall the nuts on the bolts and torque lubricated hinge attach bolts 200 to 400 inch- pounds and secure the nuts with cotter pins. Use AN950-816 or AN960-816L washers under the nut as required to obtain proper torque and engagement of the nut and cotter pin. Non lubricated hinge attach bolts should be torqued 360 to 720 inch-pounds.

NOTE: Position the washers in the same sequence as when previously installed.

(2) Pull the nose gear to its fully forward position and turn the wheel full right to clear the gear-straightening roller, allowing the lower end of the lower drag leg to be placed forward through the strut web and into position to be attached.

(3) Connect the lower drag leg to the shock strut as follows:

(a) U-1 thru U-164, except U-50 (Ref. Figure 202). On airplanes prior to U-152, except U-149 and those in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, coat the lower hinge stud (12) with metal protector (74, Table 6, 91-00-00) and secure the lower drag legs (10) to the strut, being careful to install the spacers in the same relative position from which they were removed. Torque the attaching nut to 300 inch-pounds (may be torqued to a maximum of 400 inch-pounds to align for cotter pin installation). lnstall the cotter pin. On airplanes U-149, U-152 thru U-164 and prior airplanes which are in compliance with Hawker Beechcraft (Beech) Service lnstructions No. 0516- 200, Rev. II, coat the lower hinge with grease (17, Table 6, 91-00-00) and secure the lower drag legs (10) to the strut, being careful to install the bushings and washers in the same position from which they were removed. Torque the attaching nut 200 to 400 inch-pounds and install the cotter pin.

(b) U-50, U-165 and After (Ref. Figure 203). Coat the lower hinge bolt (8) with grease (17, Table 6, 91-00-00) and secure the lower drag legs (5) to the strut, being careful to install the bushings and washers in the same position from which they were removed. When installing the lower hinge bolt on airplanes U-50 and U-165 thru U-170 only, install the spacer (11). Torque the attaching nut 200 to 400 inch-pounds and install the cotter pin.

(4) Connect the nose gear actuator to the drag brace as follows:

(a) U-1 thru U-164, except U-50, (Ref. Figure 202). Extend the actuator until the yoke bolt is aligned with the small cutout in the wheel well RH keel. To avoid rerigging the nose gear on mechanical systems, do not allow the yoke to turn while the actuator is being extended. lnstall the stop bolt (2), being careful to install spacers, washers and rollers in the same position as when removed.

(b) U-50, U-165 and After (Ref. Figure 203). Extend the actuator and align the actuator clevis bolt hole with the bolt hole in the yoke (6), then secure with bolt (10), washers, nut and cotter pin. Install one washer under the bolt head and one under the nut.

(5) Install the indicator position switch and wiring clamps to the upper drag leg.

NOTE: To maintain downlock switch adjustment, do not allow the nut on the aft side of the drag leg to turn.

(6) Connect the electrical wiring to the taxi light.

(7) Connect the nose gear steering linkage to the shock strut by attaching the steering linkage yoke to the top left side of the shock strut with the nut, washer and bolt.

(8) Attach the door operating mechanism push-pull tubes that operate the aft set of doors to the door operating A-frame by installing the nuts, washers and bolts.

(9) Attach the push-pull tubes that operate the forward set of doors to the nose gear strut by installing the nuts, washers, bushings and bolts.

(10) On airplanes U-50, U-149, U-152 and After and prior airplanes in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, lubricate the affected grease fittings with grease (17, Table 6, 91-00-00).

(11) Perform a retraction check as outlined in the applicable landing gear rigging procedure.

(12) Install the hinge bolt access covers and nose cone, if removed.

(13) Remove the airplane from the jacks (Ref. Chapter 07-10-00).

(14) Inflate the nose gear strut as indicated on the strut placard, to a strut extension of 3 to 3 1/2 inches with the weight of the airplane on the gears.

5. NOSE LANDING GEAR TORQUE KNEE

A. Removal

WARNING: Deflate the strut before disassembly.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00).

(2) Remove the shimmy damper arm from the landing gear strut by cutting the safety wire and removing the two bolts from the top of the arm. Remove the lubrication fitting (if installed) from the upper torque knee adjacent to the head of the horizontal bolt located just below and forward of the two bolts previously removed and remove the nut, washer and bolt.

(3) Remove the cotter pin, washer and clevis pin from the upper torque knee pin. Note the position of the washers on the upper torque knee pin to facilitate installation and tap the upper torque knee pin out.

(4) Remove the cotter pin, nut, washer and bolt that connect the upper and lower torque knee. Note the position of the washers on the attach bolt to facilitate installation.

(5) Remove the cotter pin, washer and clevis pin from the lower torque knee pin. Note the position of the washers on the lower torque knee pin to facilitate installation and tap the lower torque knee pin out.

B. Installation

(1) Position the lower half of the torque knee in place with the washers installed in the same position as when previously installed. Tap the lower torque knee pin in place and install the clevis pin, washer and cotter pin.

(2) Position the upper torque knee in place with the washers positioned in the same position as when previously installed and install the bolt, washers and nut that attach the upper torque knee to the lower torque knee. Torque the nut 70 to 90 inch-pounds and install the cotter pin.

(3) Position the upper end of the upper torque knee in place with the washers positioned in the same position as when previously installed, and tap the upper torque knee pin in place. Install the clevis pin, washer and cotter pin.

(4) Attach the shimmy damper arm to the landing gear strut with the horizontal bolt, washers and nut. Install the two upper bolts in the shimmy damper arm and safety wire the two bolts.

(5) Install the lubrication fitting in the upper torque knee and lubricate with grease (17, Table 6, 91-00-00).

(6) Remove the airplane from the jacks (Ref. Chapter 07-10-00).

6. NOSE LANDING GEAR DRAG BRACE (U-1 THRU U-164, EXCEPT U-50)

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00), retract the nose gear slightly to relieve down lock tension and remove the screws (one in each door) to free the aft doors from the center hinge brackets.

(2) Remove the clamps securing the down position indicator switch wiring to the upper drag leg (Ref. Figure 202).

(3) Remove the down lock switch from the upper drag leg.

NOTE: To maintain switch adjustment, do not allow the nut on the aft side of the drag leg to turn.

(4) Remove the actuator stop bolt (2). Due to the length of the bolt, it will be necessary to retract the gear approximately 1/2 of the way up until the bolt aligns with the small hole in the wheel well RH keel. Mark all rollers, washers and spacers to ensure installation in their same relative position.

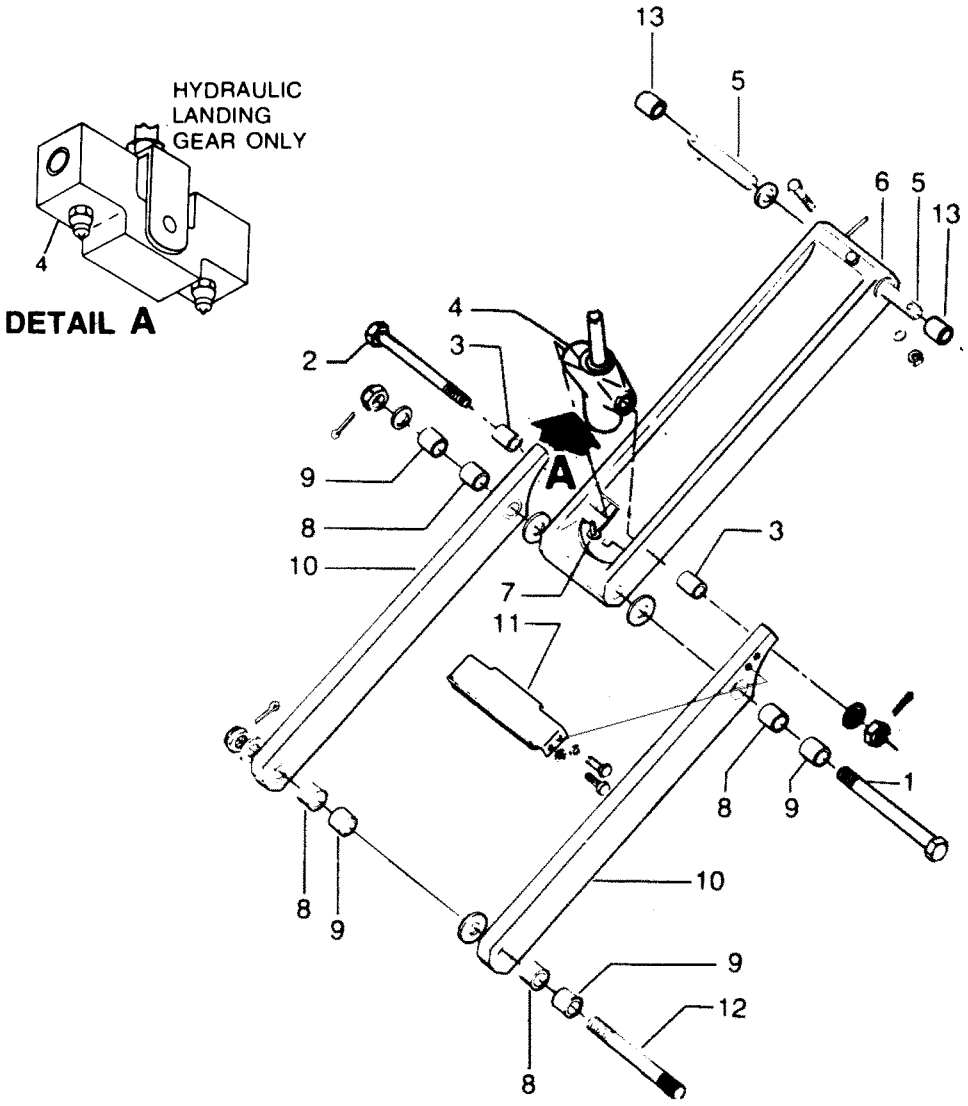
(5) Disconnect the door mechanism push pull tubes from the door operating A-frame.

(6) Remove the cotter pin, nut and spacers from one side of the lower hinge stud (12) and drive the stud out. Be careful to retain the spacers for installation in their same relative position. U-149, U-152 thru U-164 and prior airplanes which are in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II have the lower hinge stud (12) replaced with a bolt. Remove the cotter pin, nut and washer and drive the bolt out. Be careful to retain the washers and bushings for installation in their same relative positions.

(7) Retract the actuator electrically to avoid interference when removing the drag leg. To avoid rerigging the nose gear on airplanes with a mechanical landing gear system, hold the actuator yoke to prevent it from turning while the actuator is being retracted.

(8) Pull the gear to its full forward position and turn the wheel full right to clear the gear straightening roller, allowing the lower end of the drag leg to be pulled back through the strut web.

(9) Remove the two bolts securing the upper drag leg to the two pins (5). Remove the pins with Vise-Grip or equivalent pliers. Remove the drag leg assembly with the door operating A-frame still attached.



1. HINGE BOLT
2. STOP BOLT
3. BUSHING
4. YOKE
5. PINS
6. UPPER DRAG LEG
7. DOWN LOCK SWITCH
8. BUSHING
9. BUSHING
10. LOWER DRAG LEG
11. BRACKET
12. LOWER HINGE STUD
13. BUSHING

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Figure 202
Nose Gear Drag Brace Assembly (Prior to U-165, except U-50)

B. Installation

(1) The door operating A-frame should be installed on the upper drag leg before the drag leg is installed.

(2) Secure the upper end of the upper drag leg (6) with the two pins (5) and bolts. Be careful to align the pin holes with the drag leg holes when installing the pins (Ref. Figure 202).

(3) Pull the gear to its full forward position and turn the wheel full right to clear the gear straightening roller, allowing the lower end of the lower drag leg (10) to be placed forward through the strut web and into position to be attached.

(4) On airplanes prior to U-152 (except U-149), not in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, coat the lower hinge stud (12) with metal protector (74, Table 6, 91-00-00) and secure the lower drag legs (10) to the strut, being careful to install the spacers in the same relative position from which they were removed. Torque the attaching nut 200 to 400 inch-pounds and install the cotter pin.

(5) Extend the actuator until the yoke bolt hole is aligned with the small cutout in the wheel well RH keel. To avoid rerigging the nose gear, do not allow the yoke to turn while the actuator is being extended. Install the stop bolts (2), being careful to install spacers, washers and rollers in the same position as when removed.

(6) Attach the door operating mechanism push-pull tubes to the door operating A-frame.

(7) Install the indicator position switch and wiring clamps.

(8) Install the screws (one in each door) to secure the doors to the door brackets.

(9) On U-149, U-152 thru U-164 and prior airplanes which are in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, lubricate the drag leg grease fittings with grease (17, Table 6, 91-00-00).

CAUTION: *On airplanes prior to U-152 (except U-149), not in compliance with Hawker Beechcraft (Beech) Service Instructions No. 0516-200, Rev. II, coat the hinge bolts (1), stud (12) and bushings (8) with metal protector (74, Table 6, 91-00-00) at the interval specified in the Lubrication Schedule in Chapter 12 to prevent galvanic reaction between the bolts and the teflon bushings.*

7. NOSE LANDING GEAR DRAG BRACE (U-50, U-165 AND AFTER)

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00), retract the nose gear slightly to relieve downlock tension and remove the screws (one in each door) to free the aft set of the nose gear doors from the center hinge brackets.

(2) Disconnect the push-pull tubes, that operate the forward set of doors, from the nose gear strut by removing the nuts, washers, bushings and bolts.

(3) Remove the clamps securing the down position indicator switch wiring to the upper drag leg.

(4) Remove the downlock switch (3) from the upper drag leg (4) (Ref. Figure 203).

NOTE: To maintain downlock switch adjustment, do not allow the nut on the aft side of the drag leg to turn.

(5) Remove the actuator stop bolts (2) from the upper drag leg (4). Note the location of bushings and washers to ensure proper installation.

(6) Remove the bolt (10), nut and washers and disconnect the actuator clevis from the yoke (6).

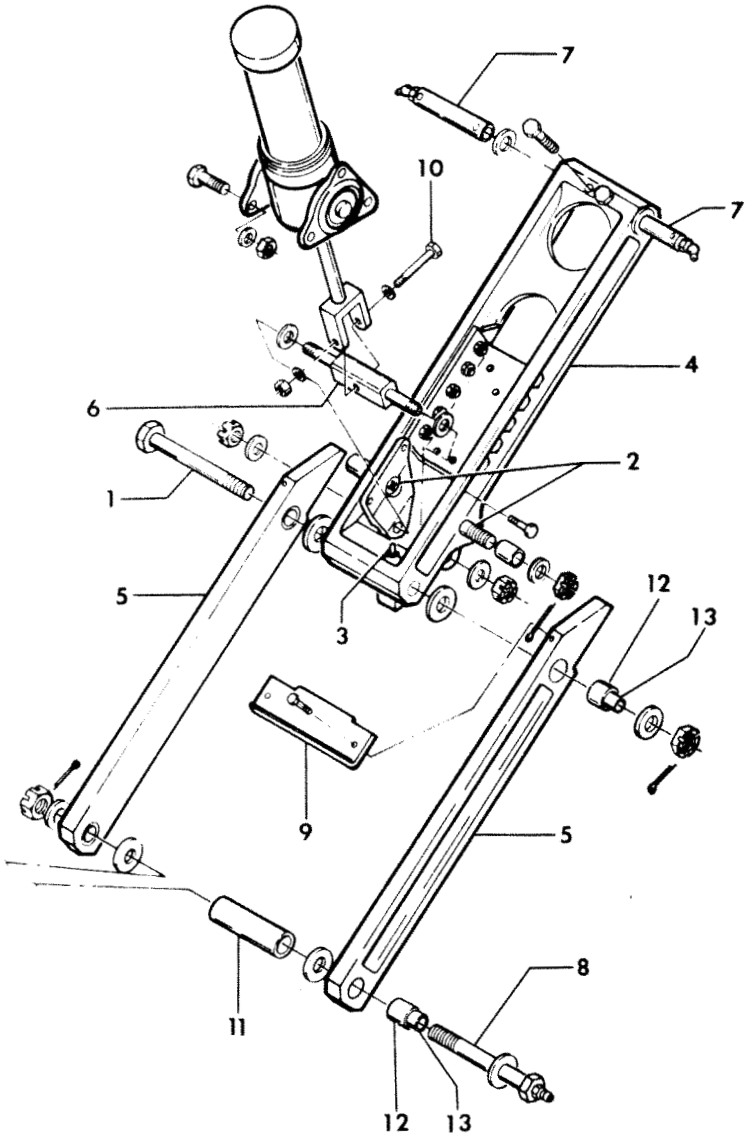
(7) Disconnect the door mechanism push-pull tubes from the door operating A-frame.

(8) Remove the cotter pin and nut from the lower hinge bolt (8) and drive the bolt out. Note the location of washers and bushings for proper installation. The spacer (11) is used on U-50 and U-165 thru U-170 only.

(9) Retract the actuator to avoid interference when removing the drag leg.

(10) Pull the nose gear to its full forward position and turn the wheel full right to clear the gear straightening roller, allowing the lower end of the drag leg to be pulled back through the strut web.

(11) Remove the two bolts securing the upper drag leg to the two pins (7). Remove the upper pins (7) with Vise-Grip or equivalent pliers and retain the washers located between the upper drag leg and the attachment fittings. Remove the drag leg assembly with the door operating A-frame still attached.



1. Bolt
2. Stop Bolts
3. Downlock Switch
4. Upper Drag Leg
5. Lower Drag Leg
6. Yoke
7. Pins
8. Lower Hinge Bolt
9. Bracket
10. Bolt
11. Spacer
12. Bushings
13. Bushings

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Figure 203
Nose Gear Drag Brace Assembly (U-50, U-165 and After)

B. Installation

(1) The door operating A-frame should be installed on the upper drag leg before the drag leg is installed.

(2) Secure the upper end of the upper drag leg (4) to the attachment fittings with the two pins (7) and bolts. Install one washer on each side of the upper drag leg. Be careful to align the pin holes with the drag leg holes when installing the pins (Ref. Figure 203).

(3) Pull the nose gear to its full forward position and turn the wheel full right to clear the gear straightening roller, allowing the lower end of the lower drag leg (5) to be placed forward through the strut web and into position to be attached.

(4) Coat the lower hinge bolt (8) with grease (17, Table 6, 91-00-00) and secure the lower drag legs (5) to the strut, being careful to install the bushings and washers in the same position from which they were removed. When installing the lower hinge bolt on airplanes U-50 and U-165 thru U-170 only, install the spacer (11). Torque the attaching nut 200 to 400 inch-pounds and install the cotter pin.

(5) Extend the actuator and align the actuator clevis bolt hole with the bolt hole in the yoke (6) and secure with bolt (10), washers, nut and cotter pin. Install one washer under the bolt head and one under the nut.

(6) Attach the door operating mechanism push-pull tubes to the door operating A-frame.

(7) Install the indicator position switch and wiring clamps to the upper drag leg.

NOTE: To maintain downlock switch adjustment, do not allow the nut on the aft side of the drag leg to turn.

(8) Connect the push-pull tubes, that operate the forward set of doors, to the nose gear strut by installing the nuts, washers, bushing and bolts.

(9) Install the screws (one in each door) to secure the aft set of doors to the center hinge brackets.

(10) Lubricate the drag leg grease fittings with grease (17, Table 6, 91-00-00). Use an Alemite Z-737 nozzle to lubricate the drag leg hinge bolt (1).

LANDING GEAR
EXTENSION AND RETRACTION
MAINTENANCE PRACTICES

1. MANUAL EXTENSION SYSTEM CHAIN

A. Lubrication

Lubricate the manual extension system chain as detailed on the Lubrication Diagram in Chapter 12.

2. NOSE GEAR ACTUATOR DRIVE SHAFT

A. Alignment Check

With the drive shaft in position on the actuator and the drive shaft support assembly attach bolts removed, determine that the drive shaft support assembly bolt holes align with the holes in the wheel well structure. If they do not, elongate the wheel well holes (Ref. Figure 201).

3. LANDING GEAR MOTOR

A. Removal

(1) Remove the carpet and the center aisle floorboard just forward of the main spar.

(2) Remove the three electrical leads from the motor terminal strip.

(3) Remove the nuts, washers and bolts attaching the motor to the landing gear gearbox and remove the motor.

B. Installation

(1) Install the motor on the gearbox with the nuts, washers and bolts.

(2) Connect the three electrical leads to the motor terminal strip.

(3) Install the center aisle floorboard and carpet just forward of the main spar.

(4) Place airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(5) Cycle the landing gear, using the electric motor driven system, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(6) Ensure all three landing gear are down and locked.

(7) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

4. LANDING GEAR MOTOR GEARBOX

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the center aisle floorboard and RH seat just forward of the main spar and the oval floorboard access cover immediately to the right of the center aisle floorboard and forward of the main spar.

(3) Position the divider link, which is identified with yellow paint, of the aft nose retract chain and manual extension system chain, just forward of the motor gearbox sprocket.

NOTE: The aft nose retract chain may be positioned by actuating the landing gear motor with the landing gear control in the full UP position through the use of the landing gear circuit breaker located on the circuit breaker panel to the right of the copilot. The manual extension system may be positioned by using the manual extension handle. The curved handle must remain in the disengaged position.

(4) Release the tension for the aft nose retract chain as described in AFT NOSE CHAIN REMOVAL and the manual extension system chain as described in MANUAL EXTENSION CHAIN REMOVAL.

(5) Disconnect the chains at the divider links.

(6) Remove the power cables from the motor gearbox. Remove the electrical wiring from the microswitch manual extension system engage lever. Remove the nut, washer and bolt attaching the manual extension cable to the spring mounted on the front of the motor gearbox.

(7) Remove the access panels, nuts, washers, taper pins and spline adapters (Ref. Figure 203, 32-30-01).

(8) Remove the nut, washer and taper pin from the spline adapter on the torque shaft adjacent to the motor gearbox, left hand and right hand.

(9) Slide the spline adapters outboard on the torque shafts to disengage the spline adapter from the motor gearbox.

NOTE: The shims used between the motor gearbox and motor gearbox mounting on the spar must be retained and installed in the same manner as when removed.

(10) Remove the nut, washer and bolt on the left hand side of the motor gearbox and the two studs and washers on the right side of the motor gearbox.

NOTE: The limit switches are not to be detached from the wiring.

(11) Slide the motor gearbox assembly away from the gearbox mounting and allow it to rest on the belly of the fuselage, then remove the up and down limit switches from the gearbox.

(12) Remove the motor gearbox assembly using caution not to damage the crossfeed fuel line directly above the motor gearbox.

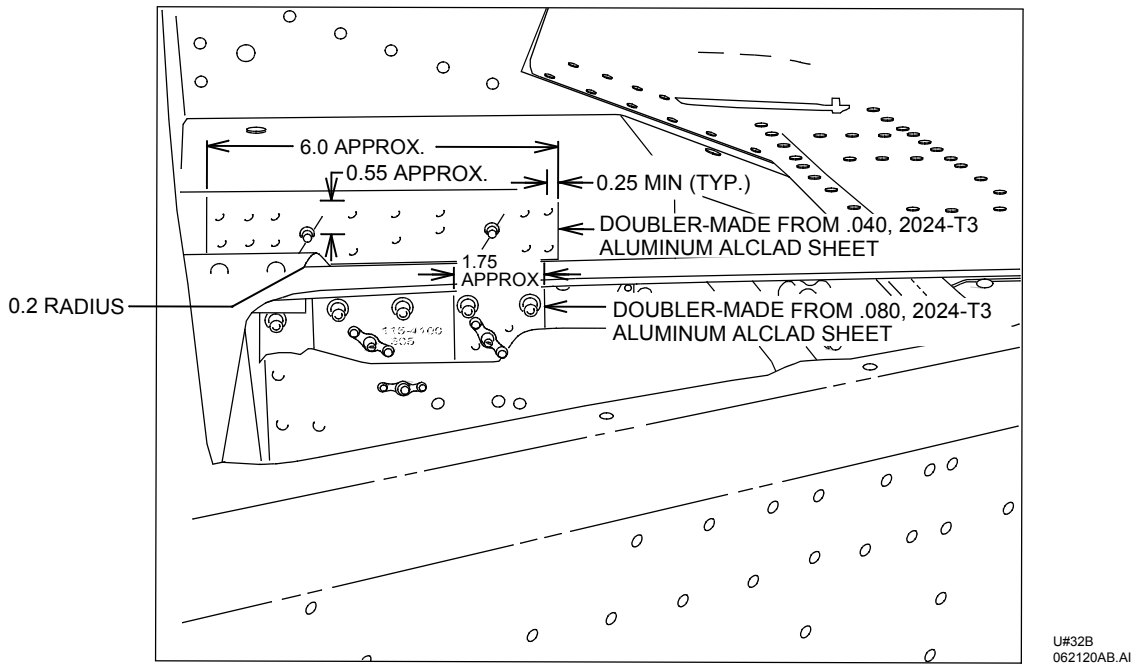


Figure 201
Modification for Alignment of the Nose Gear Actuator Drive Shaft

B. Modification for Alignment of the Nose Gear Actuator Drive Shaft

(1) Elongate the existing holes in the structure only, not the holes in the doublers, support assembly or chain guard plate (Ref. Figure 201).

(2) Trim the doublers as necessary to cover the areas indicated in the above illustration.

(3) Drill the lower doubler at the huck bolt locations to clear and to avoid removing the huck bolts.

(4) Rotate existing nut plates on the doublers in order to gain adequate edge distance between the existing rivet holes and the new rivet holes.

(5) Install all rivets with the flush head inboard.

(6) Install the nut plates with rivets (P/N MS20426AD3) and the doublers with rivets (P/N MS20426AD4) of appropriate length.

C. Installation

NOTE: To facilitate final adjustment of the up and down limit switches, install the switches and, using the jam nuts, adjust the position of the switch actuator threaded shaft to the midpoint of its travel prior to installation of the gearbox.

(1) Position the motor gearbox and install the shims in the same manner as when removed. Attach the motor gearbox to the motor gearbox mounting with the two studs and washers on the right side and the bolt washer and nut on the left side of the motor gearbox.

(2) Slide the spline adapters on the torque shafts adjacent to the motor gearbox and insert the taper pins, washers and nuts.

(3) Slide the spline adapters on the torque shafts and install the taper pins, washers, nuts and access panels (Ref. Figure 203, 32-30-01).

(4) Attach the manual extension cable to the spring mounted on the front side of the motor gearbox with the bolt, washer and nut. Attach the electrical wiring to the micro switch manual extension system lever. Attach the power cables to the motor gearbox.

(5) Connect the chains at the yellow divider links.

NOTE: Check all sprockets to determine that the chains are aligned over the sprockets properly.

(6) Adjust the tension of the aft nose retract chain and the manual extension system chain (Ref. 32-30-01).

(7) Perform the RIGGING ALL THREE LANDING GEAR procedure (Ref. 32-30-01).

(8) Install the center aisle floorboard just forward of the main spar and right hand seat and the oval floorboard access cover immediately to the right of the center aisle floorboard and forward of the main spar.

(9) Ensure all three landing gear are down and locked.

(10) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

5. MAIN LANDING GEAR ACTUATOR

A. Removal

(1) Place airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the fender assemblies from the wheel well on aircraft equipped with flotation gear.

(3) Disconnect the gear doors by removing the cotter pin, nut, washer and bolt that attach the link to the cam (Ref. Figure 211, 32-30-01).

(4) Measure the main gear actuator spring compression with the gear in the extended position (Ref. Figure 210, 32-30-01). Note this dimension to facilitate rigging the gear when the actuator is installed.

(5) Retract the gear slightly to relieve the tension on the actuator spring.

(6) Remove the access panel (Ref. Figure 203, 32-30-01). Remove the nut, washer and taper pin from the spline adapter at the universal joint and remove the spline adapter from the spline shaft (Ref. Figure 203, 32-30-01). Disconnect the main gear mechanism.

(7) Remove the nut, washer and bolt (16) that attach the universal joint (17) to the actuator pinion shaft (11) and slide the universal joint (17) off the actuator pinion shaft (11) (Ref. Figure 202).

(8) Remove the cotter pin, nut, washer and bolt that connect the actuator clevis (28) to the drag brace.

(9) Remove the screws from the clamps that attach the wire bundles to the support brackets.

(10) Remove the screw from the actuator brackets that holds the pneumatic line in place.

(11) Remove the bolts securing the forward stiffener plate to the actuator mounting bracket.

(12) Remove the bolts securing the inboard and outboard supports in which the actuator support bearings are installed.

NOTE: Extreme care should be exercised in the removal of the screw assembly from the nut assembly. Mark the nut and screw prior to separation to ensure proper engagement of the double acme threads upon assembly of the actuator (It is possible, but not permissible, to insert the screw rotated 180° out of the proper position).

(13) Screw the jackscrew out of the actuator.

(14) Work the support brackets and bearings off the actuator and remove the actuator by lifting up and forward on the lower side of the actuator and then removing downward carefully.

1. SCREW HOUSING COVER
2. ACTUATOR SCREW
3. BOLT, WASHER AND NUT
4. UPPER COVER
5. UPPER BEARING
6. BOLT, WASHER AND CASTELLATED NUT
7. SCREW HOUSING
8. HOUSING PLUG* OR VENT**
9. ACTUATOR HOUSING
10. INNER THRUST PINION BEARING
11. PINION
12. OUTER PINION BEARING
13. SHIM
14. SNAP RING
15. UNIVERSAL JOINT SPACER
16. BOLT, WASHER AND NUT
17. UNIVERSAL JOINT
18. LOWER THRUST BEARING
19. BUSHING
20. SHIM
21. HOUSING BOTTOM
22. NUT SEAL
23. ACTUATOR NUT
24. BOLT, WASHERS AND NUT
25. PLUG
26. PACKING
27. SPRING
28. CLEVIS
29. BOOT**
30. BEARING SEAL**
31. BRAKE
32. BRAKE MOUNTING BOLT, WASHERS AND NUT
33. BRAKE CLAMPING BOLT, BUSHING, WASHERS AND NUT
34. BRAKE SPRING
35. SPACER

*EARLY CONFIGURATION ACTUATORS
**LATER CONFIGURATION ACTUATORS

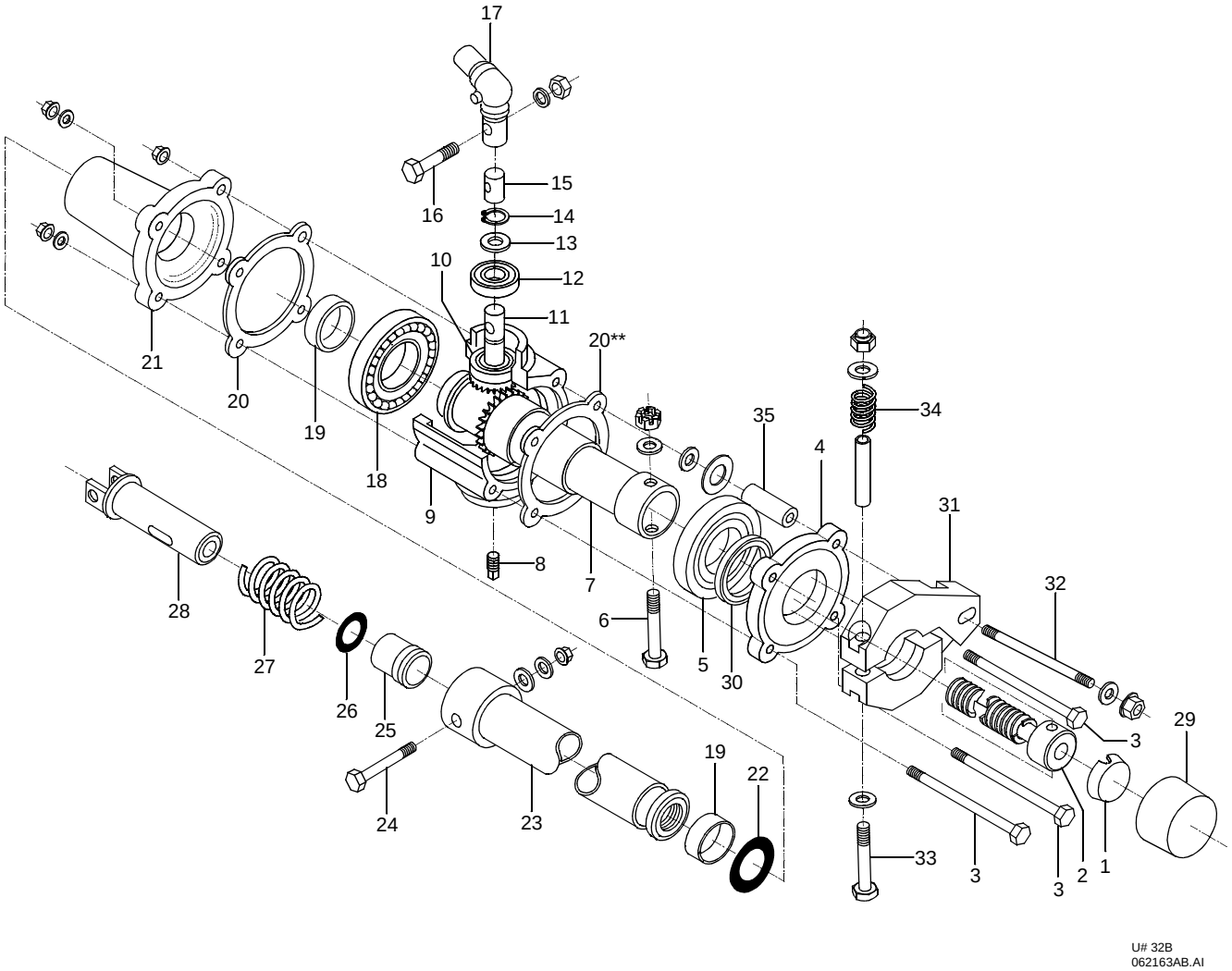


Figure 202
Main Landing Gear Actuator (Mechanical System)

B. Installation

NOTE: Inspect the actuator support bearing assemblies for corrosion or damage and ensure free rotation (Ref. Figure 208).

(1) Position the actuator and install the bolts securing the inboard and outboard supports. Shim the bearing supports until a clearance of 0.005 to 0.025 inch is obtained between the actuator and bearing (Ref. Figure 207).

NOTE: Extreme care should be exercised in the installation of the screw assembly into the nut assembly. Align the marks that were made prior to removal to ensure proper engagement of the double acme threads upon assembly of the actuator (It is possible, but not permissible, to insert the screw rotated 180° out of the proper position).

(2) Screw the jackscrew into the actuator.

(3) Check the actuator mounting brackets for distortion. Distorted brackets should be replaced. Install the forward stiffener plate on the brackets with the washers and bolts.

(4) Position the actuator jackscrew in the fully retracted position and raise the gear manually to the retracted position. Place a stand or support of sufficient height under the gear to hold the gear in the up position. Extend or retract the actuator jackscrew as necessary to determine if the clevis will align with the link (7, Figure 202, Chapter 32-10-00). If the actuator clevis does not align, remove the actuator and position the shims installed in Step (1) as necessary to provide correct alignment, being careful to maintain the required tolerances between the actuator and bearing.

(5) Position the actuator jackscrew and the landing gear in the full down position. Extend or retract the jackscrew actuator as necessary to determine if the clevis will align with the link (7, Figure 202, Chapter 32-10-00). In the event the clevis does not align with the link in the extended position and the necessary positioning of the shims would result in severe misalignment in the retracted position, an arrangement of shims should be derived which would provide the most favorable alignment possible in both positions.

(6) Install the bolt, washer and nut (16) attaching the universal joint (17) to the actuator pinion shaft (11) (Ref. Figure 202).

(7) Using screws, attach the clamps that secure the wire bundles and pneumatic line in place.

(8) Cycle the gear up and down to determine correct rig on the two gear that are still connected.

(9) The bolt that attaches the clevis to the drag leg brace should be lightly coated with grease (17, Table 6, 91-00-00). Install washers (P/N AN960-816L) under the bolt head as required and install a washer (P/N AN960-616L) under the nut. Torque the nut down snug, then back off the nut to the nearest castellation and install cotter pin. Check the bolt to determine if the bolt is free to rotate without binding.

(10) With the gear down and locked, rotate the torque shaft to obtain the same spring compression measurement as when previously installed. Reference Figure 210, 32-30-01 for correct measurement.

(11) Slide the spline adapter on the spline shaft and install the taper pin, washer and nut (Ref. Figure 203, 32-30-01).

NOTE: This actuator clearance check may be accomplished by applying clay (Ref. Figure 211, 32-30-01). Cycle the gear to the retracted position, then extend the gear and measure the thickness of the clay.

(12) Check the actuator clearance as described in CLEARANCE CHECK OF MAIN LANDING GEAR ACTUATORS in this section.

(13) Check for proper clearance between the spar and main gear, which should be 0.13 to 0.50 inch.

(14) Check the indicator lights for proper rig. If needed, perform the appropriate switch adjustment procedure (Ref. 32-60-00).

(15) Install the bolt, washer, nut and cotter pin that attach the cam to the link on the main gear doors (Ref. Figure 211, 32-30-01).

(16) Install the access panel (Ref. Figure 203, 32-30-01).

(17) Install the fender assembly on aircraft equipped with flotation gear.

(18) Cycle the landing gear, using the electric motor driven system, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(19) Ensure all three landing gear are down and locked.

(20) Remove the airplane from the jacks (Ref. Chapter 07-00-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

C. Clearance Check

Check the clearance around the actuator where it protrudes through the web of the spar whenever the landing gear is rigged or whenever the main gear, actuator, or drag leg is removed for overhaul or replacement. Since each of the aforementioned parts is shimmed for alignment with respect to the others, the position of the actuator in relation to the hole in the spar web will vary upon installation of any one of the preceding parts. Perform the clearance check as follows:

(1) Place airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Place a 1.5 inch square of clay 0.12 inch thick on top of the lower housing of the actuator.

(3) Fully retract the landing gear, then extend it to the down and locked position.

(4) Measure the thickness of the clay remaining below the cut.

(5) If the thickness just measured is less than 0.10 inch, trim the edge of the hole in the spar web only enough to obtain a clearance of 0.10 inch around the actuator. Make sure that there is adequate clearance between the actuator brake and the adjacent structure. If necessary, reposition the brake on the screw housing of the actuator to obtain this clearance.

6. NOSE LANDING GEAR ACTUATOR

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Retract the gear to 1/2 of the extended position to facilitate removal of the stop bolt, then remove the cotter pin, nut, washers, bushings and bolt securing the actuator yoke to the drag leg. Note the placement of the shim washers (if installed).

NOTE: The following Steps (3) and (4) are applicable on serials (U-1 through U-105 only). For (U-106 and after) omit Steps (3) and (4).

(3) Release tension on the chain at the idler sprocket in the nose wheel well (Ref. Figure 205, 32-30-01).

(4) Remove the bolts attaching the drive shaft support bearing to the side of the wheel well and slide the drive shaft off the actuator.

(5) Remove the two nuts and bolts, set at 90° from each other, from the drive shaft assembly and slide the shaft off the actuator spline.

(6) Remove the bolts securing the actuator support bearing to the brackets and remove the actuator from the aircraft.

B. Installation

NOTE: Inspect the actuator support bearing assemblies for corrosion or damage and ensure free rotation (Ref. Figure 209).

(1) Position the actuator and the support bearings in the support brackets and install the attaching bolts.

NOTE: Step (2) is applicable on (U-1 through U-105 only). For (U-106 and After) omit Step (2) and (3).

(2) Place the drive shaft into position on the actuator. If the drive shaft assembly attach holes do not align with the holes in the structure, elongate the holes in the structure and install doublers (Ref. Figure 201). Install the retract chain on the sprocket, and secure the support bearing with the attaching screws.

(3) Adjust the tension of the chain on (U-1 through U-105) (Ref. 32-30-01).

(4) Slide the drive shaft over the actuator spline and install the two bolts, set at 90° to each other, in the drive shaft.

(5) Connect the actuator to the nose gear drag brace by installing the stop bolt bushings, washers, nut and cotter pin in the same order as noted in disassembly. Refer to Actuator Alignment Procedure in this section if the yoke and drag brace do not operate smoothly.

(6) Rig the nose gear landing gear, (Ref. 32-30-01, RIGGING ALL THREE LANDING GEAR).

(7) Cycle the landing gear, using the electric motor driven system, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(8) Ensure all three landing gear are down and locked.

(9) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

C. Alignment

The nose gear actuator yoke should be checked for alignment with the drag brace in the following manner:

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disconnect the nose gear doors by removing the screws attaching the center hinge brackets to the left and right aft nose gear doors (Ref. Figure 204, 32-30-01).

(3) With the gear extended, remove the stop bolt and any washers between the yoke and drag brace. Locate the yoke in the proper position in relation to the drag brace.

NOTE: The yoke which attaches the actuator to the drag brace is not symmetrical about the actuator centerline. It is asymmetrical by 1/32 inch. This allows for adjustment of the alignment between the actuator and the drag brace. If it is found that the yoke does not align with the drag brace as desired, it may be rotated 180°, which should provide for improved alignment. Note the clearance between the yoke and drag brace for later reference.

(4) Insert a washer between the yoke and drag brace to determine that there is a slight resistance as it is pushed through.

(5) Insert the stop bolt attaching the drag brace to the actuator yoke and retract the gear slowly as far as possible, while still being able to measure the gap. Block the nose wheel in position and remove the stop bolt. Determine the clearance as in Step (3). If the clearance is not the same in the extended and retracted position, alignment is necessary.

NOTE: If alignment is necessary, check for cracked actuator bearing supports or other causes of misalignment.

There should be no side load against the actuator rod. If the measurement is too great, arrange any number of P/N AN960-PD816L, P/N AN960-816L or P/N AN960-816 washers on the stop bolt between the ends of the yoke and the upper drag leg members to attain alignment.

At least one washer is required on each end of the yoke.

(6) When spacing is suitable in retracted and extended positions, install the stop bolt and washers.

(7) Connect the nose gear doors.

(8) Cycle the landing gear, using the electric motor driven system, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(9) Ensure all three landing gear are down and locked.

(10) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

1. SCREW HOUSING COVER
2. ACTUATOR SCREW
3. UPPER COVER
4. GEARBOX UPPER BEARING
5. BOLT
6. SCREW HOUSING
7. HOUSING PLUG
8. ACTUATOR HOUSING
9. INNER THRUST PINION BEARING
10. PINION
11. OUTER PINION BEARING
12. SHIM
13. SNAP RING
14. GEARBOX LOWER THRUST BEARING
15. BUSHING
16. NUT SEAL
17. SHIM
18. HOUSING BOTTOM ASSEMBLY
19. BOLT
20. BUSHING (MATCHED SET)
21. ACTUATOR NUT
22. PLUG
23. PACKING
24. SPRING
25. YOKE ASSEMBLY
26. BUSHING
27. BUSHING

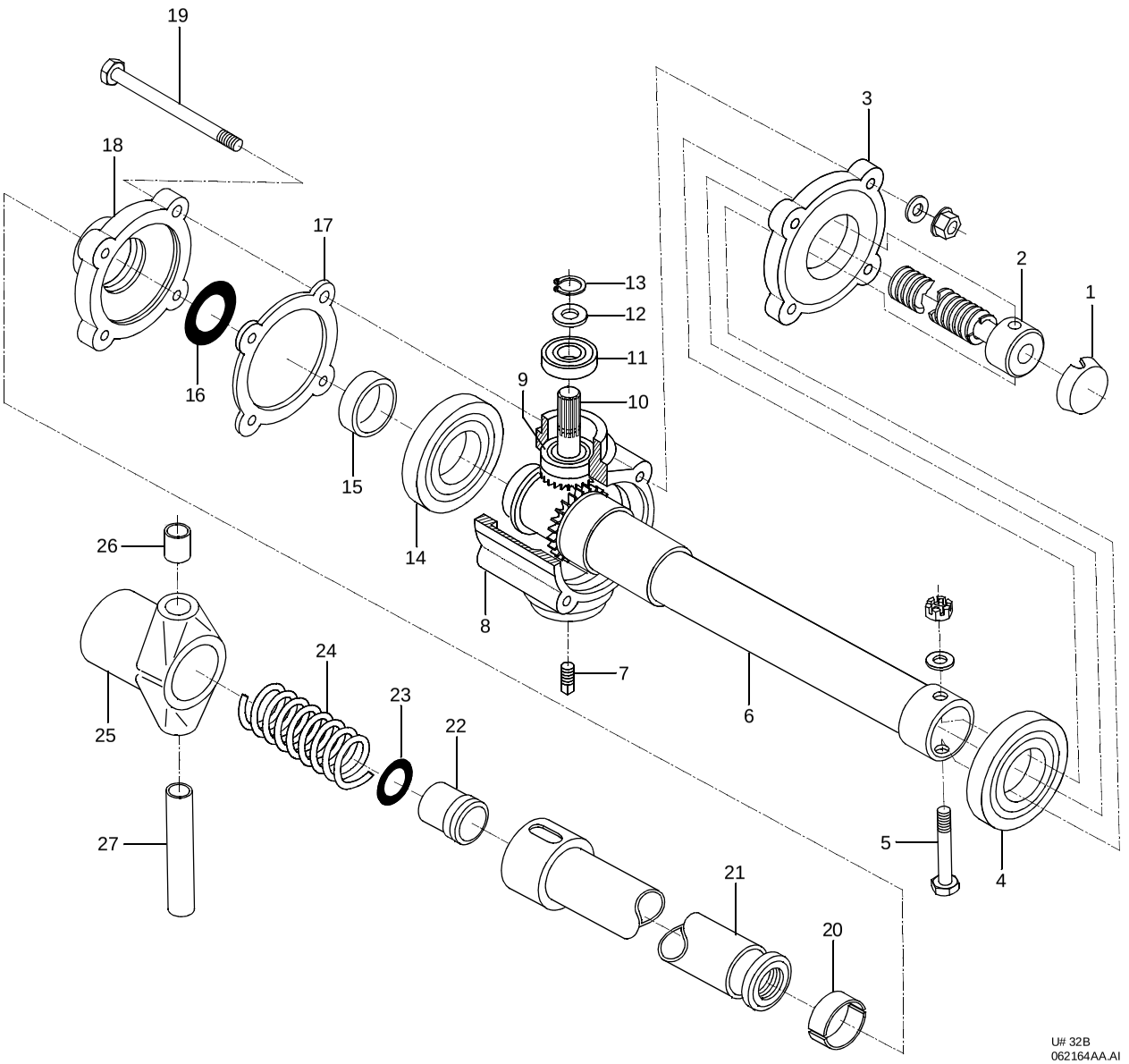


Figure 203
Nose Gear Actuator (Mechanical System)

7. NOSE AND MAIN LANDING GEAR ACTUATOR

A. Inspection

Any time the landing gear actuators are removed from the airplane for lubrication (every 1000 cycles of operation or 30 months, whichever occurs first), each actuator should be bench checked for total end play. Check each actuator as follows before disassembly for lubrication:

NOTE: Use a "Last Word" No. 711-F (product of L.S. Starrett Co. of Athol, Mass.), "Testmaster" (product of Federal Products Corp. of Providence, R.I.), or an equivalent dial indicator for these checks.

(1) Using an extension rod paralleled to the nut assembly (Ref. Figure 204), clamp a dial indicator to the nut assembly shaft so that the deflector lever of the dial indicator rests against the gear housing of the actuator. Flatten the extension rod where it contacts the nut assembly shaft to ensure firm clamping.

CAUTION: *To avoid damaging the actuator, DO NOT bottom out the actuator spring while compressing it to measure end play.*

(2) Applying approximately 100 pounds of force in either direction, move the nut assembly shaft in and out several times while measuring by dial indicator the distance over which the nut assembly may be pushed in and out of the actuator housing.

CAUTION: *Any rotation of the drive pinion, torque shaft, or nut assembly can cause an erroneous measurement.*

Check the actuator end play with the main landing gear actuator nut fully in and backed off one turn, with the nut half extended, and with the nut extended 7 inches. Measure the end play on the nose gear actuators with the actuator nut fully in and backed off one turn, with the nut half extended, and with the nut extended 10.5 inches.

(3) If the total end play measured in the preceding Step is more than 0.016 inch but less than 0.018 inch, the actuator may be returned to service, but must be checked for end play every 200 cycles. Whenever the total end play reaches 0.018 inch, the actuator should be overhauled or replaced.

(4) After the actuator has been lubricated, check the total end play to ensure that the unit has been correctly assembled following lubrication.

(5) The end play check of the input pinion gear of the screw jack actuators, both main and nose, should be accomplished with the actuators out of the airplane as explained below (Ref. Figure 206):

(a) Apply force of 125 to 150 pounds outward from the gear housing directly to the pinion shaft. If the thrust bearing is improperly installed, this will unseat the bearing and the end play will be 0.050 to 0.080 inch or more. This force may be applied by a 12 to 1 ratio lever blocking against the gearbox for a fulcrum, and applying an 11 pound force at the handle of the lever.

(b) If this check reveals proper installation, the torque shaft should be connected in the same position as removed.

(c) If the thrust bearing is found to be improperly installed in either the nose or main gear actuator, the unit should be overhauled or replaced. Ensure that the thrust bearing is installed as shown (Ref. Figure 205).

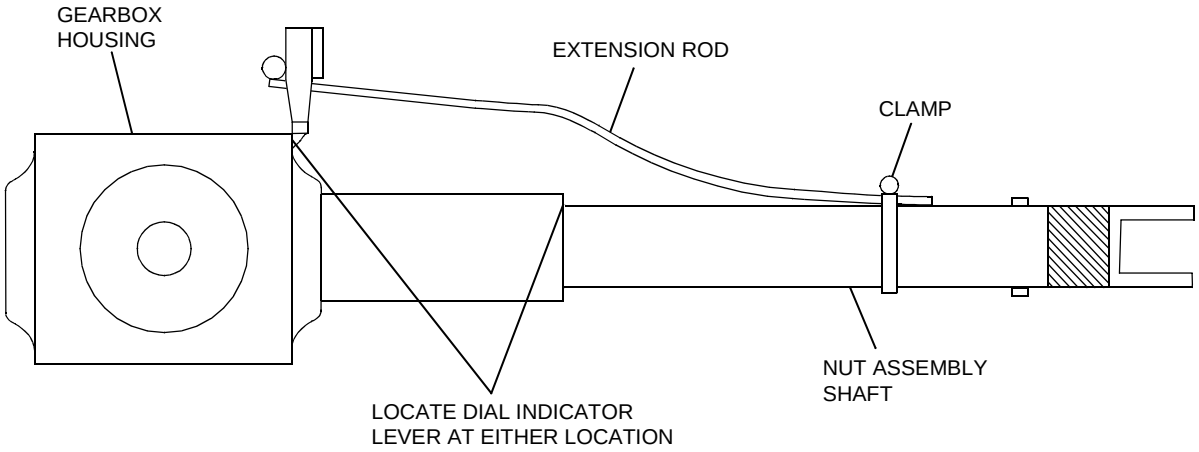


Figure 204
Landing Gear Actuator Inspection

8. MAIN LANDING GEAR ACTUATOR

A. Lubrication

NOTE: The main landing gear actuator should not be lubricated until the MAIN LANDING GEAR ACTUATOR INSPECTION has been completed.

Every 1000 cycles or 30 months remove the actuator from the airplane as described under MAIN LANDING GEAR ACTUATOR REMOVAL in this Chapter and disassemble as follows:

(1) Remove the four bolts attaching the cover (4) and housing bottom (21) to the actuator housing (9) (Ref. Figure 202).

NOTE: Be careful to mark the shims (20) to ensure installation in their same relative position.

(2) Remove the housing bottom and any shims (20) between it and the housing. Remove nut seal (22) from the housing bottom. The upper bearing may be driven out by hitting the end of the actuator nut (23) with a rawhide mallet. When the upper bearing is free from the housing, remove the bolt (6).

NOTE: Remove all old grease with solvent prior to filling.

(3) Following disassembly, pack the bearings, coat ALL internal casting surfaces of the gearbox housing and fill the gearbox housing approximately half full with grease (18, Table 6, 91-00-00).

(4) Completely assemble the actuator (Refer to Overhaul Instruction Manual, (P/N 98-36381), for Landing Gear Actuators) except for the actuator screw (2) and nut assembly (23). With the plug (25) bottomed out, fill in the nut assembly with grease (18, Table 6, 91-00-00) while taking care to

prevent air traps. With the nut in the fully retracted position, insert the nut and screw assembly into their respective end of the screw housing. Allow the excess grease to flow out of the screw ends, then finish assembling the actuator.

NOTE: Proof tests and break in are not required after lubrication, provided that the actuator is assembled with the original shims, or that new shims of the same thickness as the original shims are installed in the same location as the original shims and that no other new actuator components (with the exception of packings and seals) are installed. Refer to Overhaul Instruction Manual, (P/N 98-36381), for Landing Gear Actuators.

(5) Install the actuator in the airplane as described under MAIN LANDING GEAR ACTUATOR INSTALLATION in this Chapter.

9. NOSE LANDING GEAR ACTUATOR

A. Lubrication

NOTE: The nose landing gear actuator should not be lubricated until the NOSE GEAR ACTUATOR INSPECTION has been completed.

Every 1000 cycles or 30 months remove the actuator from the airplane as described under NOSE LANDING GEAR ACTUATOR REMOVAL in this Chapter and disassemble as follows:

(1) Remove the four bolts attaching the upper cover (3) and bottom housing assembly (18) to the actuator housing (8) (Ref. Figure 203).

(2) Remove the bottom assembly (18) and the shims (17) between it and the housing.

(3) Remove the nut seal (16) from the cover.

(4) The upper bearing (4) may be driven out by hitting the end of the actuator nut (21) with a rawhide mallet. When the upper bearing is free of the housing, remove the bolt (5), remove the actuator screw (2) from the actuator nut (21) and slide the actuator screw (2) and the actuator nut (21) out of the screw housing (6).

NOTE: Be careful to mark the shims (17) to ensure installation in their same relative position.

(5) Following disassembly, pack the bearings, coat ALL internal casting surfaces of the gearbox housing and fill the gearbox housing approximately half full with grease (18, Table 6, 91-00-00).

NOTE: Remove all old grease with solvent (38, Table 6, 91-00-00) prior to filling.

(6) Completely assemble the actuator (Refer to Overhaul Instruction Manual, P/N 98-36371, for Landing Gear Actuators) except for the actuator screw (2) and actuator nut assembly.

CAUTION: *Refer to the Overhaul Instruction Manual (P/N 98-36371) for assembly of the landing gear actuators. Be certain to install the bearings as explained in the Overhaul Manual and as illustrated in Figure 205.*

(7) With the plug (22) bottomed out, fill the nut assembly with grease (18, Table 6, 91-00-00) while taking care to prevent air traps.

(8) With the nut in the fully retracted position, insert the nut and screw assembly into their respective end of the screw housing. Allow the excess grease to flow out of the screw ends, then finish assembling the actuator.

NOTE: Proof test and break in are not required after lubrication, provided that the actuator is assembled with the original shims, or that new shims of the same thickness as the original shims are installed in the same location of the original shims and that no other new actuator components (with the exception of packings and seals) are installed. Refer to Overhaul Instruction Manual, (P/N 98-36381), for Landing Gear Actuator.

Install the actuator in the airplane as described under NOSE LANDING GEAR ACTUATOR INSTALLATION in this Chapter.

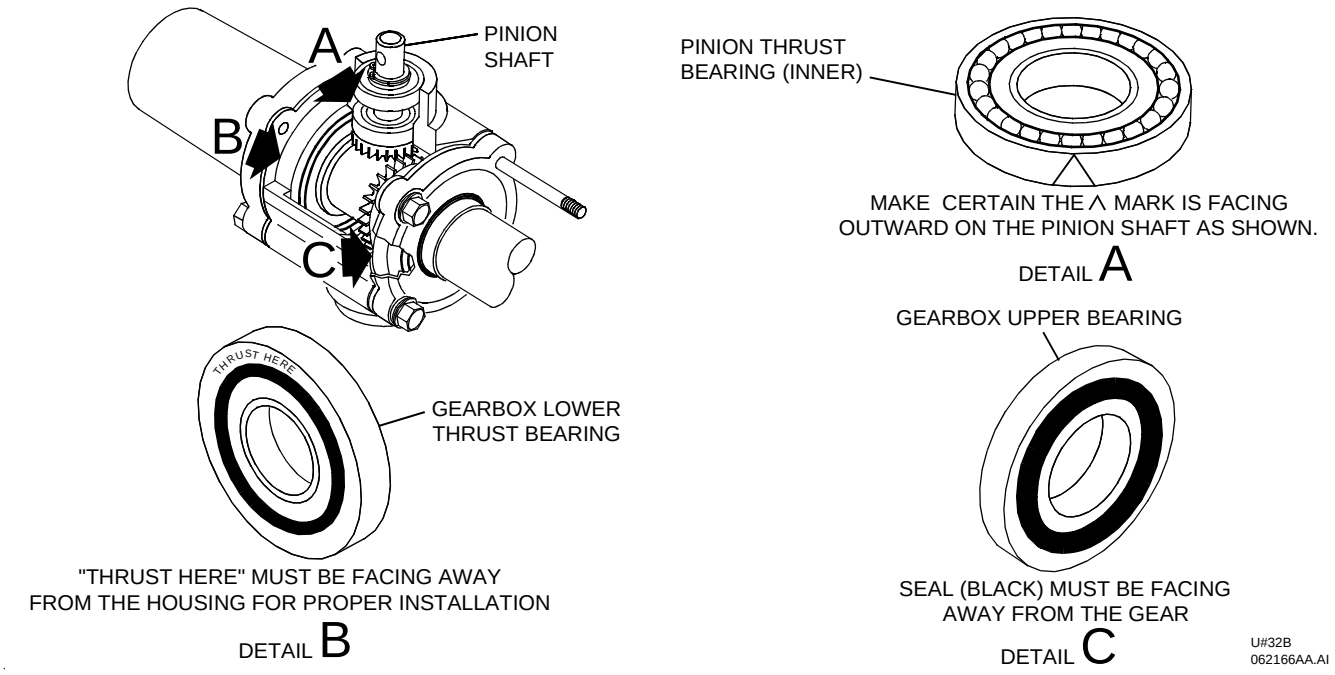


Figure 205
Pinion Thrust Bearing Installation

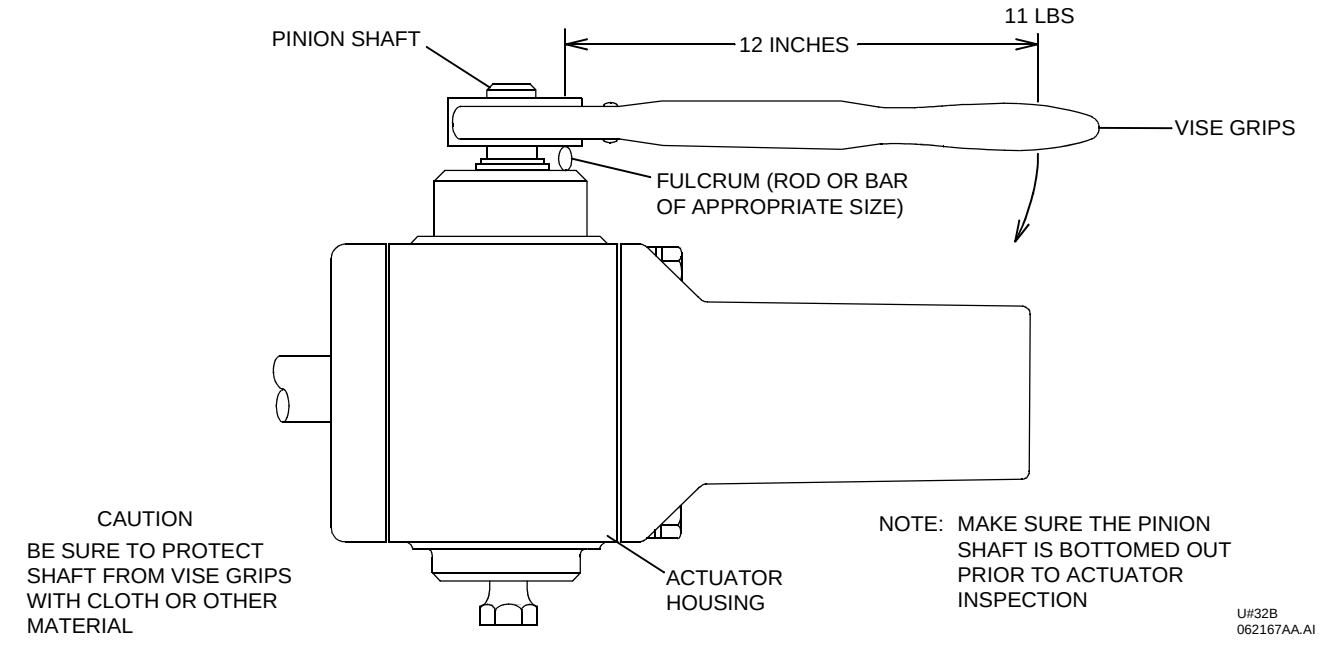


Figure 206
End Play Check Input Pinion Gear

10. MAIN LANDING GEAR INBOARD TORQUE SHAFT

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the carpet, center aisle floorboard, RH passenger seat and RH oval access panel just forward of the main spar.

(3) Remove access panel (Ref. Figure 203, 32-30-01) and pull the landing gear circuit breaker.

(4) By placing the landing gear control handle in the up position and intermittently "bumping" the landing gear circuit breaker, partially retract the landing gear until the divider link (identified by yellow paint) of the aft retract chain for the nose landing gear is in an accessible location.

(5) Release the nose gear aft retract chain tension at the first set of sprockets forward of the gearbox and separate the chain at the divider link.

(6) Remove the nut, washer and taper pin from the spline adapter (U-joint, on U-152 and After) adjacent to the landing gear gearbox and slide the spline adapter outboard on the torque shaft.

(7) With the curved handle of the manual extension system disengaged, operate the long red handle until the divider link (identified by yellow paint) of the manual extension chain is in an accessible location. Release the chain tension by loosening the two bolts in the support casting on the RH side of the manual extension drive shaft and separate the chain at the divider link.

(8) Remove the nuts, washers and taper pins from the spline adapters (Ref. Figure 203, 32-30-01). Separate the torque shafts from the spline adapters.

(9) Rotate the torque shafts as necessary to remove the nut, washer and taper pin from immediately inboard of the center carrier bearing. Access is gained through the access opening (Ref. Figure 203, 32-30-01).

(10) Move the LH torque shaft inboard, then out through the center aisle floorboard opening.

(11) Move the RH torque shaft inboard and out, after first removing the landing gear gearbox as described in LANDING GEAR MOTOR AND GEARBOX REMOVAL.

B. Installation

(1) Working through the floorboard access openings, position the torque shafts and install the landing gear gearbox as described in LANDING GEAR MOTOR AND GEARBOX INSTALLATION in this section.

(2) Working through the access openings (Ref. Figure 203, 32-30-01) install the taper pins, washers and nuts to connect the inboard torque shafts to the center carrier bearing assembly.

(3) Install the taper pins, washers and nuts with the spline adapters (Ref. Figure 203, 32-30-01).

(4) Install the taper pin, washer and nut with the spline adapter (U-joint on U-152 and After) adjacent to the landing gear gearbox.

(5) Check to be sure the aft retract chain for the nose gear is aligned on all the sprockets, then install the divider link. Take up the chain slack at the first set of sprockets forward of the gearbox. Adjust the chain tension when the landing gear are rigged.

(6) Check to be sure the manual extension system chain is aligned on all the sprockets and install the divider link. Adjust the chain (Ref. 32-30-01).

(7) Rig the landing gear as necessary (Ref. 32-30-01).

(8) Install the access panel (Ref. Figure 203, 32-30-01), then install the floorboard access panels, the carpet and passenger seats which were removed.

11. MAIN LANDING GEAR OUTBOARD TORQUE SHAFT

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the access panels, taper pins and spline adapters (Ref. Figure 203, 32-30-01).

(3) Remove the nuts, washers and taper pins located on the torque shafts, just inboard of the actuator universal joint.

NOTE: The LH torque shaft can now be removed by sliding it out through the access panel opening (Ref. Figure 203, 32-30-01).

(4) Remove the two nuts, washers and inboard taper pins from the collar which connects the halves of the RH torque shaft and remove the two separate pieces out through the access panel opening (Ref. Figure 203, 32-30-01).

B. Installation

(1) Position the two halves of the RH shaft, slide the halves together at the collar and connect them by installing the taper pins, washers and nuts through the collar and shaft end.

(2) Position the LH outboard torque shaft.

(3) Install the nuts, washers and taper pins just inboard of the actuator universal joint.

(4) Install the taper pins, washers and nuts with the spline adapters and install the access panels (Ref. Figure 203, 32-30-01).

(5) Cycle the landing gear, using the electric motor driven system, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(6) Ensure all three landing gear are down and locked.

(7) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

12. NOSE LANDING GEAR AFT RETRACT CHAIN

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the seat and floorboards on the right side of the pilot's compartment, the seats and access openings on the right side forward of the main spar and the center aisle access panels forward of the main spar.

(3) It is helpful but not necessary to remove the access panel on the bottom of the fuselage directly below the motor gearbox and retract chain if the baggage pod is not installed.

(4) Operate the retraction system intermittently using auxiliary power, until the divider link, which is identified with yellow paint, is positioned approximately two feet forward of the motor gearbox.

(5) Release the chain tension by loosening the adjustable idler sprocket (first set of sprockets forward of the motor gearbox).

(6) Disconnect the chain at the divider link and attach a feeder wire to the center end link of the chain. Lift the chain off of the crossover shaft sprocket and allow the chain to rest on the crossover shaft.

(7) Pull the chain through until the feeder wire is resting on the crossover shaft, then detach the chain from the feeder wire and leave the feeder wire over the crossover shaft under tension.

(8) Remove the chain.

B. Installation

(1) Preclean the chain with a cloth slightly dampened with stoddard solvent (38, Table 6, 91-00-00). Using a brush or spout can, lubricate the chain with SAE 30W mineral oil (83, Table 6, 91-00-00). Apply oil to the chain on the upper edges of the linkplates to obtain access to the pin bushing joints through the clearances between the roller chain linkplates. Wipe off the excess oil from the chain after lubrication.

NOTE: The chain is lubricated, as described in Step (1) every 500 hours of airplane operation.

(2) Attach the feeder wire which was retained over the crossover shaft to the center end link of the chain. Pull the feeder wire and chain over the crossover shaft, through the bulkheads and over the remaining sprockets. Detach the feeder wire from the chain and lift the chain off of the crossover shaft and onto the crossover shaft sprocket. Connect the divider link of the chain.

CAUTION: *Check the chain for alignment on the sprockets. Chain sidebars riding hard on the side of a sprocket would indicate improper installation on one of the other sprockets.*

(3) Perform the RIGGING ALL THREE LANDING GEAR procedure (Ref. 32-30-01).

(4) Cycle the landing gear, using the electric motor driven system, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(5) Install the floorboard and seat in the pilot's compartment, the seats and access openings on the right side forward of the main spar, the center aisle access panel forward of the main spar and, if removed, the access panel on the bottom of the fuselage directly below the motor gearbox.

(6) Ensure all three landing gear are down and locked.

(7) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

13. NOSE LANDING GEAR FORWARD RETRACT CHAIN

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the copilot's seat and floorboards. Remove the seats and access openings on the right side and center aisle access panel forward of the main spar. Operate the retraction system intermittently using auxiliary power until the aft retract chain divider link, which is identified with yellow paint, is positioned approximately two feet forward of the motor gearbox. Release the aft retract chain tension by loosening the adjustable idler sprocket (first set of sprockets forward of the motor gearbox).

(3) Disconnect the aft retract chain at the divider link and lift the chain off of the crossover shaft sprocket and allow the chain to rest on the crossover shaft, leaving the remaining portion of the chain in the original position.

(4) Remove the aft RH floorboard from the nose baggage compartment.

(5) Disconnect the nose gear doors by removing the screws attaching the center hinge brackets to the left and right aft nose gear doors (Ref. Figure 204, 32-30-01). Remove the forward nose gear chain covers located on the right hand side of the nose wheel well (Ref. Figure 205, 32-30-01). Remove the condenser intake grill in the nose wheel well if installed.

(6) Position the forward chain by rotating manually until the divider link, which is identified with yellow paint, is accessible. Release the tension on the forward chain by loosening the adjustable idler sprocket in the wheel well (Ref. Figure 205, 32-30-01).

(7) Disconnect the chain at the divider link and attach a feeder wire to the center end link of the chain. Pull the chain and feeder wire through and over the sprocket at the crossover shaft. Detach the chain from the feeder wire and leave the feeder wire over the sprocket under tension to facilitate installation of the chain on the sprocket.

(8) Remove the chain from the airplane.

B. Installation

(1) Preclean the chain with a cloth slightly dampened with stoddard solvent (38, Table 6, 91-00-00). Using a brush or spout can, lubricate the chain with mineral oil (83, Table 6, 91-00-00). Apply oil to the chain on the upper edges of the linkplates to obtain access to the pin bushing joints through the clearances between the roller chain linkplates. Wipe off the excess oil from the chain after lubrication.

NOTE: The chain is lubricated, as described in Step (1) every 500 hours of airplane operation.

(2) Attach the feeder wire to the center end link of the chain.

(3) Pull the chain through and align over the crossover shaft properly.

CAUTION: *Check the chain for alignment on the sprockets. Chain sidebars riding hard on the side of a sprocket would indicate improper installation on one of the other sprockets.*

NOTE: The aft drive sprocket is visible through the nose baggage compartment where the floorboard was removed.

(4) Connect the divider link of the forward chain.

(5) Lift the aft retract chain off of the crossover shaft and onto the crossover shaft sprocket. Connect the divider link and rig the aft chain with the correct tension (Ref. 32-30-01).

(6) Rig the forward chain with the correct tension (Ref. 32-30-01).

(7) Rig the landing gear (Ref. 32-30-01).

(8) Install the chain guards in the wheel well, the condenser grill if installed, the floorboards and seat in the pilot's compartment, the access openings on the right side forward of the main spar and the center aisle access panel forward of the main spar, and the nose baggage floorboard.

(9) Connect the nose gear doors by inserting the screws attaching the center hinge brackets to the left and right aft nose gear doors (Ref. Figure 204, 32-31-00).

(10) Cycle the landing gear, using the electric motor driven system, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(11) Ensure all three landing gear are down and locked.

(12) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

14. RETRACT CHAIN AND SPROCKET

A. Inspection and Replacement

The system should be inspected at the first 100 hour inspection and at normal periodic inspections thereafter.

(1) Carefully inspect the landing gear retract chain for dirt, rust, and corrosion.

CAUTION: *Under no circumstances should rust or corrosion be removed with an acid. Acid may embrittle and crack the highly heat treated links.*

(2) Clean the contaminated chains with a petroleum solvent.

(3) Replace excessively rusted or corroded chains.

(4) Visually inspect all links to assure that pin and link wear is approximately uniform. Check for excessive wear on the inner surface of the roller link sidebars and on the sides of the sprocket teeth, which would indicate a misalignment problem that should be corrected immediately to prevent undue wear of chain and sprocket teeth. Excessive sprocket tooth wear will result in "hook" shaped teeth. When sprockets are found in this condition, they should be replaced. Proper chain tension is very important and links should be removed as necessary to maintain required adjustment. It should be noted, however, that sprocket or chain wear or both can cause a chain with proper tension to jump teeth and become out of "time". When a chain is installed, all chain drives should have an initial inspection after retracting and extending the landing gear several times to assure that the chain was installed properly and the required chain tension still exists.

(5) Load the chain to approximately 18 pounds and measure the length over any given number of links. The portion of the chain being measured should be less than 1.015 times the number of links times 0.250 inch in length. If the measured portion exceeds this length, the chain is worn beyond the allowable limits and must be replaced. (Example: The chain is approximately 216 inches long and has approximately 864 links. 1.015 x 864 x 0.250 = 219.24 inches or 219.24 - 216.00 = 3.24 inches maximum length increase.) With these allowable wear and stretch limits, it becomes apparent that the chain will become loose enough to require removal of links from time to time before the chain would have to be replaced.

15. MANUAL EXTENSION SYSTEM CHAIN

A. Removal

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the right hand passenger seats and access openings on the right side and the center aisle access panel forward of the main spar.

(3) Position the divider link, which is identified with yellow paint, just forward of the motor gearbox by using the manual extension handle.

CAUTION: *The curved handle for the emergency extension system must remain in the disengaged position.*

(4) Release the chain tension by loosening the two bolts in the support casting on the right hand side of the manual extension shaft and sprocket.

(5) Disconnect the divider link and remove the chain.

B. Installation

(1) Preclean the chain with a cloth slightly dampened with solvent (38, Table 6, 91-00-00). Using a brush or spout can, lubricate the chain with mineral oil (83, Table 6, 91-00-00). Apply oil to the chain on the upper edges of the linkplates to obtain access to the pin bushing joints through the clearances between the roller chain linkplates. Wipe off the excess oil from the chain after lubrication.

NOTE: The chain is lubricated, as described in Step (1) every 500 hours of airplane operation.

(2) Route the chain over the sprockets and connect the chain at the divider link.

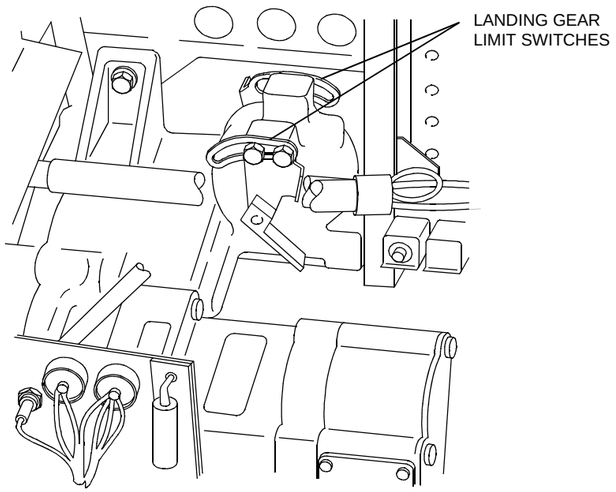
(3) Perform the RIGGING THE MANUAL EXTENSION SYSTEM procedure (Ref. 32-30-01).

(4) Cycle the landing gear, using the electric motor driven system, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

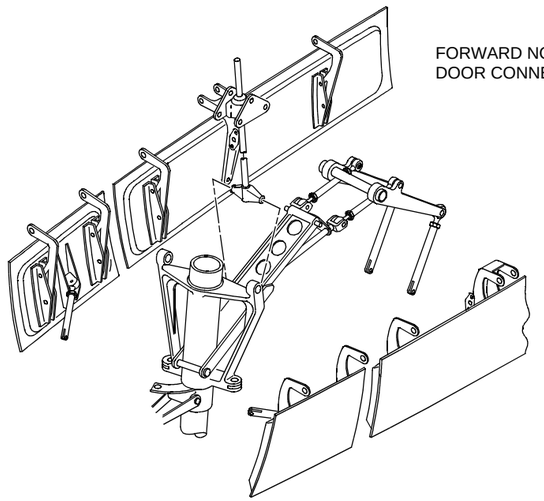
(5) Install the center aisle access panel forward of the main spar, the right hand access openings on the right side forward of the main spar and the right hand passenger seats.

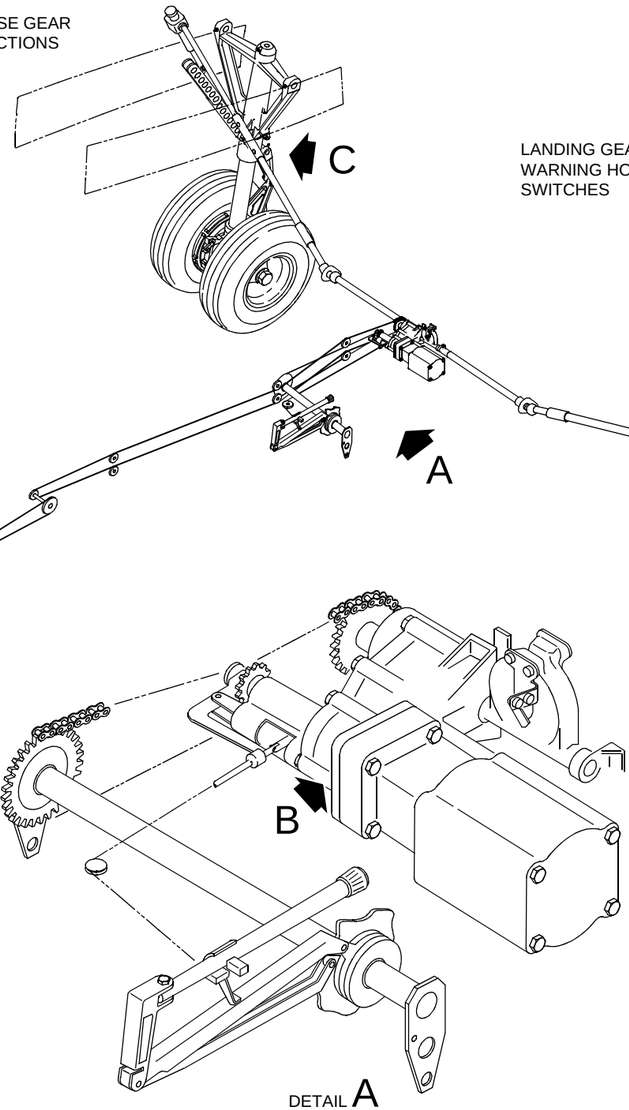
(6) Ensure all three landing gear are down and locked.

(7) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

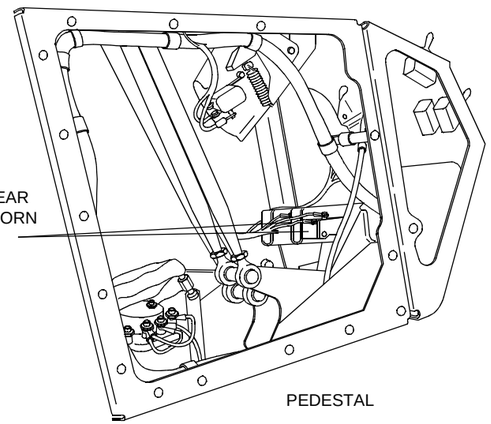


LANDING GEAR MOTOR AND GEARBOX
DETAIL B





DETAIL A



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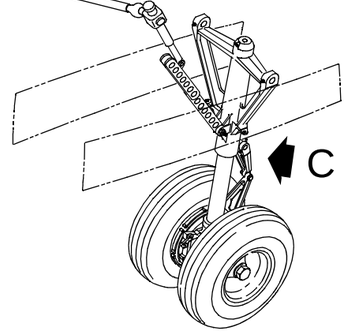

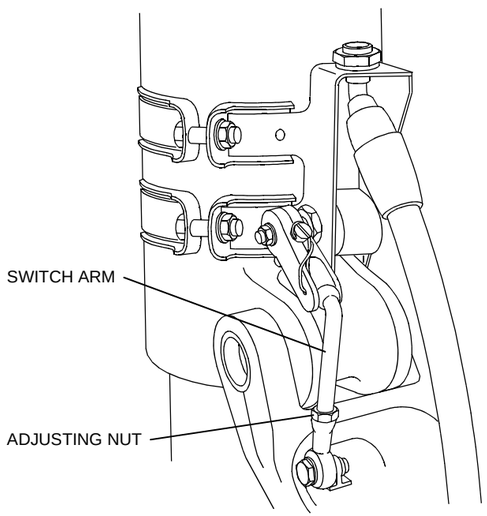

LANDING GEAR SAFETY SWITCH
DETAIL C

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Figure 207
Landing Gear Manual Extension System

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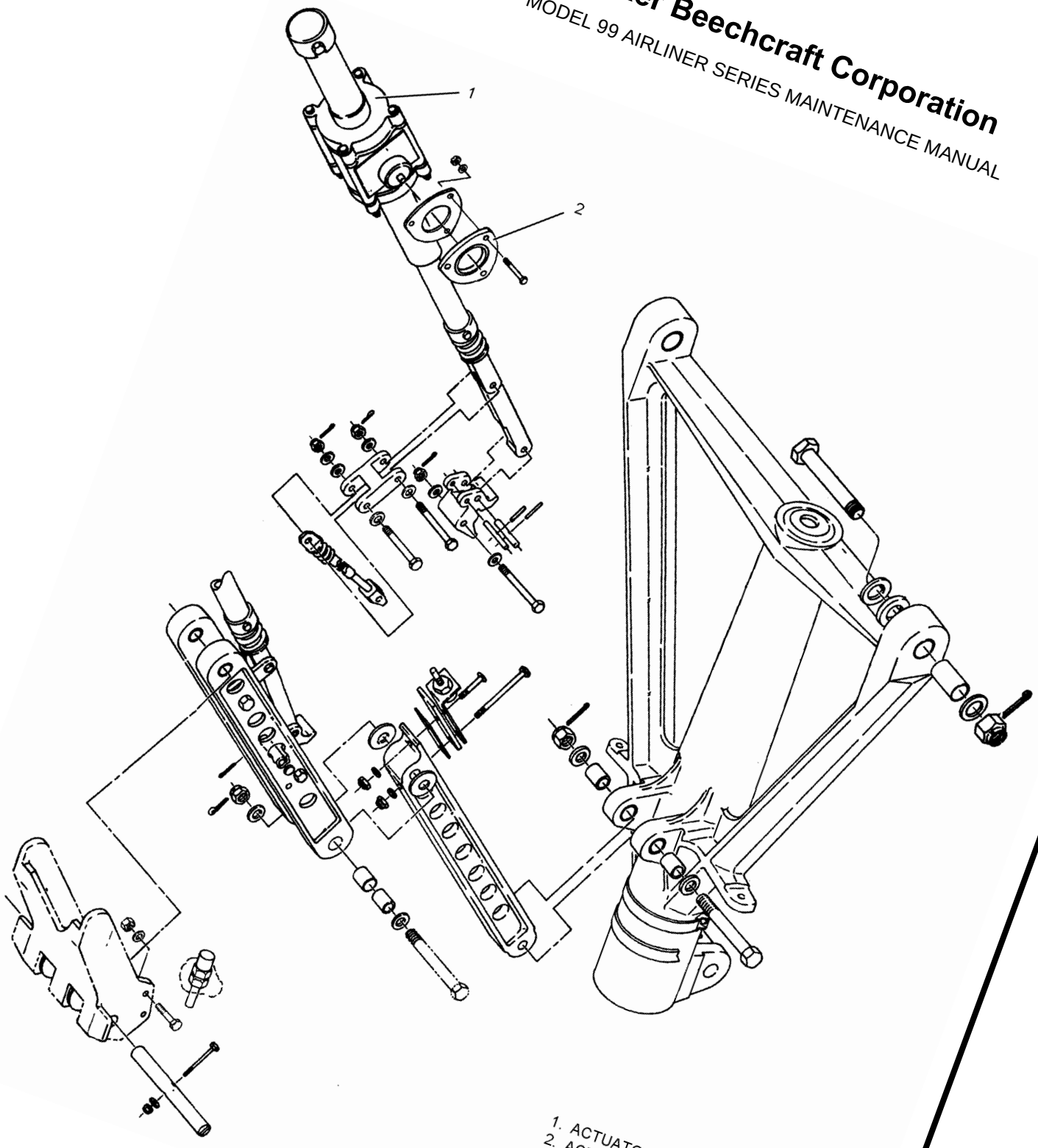


Figure 208
Main Landing Gear Actuator Support Bearing

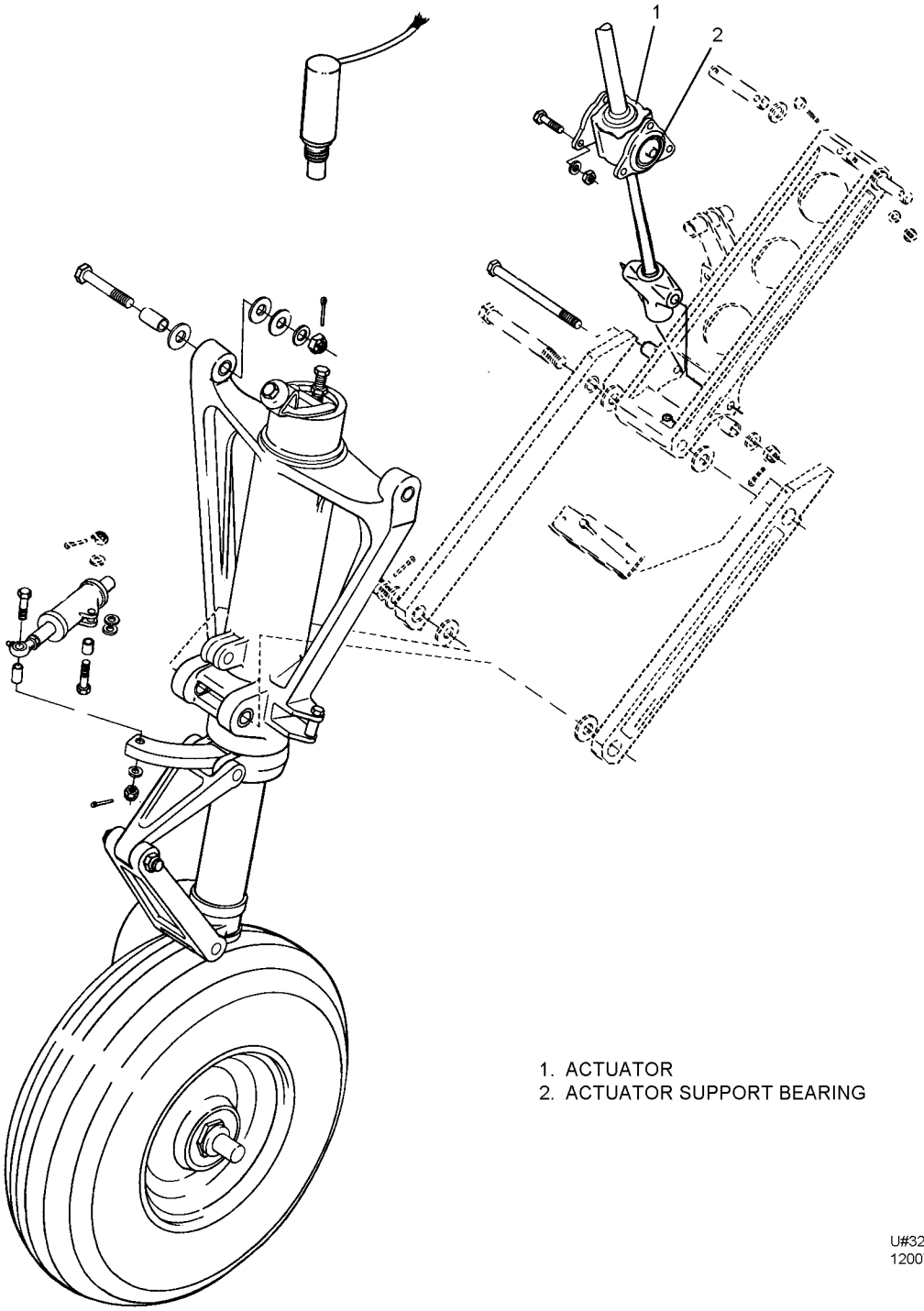


Figure 209
Nose Landing Gear Actuator Support Bearing

LANDING GEAR
MECHANICAL LANDING GEAR SYSTEM
MAINTENANCE PRACTICES

1. MECHANICAL LANDING GEAR

A. Rigging

NOTE: Emergency extension handle must be stowed while rigging electrically.

Battery voltage is not adequate to properly cycle the landing gear during rigging. When performing a rigging operation, use a 28.25 ± 0.25 volt auxiliary power unit, capable of maintaining the initial setting within 0.50 volts during the retraction and extension cycles.

During the operation, an assistant may be stationed inside the cockpit to operate the landing gear control switch. The use of an assistant is recommended in order that the mechanic outside the airplane can more readily observe the action of the landing gear during the retraction and extension cycles The recommended method for actuating the landing gear motor is through the use of the landing gear circuit breaker located on the circuit breaker panel to the right of the copilot. By applying a push pull motion to the circuit breaker the man in the cockpit can stop the landing gear in any position during either the retraction or extension cycle so that closer examination or measurements may be made.

2. REMOTE LANDING GEAR SWITCH

An alternate method for operating the landing gear from outside the airplane (if an assistant is not available) is through the application of a momentary on-off-on switch and an adequate length of three wire cord to reach from the landing gear motor to the nose wheel well area. Attach the wire to the airplane as follows (Ref. Figure 201):

(1) Remove the landing gear motor access plate from the bottom of the fuselage.

(2) Pass the terminal ends of the wire cord through the access plate and connect them as follows:

(a) Connect the UP terminal of the control switch to wire No. G5B22 terminal on the UP limit switch.

(b) Connect the DOWN terminal of the control switch to wire No. G8A22 terminal on the down limit switch, and connect the common positive wire No. G1AO terminal at the remote landing gear relay. Refer to the applicable electrical wiring diagram manual (99-0028 or 99-590030-5).

WARNING: Ensure the remote switch ON positions are properly labeled as UP and DOWN (Ref. Figure 201).

(3) Turn on the battery switch.

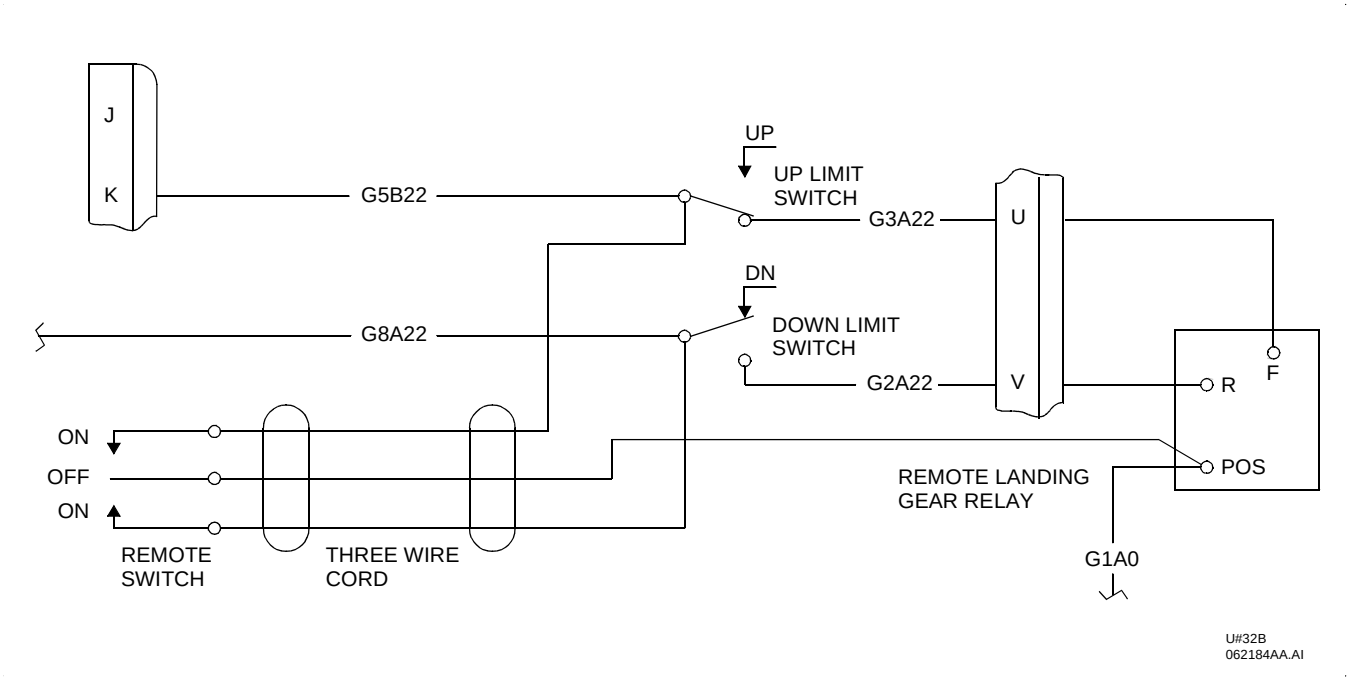
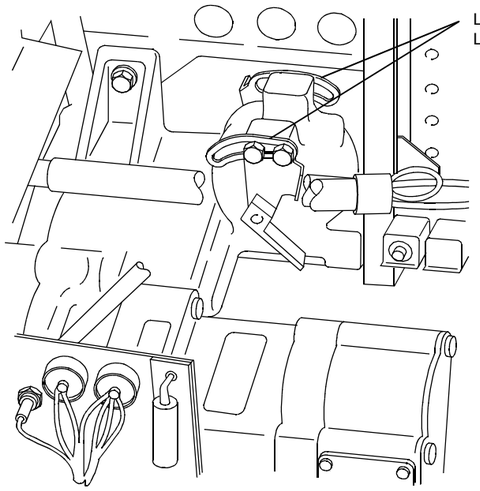
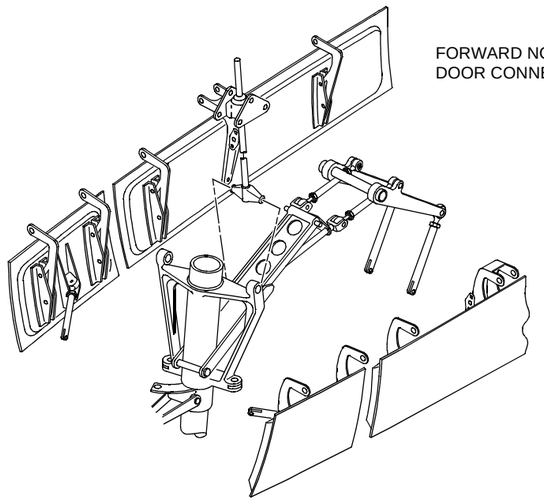


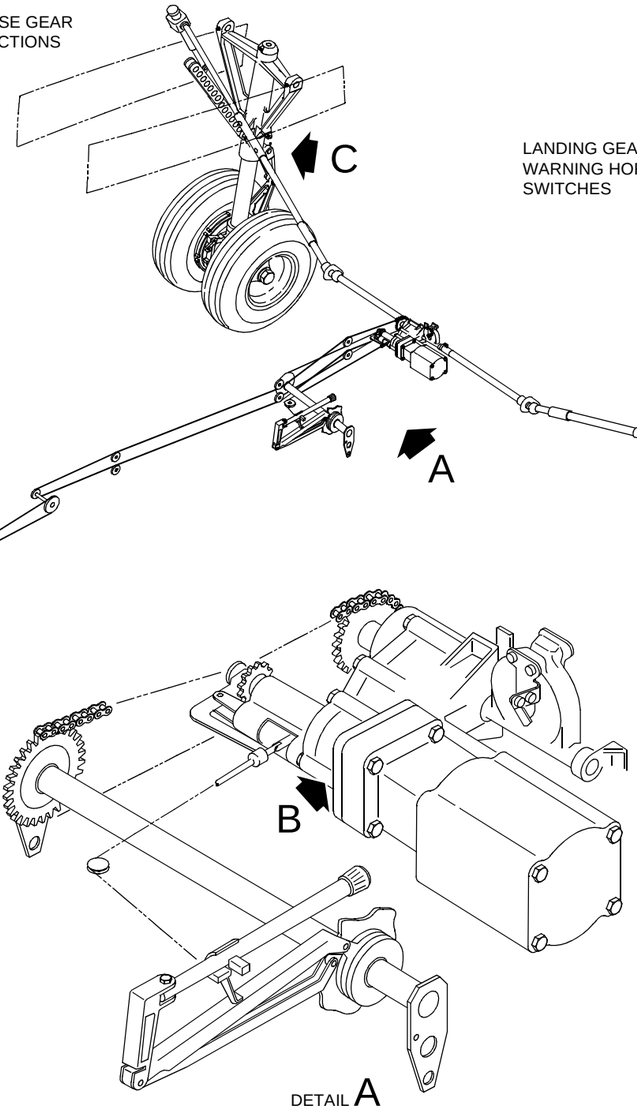
Figure 201
Remote Landing Gear Switch



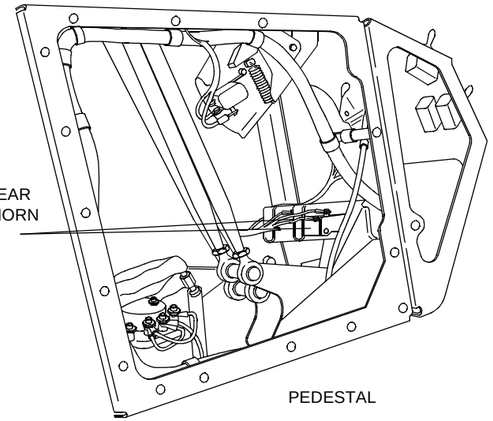
LANDING GEAR MOTOR AND GEARBOX
DETAIL B

LANDING GEAR SAFETY SWITCH
DETAIL C





DETAIL A



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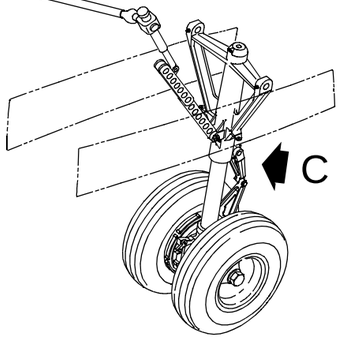

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Figure 202
Landing Gear Manual Extension System

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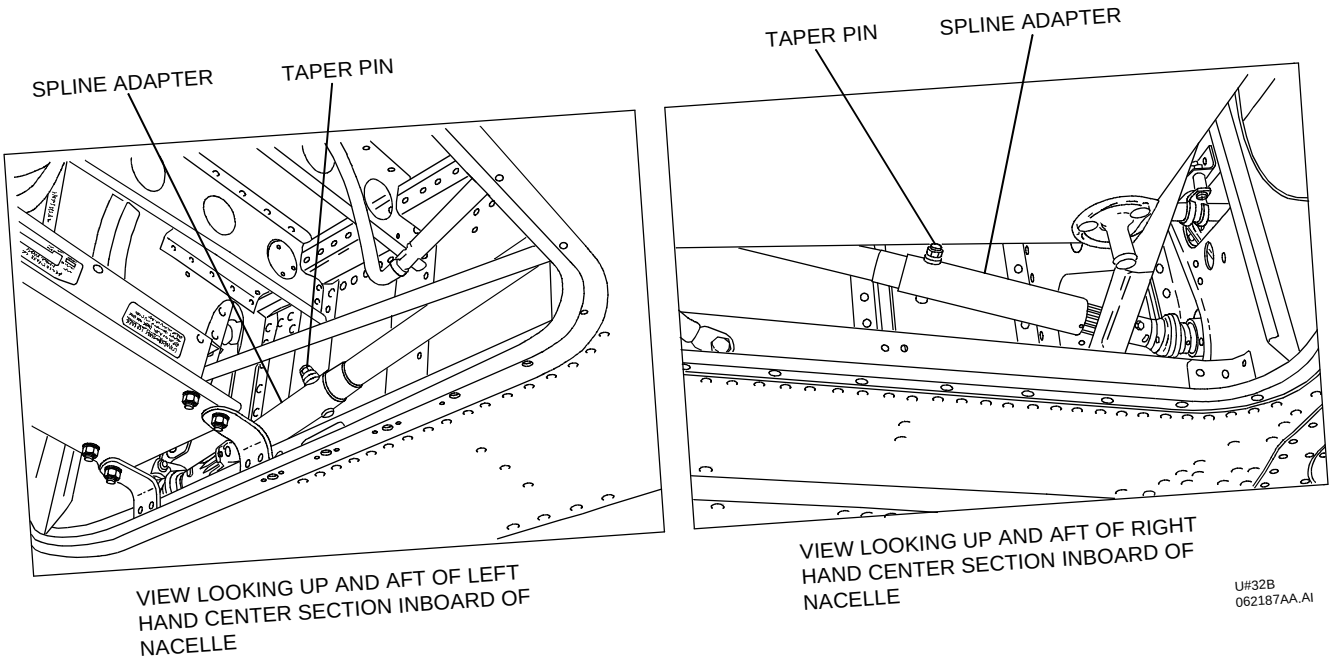

Figure 203
Access Openings and Spline Shaft Adapters

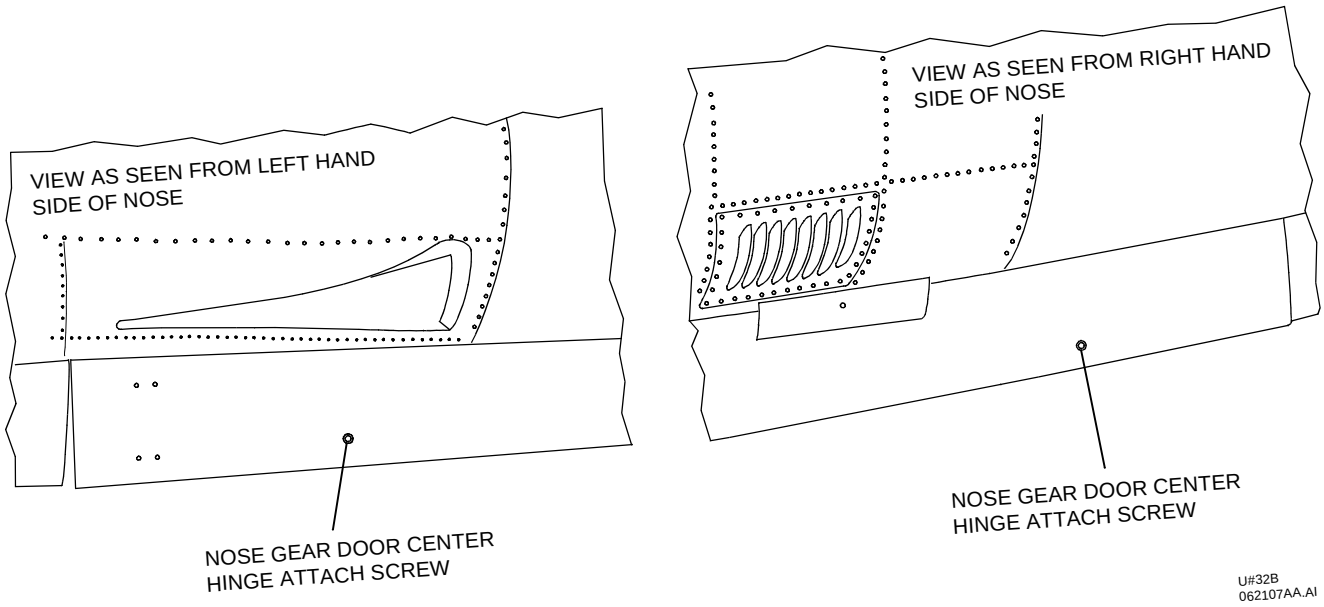

Figure 204
Nose Gear Door Attach Screws

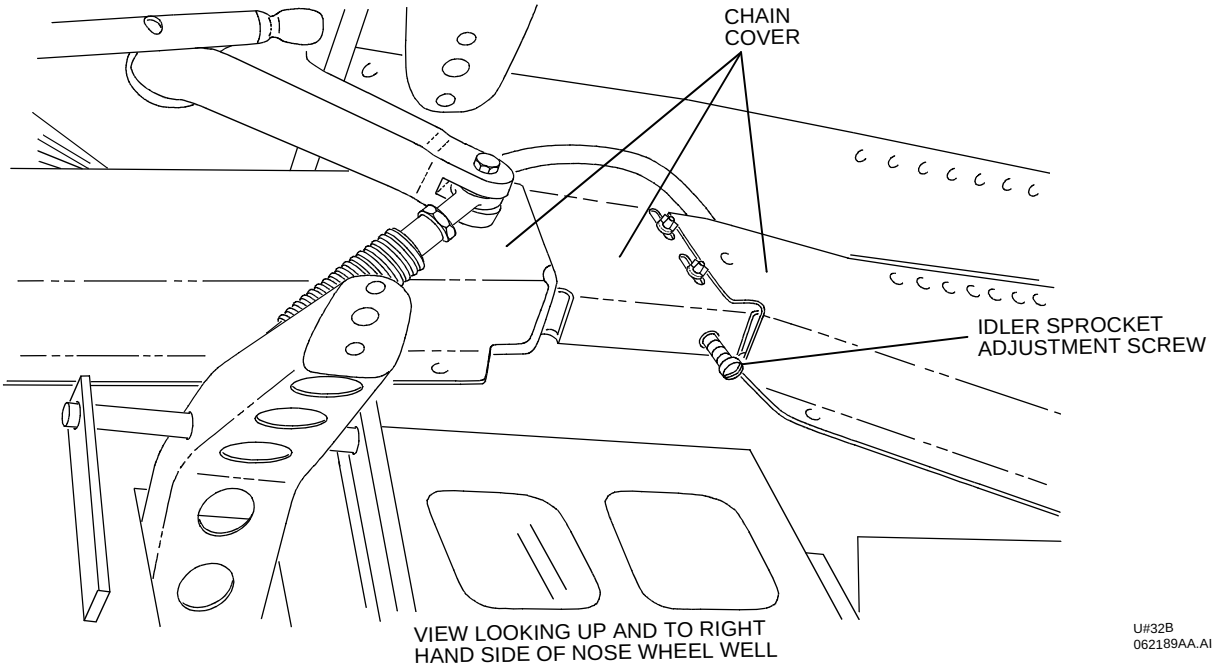

Figure 205
Chain Cover and Adjustment Screw

3. ALL THREE LANDING GEAR

A. Rigging

Rigging all three landing gear is accomplished as follows (Ref. Figure 202):

CAUTION: *Do not fully cycle landing gear electrically until gear are properly rigged.*

WARNING: Landing gear rigging shall be performed by using a 28 to 28.5 volt external power auxiliary power unit that will maintain the initial setting within 0.5 volt during the retraction and extension cycles. Battery voltage is not adequate to properly cycle the landing gear repeatedly during rigging. The external power is necessary to deliver full and consistent electrical power while making a number of landing gear cycles to ensure correct rigging.

(1) Place the airplane on jacks (Ref. Chapter 07-00-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the left hand and right hand access panels (Ref. Figure 203).

(3) Remove the taper pins from the spline adapters at the universal joints and remove the spline adapters from the spline to disconnect the main gear mechanisms.

(4) Disconnect the nose gear doors by removing the screws attaching the center hinge brackets to the left and right aft nose gear doors (Ref. Figure 204).

(5) Remove the nose gear chain covers located on the right hand side of the nose wheel well (Ref. Figure 205).

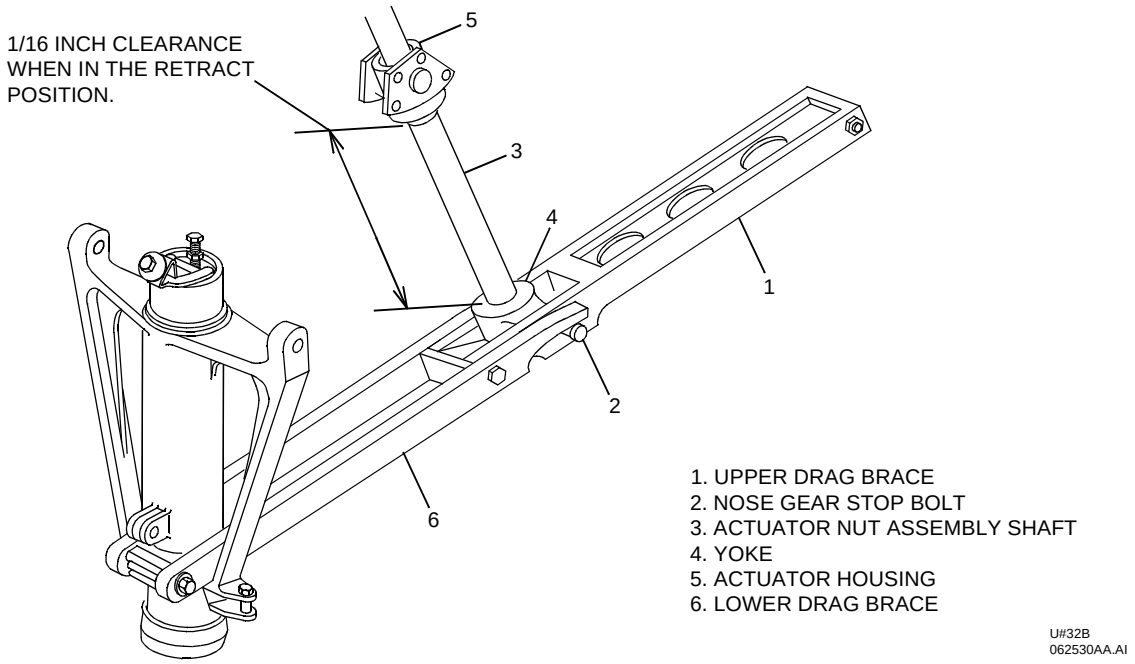

Figure 206
Nose Gear Rigging Measurement

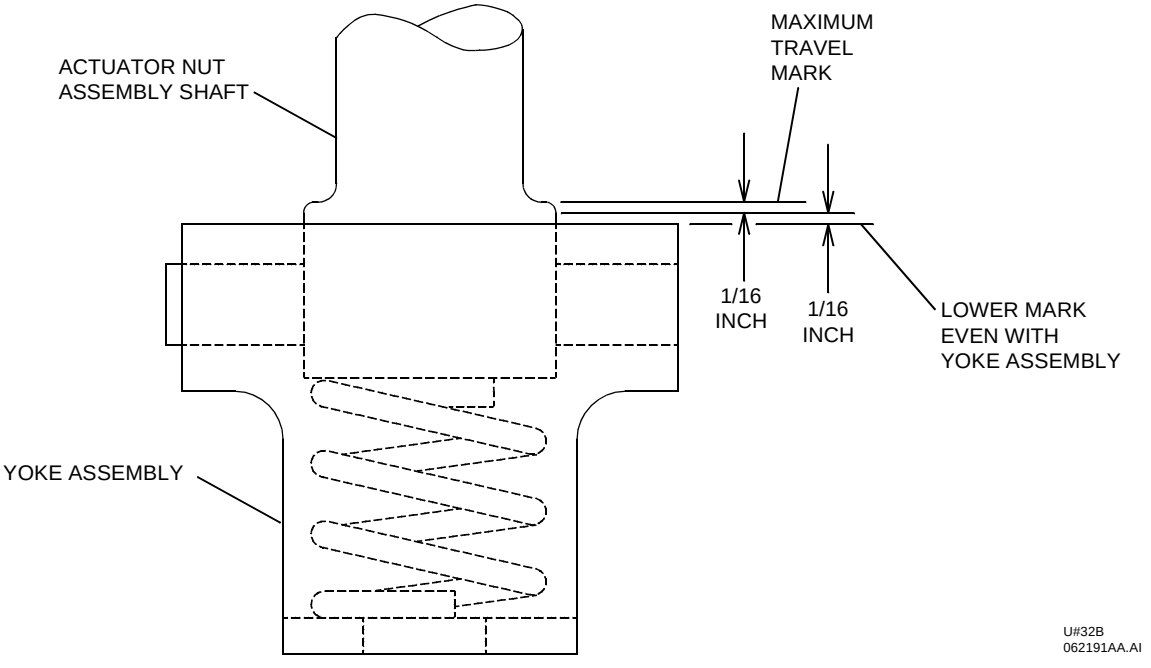

Figure 207
Measuring Nose Gear Actuator Spring Compression

(6) Remove the center aisle floorboards back to the main spar to check the retract chains for clearance with adjacent structure and proper alignment on idler sprockets.

(7) Apply a 10 pound load on the upper strand of the forward chain approximately half way between the adjustable idler sprockets and the rear of the nose wheel well. The deflection should be 1.00 +0.25/ -0.00 inch. Adjustment is accomplished by turning the screw at the bottom of the idler sprocket bracket at FS 39.00.

(8) Apply a 9 to 11 pound load to the lower strand of the aft chain at a point approximately half way between the second and third idler sprockets forward of the retract gearbox. The deflection should be 1.00 ± 0.12 inch. Adjustment is accomplished by raising or lowering the first set of idler sprockets forward of the retract gearbox.

NOTE: Removal of a link from the chains may be necessary if the adjustable idler sprockets have been adjusted to their maximum travel. Before the link is removed from the chain, the chains and sprockets should be inspected for excessive wear (Ref. 32-30-01).

(9) Remove the nose gear stop bolt (2) from the upper drag brace (1) and yoke (4) as follows (Ref. Figure 206):

(a) Remove the cotter pin, nut, and washer and slowly retract the gear to approximately 1/2 of the extended position.

(b) With the gear in this position, push the stop bolt out and through the hole in the right side of the nose wheel well keel. Let the gear down and allow to swing free.

(10) With a sharp pencil, mark a line on the actuator nut assembly shaft adjacent to the yoke assembly with the actuator spring not compressed (Ref. Figure 207).

(11) Mark two lines in increments of 1/16 inch directly above the first mark.

(12) Insert two rig stop bolts, 1 1/2 inches in length and 3/8 inch diameter with a 1/16 inch wall thickness bushing on each bolt, into opposite sides of the drag brace (Ref. Figure 208).

(13) Pull the drag brace down until the two rig stop bolts prevent further movement.

NOTE: The rig stop bolts and bushings must be inserted only far enough into the upper drag brace so that the lower drag brace comes in contact with the bolt only, and not the bushings.

(14) Extend the actuator nut assembly shaft out electrically, until the down limit switch stops the extension of the actuator.

(15) Align the holes of the yoke assembly with the holes in the upper drag brace when the gear are in the down position. If the holes do not align, turn the yoke assembly counterclockwise to shorten the actuator and clockwise to lengthen the actuator.

(16) Install the rig stop bolts and lower the gear to the completely extended position. Observe the lower line, which was marked on the actuator nut assembly shaft in Steps (10) and (11) to determine that there is no spring compression.

1. UPPER DRAG BRACE
2. STOP BOLT
3. STOP BOLT BUSHING
4. LOWER DRAG BRACE

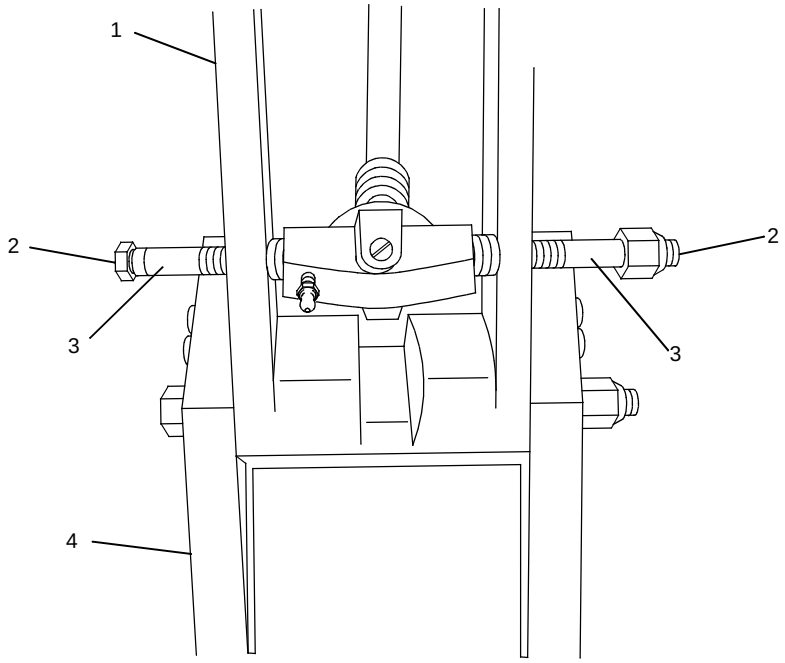

Figure 208
Stop Bolt With Bushing Not Contacting Lower Drag Brace

(17) Retract the gear slightly and insert the rig stop bolts into yoke assembly and position the bushings on the rig stop bolts so they will come in contact with the lower drag brace when in the full extended position (Ref. Figure 209).

NOTE: The actuator spring compression may be adjusted by turning the yoke assembly counterclockwise to shorten the actuator and clockwise to lengthen the actuator.

(18) Lower the gear to the complete extended position and observe the lines marked on the actuator nut assembly shaft. The yoke assembly should be even with the center line or slightly beyond, but should never go beyond the upper line. The difference in the noncompressed position and compressed position must not exceed 1/8 inch.

(19) Run the gear to the retracted position. Check the clearance between the actuator housing and yoke to maintain a minimum clearance of 1/16 (0.060) inch from bottoming out (Ref. Figure 206).

(20) Check for clearance between the shimmy damper piston rod and the floor of the nose compartment. To obtain the tolerances, adjust the up limit switch (Ref. Figure 202).

NOTE: To facilitate measuring the clearance between the actuator housing and yoke, clay may be pressed around the actuator nut assembly shaft and against the yoke. Retract the gear, then extend the gear and measure the compressed clay. This measurement should be a minimum of 1/16 (0.060) inch.

(21) Disconnect the main gear doors by removing the cotter pin, nut and bolt from the clevis and door cam assembly (Ref. Figure 211).

(22) Examine the condition of the main landing gear drag brace down lock mechanism. Examine the condition of the hook and lock plate.

1. UPPER DRAG BRACE
2. STOP BOLT
3. STOP BOLT BUSHING
4. LOWER DRAG BRACE

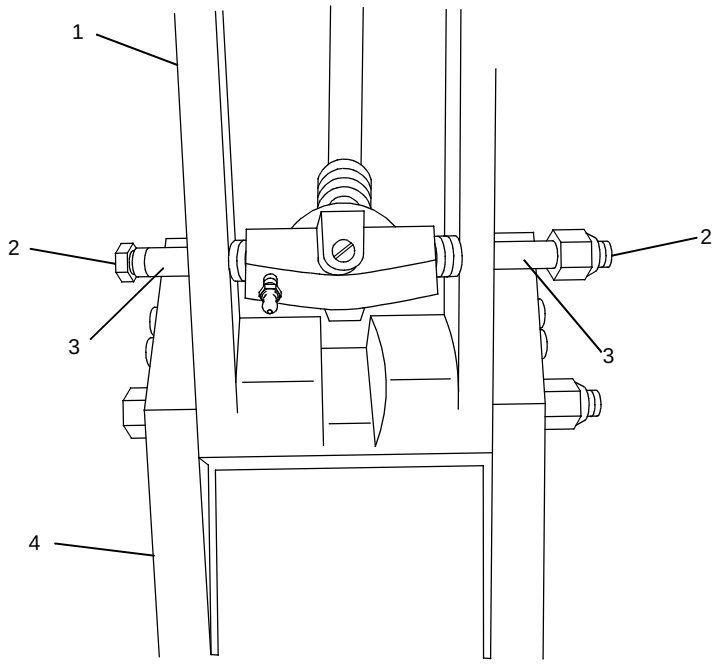

Figure 209
Stop Bolt With Bushing Contacting Lower Drag Brace

(23) Extend the main landing gear by turning the torque shafts that drive the main gear actuators. Continue to turn the torque shafts until the actuator spring is compressed to 1.30 to 1.42 inches.

(24) Measure the length of the spring in the fully compressed position from the top of the spring point (flat area at the top of the spring) to the shoulder of the clevis (Ref. Figure 210).

(25) The hook (1) must be in the full locked down position on the lock plate (3) (Ref. Figure 202, 32-60-00).

NOTE: Make sure the measurements are taken at the designated points to prevent a false reading.

(26) Lubricate the outside of the torque shaft male splines with grease (16, Table 6, 91-00-00).

(27) Connect the torque shafts and install the taper pins, washers and nuts.

(28) Carefully cycle the gear to the up position. Should either main gear be less than 0.13 inch from the main spar, rig the affected gear downward by adjusting the actuator spring compression subject to limitation (Ref. Figure 210).

(29) Apply clay to the area of the landing gear that will be closest to the main spar when the gear is in the retracted position (Ref. Figure 211).

(30) Cycle the gear to the retracted position, then extend the gear and measure the thickness of the clay. This measurement should be 0.13 to 0.50 inch.

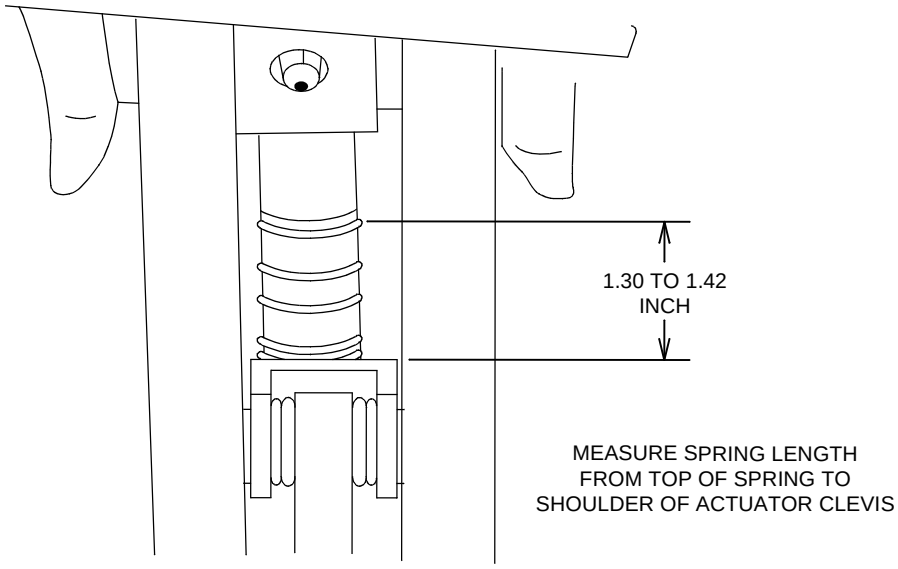

Figure 210
Measure Main Gear Actuator Spring Compression

(31) If adjusting the actuator spring compression does not allow adequate clearance, rig all three gear downward with the up limit switch located on the gearbox until the main gear are 0.13 to 0.50 inch from the main spar.

NOTE: A clearance of 0.06 inch is required between the landing gear door cam and the door actuator rollers (Ref. Figure 214).

(32) Retract the nose gear and push the nose gear door shut by hand to determine if the gear is retracted into the wheel well far enough to allow complete closure of the doors. If the doors do not clear the tire, rig the nose gear as described in Steps (10) thru (18).

(33) Connect the main gear doors by installing the bolt, nut, washer and cotter pin that attach the clevis and door cam assembly (Ref. Figure 211). The doors may need to be rigged as described in RIGGING THE MAIN LANDING GEAR DOORS and RIGGING THE FLOTATION MAIN LANDING GEAR DOORS in this section.

(34) Apply clay as previously described to the lower portion of the landing gear and retract the main gear. Extend the main gear and measure the thickness of the clay. The measurement of the clay should be 0.13 to 0.50 inch thick.

(35) Connect the nose gear door by installing the screw attaching the center hinge brackets to the left and right aft doors (Ref. Figure 204).

(36) Perform the CLEARANCE CHECK OF MAIN LANDING GEAR ACTUATORS procedure to ensure clearance between the main gear actuator and the spar web (Ref. 32-30-00).

NOTE: A minimum of 1/16 inch clearance between the switch housing and the snap ring on the switch plunger of the up and down limit switch is required to prevent damage to the switch and switch bracket.

(37) Cycle the gear several times without stopping. Check the compression of the actuator springs each time the gear are cycled to the down position. Mark the limits of travel on the down limit switch plunger in the up and down positions as a check for proper operation of the dynamic braking relay. Inconsistent travel indicates a defective relay. While the down limit switch plunger should travel approximately 1/16 inch beyond the point where the switch clicks, its travel should be consistent with every gear extension (Ref. Figures 207 and 210).

(38) Remove the two rig stop bolts from the nose gear and retract the gear to approximately 1/2 of the extended position to facilitate installation of the original stop bolt. Insert the original bolt, bushings, washer, nut and cotter pin.

(39) Install the access panels (Ref. Figure 203) and chain cover (Ref. Figure 205) and replace the center aisle floorboards.

(40) Cycle the landing gear, from the fully extended to fully retracted and back to fully extended at least one more time. Check for proper operation and rigging.

(41) Ensure all three landing gear are down and locked. Check the main landing gear drag brace down lock hook. The hook (1) must be in the full locked down position on the lock plate (3) as shown in Figure 202 Chapter 32-60-00.

(42) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

4. LANDING GEAR DOORS

The main landing gear doors consist of a single set of doors on each of the two main landing gear. The door actuators are connected to the landing gear assembly in such a manner as to open when the landing gear is in the extended position and to remain open until the gear is retracted.

The nose gear doors consist of a dual set of doors, with the smaller set at the forward portion of the wheel well. The smaller forward set of doors are connected in the same manner as the main landing gear doors, that is to open when the landing gear is in the extended position and close when the landing gear is in the retracted position. The larger aft set of nose landing gear doors are connected in such a manner as to open only to allow the gear to pass in or out of the wheel well; in other words, the doors are in the closed position when the nose landing gear is in either the fully retracted or extended position. The aft set of doors are connected mechanically to the nose landing gear. When the landing gear actuator handle is placed in either the retract or extend position, a pivot bracket mounted on the side of the nose wheel well starts to open the doors through the interconnecting rods to the door hinge and the landing gear. As the gear begins its movement the pivot bracket begins to swing either forward or backward (depending upon the position of the gear) and opens the doors to allow the gear to pass through. The pivot bracket has a 180° swing that permits the door to close after the gear is extended and opens while the gear is being retracted or extended.

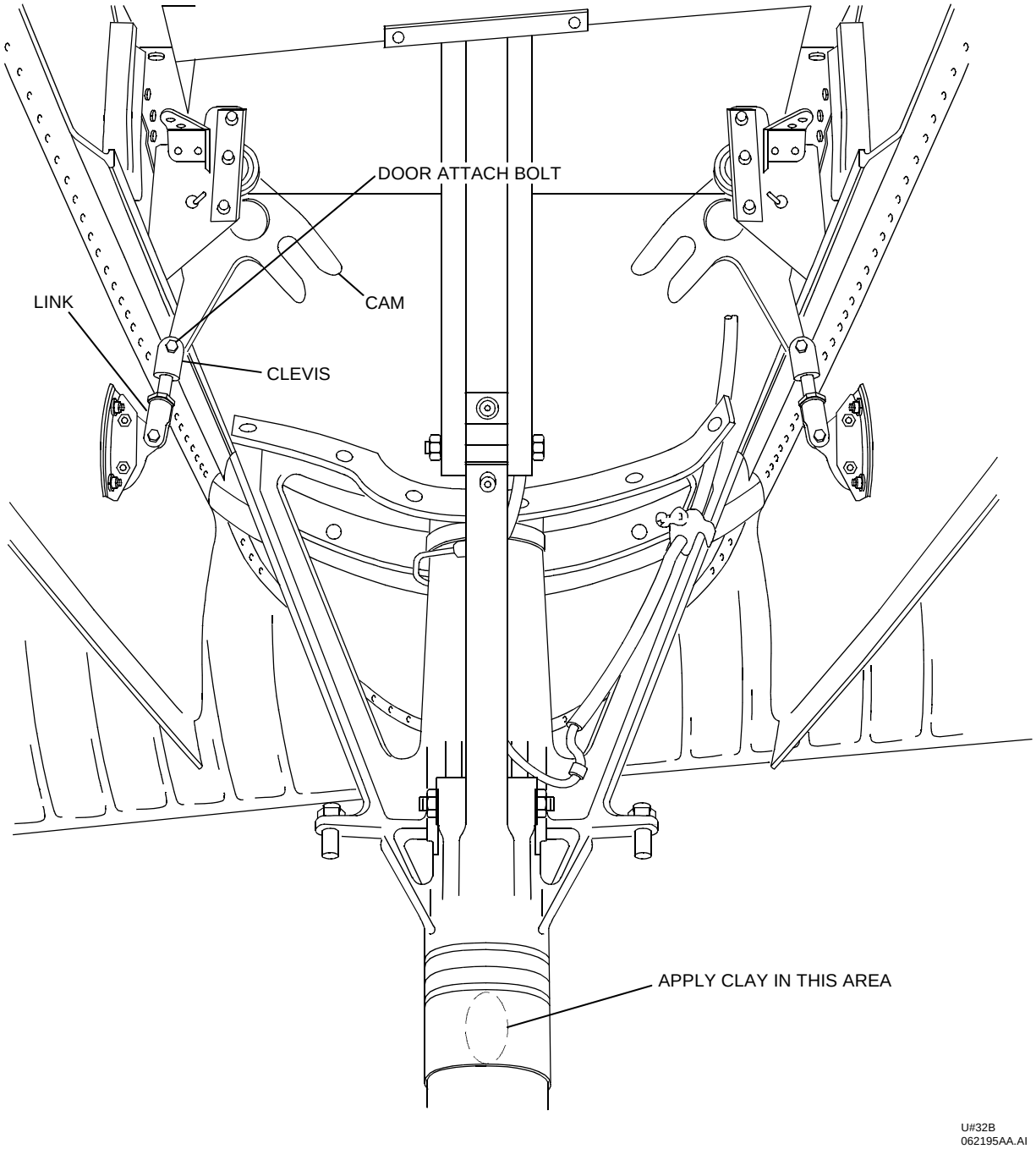

Figure 211
Main Gear Door Retract Mechanism

5. NOSE LANDING GEAR DOORS

A. Rigging

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disconnect the nose gear doors by removing the screws attaching the center hinge brackets to the left and right aft nose gear doors (Ref. Figure 204).

(3) Disconnect the small forward nose gear doors by removing the cotter pins, nuts, washers and bolts from the door to gear linkage.

(4) Retract the gear very slightly so the center line of the actuator arm at its connecting point aligns with the center line of the bolts securing the end of the actuator arm to the actuator link and the end of the link to the guide at the center hinge (Ref. Figure 212).

(5) Check that the actuator link spring is compressed to 2.60 inch measured from collar to collar.

NOTE: Lubricate the door actuators with molybdenum disulfide to prevent binding during the rigging procedure.

(6) Adjustment of the actuator link springs may be obtained by disconnecting the actuator arms from the springs by removing the cotter pins, nuts, washers and bolts. Adjust the rod end bearings in or out on the actuator springs to obtain the correct measurement of 2.60 inches as described in Step (5). Install the bolts, washers, nuts and cotter pins when the final adjustment has been obtained.

CAUTION: *To prevent damage to the hinge brackets from binding, pull the hinge down by hand before cycling the gear when the doors are disconnected.*

(7) Attach the doors to the hinge with the screws and retract the gear slowly. With the gear fully retracted, there should be a clearance of 1/4 inch between the door and tire and the door should be flush against the bottom of the fuselage. If the doors do not fit properly, lower the door, remove the cotter pin, nut, washer and bolt connecting the idler link to the door actuator arm assembly, and adjust the length of the rod by adjusting the rod end bearing.

NOTE: It is suggested that the idler link rod be adjusted in increments of only 1/4 turn while the door fit is being adjusted.

(8) Connect the idler link to the door actuator arm assembly and install the bolt, washer, nut and cotter pin.

(9) Adjust the opposite door as described in Steps (6) and (7).

NOTE: Hand forming of the doors may be required to obtain a proper fit.

(10) Rig the forward set of doors to a snug fit with the gear in the UP position by adjusting the rod end bearing on the door to gear linkage.

(11) After adjusting the doors, cycle the gear up and down several times while noting the fit of both doors in the retracted and extended positions. Make sure that the nose gear and wheel clears each door by at least 1 to 1.5 inches when the gear is cycled up.

(12) Ensure all three landing gear are down and locked.

(13) Remove the airplane from the jacks (Ref. Chapter 07-00-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

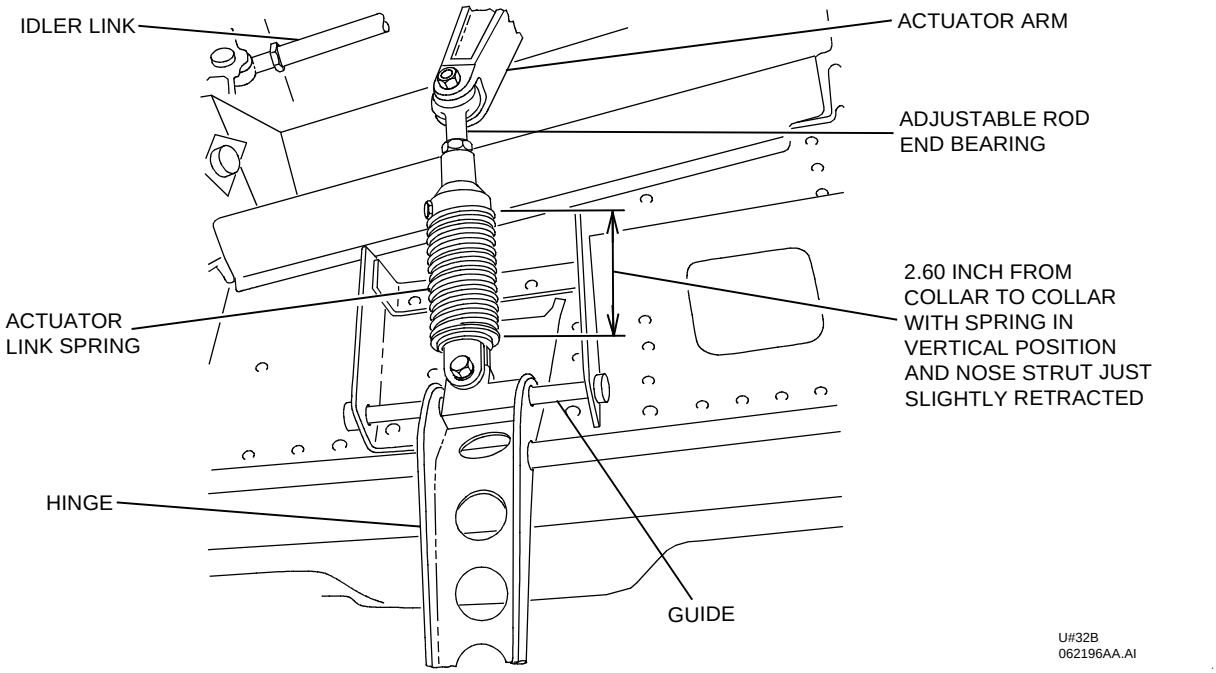

Figure 212
Nose Gear Door Adjustment

6. MAIN LANDING GEAR DOORS

A. Rigging

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disconnect the door actuating cams from the doors by removing the cotter pins, nuts, washers and bolts from the upper links. Check the cam for freedom of operation throughout its normal travel. Any tendency of binding should be freed up by adjusting the pivot bolt torque and/or lubrication of the joints (Ref. Figure 213).

(3) Check for 0.06 to 0.12 inch clearance between the door actuator roller and the lower lobe of the actuating cam by inching the gear up with the landing gear indicator circuit breaker on the copilots sub panel. If at least 0.06 inch clearance is not in evidence, remove the lower cam mounting bolt from the plate (P/N 115-810036-3) and reposition the cam pivot bolt in the support bracket slotted hole until proper clearance is attained. It will then be necessary to elongate the lower mounting holes in the support bracket to install the lower mounting bolt or replace the plate (P/N 115-810036-3) with a new one which is pilot drilled at the lower hole. Check cams for alignment and free operation; observe that the roller does not bottom out in the cam slot or at either end of the roller during retraction. Airplanes should adjust to the 0.06 to 0.12 inch dimension in order to not bottom the roller in the cam slot (Ref. Figure 214).

(4) Connect doors to cams and adjust links so that each door locks in the open position with the center point on the head of the link upper bolt 0.05 to 0.18 inch overcenter of a line between the cam hinge bolt and the lower link bolt centers (use a straight edge), when the gear is fully extended and the cam is resting against the door hinge extrusion or the wear strip if installed.

(5) Each time the linkage is adjusted, the overcenter dimension, tire clearance and closed door “snug fit” must be checked.

(6) To ensure that the door mechanism is overcenter and rigged properly, push on the door toward the gear (approximately 25 pounds at lower edge midway). The doors must be locked in the open (overcenter) position sufficiently to hold against air loads during take off and landing. If the doors fold when load is applied, repeat Steps (1) thru (6). This check is a good preflight item for the flight crews.

NOTE: If rigging airplanes prior to U-146 that DO NOT have P/N 115-810060-51 door assemblies installed proceed to Step (9). For airplanes U-146 and After and for airplanes that have P/N 115-810060-51 door assemblies installed, proceed with Step (7).

(7) Retract gear as required to check for 1.0 inch minimum clearance between tire and door. If door clearance is sufficient, inch the gear to the retracted position and check the doors for proper closed position. If doors do not close snugly, or are too tight, adjust the door travel by moving the slotted attach plate (P/N 115-810060-47) on the door and/or the link as required. Movement of the plate will require adjustment of the link to retain the overcenter position of the linkage.

(8) When rigging is satisfactory, drill an additional hole in the slotted attach plate and angles on the door to install a third bolt for positive retention of the door adjustment. Locate this bolt between the bolts through the plate slots, and in line with them. When rigging a plate that has already been

drilled, the old plate can be reused if the newly drilled hole will still provide positive retention such as a figure eight type hole. If less than two thirds a hole adjustment is made, the plate should be replaced to provide a new hole.

(9) Cycle the landing gear, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(10) Ensure all three landing gear are down and locked.

(11) Remove the airplane from the jacks (Ref. Chapter 07-00-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

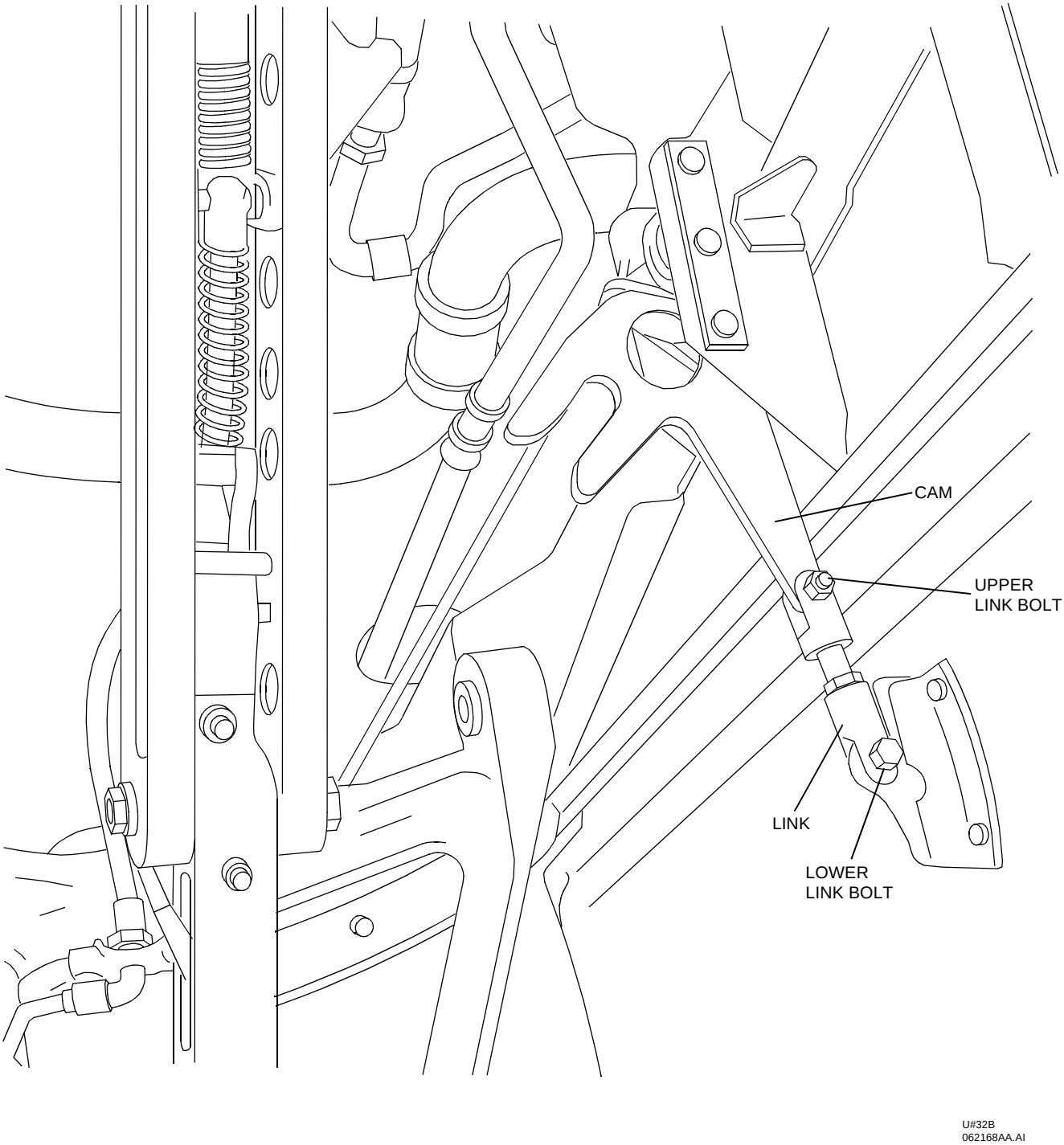

Figure 213
Main Gear Door Adjustment

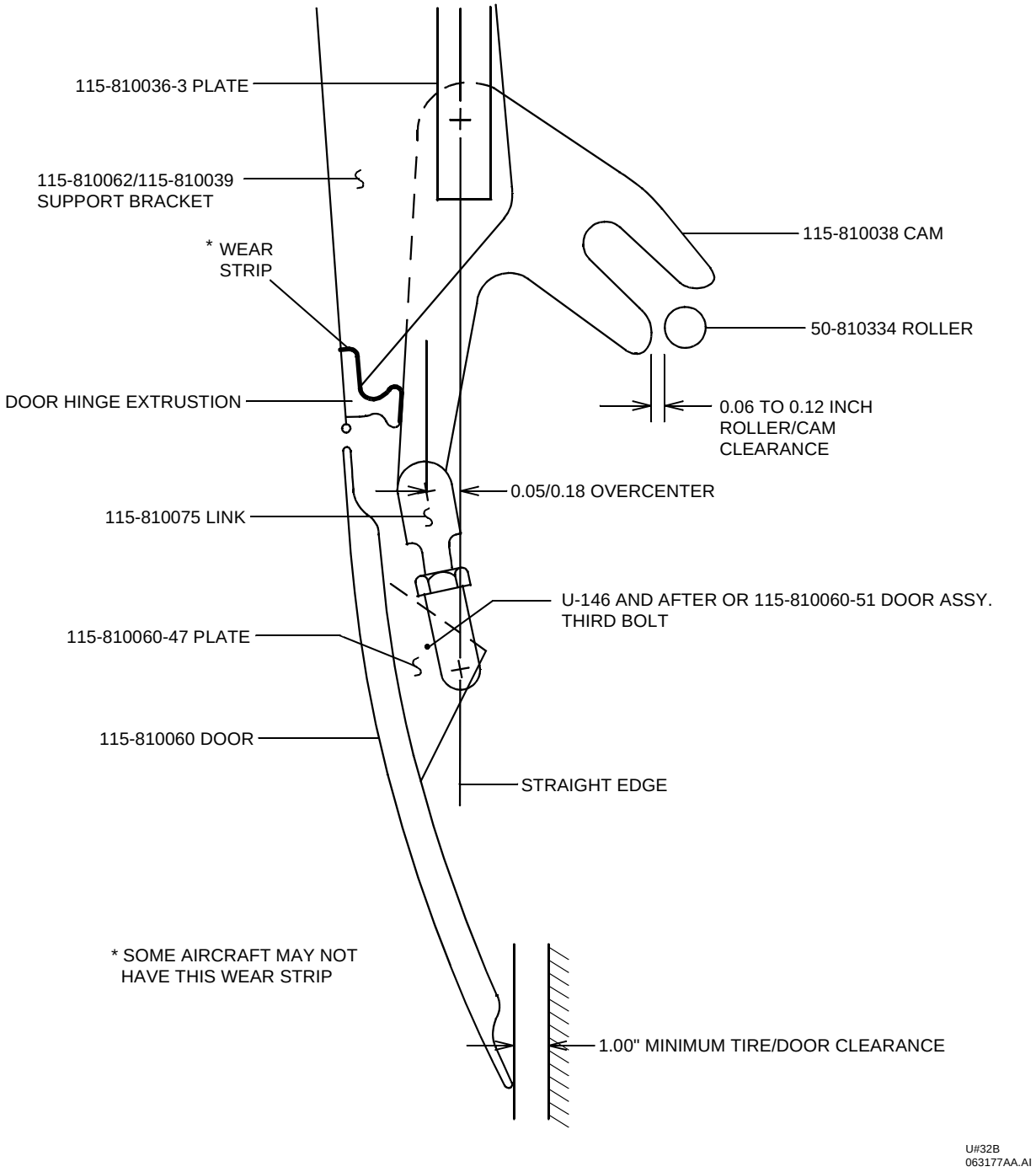

Figure 214
Main Gear Door Cam and Roller Rigging

7. FLOTATION MAIN LANDING GEAR DOORS

A. Rigging

NOTE: Before rigging of the doors is accomplished, check the door swivel for 0.010 inch maximum end play, the swivel must freely rotate. Washers (P/N AN960-516L) may be added, as required, to obtain the desired end play and free rotation (Ref. Figure 215).

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) With the doors disconnected and out of the way of the gear, retract the gear. Check and adjust the clearance between the landing gear and the main spar as outlined under RIGGING ALL THREE LANDING GEAR in this chapter.

(3) Push the doors up by hand to check for fit and clearance.

(4) Extend the gear and connect the door attach linkage so that the doors are held in an overcenter position. Adjust the link to 6.15 inches between the attachment hole centers.

CAUTION: *Observe the door linkage closely during retraction. If the actuator rod is set too long, the clevis of the rod will bottom out on the swivel attached to the bracket on the door and bend the threaded portion of the actuator rod.*

(5) Apply a force of approximately 35 pounds at the center of the door in both the inboard and outboard directions. The door should maintain an overcenter position.

(6) Retract the gear. The tire should clear the door by a minimum of 0.62 inch during the retraction cycle. If further adjustment is required, the door attach linkage should be adjusted 1/4 turn at a time (Ref. Figure 215).

(7) Cycle the gear several times and check for proper operation and a snug fit. If the door will not rig to a closed position and still maintain overcenter in the open position, it may be necessary to shim the bracket that is attached to the gear brace.

(8) Ensure all three landing gear are down and locked.

(9) Remove the airplane from the jacks (Ref. Chapter 07-00-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

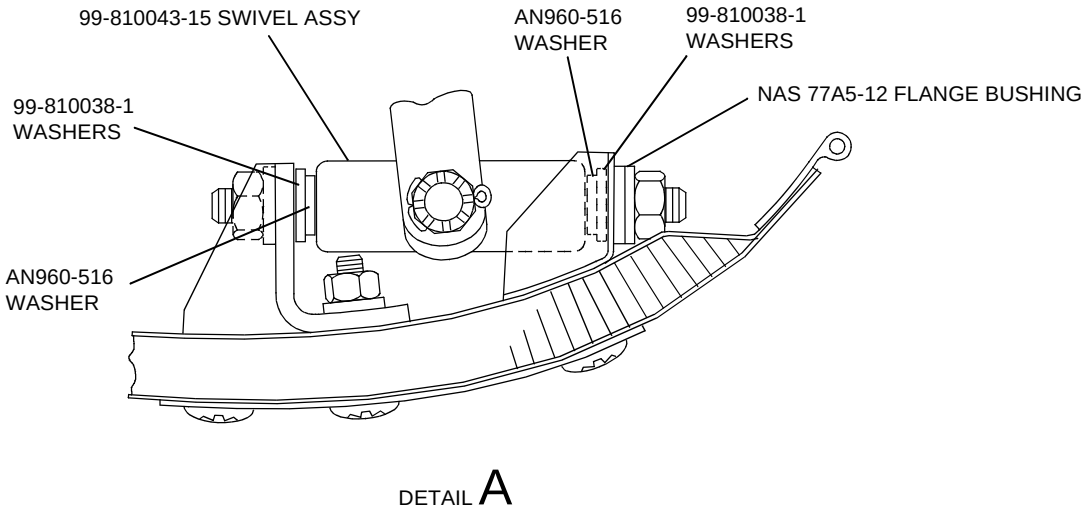

Figure 215
Flotation Main Landing Gear Door

8. LANDING GEAR MANUAL EXTENSION SYSTEM

In the event that the normal actuating switch fails to operate, the landing gear may be extended manually by employing the manual landing gear extension system. The manual extension system is actuated by two red painted levers below and to the RH side of the pilot's seat. Lift the small lever (manual extension engagement handle) and rotate it 50° clockwise. This positions the landing gear manual extension gear and chain mechanism to the operating position and opens a safety switch mounted on the main landing gear motor gearbox, removing all electrical power from the landing gear circuit. Pumping the long red handle (manual extension actuation handle) up and down will actuate the entire landing gear system through a series of gears and chains to extend the landing gear.

A. Rigging

(1) With the long red actuation handle in the stowed position, and the small curved engagement handle pulled up and rotated 50° clockwise, check to see that the safety switch mounted on the landing gear motor gearbox is in the open position.

NOTE: The safety switch must be in the open position to prevent operation of the landing gear.

(2) The switch position is set at the factory and should not normally require resetting. However, should the switch require adjustment, this may be accomplished by moving the switch on its mounting bracket.

(3) Remove the pilots floorboards to gain access to the chain.

(4) Adjust the manual extension actuating chain to deflect 0.50 ± 0.05 inch under a three pound load applied halfway between the sprocket and FS 167.0.

(5) The tension adjustment is accomplished by loosening the ratchet and sprocket shaft brackets and moving the support shaft forward until the proper tension is obtained.

(6) With the small curved engagement handle in the released position, check to be certain the sprocket shaft and gearbox are completely disengaged.

(7) With the cable disconnected, measure the distance between the smaller (driven) sprocket and the gearbox cover plate.

(8) Connect the cable and tighten the cable tension to decrease the above measurement by 1/16 +1/64/ -0 inch. Operate the small engagement handle to ensure proper operation.

(9) With the small engagement handle in the released position, operate the long red actuation handle to ensure that the manual extension system is completely disengaged.

(10) Return the long red actuation handle to the stowed position. The actuating cable should be clear of any possible interference which could apply pressure on the cable and cause the sprocket shaft to partially engage.

B. Operational Test

The manual extension system may be tested for operational capabilities in the following manner:

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Cycle the gear to the UP position, then remove all electrical power from the airplane.

(3) Remove the carpet and hardware and access panel immediately forward of the main spar.

(4) Pull the small handle at the RH side of the pilots seat UP and rotate 50° clockwise to the set position.

(5) Inspect the lever on the gearbox to confirm that it has moved sufficiently to actuate the safety switch. Rotate the small handle 50° Counterclockwise and return to the stowed position.

(6) Restore the electrical power to the airplane and, with the gear still in the UP position, perform the following check:

(a) Pull the small handle UP and rotate it 50° clockwise to the set position.

(b) Lift the long red actuation handle from its position beneath the pilots seat and pump the handle to extend the gear to the down and locked position. The gear indicator light should indicate that the gear is in the down and locked position.

(c) Release the small handle and operate the long red handle to ensure that the manual extension system is completely disengaged. Return the long red handle to the stowed position.

(d) Retract the gear by employing the gear selector handle on the instrument panel and repeat Steps (a) thru (c) and check the tension of the retract chain as described in the preceding procedure.

(7) Ensure all three landing gear are down and locked.

(8) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

LANDING GEAR
EXTENSION AND RETRACTION (HYDRAULIC LANDING GEAR)
MAINTENANCE PRACTICES

1. HYDRAULIC LANDING GEAR SYSTEM

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

WARNING: When performing an inspection or maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all landing gear are down and locked prior to removing the airplane from jacks.

When performing maintenance on the hydraulic system, cleanliness is essential for maintaining a functional and trouble free system. Any foreign materials that are allowed to enter the system are damaging to the packings, seals, valves and moving parts within the system. Such damage can result in fluid leakage and/or failure of the system components. Observe the following guidelines when working on the hydraulic system:

(1) Hydraulic storage containers should be sealed completely to prevent contamination of the hydraulic fluid during storage.

(2) Use only clean hydraulic fluid when filling the system.

(3) Use only clean hydraulic fluid when flushing the system or testing any of the system components.

(4) When any lines or hoses are disconnected and/or any components are removed from the system, plug or cap all openings to prevent the entry of any foreign material into the system.

(5) When the system is open, prevent contamination of the hydraulic fluid with other oils or water.

(6) Hands and tools should be free of contaminants (grease, oil, dirt, etc.) while working on the hydraulic system.

(7) If the system becomes contaminated, drain and flush the system with clean hydraulic fluid.

(8) To reduce the possibility of leaks in the hydraulic system, care should be taken when installing hydraulic fittings or components. Any time a fitting or component is loosened or removed, discard the packing and install the fitting or component with a new packing. To prevent damage to the packing, coat the packing and the threads of the fitting or component with hydraulic fluid before installing the packing; do not contaminate the packing after it is lubricated.

A. Servicing

WARNING: If the accumulator charging gage reads 400 psi or less place the airplane on jacks when charging the accumulator. If the nose gear actuator unlocks while charging the accumulator, the accumulator may have internal damage and should be replaced.

Servicing the hydraulic landing gear system consists primarily of maintaining the correct fluid level and maintaining the correct accumulator precharge. Charge the accumulator to 600 ± 30 psig (U-1 thru U-164, except U-50) or to 750 ± 38 psig (U-50, U-165 and After) with dry compressed air or nitrogen.

A fill can, located just inboard of the LH nacelle and forward of the main spar, contains a cap and dipstick assembly, marked FULL and ADD, for convenience of maintaining the correct system fluid level. Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING) and cycle the landing gear. After the landing gear has been cycled once, the fluid level may return to ADD. Refill the fill can and recycle the landing gear until the FULL level is maintained after two cycles. Use only hydraulic fluid (9, Table 6, 91-00-00) when servicing the system. Hydraulic system capacity is 233 cu. in. of hydraulic fluid; usable is 194 cu. in. (231 cu. in. is equal to 1 gallon).

Replace the system filter and clean the finger screens in the power pack Gear up and down ports at the intervals shown in Chapter 12-10-00.

B. Purge Modification (U-50, U-165 and After)

The following modifications to the hydraulic plumbing of airplanes U-50, U-165 and After will allow the hydraulic system to be bled without disconnecting the plumbing.

The following parts are required for this modification.

MODIFICATION PART LIST

Part Number	QTY
AN804-4S (Tee)	1
AN929-4S (Cap Assembly)	3
AN924-4S (Nut)	1
99-580140-1 (Line)	1
AN804-6S (Tee)	1
AN929-6S (Cap Assembly)	6
AN924-6S (Nut)	1
99-580141-1 (Line)	1
AN824-6S (Tee)	1
6R6X SS (Tee)	4
4R6X SS (Tee)	2

(1) Remove all electrical power from the airplane.

(2) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(3) Modify the nose gear emergency extend plumbing as follows:

(a) Remove the nose gear emergency extend line (color coded brown) connected to the elbow in the actuator hose and to the bulkhead fitting on the bracket located adjacent to the actuator on the RH wheel well keel.

(b) Disconnect the line from the other end of the bulkhead fitting and remove the bulkhead fitting from the bracket.

(c) Replace the bulkhead fitting removed in Step (b) with a tee (P/N AN804-4S) and cap assembly (P/N AN929-4S). Install the tee with the cap assembly pointing inboard using a nut (P/N AN924-4S) and the washer removed in Step (b).

(d) Install a line (P/N 99-580140-1) in place of the line removed in Step (a).

(4) Modify the nose gear normal extend plumbing as follows:

(a) Remove the nose gear normal extend line (color coded white) connected to the elbow in the actuator hose and to the bulkhead fitting on the bracket located adjacent to the actuator on the RH wheel well keel.

(b) Disconnect the line from the other end of the bulkhead fitting and remove the bulkhead fitting from the bracket.

(c) Replace the bulkhead fitting removed in Step (b) with a tee (P/N AN804-6S) and cap assembly (P/N AN929-6S). Install the tee with the cap assembly pointing inboard using a nut (P/N AN924-6S) and the washer removed in Step (b).

(d) Install a line (P/N 99-580141-1) in place of the line removed in Step (a).

(5) Modify the nose gear retract plumbing as follows:

(a) Follow the retract hose aft from the actuator trunnion and locate the union between the actuator hose and the rigid line.

(b) Disconnect the retract hose and line from the union.

(c) Replace the union with a tee (P/N AN824-6S) and cap assembly (P/N AN929-6S). Install the tee with the cap assembly pointing inboard.

(6) Modify the LH and RH main gear plumbing individually as follows:

(a) Follow the extend and retract hoses (color coded white and blue) from the actuator to the bulkhead fittings.

(b) Disconnect the hoses from the fittings and install a tee (P/N 6R6X SS) and cap assembly (P/N AN929-4S) between each fitting and hose end.

(c) Install the tees with the cap assemblies pointing forward.

(d) Follow the emergency extend hose (color coded brown) from the actuator to the bulkhead fitting.

(e) Disconnect the hose from the fitting and install a tee (P/N 4R6X SS) and cap assembly (P/N AN929-4S) between the fitting and hose end. Install the tee with the cap assembly pointing forward.

(7) Bleed the hydraulic system as described under LANDING GEAR SYSTEM BLEEDING (U-50, U-165 and After) procedures in this Chapter.

2. LANDING GEAR SYSTEM SHUTTLE VALVE

A. Operational Test

The following procedure checks all three actuator shuttle valves. This check may also be accomplished by testing all three actuators individually using the procedures described under ACTUATOR SHUTTLE VALVE FUNCTIONAL TEST.

There is a shuttle valve at the top of each actuator. When hydraulic fluid enters an actuator through the secondary extend port, the shuttle valve piston moves to close the primary port, thus resisting the flow of hydraulic fluid through the primary port. Excessive leakage through the shuttle valve(s) may prevent the landing gear from safely locking down during hand pump operations (Ref. Figure 201).

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Retract the landing gear to the fully up position using the power pack or hand pump.

(3) Using the hand pump, only partially extend the landing gear. Actuate the hand pump approximately 5 to 20 strokes, but stop before any landing gear lock down.

(4) Choose a convenient location to open the normal extend plumbing system.

NOTE: Cleanliness is essential when performing maintenance on any hydraulic system. If all three landing gear are permitted to lock down, the fluid flow rate increases. If fluid flow is measured after the landing gear lock down, the quantity shall not exceed 15 drops per minute (20 drops equals one milliliter).

(5) Open the normal extend plumbing line at the location selected. Connect a tube or hose to each open end of the line. Place the other end of each hose into a container to catch any hydraulic fluid that drains out.

(6) Continue to extend the landing gear by using the hand pump. Residual drops of fluid can be expected because the hydraulic lines are open, but if hydraulic fluid drips almost every time the hand pump is actuated, there is excessive leakage across the actuator shuttle valve into the normal extend plumbing lines. This flow must be measured before all three landing gear lock down.

(7) If excessive leakage is noted, the actuators must be checked individually to determine which one, two, or all three actuators have excessive shuttle valve leakage. Refer to the ACTUATOR SHUTTLE VALVE FUNCTIONAL TEST for this procedure. Replace components as required.

(8) Connect all plumbing lines. Bleed the hydraulic system.

(9) Cycle the landing gear with the hand pump, then with the power pack to check operation.

(10) Remove airplane from jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

B. Functional Test

This procedure may be performed with the actuator on the bench or installed in the airplane (Ref. Figure 201).

(1) Connect a hand pump and pressure gauge to the hydraulic actuator's secondary extend port.

(2) The hydraulic actuator's primary extend port and the retract port must be open.

(3) Slowly increase pressure to the secondary extend port while observing the pressure gauge. The shuttle valve will move to close the primary extend port. The actuator rod will extend.

(4) When the rod is fully extended, increase pressure to 650 psi.

(5) With 650 psi applied to the secondary extend port, maximum leakage from the primary extend port shall be 10 drops per minute (20 drops equals one milliliter).

(6) If there is excessive shuttle valve leakage, the actuator must be overhauled or replaced.

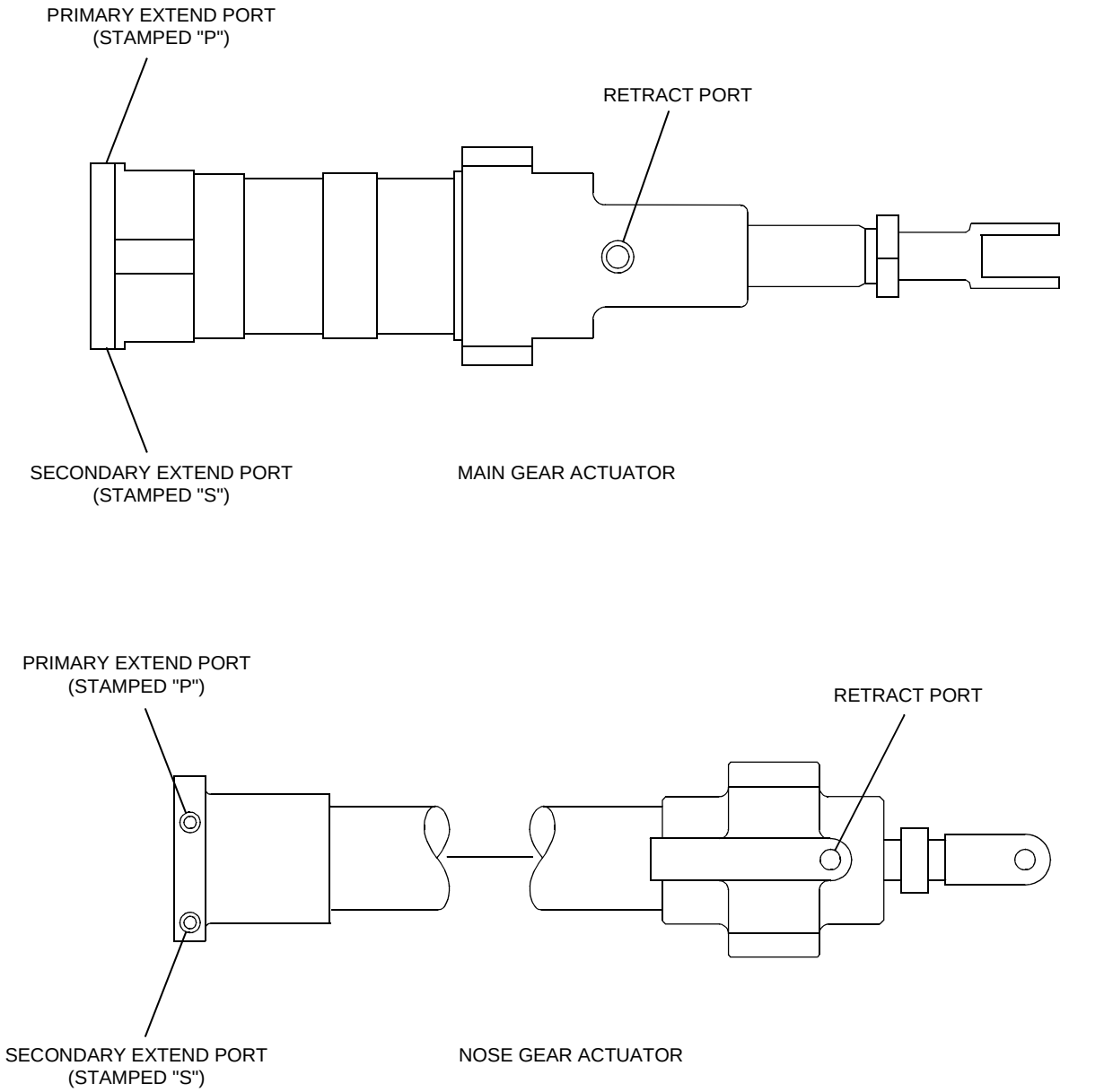

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Figure 201
Gear Actuator Assemblies

3. LANDING GEAR SYSTEM

A. Bleeding (U-1 thru U-164, except U-50 and U-50, U-165 and After Not Modified as Detailed Under Hydraulic System Purge Modification)

The landing gear hydraulic system should not need to be filled and bled unless the power pack is removed and any of the system lines are opened at the actuators, resulting in a considerable loss of fluid and entry of air into the system.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

WARNING: If the nose gear actuator unlocks while the accumulator is being charged, the accumulator may have internal damage and should be replaced.

(2) Charge the accumulator to 600 ± 30 psig (U-1 thru U-164, except U-50) or to 750 ± 38 psig (U-50,U -165 and After) with dry compressed air.

(3) Disconnect each landing gear actuator from its respective drag brace.

(4) Disconnect the emergency extension lines (color coded brown) from the actuators and cap the actuator ports and lines.

NOTE: While bleeding the hydraulic system, always maintain the proper level of hydraulic fluid in the fill can. Avoid overfilling the system as any excess hydraulic fluid will be dumped overboard.

(5) Check the fluid level in the external fill can and add hydraulic fluid (9, Table 6, 91-00-00) if required.

(6) Remove the cap from the nose gear emergency extend line (color coded brown) in the nose wheel well. Connect a hose to the line and place the other end of the hose in a catch container.

(7) Connect an external power supply to the airplane and adjust the voltage to a positive 28 ± 0.25 volts.

(8) Alternately retract and extend the landing gear several times with the emergency extension hand pump to force air out of the emergency extend plumbing as follows:

(a) After removing the safety devices, pull up on the red knob on the service valve to place the valve in the gear up position. Install the block attached to the service valve under the knob to ensure that the service valve remains in the gear up position.

(b) Unstow the emergency extension hand pump handle and retract the landing gear with the hand pump.

WARNING: As a safety precaution, do not place the service valve in the gear down position before the 2 ampere control circuit breaker is pulled.

(c) Pull the 2 ampere control circuit breaker. Remove the block from under the red knob and push down on the knob to place the service valve in the gear down position. Using the emergency extension hand pump, extend the landing gear.

(9) Remove the hose from the line and cap the line.

(10) Remove the cap from the emergency extend plumbing (color coded brown) in either the LH or RH main gear wheel well. Connect a hose to the line and place the other end of the hose in a catch container and bleed the line as outlined in Step (8).

(11) Remove the hose from the line and cap the line.

(12) Remove the cap from the emergency extend plumbing (color coded brown) in the other main wheel well. Connect a hose to the line tee and place the other end of the hose in a catch container and bleed the system as outlined in Step (8).

(13) Remove the hose from the line and cap the line.

(14) Place the emergency extension hand pump handle in the stowed position. Ensure that the service valve is in the gear down position with the safety devices in position.

(15) Connect all three emergency extend lines to the actuators.

(16) Disconnect the retract lines (color coded blue) from the actuators and cap the actuator ports and the lines.

(17) Remove the cap from the nose gear retract line (color coded blue) in the nose wheel well. Connect a hose to the line tee and place the other end of the hose in a catch container.

CAUTION: *If the retract line has not been purged after 30 seconds of continuous running of the power pack, allow the power pack to cool 15 minutes before continuing.*

(18) Place the landing gear control switch in the up position and allow the power pack to run until all the air is bled from the retract line. Shutdown the power pack by pulling the 2 ampere circuit breaker.

(19) Remove the hose from the line and cap the line.

(20) Remove the cap from the retract line (color coded blue) in either the LH or RH main gear wheel well. outlined in Step (18). Connect a hose to the line and place the other end of the hose in a catch container and bleed the line as outlined in Step (18).

(21) Remove the hose from the line and cap the line.

(22) Remove the cap from the retract line (color coded blue) in the other main wheel well. Connect a hose to the line and place the other end of the hose in a catch container and bleed the line as outlined in Step (18).

(23) Remove the hose from the line and cap the line.

(24) Connect all three retract lines to the actuators.

(25) Place the landing gear control switch in the up position and retract the actuators.

(26) Check the fluid level in the external fill can and add hydraulic fluid (9, Table 6, 91-00-00) if required. Avoid overfilling the system as any excess hydraulic fluid will be dumped overboard.

(27) Disconnect the normal extend lines (color coded white) from each actuator and cap the actuator ports and the lines.

(28) Remove the cap from the nose gear normal extend line (color coded white) on the nose wheel well. Connect a hose to the line and place the other end of the hose in a catch container.

CAUTION: *If the extend line has not been purged after 30 seconds of continuous running of the power pack, allow the power pack to cool 15 minutes before continuing.*

(29) Place the landing gear control switch in the down position and allow the power pack to run until all the air is bled from the extend line. Shutdown the power pack by pulling the 2 ampere circuit breaker.

(30) Remove the hose from the line and cap the line.

(31) Remove the cap from the normal extend line (color coded white) in either the LH or RH main gear wheel well. Connect a hose to the line and place the other end of the hose in a catch container and bleed the line as outlined in Step (29).

(32) Remove the hose from the line and cap the line.

(33) Remove the cap from the normal extend line (color coded white) in the other main wheel well. Connect a hose to the line and place the other end of the hose in a catch container and bleed the line as outlined in Step (29).

(34) Remove the hose from the line and cap the line.

(35) Connect all three normal extend lines to the actuators.

(36) Using the power pack, retract and extend the actuators two times.

(37) With the actuators extended, connect the actuators to their respective drag braces.

(38) Using the power pack, cycle the landing gear up and down twice.

(39) Check the fluid level in the external fill can and add hydraulic fluid (9, Table 6, 91-00-00) if required. Avoid overfilling the system as any excess hydraulic fluid will be dumped overboard.

(40) Disconnect the external power supply from the airplane.

CAUTION: *Before removing the airplane from the jacks, make sure that the hand pump handle is in the stowed position, that the red knob on the service valve is pushed down with the safety device in position. and that the landing gear is in the down and locked position.*

B. Bleeding (U-50, U-165 and After)

The landing gear system should not need to be filled and bled unless the power pack is removed and any of the system lines are opened at the actuators, resulting in a considerable loss of fluid and air entry into the system. Before this procedure, the system plumbing must be modified as detailed under HYDRAULIC SYSTEM PURGE MODIFICATION.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

WARNING: If the nose gear actuator unlocks while the accumulator is being charged, the accumulator may have internal damage and should be replaced.

(2) Charge the accumulator to 750 ± 38 psig with dry compressed air or nitrogen.

NOTE: While bleeding the hydraulic system, always maintain the proper level of hydraulic fluid in the fill can. Avoid overfilling the system as any excess hydraulic fluid will be dumped overboard.

(3) Check the fluid level in the external fill can and add hydraulic fluid (9, Table 6, 91-00-00) if required.

(4) Remove the cap from the tee in the nose gear emergency extend plumbing (color coded brown) located in the nose wheel well. Connect a hose to the tee and place the other end of the hose in a catch container.

(5) Connect an external power supply to the airplane and adjust the voltage to a positive 28 ± 0.25 volts.

(6) Alternately retract and extend the landing gear several times with the emergency extension hand pump to force air out of the emergency extend plumbing as follows:

(a) After removing the safety devices, pull up on the red knob on the service valve to place the valve in the gear up position. Install the block attached to the service valve under the knob to ensure that the service valve remains in the gear up position.

(b) Unstow the emergency extension hand pump handle, and retract the landing gear with the hand pump.

WARNING: As a safety precaution, do not place the service valve in the gear down position before the 2 ampere circuit breaker is pulled.

(c) Pull the 2 ampere circuit breaker. Remove the block from under the red knob, and push down on the knob to place the service valve in the gear down position.

(7) Remove the hose from the tee and replace the cap.

(8) Remove the cap from the tee located in the emergency extend plumbing (color coded brown) in either the LH or RH main gear wheel well. Connect a hose to the tee, place the other end of the hose in a catch container, and bleed the line as outlined in Step (6).

(9) Remove the hose from the tee and replace the cap.

(10) Remove the cap from the tee in the emergency extend plumbing (color coded brown) located in the other main gear wheel well. Connect a hose to the tee, place the other end of the hose in a catch container, and bleed the system as outlined in Step (6).

(11) Remove the hose from the tee and replace the cap.

(12) Place the emergency extension hand pump handle in the stowed position. Ensure that the service valve is in the gear down position with the safety devices in position.

(13) Remove the cap from the tee in the nose gear retract plumbing (color coded blue) located in the nose wheel well. Connect a hose to the tee, and place the other end of the hose in a catch container.

CAUTION: *If the retract line has not been purged after 30 seconds of continuous running of the power pack, allow the power pack to cool 15 minutes before continuing.*

(14) Place the landing gear control switch in the up position, and allow the power pack to run until all the air is bled from the retract line. Shutdown the power pack by pulling the 2 ampere circuit breaker.

(15) Remove the hose from the tee and replace the cap.

(16) Remove the cap from the tee located in the retract plumbing (color coded blue) in either the LH or RH main gear wheel well. Connect a hose to the tee, place the other end of the hose in a catch container, and bleed the line as outlined in Step (14).

(17) Remove the hose from the tee and replace the cap.

(18) Remove the cap from the tee in the retract plumbing (color coded blue) located in the other main gear wheel well. Connect a hose to the tee, place the other end of the hose in a catch container, and bleed the line as outlined in Step (14).

(19) Remove the hose from the tee and replace the cap.

(20) Place the landing gear control switch in the up position, and retract the landing gear switch.

(21) Check the fluid level in the external fill can and add hydraulic fluid (9, Table 6, 91 -00-00) if required. Avoid overfilling the system as any excess hydraulic fluid will be dumped overboard.

(22) Extend the landing gear.

(23) Remove the cap from the tee in the nose gear normal extend plumbing (color coded white) located in the nose wheel well. Connect a hose to the tee and place the other end of the hose in a catch container.

CAUTION: *If the extend line has not been purged after 30 seconds of continuous running of the power pack, allow the power pack to cool 15 minutes before continuing.*

(24) Place the landing gear control switch in the down position, and allow the power pack to run until all the air is bled from the extend line. Shutdown the power pack by pulling the 2 ampere circuit breaker.

(25) Remove the hose from the tee and replace the cap.

(26) Remove the cap from the tee located in the normal extend plumbing (color coded white) in either the LH or RH main gear wheel well. Connect a hose to the tee, place the other end of the hose in a container, and bleed the line as outlined in Step (24).

(27) Remove the hose from the tee and replace the cap.

(28) Remove the cap from the tee in the normal extend plumbing (color coded white) located in the other main gear wheel well. Connect a hose to the tee, place the other end of the hose in a catch container, and bleed the line as outlined in Step (24).

(29) Remove the hose from the tee and replace the cap.

(30) Using the power pack, cycle the landing gear up and down twice.

(31) Check the fluid level in the external fill can and add hydraulic fluid (9, Table 6, 91 -00-00) if required. Avoid overfilling the system as any excess hydraulic fluid will be dumped overboard.

(32) Disconnect the external power supply from the airplane.

CAUTION: *Before removing the airplane from the jacks, make sure that the hand pump handle is in the stowed position, that the red knob on the service valve is pushed down with the safety device in position, and that the landing gear is in the down and locked position.*

(33) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

4. POWER PACK MOTOR

A. Removal

(1) Remove the carpet and the center aisle floorboard just forward of the main spar (just aft of the main spar on airplanes U-50, U-165 and After).

(2) Remove the two wires from the motor terminal strip and identify to facilitate installation.

(3) Remove plumbing as necessary and cap to eliminate contamination.

NOTE: As the motor is removed, be sure to retain the pump shaft to motor shaft coupler for motor installation. Airplanes U-50, U-165 and After and some earlier airplanes do not have the coupler installed.

(4) To remove motor (P/N 115-380002), remove the four bolts and washers securing the motor to the power pack and remove the motor. To remove motor (P/N 115-380002-5), remove the two bolts securing the motor to the motor mounting plate and remove the motor. Remove the four bolts and washers to remove the mounting plate from the power pack.

B. Installation

(1) Lubricate the pump spline, motor spline, and coupler (if installed) with lubricating grease (12, Table 6, 91-00-00) prior to installation.

(2) Install the coupler (if installed) on the motor spline before installation of the motor.

NOTE: To ensure proper attachment and to avoid possible damage to the power pack, install the motor with the terminals opposite the pressure switch, and use bolts with only one washer under each bolt head when attaching the motor or motor mounting plate.

(3) To install motor (P/N 115-380002), mate the motor spline with the pump shaft and secure the motor to the power pack with the four bolts and washers. To install motor (P/N 115-380002-5), install the motor mounting plate (if removed) on the power pack with the four bolts and washers, mate the motor spline with the pump shaft, and secure the motor with the two mounting bolts. Torque the mounting bolts to 35 inch-pounds.

NOTE: The positive lead is connected to the “A” terminal nearest the power pack and the negative lead to the “C” terminal in the center of the terminal strip.

(4) Attach the two wires to the motor terminal strip.

(5) Remove protective caps and install the previously removed plumbing.

(6) Install the center aisle floorboard and carpet.

(7) Check fill can and add hydraulic fluid (9, Table 6, 91-00-00).

(8) Perform THREE POINT JACKING procedure (Ref. Chapter 07-10-00). All tires must be clear of the floor.

(9) Cycle the gear. After the gear have been cycled once, the fluid level may return to ADD. Fill the fill can and cycle the gear until the FULL level is maintained after two cycles.

(10) Perform LOWERING THE AIRPLANE AFTER THREE POINT JACKING procedure (Ref. Chapter 07-10-00).

5. POWER PACK PRESSURE SWITCH

A. Removal

(1) Remove the carpet and the center aisle floorboard just forward of the main spar (just aft of the main spar on airplanes U-50, U-165 and After).

NOTE: Shop towels will be needed to absorb the system fluid which will escape from the port when the switch is removed.

(2) Unscrew the power pack pressure switch (4) and remove it from the power pack (Ref. Figure 202).

B. Installation

NOTE: The new packing installed with the new power pack pressure switch (4) should be lubricated with petrolatum (27, Table 6, 91-00-00) prior to installation of the switch.

(1) Install the power pack pressure switch (4) by screwing it into the power pack (Ref. Figure 202).

(2) Install the center aisle floorboard and carpet.

6. GEAR SELECTOR SOLENOID

A. Removal

(1) Place airplane on jacks (Ref. Chapter 07-00-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) On airplanes prior to U-165, except U-50, remove the seats from the RH seat deck and the carpet and floorboard access openings just forward and just aft of the main spar. Remove the heat and vent duct that is installed over the gear selector solenoid (1) (Ref. Figure 202).

(3) On airplanes U-50, U-165 and After, remove the center aisle floorboard just aft of the main spar.

NOTE: Shop towels will be needed to absorb the system fluid which will escape from the port when the switch is removed.

(4) Cut the safety wire and unscrew the gear selector solenoid (1) from the power pack.

B. Installation

NOTE: The new packing installed with the new solenoid, should be lubricated with petrolatum (27, Table 6, 91-00-00) prior to installation of the solenoid.

(1) Place airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Install the gear selector solenoid (1) by screwing it into the power pack and safety wire. Install hand tight only (Ref. Figure 202).

(3) On airplanes prior to U-165, except U-50, install the heat and vent duct. Install the floorboard access openings, carpeting and seats previously removed.

(4) On airplanes U-50, U-165 and After, install the center aisle floorboard just aft of the main gear.

(5) Cycle the landing gear using the airplane hydraulic power pack, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(6) Ensure all three landing gear are down and locked.

(7) Remove the airplane from jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

7. FLUID LEVEL SENSOR

A. Removal

(1) Remove the carpet and the center aisle floorboard just forward of the main spar (just aft of the main spar on airplanes U-50, U-165 and After).

CAUTION: *When removing wiring from the sensor, be careful to avoid damage to sensor.*

(2) Remove the electrical wiring from the fluid level sensor (2) (Ref. Figure 202).

(3) Unscrew the fluid level sensor (2) from the power pack.

1. GEAR SELECTOR SOLENOID
2. FLUID LEVEL SENSOR
3. MOTOR
4. POWER PACK PRESSURE SWITCH
5. FINGER SCREENS
6. SYSTEM FILTER

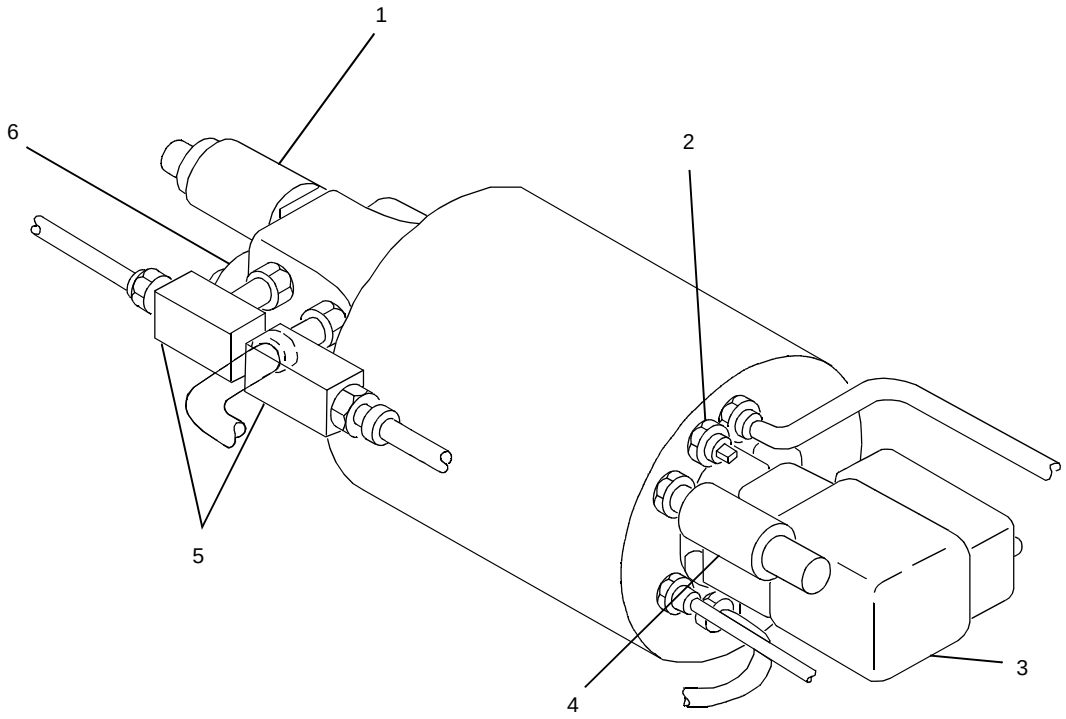

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Figure 202
Hydraulic Power Pack

B. Function Test (Power Pack P/N 99-388002- 1) (U-1 thru U-164, Except U-50)

NOTE: Power packs (P/N 99-388002-5) use an optically operated fluid level sensor with an integrated self test circuit. The integral self test circuit is energized by a switch on the instrument panel and functionally tests the sensing unit's internal circuit.

A resistor having 180 ohms resistance within a tolerance of 20% and a power rating of at least 3 watts is suggested.

The fluid level sensor is of the thermistor type and should be checked for proper operation before installation. A test harness may be fabricated with a 1/4 ampere fuse (2) (or other suitable current protective device), a switch (3), 327 lamp (4), 180 ohm resistor (5), and a pair of clip leads (6) (Ref. Figure 203).

The lamp (4) and resistor (5) are to be connected in parallel, then in series with the fuse (2) and switch (3). The sensor (7) may be tested as follows:

CAUTION: *Because the sensor will heat rapidly, serious injury may occur if contact is made during the test.*

(1) Visually check for stripped or damaged threads, broken insulation, a broken terminal, or evidence of overheating, and replace if any of the above conditions are noted.

(2) Connect the fluid level sensor across a 24 to 28 vdc power source as shown in Figure 203. Connect one of the clip leads to the terminal and the other to the shell of the sensor.

(3) Suspend the fluid level sensor in a container of hydraulic fluid (9, Table 6, 91-00-00). Apply power to the sensor for at least 5 minutes. The lamp must NOT illuminate brightly.

(4) Remove the fluid level sensor from the fluid and suspend it in free air. The lamp MUST come on brightly within 5 minutes. Open the circuit as soon as the lamp comes on.

C. Installation

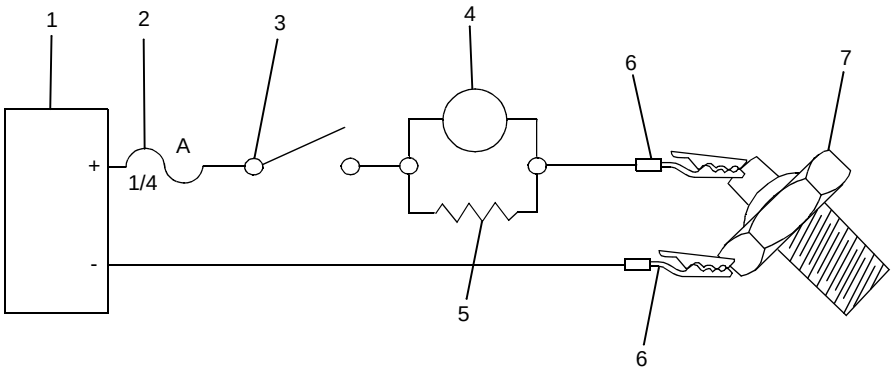
(1) Remove any oil from the threads on the fluid level sensor (2) and power pack (Ref. Figure 202).

(2) Before installing a thermistor type fluid level sensor, apply primer (113, Table 6, 91-00-00) and sealing compound (114, Table 6, 91-00-00) to the threads of the sensor.

(3) Screw the fluid level sensor (2) into the power pack. Install the sensor hand tight only.

(4) Install the electrical wiring to the fluid level sensor (2).

(5) Install the center aisle floorboard carpet.



1. 24 TO 28 VDC POWER SOURCE
2. 1/4 AMPERE FUSE
3. SWITCH
4. 327 LAMP
5. 180 OHM, 3 WATT RESISTER
6. CLIP LEADS
7. LIQUID LEVEL SENSOR UNDER TEST

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Figure 203
Fluid Level Sensor Test Harness (U-1 thru U-164, Except U-50)

8. POWER PACK

A. Removal

(1) Place airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the carpet and the center aisle floorboard (just forward of the main spar on airplanes U-50, U-165 and After).

CAUTION: *Take care to avoid damaging the sensor while disconnecting the electrical wiring.*

(3) Remove and identify the electrical wiring from the motor, hydraulic fluid level sensor, uplock pressure switch and gear select solenoid.

CAUTION: *As the hydraulic lines are disconnected from the power pack, plug or cap all openings to prevent entry of foreign material into the lines or power pack.*

NOTE: A shop towel may be used to absorb the system fluid which will escape from the ports during line removal.

(4) Remove the screw attaching the binding jumper to the motor end of the power pack. Remove and identify the plumbing from the valve body and the motor end of the power pack. Cap the ports to avoid contamination.

(5) To remove the power pack from the mounting bracket, remove the three bolts and washers from each end of the power pack. Remove the power pack.

B. Installation

(1) Position the power pack in the airplane.

CAUTION: *To ensure proper attachment and to avoid possible damage to the power pack, use bolts with only one washer under each bolt head.*

(2) Attach the power pack to the mounting bracket by installing the three bolts and washer on each end of the power pack.

(3) Remove the caps and connect the plumbing, as previously identified, to the power pack.

CAUTION: *To avoid possible damage to the special nut, use only an MS35207-261 screw.*

(4) Connect the electrical wiring, as previously identified, to the electrical components of the power pack. Secure the bonding jumper to the motor end of the power pack with the MS35207-261 screw.

(5) Fill and bleed the hydraulic landing gear system. Cycle the landing gear using the airplane hydraulic power pack as many times as needed to bleed the system. Check for proper operation and check rigging.

(6) Ensure all three landing gear are down and locked.

(7) Install the center aisle floorboard carpet.

(8) Remove the airplane from jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

9. MAIN LANDING GEAR ACTUATOR

A. Removal

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from the jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

WARNING: When performing an inspection or maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all landing gear are down and locked prior to removing the airplane from jacks.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the fender assemblies from the wheel well on airplanes equipped with flotation gear.

(3) Disconnect the gear doors by removing the cotter pin, nut, washer and bolt that attach the link to the door actuating cam. On airplanes equipped with flotation gear, disconnect the gear doors by removing the cotter pin, nut, washers, bushing and bolt that attach the door link to the door swivel.

CAUTION: *As the hydraulic lines are disconnected, plug or cap all openings to prevent entry of foreign material into the lines or actuator.*

(4) Remove the three hydraulic lines from the actuator and identify to facilitate installation.

(5) Remove the cotter pin, nut, washer and bolt that connect the actuator clevis to the drag brace.

(6) Remove the screws from the clamps that attach the wire bundles to the support brackets.

(7) Remove the bolts attaching any lines to the actuator mounting brackets.

(8) Remove the remaining bolts securing the forward stiffener plate to the actuator mounting brackets.

(9) Remove one of the actuator support brackets in which the actuator support bearings are installed. Work the actuator from the remaining mounting bracket and bearing. Remove the actuator by lifting up and forward on the lower side of the actuator and then removing downward carefully.

B. Installation

NOTE: Inspect the actuator support bearing assemblies for corrosion or damage and ensure free rotation (Ref. Figure 204).

(1) Locate the actuator in the correct position and install the removed actuator support bracket with the attaching bolts. Shim the bearing supports as required until a clearance of 0.005 to 0.025 inch is obtained between the actuator and bearing.

(2) Install the forward stiffener plate to the actuator mounting brackets. Secure any clamps that were previously removed.

(3) Check for actuator clevis side load and main gear drag leg assembly alignment as described under MAIN LANDING GEAR RIGGING (Ref. 32-31-01).

(4) Lightly coat the bolt that attaches the clevis to the drag leg brace with grease (17, Table 6, 91-00-00). Install washer (P/N AN960-816L) under the bolt head as required and install a washer (P/N AN960-616L) under the nut. Torque the nut down snug, then back off the nut to the nearest castellation and safety. Check the bolt to determine if the bolt is free to rotate without binding.

NOTE: The hydraulic actuators are adjusted by the vendor for a nominal stroke length and should not have to be adjusted. However, if the clearance measured in Step e. is not within tolerance, refer to the procedure pertaining to actuator stroke adjustment under MAIN LANDING GEAR RIGGING (Ref. 32-31-01).

(5) With the landing gear in the retracted position, check for a clearance of 0.375 $\pm$ 0.125 inch (U-1 thru U-164, except U-50) or 0.38 +0.12/ -0.25 inch (U-50, U-165 and After) between the spar and main gear. On airplanes equipped with flotation gear, check for a clearance of 0.42 $\pm$ 0.12 inch. This check may be accomplished by applying clay as shown in Figure 204, Chapter 32-31-01. Cycle the gear to the retracted position, then extend the gear and measure the thickness of the clay.

(6) Check the indicator lights for proper rig.

(7) Connect the gear doors by attaching the link to the door actuating cam with the bolt, washer, nut, and cotter pin. On airplanes equipped with flotation gear, connect the gear doors swivel with the bolt, bushing, washers, nut and cotter pin.

(8) Install the fender assembly on airplanes equipped with flotation gear.

(9) Fill and bleed the hydraulic landing gear system. Cycle the landing gear using the power pack as many times as needed to bleed the system, check for proper operation and check rigging.

(10) Ensure all three landing gear are down and locked.

(11) Remove the airplane from jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

10. NOSE LANDING GEAR ACTUATOR

A. Removal

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from the jacks.

CAUTION: *Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.*

WARNING: When performing an inspection or maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all landing gear are down and locked prior to removing the airplane from jacks.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disconnect the aft set of gear doors by removing the screws that attach the door to the center hinge brackets.

CAUTION: *As the hydraulic lines are disconnected, plug or cap all openings to prevent entry of foreign material into the lines or actuator.*

(3) Remove the three hydraulic lines from the actuator and identify.

(4) Remove the cotter pin, nut, and bolt and disconnect the actuator clevis from the yoke.

(5) Remove the electrical plug for the downlock switch on the actuator housing.

(6) Remove the bolts securing the actuator support bearing to the brackets, and remove the actuator from the airplane.

B. Installation

NOTE: Inspect the actuator support bearing assemblies for corrosion or damage and ensure free rotation (Ref. Figure 205).

(1) Position the actuator in the actuator support bearing brackets and secure with the attaching bolts.

(2) Install the three hydraulic lines to the actuator as identified during removal.

(3) Check for actuator clevis side load and nose gear drag leg assembly alignment as described under NOSE LANDING GEAR RIGGING (Ref. 32-31-01).

(4) Install the actuator clevis on the yoke and secure with the bolt, nut, and cotter pin.

(5) Connect the plug from the down lock switch on the actuator to the terminal in the wheel well.

NOTE: The hydraulic actuator is adjusted by the vendor for a nominal stroke length and should not have to be adjusted; however, if the clearance measured in Step (6) is not within tolerance, refer to the procedure pertaining to actuator stroke adjustment under NOSE LANDING GEAR RIGGING (Ref. 32-31-01).

(6) With the landing gear in the retracted position, check for a clearance of 0.25 ± 0.125 inch between the shimmy damper piston rod and the nose baggage compartment floor. This check may be accomplished by applying clay to the nose baggage compartment floor in the area that will be adjacent to the shimmy damper piston rod. Cycle the gear to the retracted position, then extend the gear and measure the thickness of the clay.

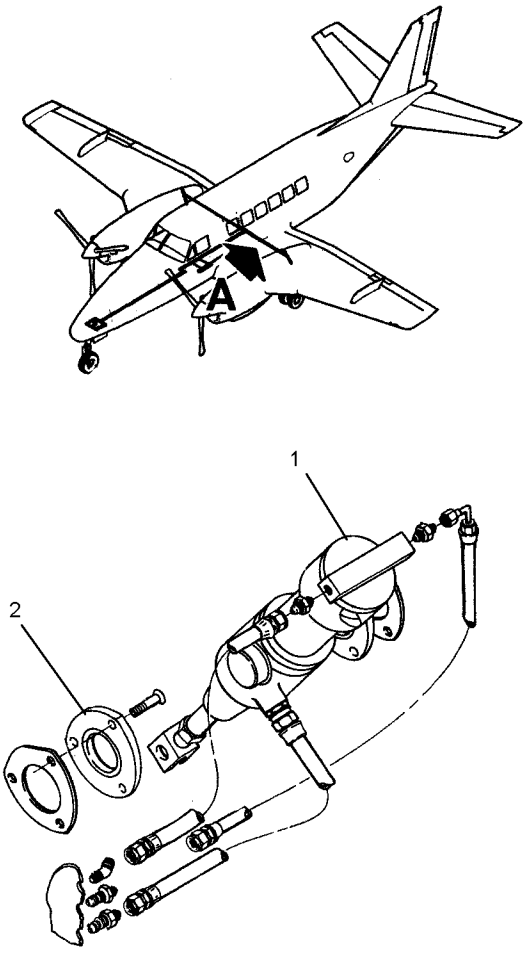
(7) Check the position indicator switches for proper rig (Ref. 32-60-00).

(8) Connect the nose gear doors.

(9) Fill and bleed the hydraulic landing gear system. Cycle the landing gear using the airplane hydraulic power pack as many times as needed to bleed the system. Check for proper operation and check rigging.

(10) Ensure all three landing gear are down and locked.

(11) Remove the airplane from jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).



1. ACTUATOR
2. ACTUATOR SUPPORT BEARING

DETAIL A

U#32B
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Figure 204
Main Landing Gear Actuator Support Bearing

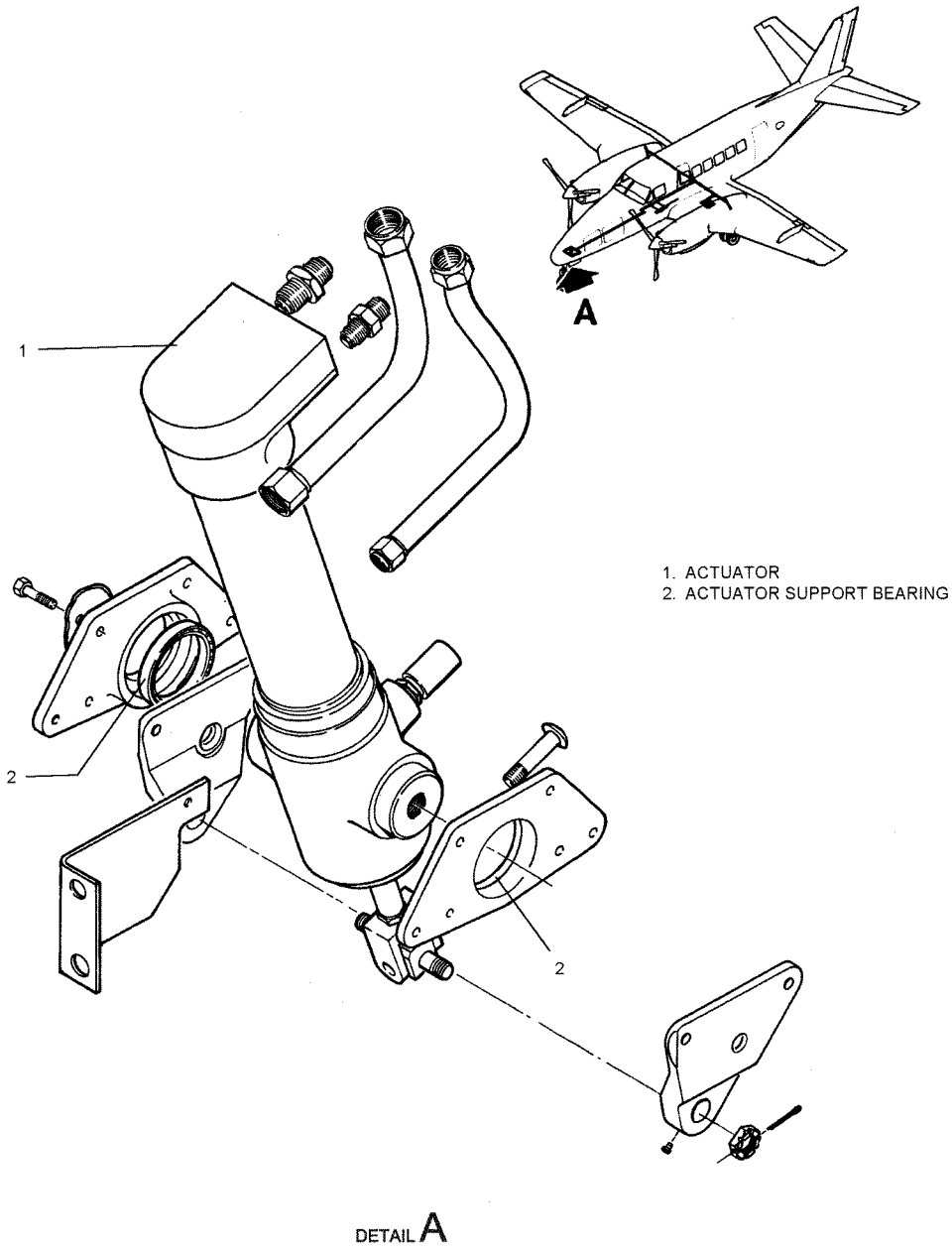


Figure 205
Nose Landing Gear Actuator Support Bearing

LANDING GEAR
HYDRAULIC LANDING GEAR SYSTEM
MAINTENANCE PRACTICES

1. HYDRAULIC LANDING GEAR SYSTEM

A. Rigging

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from jacks.

CAUTION: Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.

WARNING: When performing an inspection or maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all landing gear are down and locked prior to removing the airplane from jacks.

***CAUTION:** If the red knob on the service valve is pushed down while the landing gear is retracted with electrical power on and the landing gear control handle in the down position, the landing gear will extend immediately.*

The service valve, located adjacent to the power pack under the center aisle floorboard, should be used in conjunction with the emergency extension hand pump to raise and lower the landing gear while rigging the landing gear. The landing gear should not be cycled electrically until the landing gear is properly rigged. The service valve is normally in a mode to extend the landing gear (red knob pushed down). With the airplane on jacks and an external power supply of 28 $\pm$ 0.25 volts connected, the landing gear can be retracted by pulling the red knob on the service valve up and then operating the emergency extension hand pump. On airplanes U-50, U-165 and After, it will be necessary to pull the detent pin and move the safety device before the knob can be pulled up. To prevent the knob from being pushed down while the airplane is being serviced, slide the block attached to the service valve under the knob. After the landing gear is retracted, position the hand pump handle in the stowed position and turn off the electrical power. To extend the landing gear, push the red knob on the service valve down and extend the landing gear with the emergency extension hand pump; the electrical power does not have to be on to extend the landing gear.

NOTE: When cycling the landing gear with the power pack, allow a one minute cooling period between cycles, and after five cycles, allow a five minute cooling period.

Before removing the airplane from the jacks, make sure the emergency extension handle is in the stowed position and that the red knob on the service valve is pushed down. On airplanes U-50, U-165 and After, install the safety device over the red knob after the knob is pushed down. Retract and extend the landing gear with the hydraulic power pack to assure the landing gear is in the down and locked position before removing the airplane from the jacks.

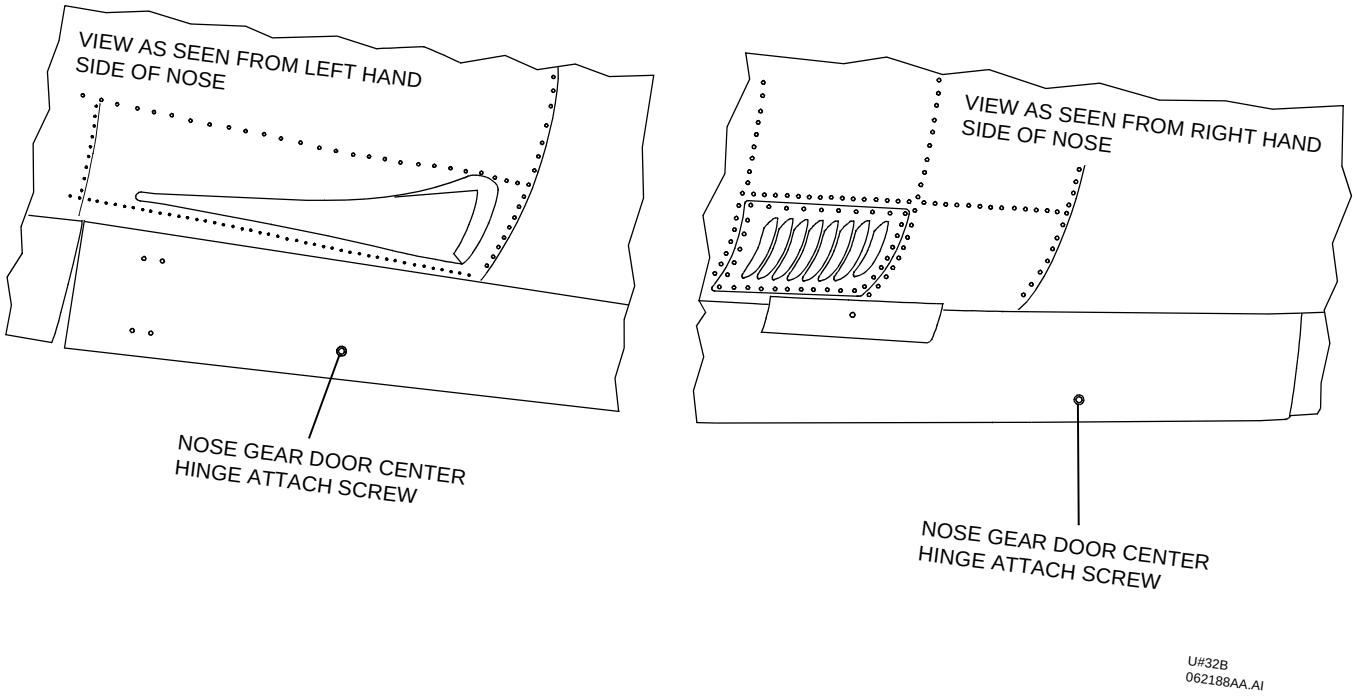


Figure 201
Nose Gear Door Attach Screws

2. NOSE LANDING GEAR (PRIOR TO U-165, EXCEPT U-50)

A. Rigging

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from jacks.

CAUTION: Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.

WARNING: When performing an inspection or maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all landing gear are down and locked prior to removing the airplane from jacks.

***CAUTION:** Do not fully cycle the landing gear electrically until the gear is properly rigged. Use of the emergency extension hand pump is recommended for initial rigging.*

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Connect an external power supply to the airplane and adjust the voltage to 28 ± 0.25 volts.

(3) Remove the screws (one in each door) to free the aft set of doors from the center hinge brackets (Ref. Figure 201).

(4) Disconnect the push pull tubes that operate the forward set of doors from the nose gear strut by removing the nuts, washers, bushings and bolts.

***CAUTION:** Ensure that the lines used in connecting the pressure gage to the airplane's hydraulic system are free of dirt and other contaminants. Purge all air from the pressure gage lines prior to connection to the hydraulic system.*

(5) Install a pressure gage (with a scale up to at least 1000 psi) in parallel with the extension side of the landing gear hydraulic system.

(6) Disconnect the nose landing gear actuator clevis from the yoke, and disengage the clevis by partially retracting the landing gear actuator.

(7) Pull the nose landing gear drag brace down until the stop bolt prevents further movement.

(8) Remove the safety wire and loosen the actuator clevis jam nut.

(9) Extend the actuator by applying 500 to 700 psig to the actuator extend port in order to positively lock the actuator in the extended position.

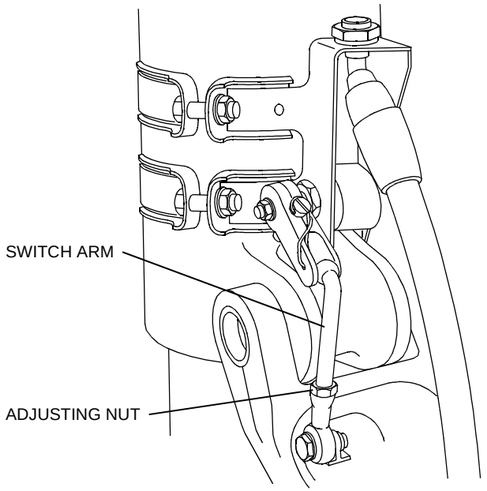
(10) Align the hole in the actuator clevis with the hole in the yoke (3) by turning the actuator piston rod (2) while the clevis straddles the yoke (3), When the holes are aligned, rotate the actuator piston rod an additional 1/2 turn, extending the actuator clevis outward (Ref. Figure 203).

(11) Tighten the actuator clevis jam nut to 60 inch-pounds, then safety wire the jam nut to the key washer.

(12) To eliminate any side load on the actuator, the alignment of the actuator and yoke assembly to the upper (1) and lower (5) drag leg assembly must be checked before securing the actuator (2) and yoke (3) assembly to the upper drag leg (1). Check for proper alignment as follows:

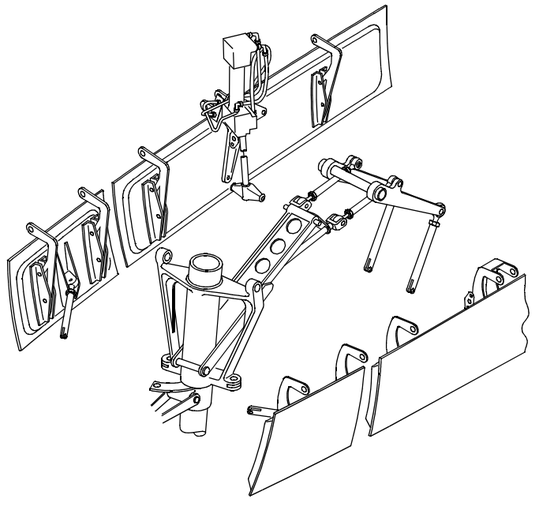
(a) Remove the cotter pin, nut, bushings, and washer from the stop bolt (4). Slowly retract the nose gear to approximately 1/2 of the extended position. With the gear in this position, push the stop bolt (4) out and through the hole in the right side of the nose wheel well keel. Let the gear down and allow it to swing free.

(b) Position the actuator (2) and yoke (3) assembly to install with the upper drag leg (1). Measure the space between the yoke (3) and upper drag leg (1) on each side of the yoke (3).



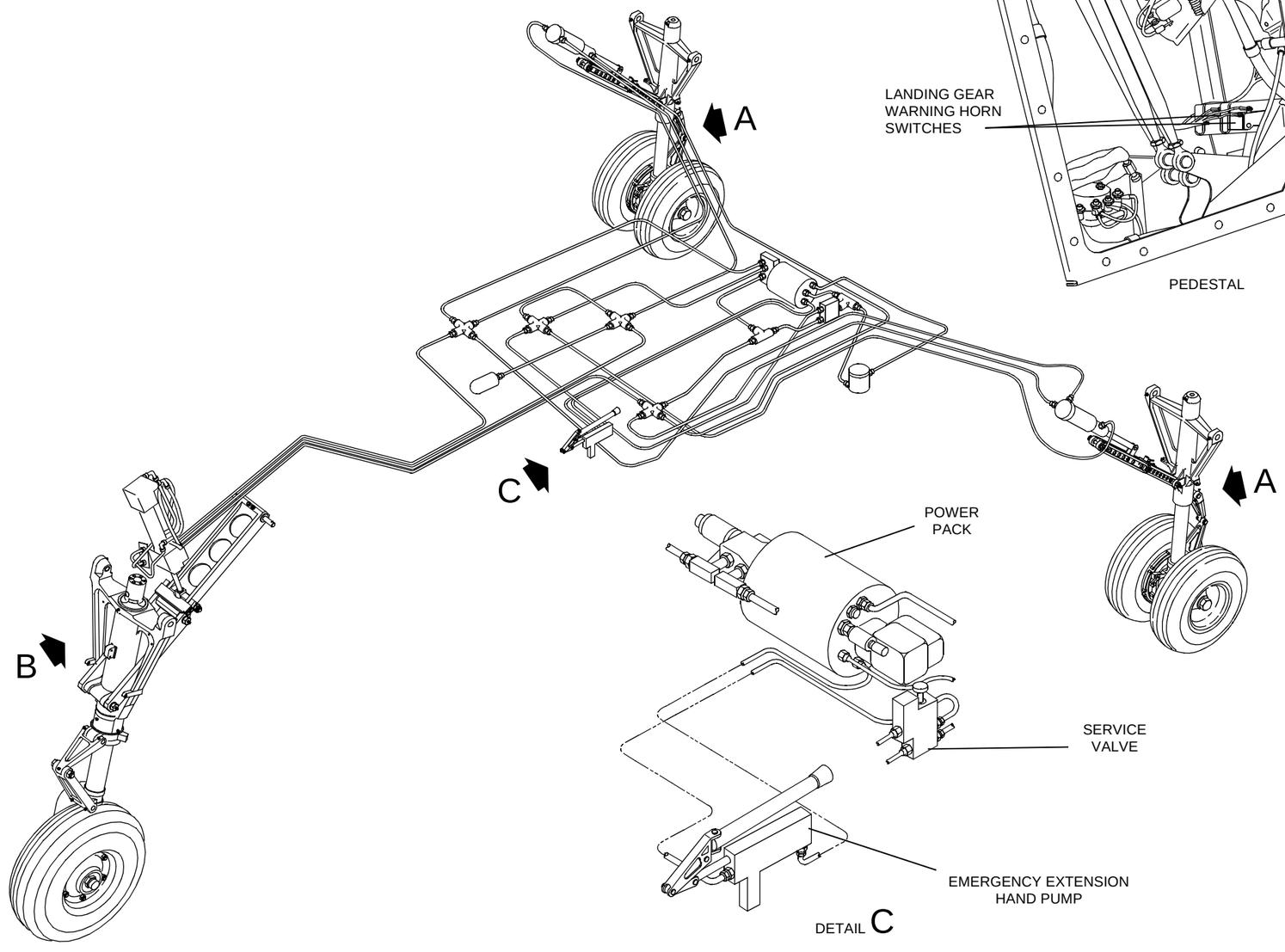
LANDING GEAR SAFETY SWITCH

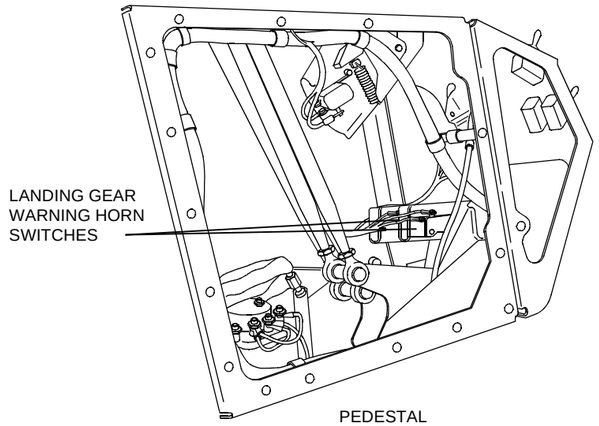
DETAIL A



FORWARD NOSE GEAR DOOR CONNECTIONS

DETAIL B





PEDESTAL

DETAIL C

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Figure 202
Hydraulic Landing Gear System

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NOTE: To retract the nose gear actuator, 200 to 400 psi of hydraulic pressure must be applied to the retract port to unlock the internal locking mechanism.

(c) Retract the actuator to its full retract position.

(d) Manually push the nose gear to the retracted position and locate the actuator and yoke assembly to install with the upper drag leg. Measure the space between the yoke and drag leg on each side of the yoke.

NOTE: The yoke is offset to facilitate alignment and may be installed with the short lug on either side. If it is necessary to rotate the yoke for proper alignment, the yoke should be removed from the actuator clevis and installed in the desired position to maintain actuator and yoke assembly hole alignment with the holes in the upper drag brace.

(e) Install washers as required to obtain proper alignment. A minimum of one washer must be maintained on each side of the yoke.

NOTE: The yoke is equipped with grease fittings and should be lubricated with grease (17, Table 6, 91-00-00) every 100 hours.

(f) "Bump" the gear to approximately 1/2 of the extended position, insert the stop bolt through the hole in the right side of the nose wheel well keel, and secure the actuator and yoke assembly to the upper drag leg. Install the bushings, washer, nut and cotter pin.

(13) Apply clay to the underside of the nose baggage compartment floor in the area that will be adjacent to the shimmy damper piston rod when the nose gear is fully retracted.

(14) Cycle the gear to the retracted position, then extend the gear and measure the thickness of the clay. The clearance between the shimmy damper piston rod and the baggage compartment floor should be 0.250 ± 0.125 inch.

(15) The stroke length of the actuator is preset by the vendor and should not require adjustment; however, if the clearance measured in Step (14) is not within tolerance, the actuator stroke length may be adjusted as follows:

(a) Disconnect the two hydraulic lines from the end cap. Plug the disconnected lines and the ports in the end cap to prevent contaminants and foreign matter from entering the system.

CAUTION: *If more than two complete turns of the actuator end cap are required to obtain the proper actuator stroke, contact Hawker Beechcraft Corporation, Commercial Service Department. The end cap must be rotated in 360 degree rotations only, to ensure alignment of the hydraulic plumbing to the shuttle valve in the end cap.*

(b) Remove the safety wire, loosen the end cap jam nut and rotate the end cap one full turn at a time until the proper adjustment is obtained. When the end cap is rotated out, the stroke length is increased, conversely, when the end cap is rotated in, the stroke length is decreased.

(c) After the proper adjustment is obtained, tighten the end cap jam nut to 50 foot-pounds and safety wire.

(d) Connect the two hydraulic lines which were previously disconnected.

(16) Adjust the nose gear drag brace downlock and the up position switches (Ref. 32-60-00).

(17) Connect the push pull tubes, that operate the forward set of doors, to the nose gear strut with the bolts, bushings, washers and nuts.

(18) Push the aft set of nose gear doors closed, and install the screws (one in each door) to attach the doors to the center hinge brackets.

(19) Cycle the landing gear using the airplane hydraulic power pack, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(20) Ensure all three landing gear are down and locked.

(21) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

1. UPPER DRAG LEG
2. ACTUATOR PISTON ROD
3. YOKE
4. STOP BOLT
5. LOWER DRAG LEG

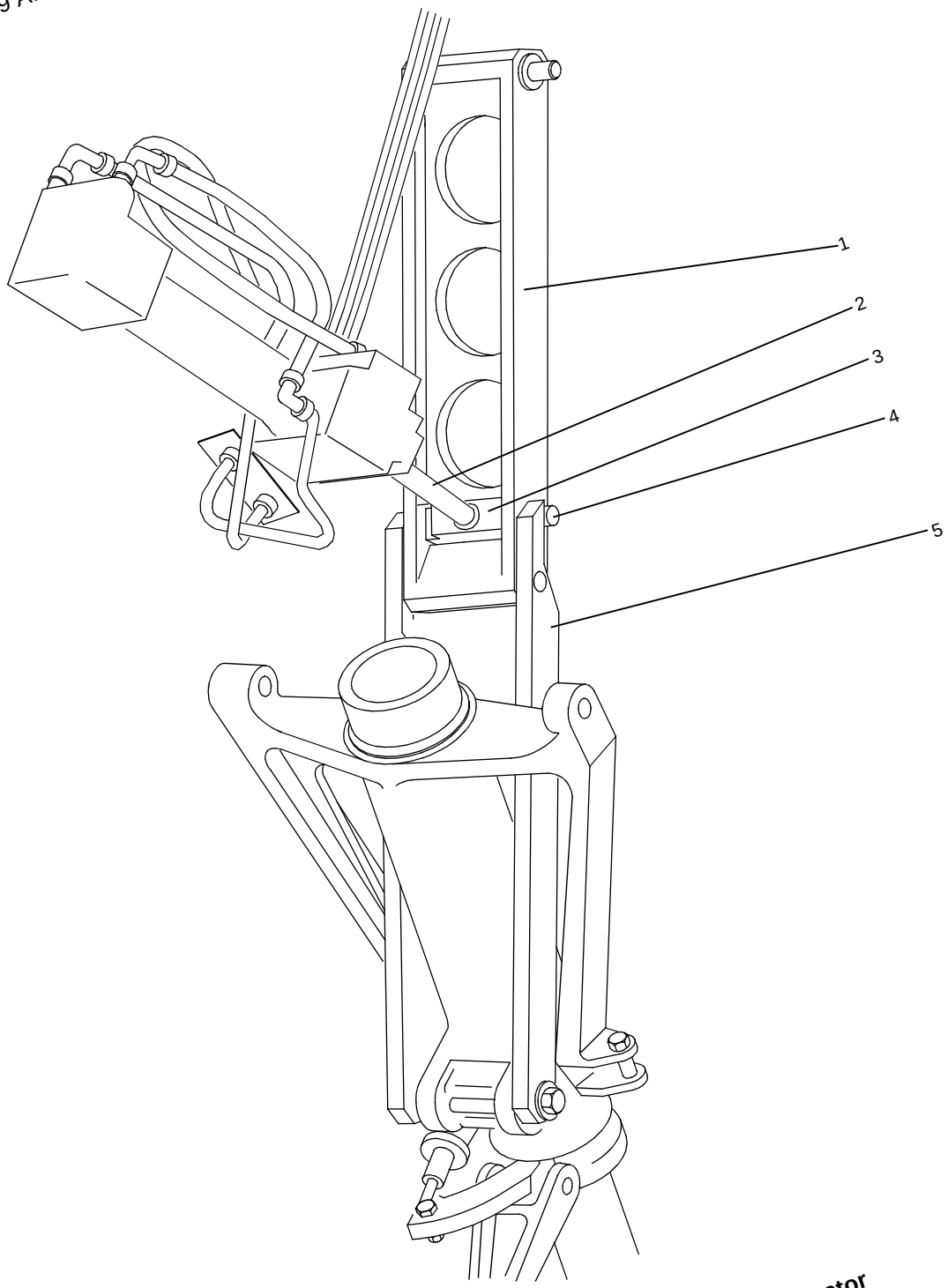

Figure 203
Nose Gear Drag Brace and Actuator

3. NOSE LANDING GEAR RIGGING (U-50, U-165 AND AFTER)

A. Rigging

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from jacks.

CAUTION: Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.

WARNING: When performing an inspection or maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all landing gear are down and locked prior to removing the airplane from jacks.

CAUTION: *Do not fully cycle the landing gear electrically until the gear is properly rigged. Use of the emergency extension hand pump is recommended for initial rigging.*

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Connect an external power supply to the airplane and adjust the voltage to 28 ± 0.25 volts.

(3) Remove the screws (one in each door) to free the aft set doors from the center hinge brackets.

(4) Remove the nuts, washers, bushings and bolts connecting the nose gear strut to the push-pull tubes that operate the forward set of doors.

CAUTION: *Ensure that the lines which are to be used in connecting the pressure gage to the airplane's hydraulic system are free of dirt and other contaminants. Purge all air from the pressure gage lines prior to connection to the hydraulic system.*

(5) Install a pressure gage (with a scale up to at least 1000 psi) in parallel with the extension side of the landing gear hydraulic system.

(6) Disconnect the nose landing gear actuator clevis from the yoke, and disengage the clevis by partially retracting the landing gear actuator.

(7) Pull the nose landing gear drag brace down until the stop bolts and bushings prevent further movement.

(8) Remove the safety wire and loosen the actuator clevis jam nut.

(9) Extend the actuator by applying 500 to 700 psig to the actuator extend port in order to positively lock the actuator in the extended position.

(10) Align the hole in the actuator clevis with the hole in the yoke by turning the actuator piston rod while the clevis straddles the yoke. When the holes are aligned, rotate the actuator piston rod an additional 1/2 turn, extending the actuator clevis outward.

(11) Tighten the actuator clevis jam nut to 60 inch-pounds and safety wire the jam nut to the key washer.

(12) To eliminate any side load on the actuator, check the alignment of the actuator and yoke to the drag brace as follows

CAUTION: *To retract the nose gear actuator, 200 to 400 psi of hydraulic pressure must be applied to the retract port to unlock the internal locking mechanism.*

Retract the actuator to fully retracted position.

(a) Manually push the nose gear to the retracted position. Place the yoke in position for attachment to the actuator clevis, and check the horizontal alignment of the actuator clevis holes and the attachment hole in the yoke. If the holes do not align, measure the amount of misalignment.

(b) With the nose gear fully extended, place the yoke in position for attachment to the actuator clevis and measure the amount of horizontal misalignment, if any.

(13) If misalignment is determined in Step (12), remove the yoke plates and yoke from the upper drag leg. Separate the yoke plates and yoke by removing the cotter pins and nuts from the yoke.

(14) Shift the 100951-X-063-UD and 100951-X-032-UD washers, using a combination as required, from side to side to obtain the proper yoke alignment. A minimum of one washer is required on each side of the yoke and under each nut. When attaching the yoke to the yoke plates, tighten the nuts down snug, then back off to the nearest castellation and install the cotter pins. The yoke must be free to rotate without binding. After tightening the nuts, the distance between the outer surfaces of the yoke plates should measure 3.422 inches. A maximum end play of 0.04 inch between the yoke and plates is acceptable.

(15) Attach the yoke plates to the upper drag leg with the attaching bolts. Repeat Step (9) to ensure proper alignment of the actuator clevis and yoke.

(16) After proper alignment of the actuator clevis and yoke is obtained, attach the actuator clevis to the yoke with the attaching bolt, washers and nut.

(17) Using an Alemite grease nozzle (P/N 314150), lubricate the grease fittings in the yoke plates with grease (17, Table 6, 91-00-00).

(18) Apply clay to the underside of the nose baggage compartment floor in the area that will be adjacent to the end of the shimmy damper piston rod when the nose gear is fully retracted.

(19) Cycle the gear to the retracted position, then extend the gear and measure the thickness of the clay. The clearance between the shimmy damper piston rod and the baggage compartment floor should be 0.250 ± 0.125 inch.

(20) The stroke length of the actuator is preset by the vendor and should not require adjustment; however, if the clearance measured in Step (18) is not within tolerance, the actuator stroke length may be adjusted as follows:

(a) Disconnect the two hydraulic lines from the end cap. Plug the disconnected lines and the ports in the end cap to prevent contaminants and foreign matter from entering the system.

CAUTION: *If more than two complete turns of the actuator end cap are required to obtain the proper actuator stroke, contact Hawker Beechcraft Corporation, Commercial Service Department. The end cap must be rotated in 360 degree rotations only, to ensure alignment of the hydraulic plumbing to the shuttle valve in the end cap.*

(b) Remove the safety wire, loosen the end cap jam nut and rotate the end cap one full turn at a time until the proper adjustment is obtained. When the end cap is rotated out, the stroke length is increased, conversely, when the end cap is rotated in, the stroke length is decreased.

(c) After the proper adjustment is obtained, tighten the end cap jam nut to 50 foot-pounds and safety wire.

(d) Connect the two hydraulic lines which were previously disconnected.

(21) Adjust the nose gear drag brace downlock and the up position switches (Ref. 32-60-00).

(22) Install the bolts, bushings, washers and nuts securing the nose gear strut to the push pull tubes that operate the forward set of doors.

(23) Push the aft set of nose gear doors closed and install the screws (one in each door) to attach the doors to the center hinge brackets.

(24) Cycle the landing gear using the airplane hydraulic power pack, From the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(25) Ensure all three landing gear are down and locked.

(26) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

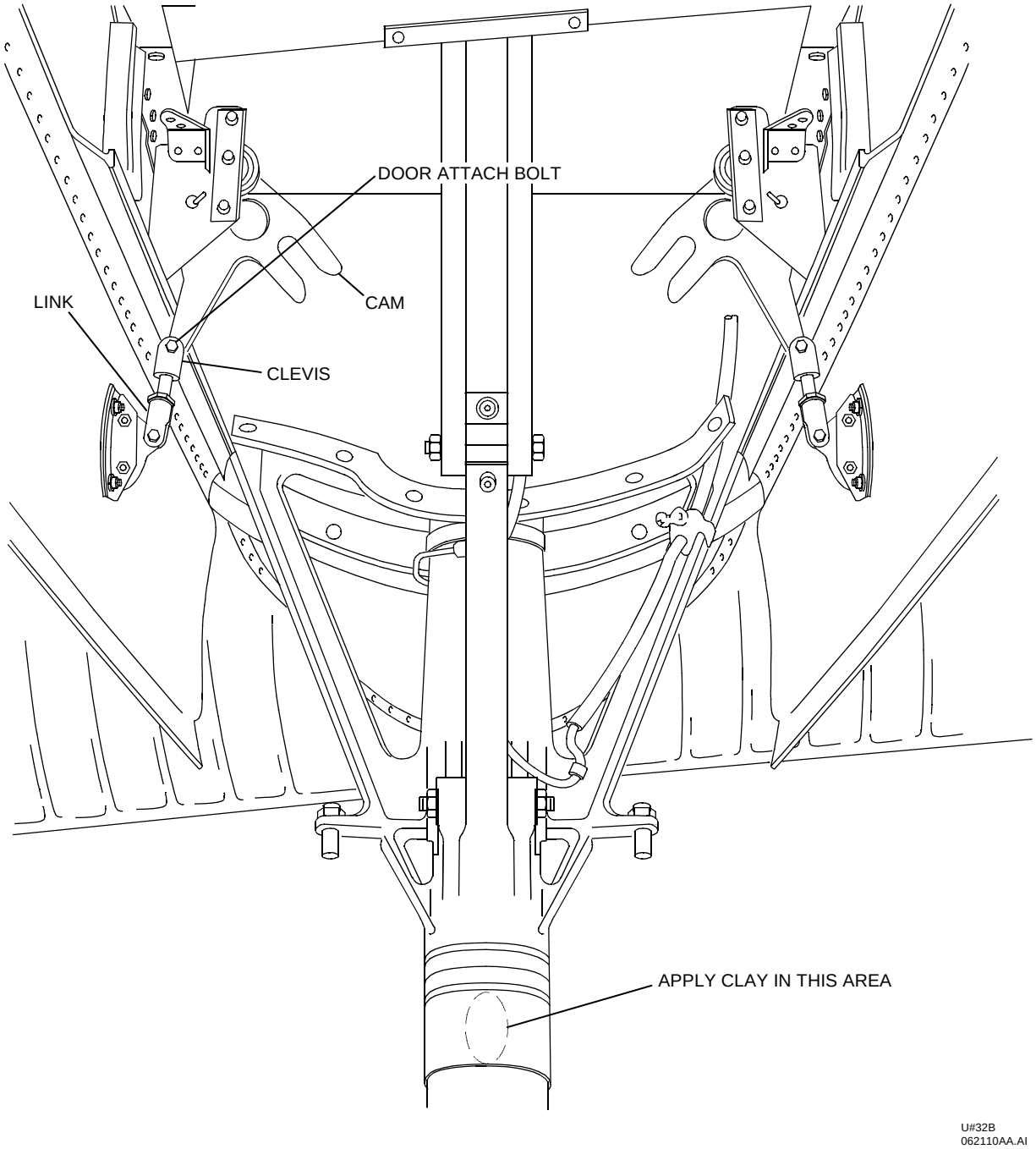

Figure 204
Main Gear Door Retract Mechanism

4. MAIN LANDING GEAR

A. Rigging

WARNING: When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gear are down and locked prior to removing the airplane from jacks.

CAUTION: Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the interest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations must be compensated for prior to jacking the airplane.

WARNING: When performing an inspection or maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance on the landing gear system.

Stay clear of the wheel wells, landing gear, and gear doors while the landing gear is in operation.

After performing maintenance on the landing gear system, the landing gear must be cycled using the hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once, checking for proper operation and rigging. Ensure that all landing gear are down and locked prior to removing the airplane from jacks.

***CAUTION:** Do not fully cycle the landing gear electrically until the gear is properly rigged. Use of the emergency extension hand pump is recommended for initial rigging.*

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Connect an external power supply to the airplane and adjust the voltage to 28 ± 0.25 volts.

(3) Disconnect the landing gear doors and secure the doors out of the way with safety wire.

(4) Disconnect the actuator from the drag leg and the drag leg from the landing gear strut. Check the alignment of the lower drag leg with the attachment fitting on the landing gear strut. There shall be no side loads on the drag leg when the landing gear is in the extended or retracted position. Correct misalignment by shifting washers (P/N AN960-10161) from side to side of the wheel well attach fitting and upper drag legs. A minimum of one washer should be maintained between the upper drag legs and the wheel well attach fitting. After the proper alignment is obtained, connect the lower drag leg to the landing gear strut.

***CAUTION:** Ensure that the lines which are to be used in connecting the pressure gage to the airplane's hydraulic system are free of dirt and other contaminants. Purge all air from the pressure gage lines prior to connection to the hydraulic system.*

(5) Install a pressure gage (with a scale up to at least 1000 psi) in parallel with the extension side of the landing gear hydraulic system.

(6) Extend the actuator by applying 500 to 700 psig to the actuator extend port in order to positively lock the actuator in the extended position.

CAUTION: *To ensure correct operation of the gear it is imperative that the “H” link and lock link are in a "hard lock" position (the end of the lock plate making contact with the back of the lock hook).*

(7) Loosen the jam nut above the actuator clevis and adjust the clevis so it will engage both the "H” link and the lock link of the drag brace.

(8) With the actuator clevis holes aligned with the holes in the "H” link and lock link, tighten the jam nut and safety wire to the key washer.

(9) The alignment of the actuator clevis to the drag leg "H” link and lock link must be checked before securing the actuator clevis to the drag leg. With the landing gear in the down position and the actuator clevis hole aligned with the "H” link and lock link holes, the clevis should slip into the "H" link without causing a side load on the actuator. Add or remove shim (P/N 115-810669) under the actuator bearing supports to correct actuator misalignment.

(10) Attach the actuator clevis to the drag leg with the bolt, washers, nut and cotter pin. Place the washers in the same positions as when removed. Tighten the nut down snug, then back off to the nearest castellation and insert the cotter pin. The bolt should be free to rotate without binding. Lubricate the joint with grease (17, Table 6, 91-00-00).

(11) Apply clay to the area of the landing gear strut that will be adjacent to the main spar when the gear is in the retracted position (Ref. Figure 204).

(12) Cycle the gear to the retracted position, then extend the gear and measure the thickness of the clay. The clearance between the spar and main gear should be 0.375 $\pm$ 0.125 inch (airplanes prior to U-165 except U-50) or 0.38 +0.12/ -0.25 inch (airplanes U-50, U-165 and After). On airplanes equipped with flotation gear, check for a clearance of 0.42 $\pm$ 0.12 inch.

(13) The stroke length of the actuator is preset by the vendor and should not require adjustment; however, if the clearance measured in Step (12) is not within tolerance, the actuator stroke length may be adjusted as follows:

(a) Disconnect the two hydraulic lines from the end cap. Plug the disconnected lines and the ports in the end cap to prevent contaminants and foreign matter from entering the system.

CAUTION: *If more than two complete turns of the actuator end cap are required to obtain the proper actuator stroke, contact Hawker Beechcraft Corporation, Commercial Service Department. The end cap must be rotated in 360 degree rotations only to ensure alignment of the hydraulic plumbing to the shuttle valve in the end cap.*

(b) Remove the safety wire and loosen the lock screw located on the trunnion casting. Rotate the actuator cylinder and end cap one full turn at a time until the proper adjustment is obtained. When the cylinder and end cap are rotated out, the stroke length is increased and when rotated in, the stroke length is decreased. When increasing the stroke length, do not unscrew the cylinder tube beyond the "maximum stroke indicator line".

(c) After the proper adjustment is obtained, tighten the lock screw and safety wire. Do not over tighten the lock screw.

(d) Connect the two hydraulic lines which were previously disconnected.

(14) Adjust the main gear drag brace downlock and the up position switches (Ref. 32-60-00).

(15) Connect the landing gear doors.

(16) Cycle the landing gear using the power pack, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and check rigging.

(17) Ensure all three landing gear are down and locked.

(18) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

5. LANDING GEAR DOORS

The main landing gear doors consist of a single set of doors on each of the two main landing gear. The door actuators are connected to the landing gear assembly in such a manner as to open the doors when the landing gear is in the extended position and to remain open until the gear is retracted.

The nose gear doors consist of a dual set of doors, with the smaller set at the forward portion of the wheel well. The smaller forward set of doors are connected in the same manner as the main landing gear doors, that is, to open when the landing gear is in the extended position and close when the landing gear is in the retracted position. The larger aft set of nose landing gear doors are connected in such a manner as to open only to allow the gear to pass in or out of the wheel well; in other words, the doors are in the closed position when the nose landing gear is in either the fully retracted or extended position. The aft set of doors are connected mechanically to the nose landing gear drag brace. When the landing gear control handle is placed in either the retractor extend position, a pivot bracket mounted on the side of the nose wheel well starts to open the doors through the interconnecting rods to the door hinge and the landing gear. As the gear begins its movement, the pivot bracket begins to swing either forward or backward (depending upon the position of the gear) and opens the doors to allow the gear to pass through. The pivot bracket has a 180° swing that permits the door to close after the gear is extended and opens while the gear is being retracted or extended.

6. NOSE LANDING GEAR DOORS

A. Rigging

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Remove the screws (one in each door) to free the aft set of doors from the center hinge brackets (Ref. Figure 201).

(3) Check the alignment of the center hinge brackets and the holes in the aft set of doors; adjust the washers (P/N AN960C-17) shown in Figure 205 as necessary to achieve proper alignment.

(4) Disconnect the push pull tubes that operate the forward set of doors from the nose gear strut by removing the nuts, washers, bushings and bolts.

NOTE: Lubricate the door actuators with grease (18, Table 6, 91-00-00) to prevent binding during the rigging procedures.

(5) With the nose retracted, check and adjust the clearance between the shimmy damper piston rod and nose baggage compartment floor as outlined under NOSE LANDING GEAR RIGGING in this chapter.

(6) Adjust the idler links to 7.50 inches between pivot points (Ref. Figure 205).

(7) Measure distance "A" shown in Figure 205 with the gear fully retracted and with the gear fully extended. The distance between the pivot points must be within ± 0.09 inch from the gear down to the gear up position. Adjust the idler link length as required to obtain the above tolerance.

(8) Retract the nose gear slightly so the center line of attach points “A”, “B” and "C" (Ref. Figure 206) align. With the mechanism aligned, check that the actuator link spring is compressed to 2.60 inch measured from collar to collar.

(9) Adjustment of the actuator link springs may be accomplished by disconnecting the actuator links from the actuator arms. Adjust the actuator link rod end bearings in or out to obtain the correct measurement of 2.60 inch as described in Step (8).

CAUTION: *To prevent damage to the hinge brackets from binding, pull the hinge down by hand before cycling the gear when the doors are disconnected.*

NOTE: It is suggested that the idler links be adjusted in increments of only 1/4 turn while the door fit is being adjusted.

(10) Cycle the nose gear at approximately 15° intervals. At each interval check that the mechanism and center hinge brackets are operating freely and not binding. Adjust the idler links and actuator links as required to assume free travel.

(11) With the nose gear fully extended, attach one door at a time to the center hinge brackets and check for a snug fit. Adjust the idler links as necessary to obtain a proper fit.

(12) With the nose gear fully retracted, attach the LH door to the center hinge bracket. Check for a snug door fit and a minimum clearance of 0.25 inch between the door and tire. Adjust the idler link as necessary to obtain a proper fit. Adjustment of the idler link in this position may change the door fit when the nose gear is in the extended position. Repeat this step for the RH door.

(13) Rig the forward set of doors to a snug fit with the gear in the retracted position by adjusting the rod end bearing on the door to gear linkage.

(14) After adjusting the doors, cycle the nose gear up and down several times, noting the fit of both sets of doors. Make sure that the nose gear and tire clear each door by at least 1 to 1 1/2 inches when the gear is cycled up.

(15) Ensure all three landing gear are down and locked.

(16) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

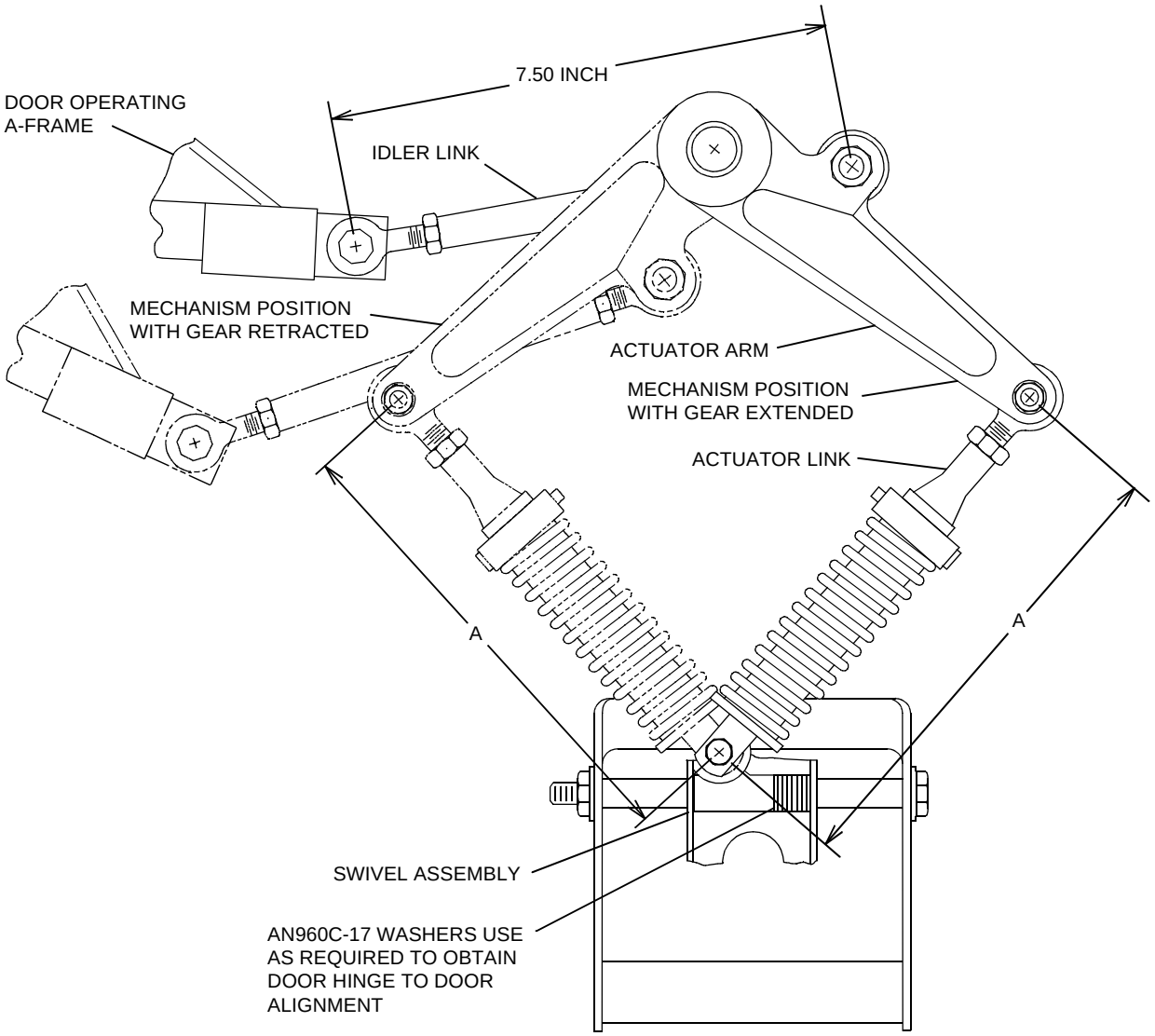

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Figure 205
Aft Nose Gear Door Rigging

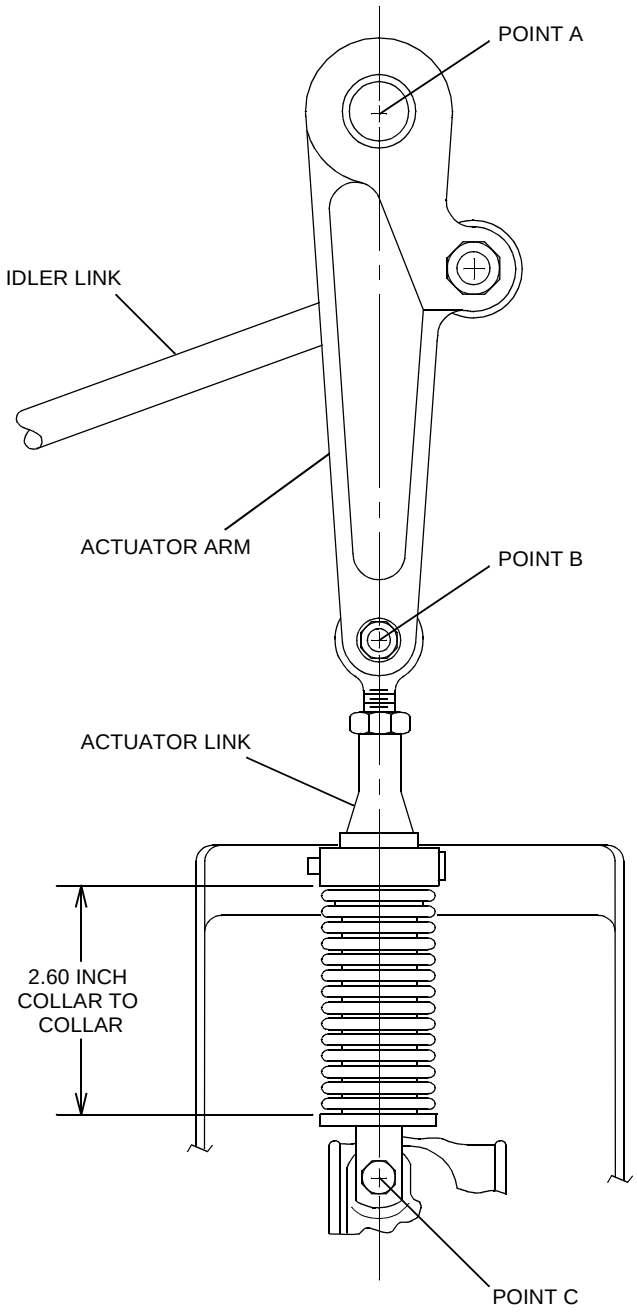

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Figure 206
Aft Nose Gear Door Actuator Link Rigging

7. MAIN LANDING GEAR DOORS

A. Rigging

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disconnect the door actuating cams from the doors by removing the cotter pins, nuts, washers and bolts from the upper links (Ref. Figure 207).

(3) With the landing gear retracted to the position shown in Figure 208, check for a clearance of 0.06 to 0.12 inch between the door actuator roller and the lower lobe of the actuation cam. If the clearance is less than 0.06 inch, hold the cam against the door hinge extrusion or wear strip, while removing the lower mounting bolt from cam plate, and reposition the cam pivot bolt in the slotted hole of the support bracket until the proper clearance is obtained. It will then be necessary to elongate the lower mounting holes in the support bracket for installation of the lower mounting bolt or to replace the plate with a new one that is pilot drilled at the lower hole. Airplanes equipped with the wear strip may be adjusted to a clearance of 0.06 inch. On earlier airplanes without the wear strip, the 0.06 inch clearance may not be obtainable because the roller will bottom out in the actuator cam slot, but under no circumstances must the clearance exceed 0.12 inch.

(4) Check the cams for proper alignment and freedom of movement and that the cam slot does not bottom out on the actuator roller when the gear is cycled.

NOTE: Each time the door linkage is readjusted, the overcenter dimension, the tire clearance, and the "snug" fit of closed doors must be rechecked.

(5) With the upper edge of a straight edge at the centerline of the center cam mounting bolt and the lower edge at the centerline of the lower link attaching bolt, check that each door locks in the open position with the center line of the upper link bolt 0.05 to 0.18 inch overcenter at the attaching point of the cam and link, when the gear are fully extended with the cam resting against the door hinge extrusion or wear strip. If necessary, adjust the length of the link until the door mechanism is properly overcenter (Ref. Figure 208).

(6) Retract the gear as required to check for a minimum clearance of one inch between the tire and door. If door clearance is sufficient, retract the gear and check that the doors are properly closed. If the doors do not close flush or are too tight, adjust the door linkage by moving the slotted attach plate on the door or by adjusting the link as required to obtain a proper fit.

(7) To ensure that the door is properly rigged and that the door linkage is overcenter, apply 25 pounds of pressure toward the gear at the lower center edge of the door. This is to ensure that each door is sufficiently locked in the open (overcenter) position to withstand the air load present during takeoff and landing. Perform Steps (2) thru (7) if the door folds while under this 25 pound load.

(8) When rigging is satisfactory, drill an additional hole in the slotted attach plate and angles on the door to install a third bolt for positive retention of the door adjustment. Locate this bolt between the bolts through the plate slots, and in line with the bolts in the plate slots. When rigging a plate that has already been drilled, the old plate can be reused if the drilled hole will provide positive retention, such as a hole in the shape of a figure eight. However, if the circumference of the new hole in the plate overlaps the circumference of the old hole in the plate by two thirds or more, the plate should be replaced to provide a new hole.

(9) Cycle the landing gear using the power pack, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and check rigging.

(10) Ensure all three landing gear are down and locked.

(11) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

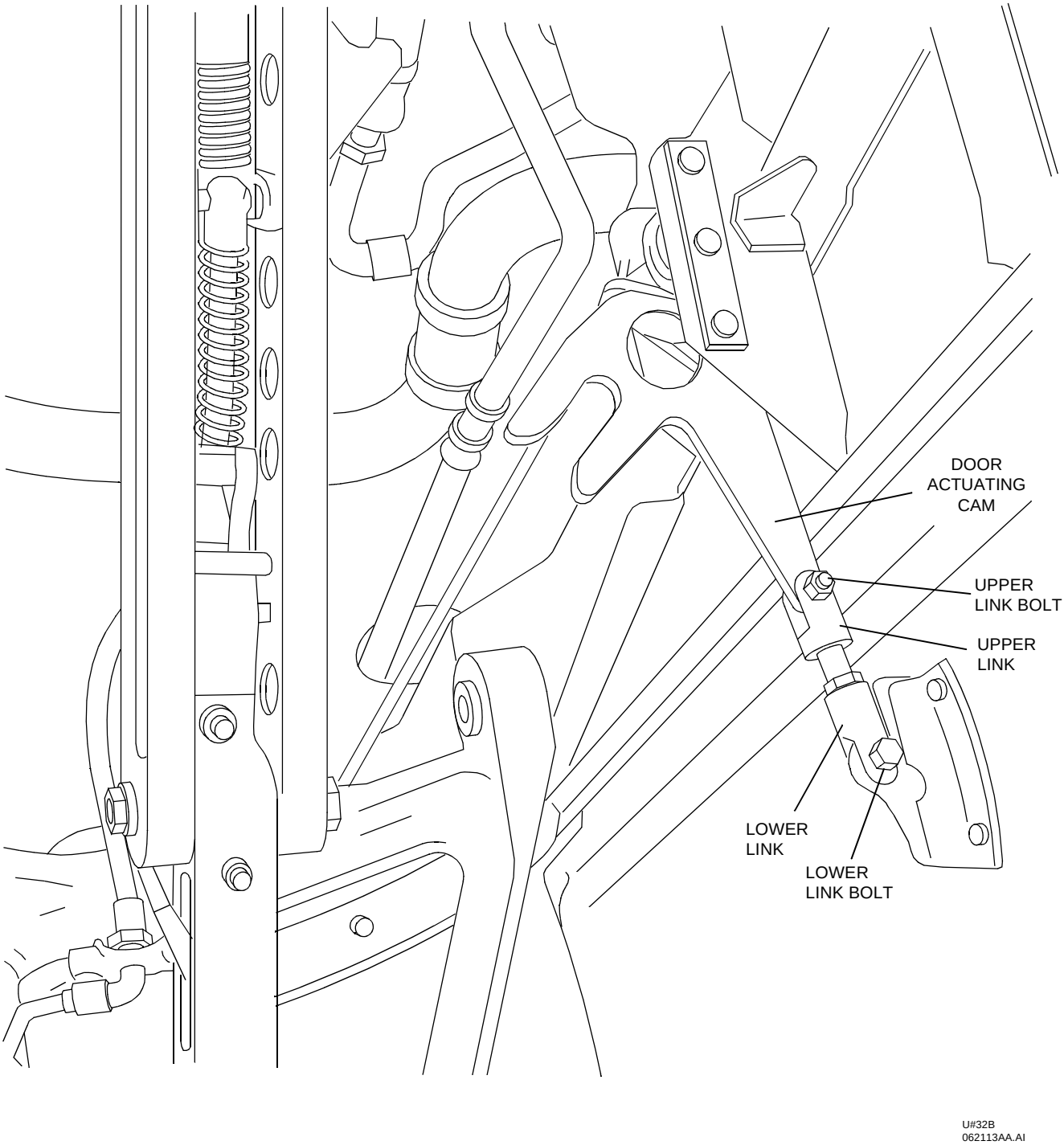


Figure 207
Main Gear Door Adjustments

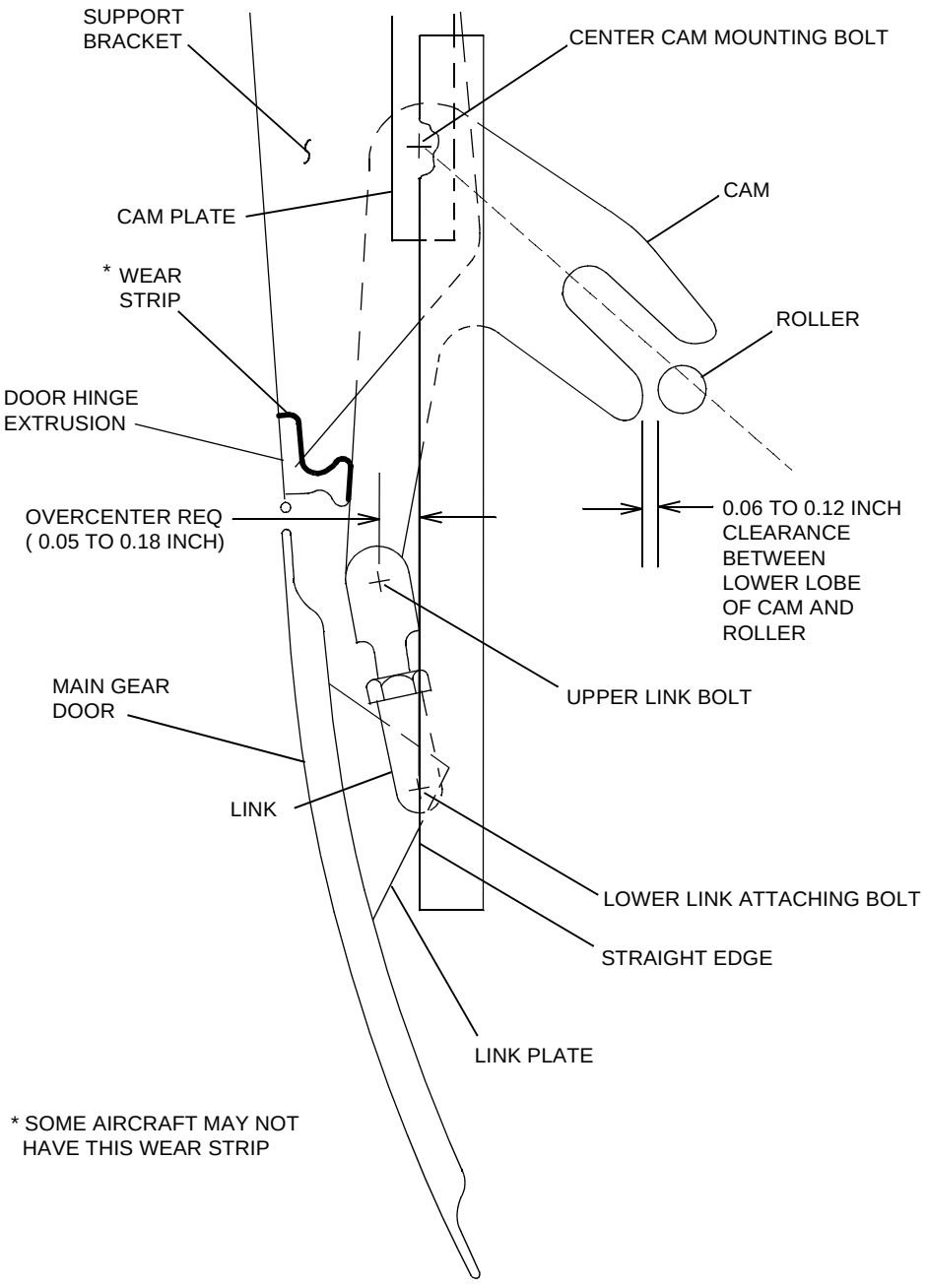


Figure 208
Main Gear Door Cam and Roller Rigging

8. FLOTATION MAIN LANDING GEAR DOORS

A. Rigging (Prior to U-165, Except U-50 Without Kit No. 101-8013-1S Installed)

NOTE: Before rigging of the doors is accomplished, check the door swivel for 0.010 inch maximum end play. Washers (P/N AN960-516L) may be added as required, to obtain the desired end play (Ref. Figure 209).

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) With the doors disconnected and out of the way, retract the gear. Check and adjust the clearance between the landing gear and main spar as outlined under MAIN GEAR RIGGING in this chapter.

(3) Push the doors up by hand to check for fit and clearance.

(4) Extend the gear and connect the door attach linkage so that the doors are held in an overcenter position. Adjust the link to 6.15 inches between the attachment hole centers.

(5) Apply a force of approximately 35 pounds at the center of the door in both the inboard and outboard directions. The door should maintain an overcenter position.

CAUTION: *Observe the door linkage closely during retraction. If the actuator rod is set too long, the clevis of the rod will bottom out on the swivel attached to the bracket on the door and bend the threaded portion of the actuator rod.*

(6) Retract the gear. The tire should clear the door by a minimum of 0.62 inch during the retraction cycle. If further adjustment is required, the door attach linkage should be adjusted 1/4 turn at a time (Ref. Figure 209).

(7) Cycle the gear several times and check for proper operation and a snug fit. If the door will not rig to a closed position and still maintain overcenter in the open position, it may be necessary to shim the bracket that is attached to the gear brace.

(8) Cycle the landing gear using the airplane hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(9) Ensure all three landing gear are down and locked.

(10) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

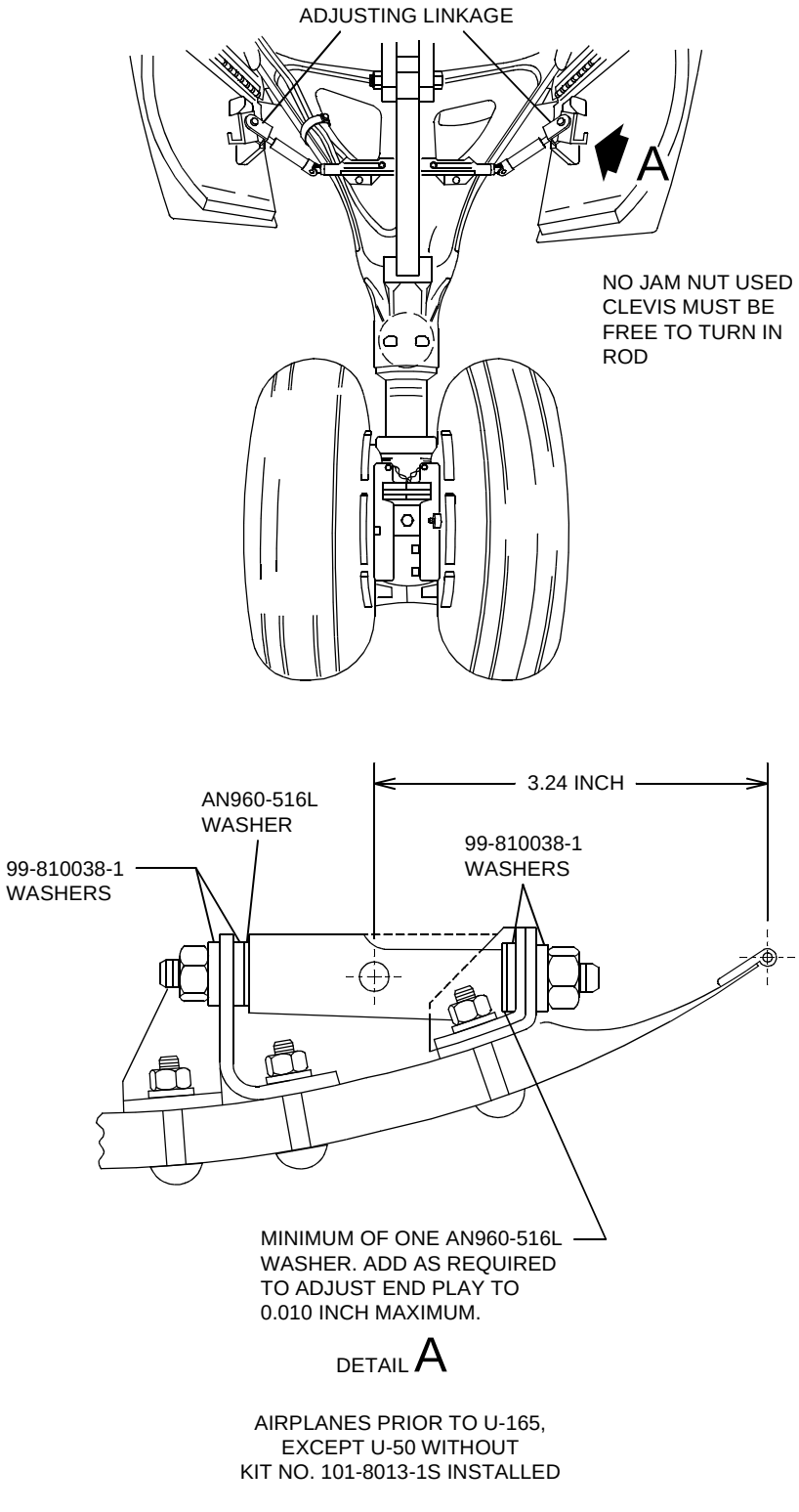


Figure 209
Flotation Gear Doors

9. FLOTATION MAIN LANDING GEAR DOORS

A. Rigging (U-50, U-165 and After, and Earlier Airplanes With Kit No. 101-8013-1S Installed)

NOTE: Before rigging of the doors is accomplished, check the door swivels for 0.010 inch maximum end play (Ref. Figure 210). The swivels should be free to rotate.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) With the doors disconnected and out of the way, retract the gear. Check and adjust the clearance between the landing gear and main spar as outlined under MAIN GEAR RIGGING in this chapter.

(3) Push the doors up by hand to check for fit and clearance.

(4) Extend the landing gear and adjust the door links to 6.15 inches between the attachment hole centers. Adjust the links as necessary so the doors will close snugly when the landing gear is in the UP position. Do not overtension the doors.

NOTE: During the hookup of the door links, check each joint for freedom of movement. No tight joints are permitted at the locations shown in Figure 211.

(5) Retract the landing gear and check that the tires clear the doors by a minimum of 0.62 inch during the retraction cycle. If further adjustment is required, the door link clevis should be adjusted 1/4 turn at a time.

(6) With the landing gear in the DOWN position, apply a maximum of 35lbs. of force to the lower edge of the door in both an inboard and outboard direction. The door must stay locked.

(7) Use a maximum of three shims (P/N 99-810000-19) between the top brace forging and the bracket assembly, if required to accomplish Step (6).

(8) Cycle the gear slowly to the following approximate door positions: open, partly closed, and closed. Check each joint for freedom of movement. No tight joints are permitted.

(9) Cycle the gear several times and check for proper operation and snug fitting doors.

(10) After the doors are rigged, secure the jam nut on the link assembly and safety as shown in Figure 211.

(11) Cycle the landing gear using the power pack, from the fully extended to fully retracted and back to fully extended at least once. Check for proper operation and check rigging.

(12) Ensure all three landing gear are down and locked.

(13) Remove the airplane from the jacks (Ref. Chapter 07-00-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

10. LANDING GEAR MANUAL EXTENSION

A. Operational Test

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) With the landing gear in the retracted position, pull the landing gear circuit breaker and place the landing gear control handle in the down position.

(3) Unstow the pump handle from the securing clip and pump the handle up and down until the three green down and locked indicator lights come on.

(4) Continue to pump until further pressure buildup is felt on the pump handle, then stow the handle under the securing clip.

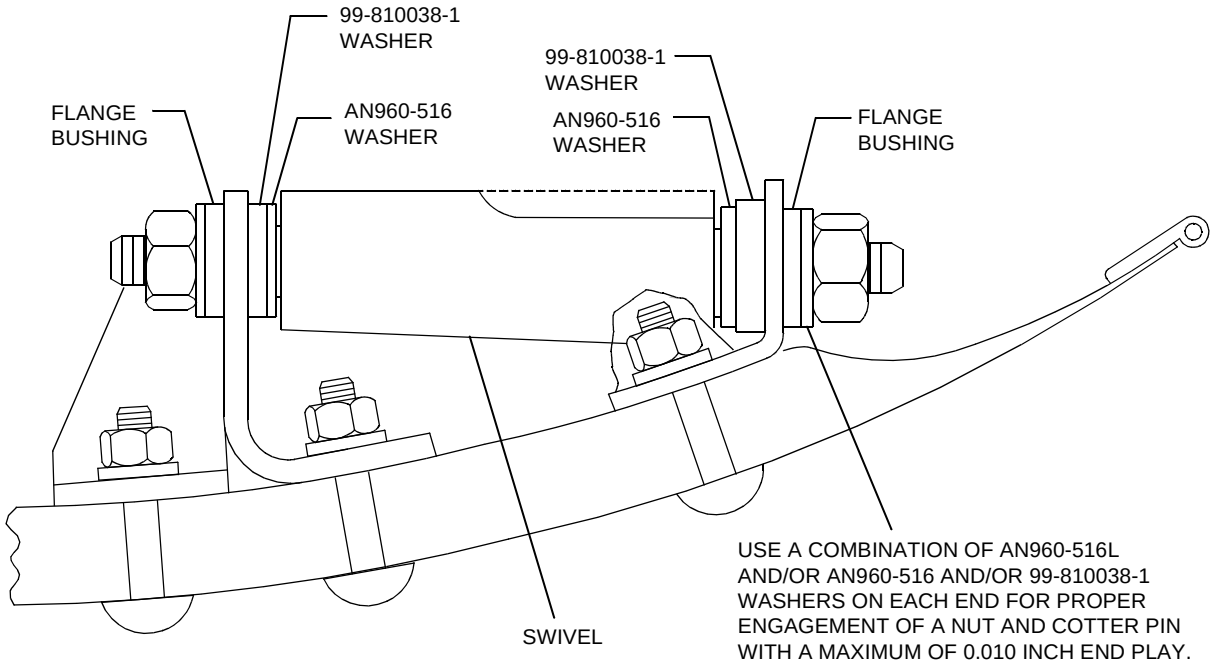
(5) On airplane serials U-50, U-219 and After, and on earlier airplane serials that have installed Kit No. 99-8025-3S or 99-8025-5S, secure the pump handle with safety wire. Earlier airplane serials which have not installed Kit No. 99-8025-3S or 99-8025-5S should do so at this time.

CAUTION: *After a landing gear PRACTICE manual extension, the pump handle MUST be stowed flush against the floor under the securing clip and be safety wired in place to assure proper operation of the normal system for a subsequent retraction.*

(6) After manual extension, cycle the landing gear using the airplane hydraulic power pack, from fully extended to fully retracted and back to fully extended at least once. Check for proper operation and rigging.

(7) Ensure all three landing gear are down and locked.

(8) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).



U-50, U-165 AND AFTER
AND EARLIER AIRPLANES
WITH KIT NO. 101-8013-1S INSTALLED

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Figure 210
Flotation Gear Door Swivel

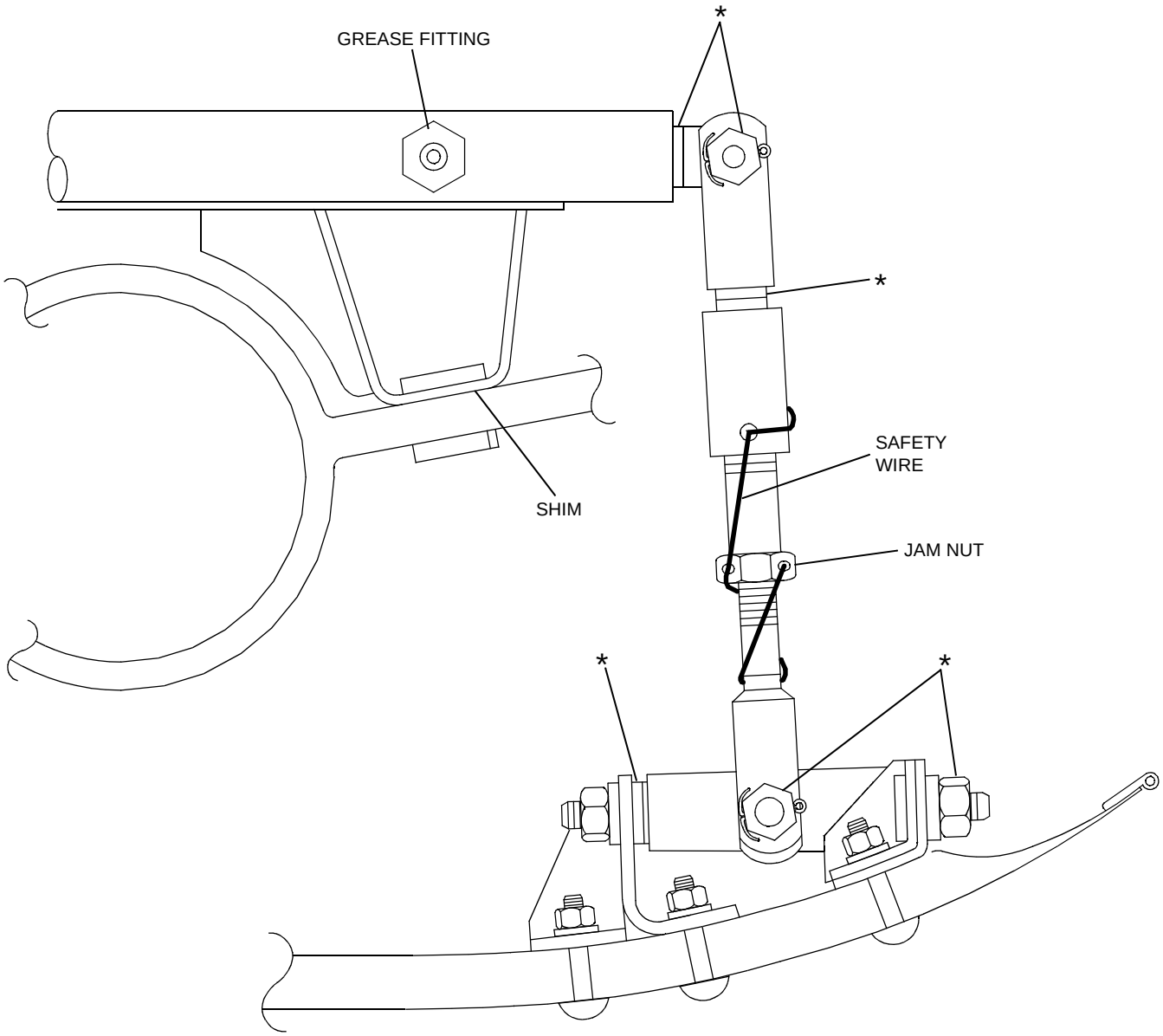


Figure 211
Flotation Gear Door Link Hookup

LANDING GEAR
WHEELS AND BRAKES
DESCRIPTION AND OPERATION

1. HYDRAULIC BRAKE SYSTEM

The dual hydraulic brakes are operated by depressing either the pilot's or copilot's rudder pedals. A shuttle valve adjacent to each set of pedals permits changing braking action from one set of pedals to the other. The depression of either set of pedals compresses the piston rod in the master cylinder attached to each pedal. The hydraulic pressure resulting from the movement of the pistons in the master cylinders is transmitted through flexible hoses and fixed aluminum tubing to the disc brake assemblies on the main landing gear wheels. This pressure forces the brake pistons to press against the linings and discs of the brake assembly. Early brake assemblies contain an automatic adjuster that retracts the brake linings to a preset clearance by use of compression springs upon release of the hydraulic pressure against the brake piston.

Dual parking valves are installed adjacent to the rudder pedals between the master cylinders of the pilot's rudder pedals and the wheel brakes. After the pilot's brake pedals have been depressed to build up pressure in the brake lines, both valves can be closed simultaneously by pulling out the parking brake handle on the left subpanel. This closes the valves to retain the pressure that was previously pumped into the brake lines. The parking brake is released when the brake pedals are depressed and the parking brake control is pushed in.

Brake system servicing is limited primarily to maintaining the hydraulic fluid level in the reservoir mounted in the upper LH corner of the aft bulkhead of the nose baggage compartment. A dip stick is provided for measuring the fluid level. When the reservoir is low on fluid, add a sufficient quantity of hydraulic fluid (9, Table 6, 91-00-00) to fill the reservoir to the full mark on the dipstick. For bleeding of the brakes, refer to BLEEDING THE BRAKE SYSTEM in this chapter. The only other requirements related to servicing involves the wheel brakes themselves. Although brakes on earlier airplanes were equipped with an automatic adjuster that eliminates the need for periodic adjustment of brake clearance, the brake must be checked periodically for indication of excessive wear. For detailed information relating to the proper inspection and repair procedures, refer to the applicable Hawker Beechcraft Maintenance Instructions for Wheels, Brakes and Tires (Publication Number 98-33818E or 98-35161B).

LANDING GEAR
WHEELS AND BRAKES
TROUBLESHOOTING

1. BRAKE SYSTEM

Troubleshooting - Brake System

INDICATION	PROBABLE CAUSE	REMARKS
1. Solid pedal and no brakes.	a. Brake lining worn beyond allowable limit.	
2. Spongy brake.	a. Air in system.	
3. Unable to hold pressure.	a. Leak in brake system.	a. Visually check entire system for evidence of leaks.
		b. Check master cylinder seals; replace if scored.
4. Parking brake will not hold.	a. Air in system.	
	b. Defective parking brake valve.	
5. Brakes grab.	a. Stones or foreign matter locking brake disc.	
	b. Warped or bent disc.	

LANDING GEAR
WHEELS AND BRAKES
MAINTENANCE PRACTICES

1. BRAKE SYSTEM

A. Bleeding

Brake system bleeding will be required whenever the system is opened at any point between the master cylinder and the wheel brake assembly, whenever the brakes become spongy in service, or whenever the parking brake will no longer hold. In the latter instance, the system should be further checked for leakage.

B. Gravity Bleeding

This method of bleeding is done from the master cylinder down to the brake assembly. With the pilot's right brake applied, open the bleeder valve on the respective brake assembly to allow the trapped air to escape. When the fluid is free of bubbles, apply the copilot's right brake. When both the pilot's and copilot's right brake pedal feels firm and there is no trace of air bubbles, repeat the procedure for the left brake pedal. During the bleeding procedure replace hydraulic fluid as necessary in the brake reservoir.

C. Pressure Bleeding

The most efficient method of bleeding the brake system is back bleeding with a pressure pot. By this method the system is bled from the wheel cylinder up in the following manner (Ref. Figure 201):

(1) Attach an extension line approximately three feet long to the reservoir vent line fining located on the upper aft bulkhead in the nose baggage compartment.

(2) Place the other end of the extension line into a clean container.

(3) Connect the pressure pot lines to the bleeder valve port on each brake assembly.

(4) Maintain a pressure of approximately 15 pounds on the pressure pot and open the pressure pot bleeder valve. Bleed the pilot's brake until all evidence of sponginess is gone. Raise and lower the gear two or three times to remove trapped air in the wheel cylinder. When the pilot's brake functions satisfactorily, close the valve on the pressure pot and pump the copilot's brake pedal to change the shuttle valve position so that fluid will be routed through the copilot's brake system. Bleed the copilot's brakes in the same manner as the pilot's brakes were bled, close the bleeder valve and repeat the process with the other wheel.

(5) Remove the pressure pot lines and the extension line.

(6) Remove the hydraulic reservoir cap and add hydraulic fluid (9, Table 6, 91-00-00) to the FULL level on the dipstick.

(7) Check the brake operation; pedal pressure should be equal on both brakes.

NOTE: Do not cap the vent port in the nose baggage compartment. It must remain open in order for the lines to breathe.

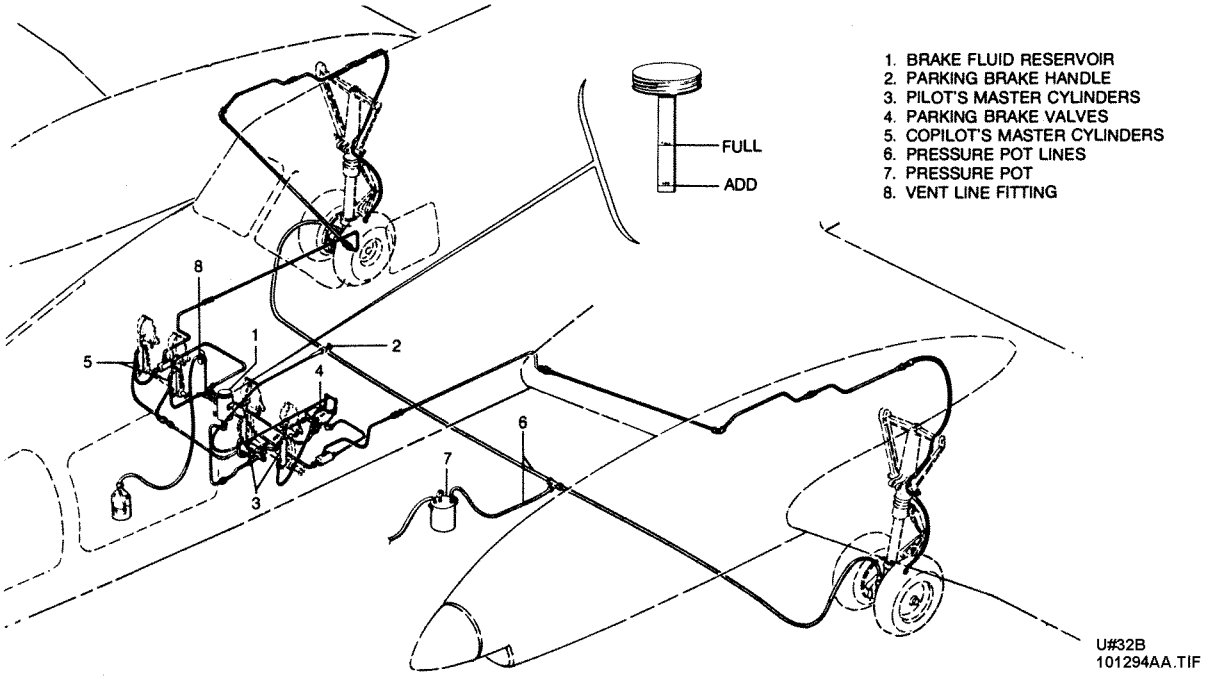
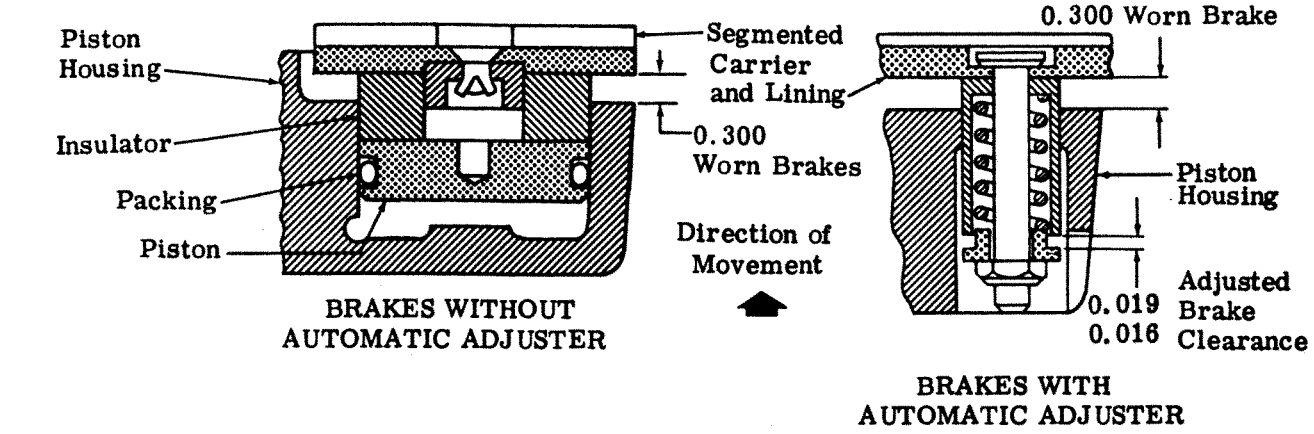
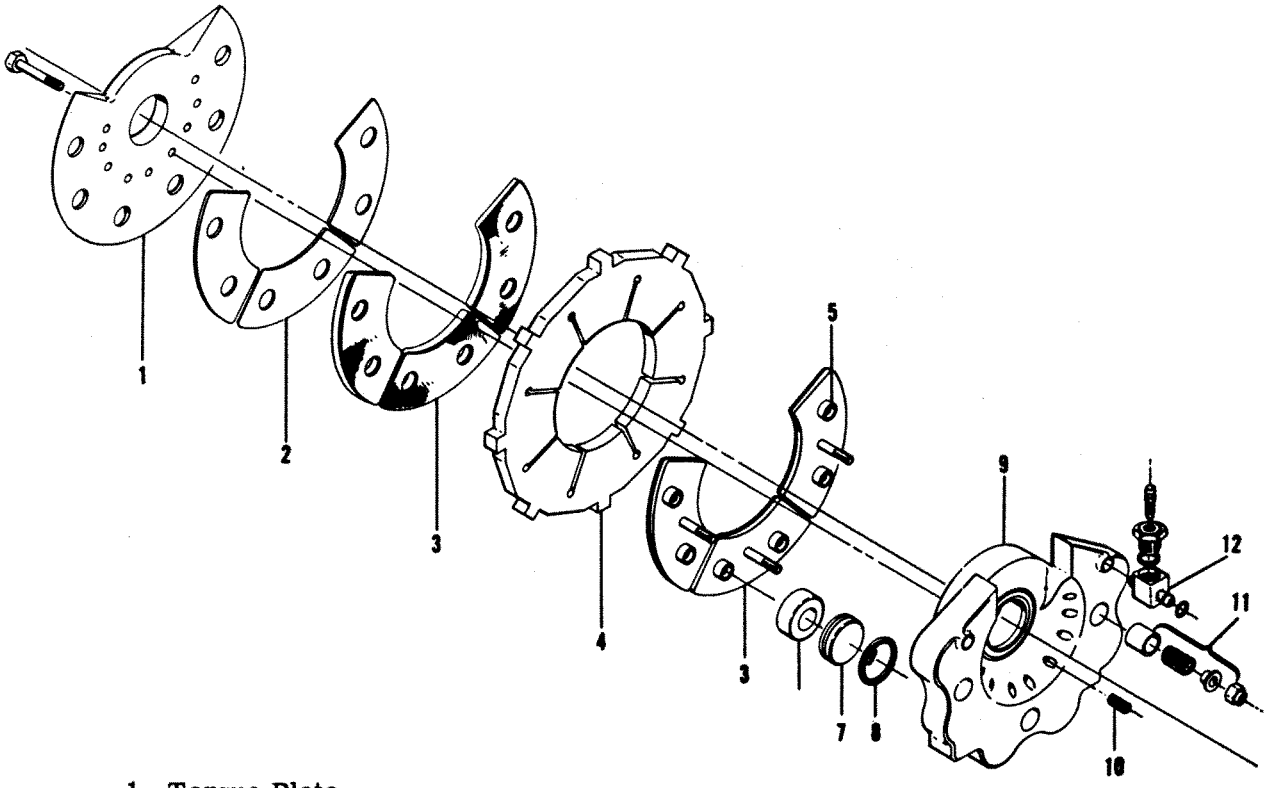


Figure 201
Bleeding the Brakes





1. Torque Plate
2. Insulator
3. Segmented Carrier And Lining
4. Brake Disc
5. Torque Buttons
6. Piston Insulator
7. Piston
8. Packing
9. Piston Housing
10. Thread Insert
11. Automatic Adjuster Assembly
12. Brake Line Manifold

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Figure 202
Brake Assembly (Half of Duel Brake Shown)

2. BRAKE WEAR

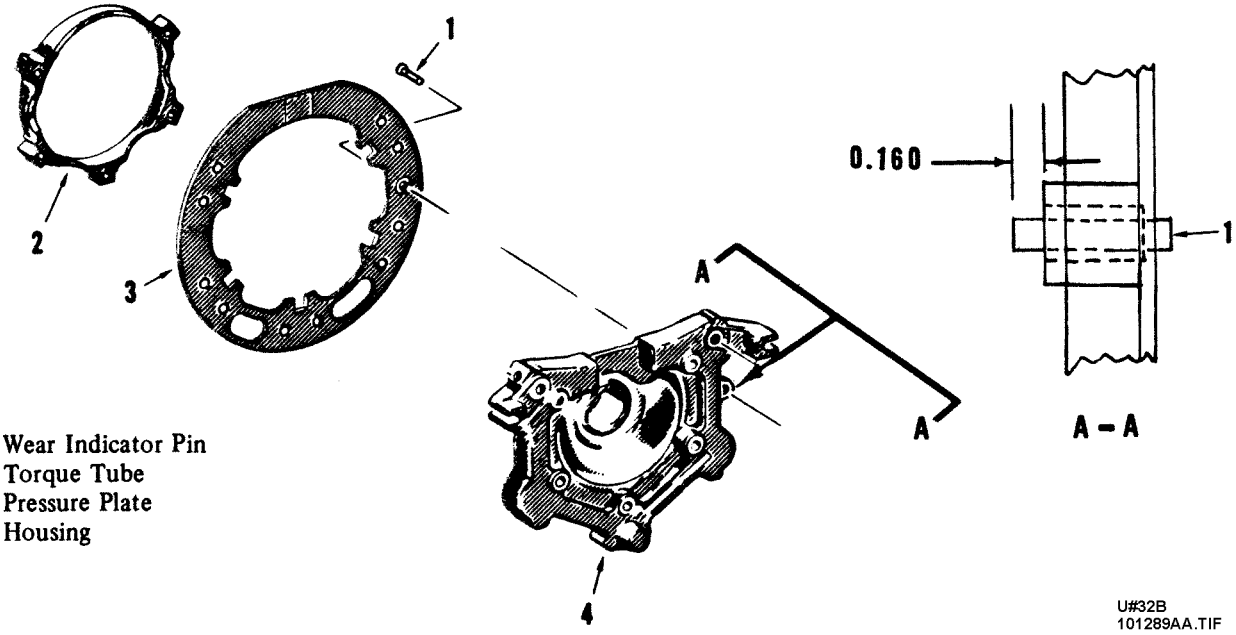
A. Limits

NOTE: Lining adjustment is automatic on some airplanes and the brake adjuster setting should not be tampered with.

To check for a worn brake condition on all airplanes (with or without self adjusters), measure the distance between the segmented carrier and lining assembly and the piston housing. When the distance is 0.300 inch or more, the brake is ready for a lining inspection. If the thickness of the segmented carrier and lining assembly is less than 0.198 inch, overhaul the brake as indicated in the Hawker Beechcraft Maintenance Instructions for Wheels, Brakes and Tires used on the Model 99 Airliner (Publication Number 98-33818E). On automatic adjusting brakes, a brake clearance of 0.016 to 0.019 inch should be maintained between the spring retainer and the automatic adjusting housing to prevent brake "chatter" and to assure proper braking response to pedal movement (Ref. Figure 202).

B. Limits (Flotation Gear)

The brake contains a built-in wear indicator pin (1) for ease in determining the extent of brake wear (Ref. Figure 203). On a new brake, the wear indicator pin extends 0.160 inch beyond the face of the pin bushing (with brake applied). The brake should be removed from service and overhauled when the end of the wear indicator pin is flush with the face of the pin bushing (with brake applied). Overhaul the brake as indicated in the Hawker Beechcraft Maintenance Instructions for Main Wheels, Brakes and Tires and Nose Wheel and Tire (Publication Number 98-35161B).



1. Wear Indicator Pin
2. Torque Tube
3. Pressure Plate
4. Housing

Figure 203
Brake Wear Limits (Flotation Gear)

3. BRAKE MASTER CYLINDER

A. Linkage Adjustment

(1) Loosen the lock nut on the piston shaft.

(2) Remove the pin from the clevis on the back of the rudder pedal and slide the clevis free of the lug.

(3) Adjust the length of the shaft by turning the clevis on or off the shaft. Adjust the clevis on the shaft so that the distance between the center line of the clevis hole and the center line of the cylinder attach hole is nine inches. The length of the shaft should be adjusted so that the piston does not "bottom" when the brake is applied. A maximum of seven threads on the shaft should be exposed.

(4) Install the clevis and pin, then safety the pin and tighten the lock nut.

(5) Check the upper and lower attachment points for possible interference between the clevis and attachment lugs which could cause binding and subsequent damage. Any interference would occur only when the rudder pedals are at full travel in either direction.

B. Removal

(1) Remove the front floorboard.

(2) Disconnect and plug the brake lines at the cylinder.

(3) Remove the master cylinder attaching bolts at the upper and lower end of the master cylinder. Remove the cylinder assembly.

C. Installation

(1) Place the brake master cylinder in position on the rudder pedal and install the attaching bolts.

(2) Connect the brake lines to the master cylinder.

(3) Replace the floorboard.

(4) Bleed the brake system as outlined in BLEEDING THE BRAKE SYSTEM.

4. PARKING BRAKE CONTROL

A. Adjustment

NOTE: Overhaul of the parking brake valves is not recommended. If leakage is encountered, replace the valve.

When a parking brake valve has been replaced, it is important that full travel of the parking brake valve be maintained. This may be assured by the following adjustment:

(1) Loosen the phenolic block bolts and reposition the outer housing of the brake control in the blocks until pushing in the brake control fully opens the parking brake valve.

(2) Push forward on the valve arm assemblies until the parking brake valves are fully closed, then check that the control handle protrudes approximately 1 $\pm$ 1/16 inch from the left subpanel.

(3) Pull the valve arm assemblies aft until the parking brake valves are fully open, then check that the control handle is depressed to within 1/16 inch of the subpanel.

(4) The arm travel from the fully closed to the fully open position of the parking valve should be approximately 1 ± 1/16 inch.

5. MAIN WHEEL AND BRAKE

A. Removal

(1) Remove the axle nut, outer seal, and the bearing cone, then remove the wheel and tire from the axle.

CAUTION: *Use care, when removing all internal and external main wheel components to avoid damaging the axle surface and/or nicking the axle threads.*

(2) After the main wheel and tire have been removed, disconnect the hydraulic line and cap all open lines and ports.

(3) Slide the brake and spacer off the axle.

(4) The opposite wheel and brake may be removed in the same manner.

B. Installation

(1) Pack grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00) into bearing cones and smear grease on ends of rollers. Do not overlubricate. Spread a thin coat of grease on the surface of the bearing cups.

CAUTION: *Aeroshell Grease 5 and MIL-G-81322 are not compatible. DO NOT MIX. If grease is changed, make certain that all affected components are thoroughly cleaned before lubrication.*

(2) Check the axle and nut for burrs or rough threads.

(3) Apply grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00) to the axle threads and all bearing surfaces of washers and nuts.

(4) Mount one brake in place on the axle, making sure that the brake spacer bottoms in the strut. Apply a thin coat of grease (same as used on wheel bearings) on the inside of the spacer and on the axle at assembly.

(5) Mount the wheel and tire on the axle. Check to ensure that the disc lugs engage with the torque key lined slots on the inner wheel half flange.

(6) Install the bearing cone, outer seal, and axle nut.

(7) Tighten the axle nut 250 to 300 inch-pounds torque while rotating the wheel to ensure proper seating of the bearings.

(8) Back off the axle nut to zero torque, then torque the nut 125 to 145 inch-pounds while rotating the wheel.

(9) Install the cotter pin. If the holes do not align, tighten the nut to the next available keying position.

(10) Remove the caps from the brake inlet ports and carefully install the bleed manifold in the front port and the hydraulic line inlet manifold at the rear port.

(11) Install the opposite brake assembly and carefully align the inlet ports with the manifold assemblies installed in the first brake. Apply a thin coat of grease (same as used on wheel bearings) on the inside of the brake spacer and on the axle at assembly.

(12) Repeat Steps (5) through (9) to install the opposite wheel and tire.

(13) Bleed the brake system as outlined in BLEEDING THE BRAKE SYSTEM.

6. NOSE WHEEL

A. Removal

(1) Disconnect and remove the static dissipation cable at the lower static cable mounting nut.

NOTE: The static dissipation cable is not installed on later model airplanes.

(2) Remove the axle nut and spacer.

CAUTION: *Use care when removing all internal and external nose wheel components to avoid damaging the axle surface and/or nicking the axle threads.*

(3) Remove the outer grease seal and the bearing cone, and remove the wheel and tire from the axle.

(4) Remove the inner bearing cone and the grease seal.

B. Installation

(1) Pack grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00) into bearing cones and smear grease on ends of rollers. Do not overlubricate. Spread a thin coat of grease on the surface of the bearing cups.

CAUTION: *Shell Grease 5 and MIL-G-81322 are not compatible, DO NOT MIX. If grease is changed, make certain that all affected components are thoroughly cleaned before relubrication.*

(2) Check the axle and nut for burrs or rough threads.

(3) Carefully place a seal on the axle against the inboard spacer with the legend "This side toward bearing" facing the proper direction. Place the inboard bearing cone on the axle against the seal, place the wheel and tire on the axle, and seat the bearing cone in the cup. Install the outboard bearing cone on the axle and seat the cone in the outboard bearing cup.

(4) Install a seal on the axle with the legend "This side toward bearing" against the bearing cone. Install the outboard spacer.

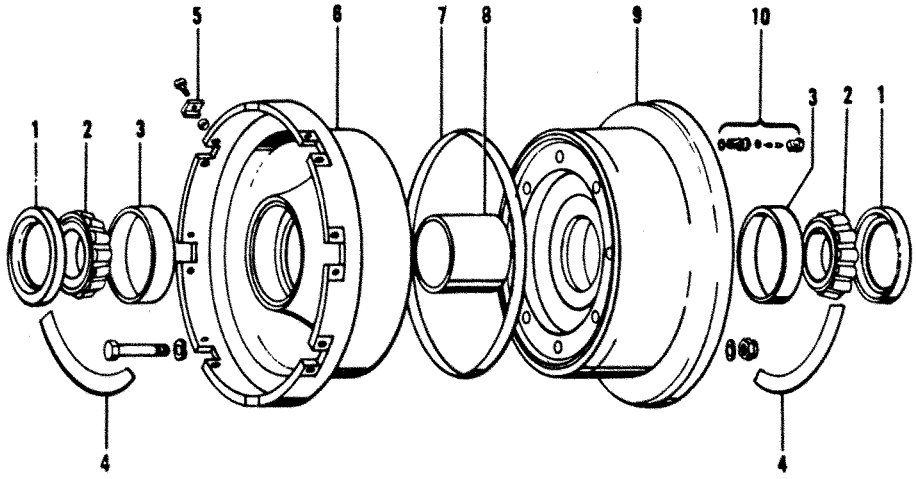
(5) Apply a light coat of grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00) to the threads of the axle and the nut, and install the nut on the axle.

(6) Tighten the axle nut 150 to 200 inch-pounds torque while rotating the wheel to insure proper seating of the bearings.

(7) Back off the axle nut to zero torque, then torque the nut to 30 inch-pounds while rotating the wheel. Check to see that there is no side motion of the wheel.

(8) Install the cotter pin. If the holes do not align, tighten the nut to the next available keying position.

(9) Install the static dissipation cable (if applicable) with 8.43 ± 0.12 inch extending beyond the lower static cable mounting nut.



1. Bearing Seal
2. Bearing
3. Bearing Race
4. Identification Plate
5. Torque Key
6. Inner Wheel Half
7. Packing
8. Sleeve
9. Outer Wheel Half
10. Air Valve Assembly

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Figure 204
Main Wheel Assembly

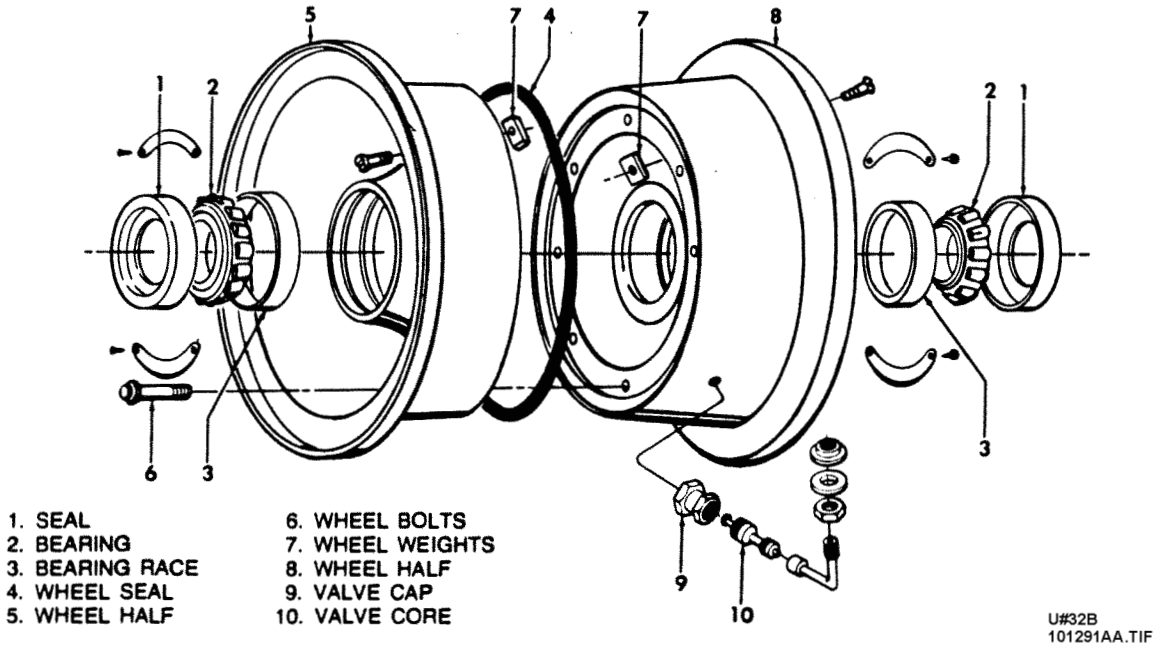


Figure 205
Nose Wheel Assembly

7. MAIN WHEEL AND BRAKE (FLOTATION GEAR)

A. Removal

(1) Remove the nut securing each wheel and tire to the axle. Remove each wheel and tire from the axle, being careful not to drop or damage the bearing seals and cones.

CAUTION: *Use care when removing all internal and external main wheel components to avoid damaging the axle surface and/or nicking the axle threads.*

(2) Disconnect the brake hydraulic line from the inlet swivel fitting and cap the line.

(3) Remove the brake self-locking nut, washers, and bolt.

(4) Separate and remove the brake assembly from the strut, being careful not to drop or damage the inlet swivel fitting.

B. Installation

(1) Position the inboard brake assembly on the shock strut. Apply a thin coat of grease (same as used on wheel bearings) to the inside diameter of the brake housing on the axle at assembly.

(2) Lubricate packings with grease (17, Table 6, 91-00-00) and install packings in the grooves in the inlet fitting. Lightly lubricate surfaces next to the grooves.

(3) Install the inlet fitting at the adapter on the inboard brake assembly, being careful not to displace or damage the packings.

(4) Position the outboard brake assembly on the shock strut. Carefully mate the inlet adapter on the outboard brake with the inlet fitting, do not displace or damage the packings.

(5) Make sure that both brake assemblies are correctly seated on the strut and axle and that the inlet fitting swivels freely.

(6) Secure the brake assemblies together with the attaching bolts, washers, and self-locking nuts, installing one washer under each bolt head and one washer under each nut.

(7) Tighten the self-locking nuts to 120 inch-pounds torque.

(8) Connect the hydraulic brake fluid line to the inlet fitting.

(9) Lubricate wheel bearing cups with a light coat of grease (17, Table 6, 91-00-00) or preferred grease (112, Table 6, 91-00-00).

CAUTION: *Aeroshell Grease 5 and MIL-G-81322 are not compatible, DO NOT MIX. If grease is changed, make certain that all affected components are thoroughly cleaned before relubrication.*

(10) Pack bearing cones with grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00).

(11) Lubricate lips of bearing seals with a light coat of grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00).

(12) Check the axle and nut for burrs or rough threads.

(13) Apply grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00) to threads and all bearing surfaces of washers and nuts.

(14) Install the inboard bearing seal and the bearing cone on the axle. The lips of the bearing seal must be toward the wheel.

(15) Place the wheel and tire on the axle and install the outboard bearing cone and seal. Check to ensure engagement of brake disc lugs in the torque key lined slots of the inner wheel half flange.

CAUTION: *Make sure that the bearing seals are properly seated in the wheel against the bearing cups and that the seals are not damaged.*

(16) Place the axle nut on the axle and tighten 150 to 200 inch-pounds torque while rotating the wheel to insure proper seating of the bearings.

(17) Back off the axle nut to zero torque, then torque the nut to 40 inch-pounds while rotating the wheel.

(18) Install the cotter pin. If the holes do not align, tighten the nut to the next available keying position.

(19) Repeat Steps (9) through (18) to install the opposite wheel and tire.

(20) Bleed the brake system as described in BLEEDING THE BRAKE SYSTEM.

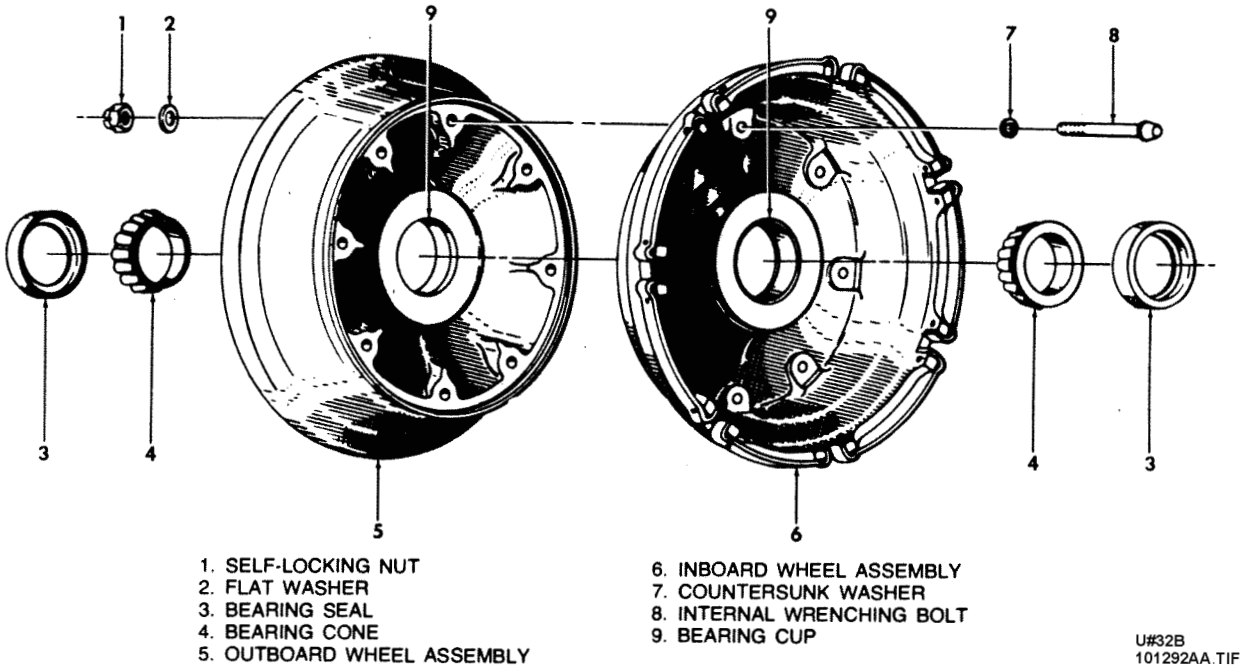


Figure 206
Main Wheel Assembly (Flotation Gear)

8. NOSE WHEEL (FLOTATION GEAR)

A. Removal

(1) Disconnect the static dissipation cable at the lower static cable mounting nut.

NOTE: The static dissipation cable is not installed on later model airplanes.

(2) Remove the axle nut and spacer.

CAUTION: *Use care when removing all internal and external nose wheel components to avoid damaging the axle surface and/or nicking the axle threads.*

(3) Remove the wheel and tire from the axle, being careful not to damage the closure rings, felt seal rings, and bearing cones.

(4) Remove the retaining rings, closure rings, felt seal rings, and bearing cones.

B. Installation

(1) Pack bearing cones and lubricate bearing cups with grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00).

CAUTION: *Aeroshell Grease 5 and MIL-G-81322 are not compatible, DO NOT MIX. If grease is changed, make certain that all affected components are thoroughly cleaned before relubrication.*

(2) Lubricate felt seal rings with SAE No. 10 machine oil.

(3) Install bearing cones in the wheel. Install one closure ring against each bearing cup, then one felt seal ring and the other closure ring on each side of the wheel. The flanged ends of all closure rings must face the bearings.

(4) Secure bearings and seals in place with retaining rings.

(5) Check the axle and nut for burrs or rough threads.

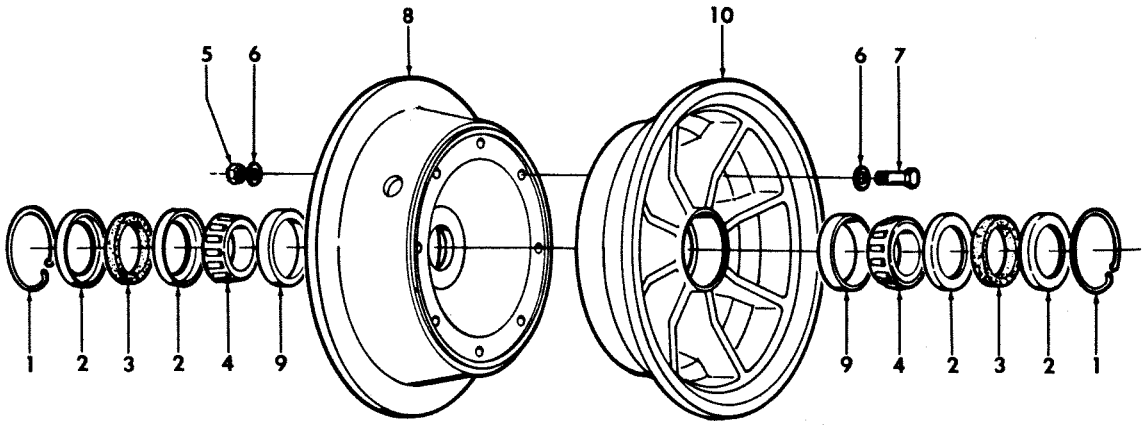
(6) Apply a light coat of grease (17, Table 6, 91-00-00) or the preferred grease (112, Table 6, 91-00-00) to the threads of the axle and the nut, and place the wheel and tire, spacer and nut on the axle.

(7) Tighten the axle nut 150 to 200 inch-pounds torque while rotating the wheel to insure proper seating of the bearings.

(8) Back off the axle nut to zero torque, then torque the nut to 30 inch-pounds while rotating the wheel. Check to see that there is no side motion of the wheel.

(9) Install the cotter pin. If the holes do not align, tighten the nut to the next available keying position.

(10) Install the static dissipation cable (if applicable) with 8.43 ± 0.12 inch extending beyond the lower static cable mounting nut.



1. RETAINING RING
2. BEARING CLOSURE RING
3. FELT SEAL
4. BEARING CONE
5. NUT
6. WASHER
7. BOLT
8. WHEEL HALF
9. BEARING CUP
10. WHEEL HALF

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Figure 207
Nose Wheel Assembly (Flotation Gear)

LANDING GEAR
BRAKE DEICER (U-50, U-165 AND AFTER)
MAINTENANCE PRACTICES

1. BRAKE DEICE SHUTOFF VALVE

NOTE: Make sure that the four flexible hoses for the brake deicer are the same color and vendor on both main wheels. The Flexfab (Gray) hoses and the Bemco (Red) hoses must not be mixed on any one airplane when maintenance is performed.

A. Removal

(1) Remove the access panel from the top side of the nacelle just aft of the main spar.

(2) Disconnect the electrical wiring to the shutoff valve solenoid and to the annunciator light switch. Tag and identify the wires.

(3) Remove the tubes from the inlet and outlet sides of the shutoff valve.

(4) Remove the shutoff valve from the mounting bracket.

B. Installation

(1) Attach the shutoff valve to the mounting bracket.

(2) Connect the tubes to the inlet and outlet sides of the shutoff valve.

(3) Connect the electrical wiring to the shutoff valve solenoid and to the annunciator light switch.

(4) Install the access panel to the nacelle.

2. DISTRIBUTOR MANIFOLD

A. Removal

(1) Remove the main wheel and brake as indicated in Chapter 32-40-00.

(2) Disconnect the deicer hose from the journal of the distributor manifold.

(3) Remove the two bolts, washers, and nuts securing the distributor manifold to the piston and axle assembly.

B. Installation

NOTE: Ensure that the orifices in the distributor manifold are clean and unobstructed prior to installation of the manifold.

(1) Secure the distributor manifold to the piston and axle assembly with the two retaining bolts, washers, and nuts.

NOTE: Tighten the top retaining bolt only until snug, taking care not to overtorque. It should be possible to rotate the washer by finger when the bolt is properly tightened.

(2) Connect the bleed air hose to the journal of the distributor manifold.

(3) Install the main wheel and brake as indicated in Chapter 32-40-00.

3. DISTRIBUTOR MANIFOLD (FLOTATION GEAR)

A. Removal

(1) Remove the main wheel and brake as indicated in Chapter 32-40-00.

(2) Disconnect the deicer hose from the journal of the distributor manifold.

(3) Remove the five bolts, washers and nuts securing the distributor manifold to the wheel and brake assembly.

B. Installation

NOTE: Ensure that the orifices in the distributor manifold are clean and unobstructed prior to installation of the manifold.

(1) Secure the distributor manifold to the wheel and brake assembly with the five retaining bolts, washers, and nuts. Tighten the self-locking nuts evenly and torque to 120 inch-pounds.

(2) Connect the bleed air hose to the journal of the distributor manifold.

(3) Install the main wheel and brake as indicated in Chapter 32-40-00.

LANDING GEAR
STEERING
DESCRIPTION AND OPERATION

1. GENERAL

Direct linkage to the rudder pedals allows the nose wheel to be turned 14° to the left of center and 12° to the right. When adder control is augmented by a wheel brake, the nose wheel deflection can be increased considerably. A spring mechanism in the steering linkage dampens the transmission of excessive shock loads to the rudder pedals. When retracted, the nose wheel is automatically centered and the steering linkage becomes inoperative.

LANDING GEAR
STEERING MECHANISM
MAINTENANCE PRACTICES

1. STEERING MECHANISM

A. Removal

(1) Remove the screws (one in each door) to free the aft doors from the center hinge brackets.

(2) Remove the cotter pin, nut, washer and bolt from the intermediate control tube (11) and centering cam assembly (10) (Ref. Figure 201).

(3) Remove the three bolts and washers that attach the centering cam assembly (10) to the centering cam bracket.

(4) Disconnect the disconnect centering actuator (9) from the bottom of the floorboard by removing the nuts, washers and bolts that attach the disconnect centering actuator bracket to the floorboard.

(5) Disconnect the electrical wiring from the disconnect centering actuator motor.

(6) Remove the cotter pin, nut, washer and bolt that attach the disconnect tube (8) to the forward idler arm (7). Remove the aft portion of the steering mechanism from the airplane.

(7) Remove the cotter pin, nut, washer and bolt that attach the forward control tube (6) to the forward idler arm (7).

(8) Remove the cotter pin, nut, washier and bolt that attach the steering mechanism to the top of the strut.

(9) Remove the four bolts and washers that attach the support (1) to the inner wheel well structure.

(10) Remove the forward portion of the steering mechanism from the airplane.

B. Installation

(1) Position the forward portion of the steering mechanism in place and attach the support (1) to the inner wheel well structure with the four bolts (Ref. Figure 201).

(2) Install the bolt, washer, nut and cotter pin that attach the steering mechanism to the top of the strut.

(3) Install the bolt, washer, nut and cotter pin that attach the forward control tube (6) to the forward idler arm (7).

(4) Position the aft portion of the steering mechanism in place and install the bolt, washer, nut and cotter pin that attach the disconnect tube (8) to the forward idler arm (7).

(5) Attach the disconnect centering actuator (9) to the bottom of the floorboard by installing the bolts, washers and nuts that attach the disconnect centering actuator to the floorboard.

(6) Connect the electrical wiring to the disconnect centering actuator motor.

(7) Install the three bolts and washers that attach the centering cam assembly (10) to the centering cam bracket.

(8) Install the bolt, washer, nut and cotter pin that attach the intermediate control tube (11) to the centering cam assembly (10).

(9) Adjust the forward control tube (6) to center the nose wheel when the rudder pedals are centered and test the system for binding through full rudder travel in both directions.

(10) Install the screws (1 in each door) to secure the doors to the door brackets.

2. NOSE LANDING GEAR STEERING DISCONNECT MECHANISM

To remove the steering action from the rudder pedals when the airplane is in flight, a disconnect actuator is incorporated in the nose gear steering mechanism. When the weight is removed from the left hand landing gear strut (during the retract cycle), the landing gear safety switch closes causing the actuator to automatically disconnect the nose wheel steering from the rudder pedals. The nose gear steering disconnect mechanism may be rigged in the following manner:

(1) Set the nose gear steering for full left and right travel as coordinated with the rudder.

NOTE: The red marks on the front of the nose strut are maximum tow positions and not steering positions.

(2) Center the nose wheel and connect the actuator.

NOTE: The internal rigging of the actuator is preset. Rigging of the actuator in the field is not necessary.

(3) Test for full nose wheel travel in the steering mode with the actuator connected by working the rudder pedals.

(4) With the airplane on jacks, actuate the landing gear slowly and check the centering action of the entire mechanism. If adjustment is required, adjust the length of the connecting rods in the steering mechanism as necessary to obtain steering travel to coincide with the rudder travel.

3. DISCONNECT ACTUATOR

A. Removal

Removal of the landing gear disconnect actuator is accomplished as follows:

(1) With the airplane on jacks and the lading gear in the extended position, disconnect the aft set of nose landing gear doors for easier access to the actuator.

(2) Disconnect the actuator from the nose gear steering linkage.

(3) Disconnect the actuator electrical leads from the airplane wiring.

(4) Remove the bolt that attaches the actuator to the actuator support bracket and remove the actuator.

B. Installation

(1) Position the actuator and install the bolt that attaches the actuator to the actuator support bracket. Center the actuator with the actuator support bracket by installing AN960-10 or AN960-10L washers, as required, between the support bracket and the actuator.

(2) Connect the actuator to the steering disconnect linkage. Center the actuator with the steering disconnect linkage by installing AN960-416 or AN960-416L washers, as required between the bushings and the actuator.

(3) Connect the actuator electrical leads to the airplane wiring.

NOTE: Check the steering mechanism for proper rig as described in this chapter under NOSE LANDING GEAR STEERING DISCONNECT MECHANISM.

(4) Connect the aft nose gear doors to the center hinge brackets.

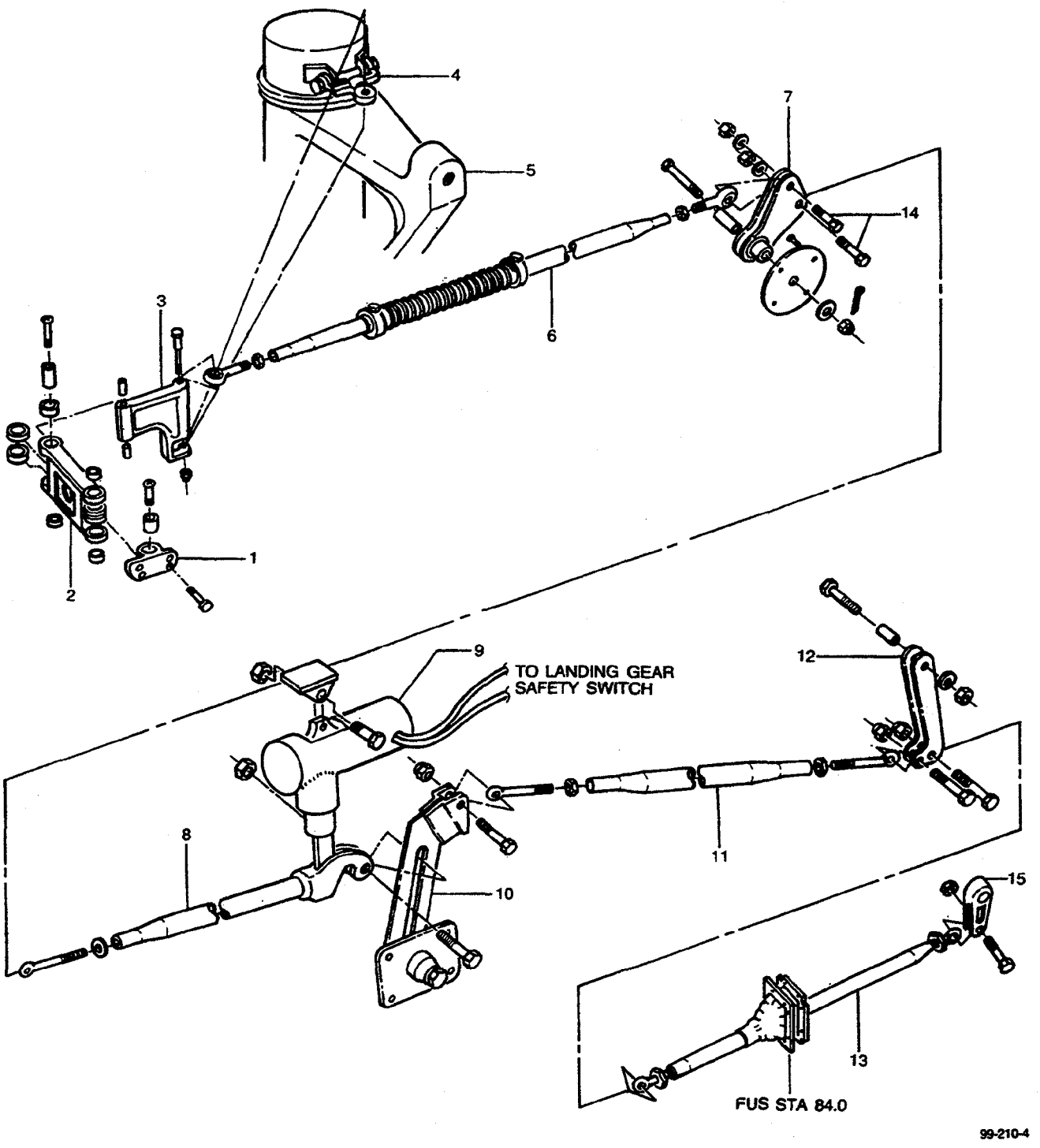

Figure 201
Nose Gear Steering

LANDING GEAR
POSITION AND WARNING
DESCRIPTION AND OPERATION

1. GENERAL

A downlock position indicator switch is attached to each drag leg assembly, on the mechanical landing gear system, and to each landing gear drag leg assembly and the nose gear actuator housing on the hydraulic system. This switch will cause the three green down-position indicator lights to come on when the landing gear is down and locked. A gear up position switch, located in each wheel well, actuates a red "gear in transit" light located in the landing gear control handle, as soon as the landing gear begins to extend. The light will remain illuminated until the gear is down and locked. The in transit light again illuminates when the gear begins to retract and remains illuminated until the gear is up and locked. The landing gear position lights and the landing gear control handle light may be tested by pressing the GEAR LIGHT TEST switch next to the gear handle. A dimming rheostat is used to regulate the intensity of the lights only when navigation lights are illuminated.

The red light also comes on with the warning horn anytime the gears are not down and locked when either power lever is retarded to less than 79% $\pm$ 2% gas turbine (N_1) speed. A HORN SILENCE button, located adjacent to the landing gear control handle, will silence the warning horn until the landing gear or power levers are recycled, however, the landing gear warning horn will sound anytime the flaps are extended beyond the approach position and the landing gear is not down and locked. The HORN SILENCE button has no control over the warning horn in this situation. The warning horn is actuated by the down-position switches, which are wired in parallel with each other and in series with the power lever warning horn switches, located in the pedestal. On U-50, U-124 and After and on previous airplanes with Kit 99-3017S installed in compliance with Service Instructions No. 0318-350, the in transit light in the landing gear handle will remain illuminated after the warning horn silence circuit has been activated.

CAUTION: *Never rely on the safety switch to keep the gear down while taxiing or on landing or takeoff roll. Always check the position of the landing gear switch.*

To prevent accidental gear retraction on the ground, a safety switch on the right main strut breaks the control circuit whenever the strut is compressed.

LANDING GEAR
POSITION AND WARNING
TROUBLESHOOTING

1. LANDING GEAR WARNING HORN AND SWITCHES

Troubleshooting - Landing Gear Warning Horn and Switches

INDICATION	PROBABLE CAUSE	REMARKS
(1) Warning horn inoperative or malfunctioning.	(a) Horn circuit breaker tripped	
	(b) Open or grounded circuit.	Check continuity: (1) Circuit breaker to horn, throttle switches, LH gear down lock warning switch, and ground. (2) From throttle switches to nose gear down lock warning switch and ground. (3) From throttle switches to RH gear down lock warning switch and ground.
	(c) Throttle switches defective or out of adjustment.	
	(d) Main or nose gear down lock warning switches defective or out of adjustment.	
	(e) Flasher stuck open.	Check flasher continuity.
(2) Circuit breaker tripping.	(a) Grounded circuit.	Check for ground between circuit breaker and warning horn.

2. LANDING GEAR POSITION INDICATORS AND SWITCHES

Troubleshooting - Landing Gear Position Indicators and Switches

INDICATION	PROBABLE CAUSE	REMARKS
(1) All indicators inoperative.	(a) Landing gear indicator circuit breaker tripped.	
	(b) Open circuit between circuit breaker and indicators.	
(2) One indicator completely inoperative.	(a) Defective indicator.	
	(b) Open circuit.	

Troubleshooting - Landing Gear Position Indicators and Switches (Continued)

INDICATION	PROBABLE CAUSE	REMARKS
	(c) Defective indicator switch.	

3. LANDING GEAR LIMIT AND SAFETY SWITCHES

Troubleshooting - Landing Gear Limit and Safety Switches

INDICATION	PROBABLE CAUSE	REMARKS
(1) Landing gear motor fails to shut off when gear is retracted and “in transit” light stays off.	(a) Up limit switch out of adjustment (mechanical system only).	
	(b) Defective up limit switch (mechanical system); pressure switch defective (hydraulic system).	
(2) Landing gear fails to retract.	(a) Safety switch not closing.	
	(b) Up limit switch stuck (mechanical system); pressure switch defective (hydraulic system).	
(3) Landing gear fails to retract completely but “in-transit” light illuminates.	(a) Up limit switch opens early (mechanical system); Pressure switch defective (hydraulic system).	
(4) Positive down locks do not fully engage.	(a) Down limit switch opens early.	
(5) Landing gear motor fails to shut off when gear is extended.	(a) Defective down limit switch.	

LANDING GEAR
POSITION AND WARNING
MAINTENANCE PRACTICES

1. NOSE AND MAIN LANDING GEAR UP POSITION SWITCH

A. Adjustment (U-1 thru U-164, except U-50)

Before adjusting any switch, the landing gear must be rigged correctly as instructed in Chapter 32-30-01 or 32-31-01.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disable the landing gear doors by disconnecting from the actuating cams.

(3) Ensure that the landing gear is fully retracted before adjusting the switch.

(4) Loosen the locking nuts that hold the switch in position. Adjust the switch until the in transit light in the landing gear control switch handle extinguishes.

NOTE: Excessive compression or bottoming out of the switch plunger may cause damage to the switch.

(5) Adjust the switch (in) an additional 0.050 to 0.080 inch beyond the actuation point and tighten the locking nuts to hold the switch in position and safety wire.

(6) Operate the gear through at least one complete cycle and check for proper operation of the in transit lights located in the landing gear control switch handle.

(7) Connect the landing gear doors to the actuating cams.

(8) Ensure all three gear are down and locked.

(9) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

B. Adjustment (U-50, U-165 and After)

Before adjusting any switch, the landing gear must be rigged correctly as instructed in Chapter 32-31-01.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disable the landing gear doors by disconnecting from the actuating cams.

(3) Ensure that the landing gear is fully retracted before adjusting the switch.

NOTE: Determining the actuation point of the up position switches by the presence of continuity is the only approved method of ascertaining the actuation point of these switches.

(4) Refer to the appropriate MODEL 99 AIRLINER Wiring Diagram Manual and connect an ohmmeter or continuity checker between the following points:

(a) Nose Gear Switch - pins “7” and “9” of the nose wheel well terminal board.

(b) LH Main Gear Switch - pin “D” of the wheel well receptacle and pin “11A” of the main cabin crossover harness receptacle.

(c) LH Main Gear Switch - pin “D” of the wheel well receptacle and pin “12A” of the main cabin crossover harness receptacle.

(5) Loosen the locking nuts that hold the switch in position. Adjust the switch until the ohmmeter or continuity checker indicates continuity.

NOTE: Excessive compression or bottoming out of the switch plunger may damage the switch.

(6) Adjust the switch (in) an additional 0.050 to 0.080 inch beyond the actuation point and tighten the locking nuts to hold the switch in position and safety wire.

(7) Disconnect the ohmmeter or continuity checker.

(8) Operate the gear through at least one complete cycle, and check for proper operation of the in transit lights located in the landing gear control switch handle.

(9) Connect the landing gear doors to the actuating cams.

(10) Ensure all three gear are down and locked.

(11) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

2. NOSE GEAR DRAG BRACE DOWNLOCK SWITCH

A. Adjustment (U-1 thru U-151, except U-50)

Before adjusting the downlock switch, the landing gear must be rigged correctly as instructed in Chapter 32-30-01 or 32-31-01. With the landing gear down and the actuator locked, adjust the switch as follows:

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disable the landing gear doors by disconnecting from the actuating cams.

NOTE: Determining the actuation point of the downlock switch by the presence of continuity is the only approved method of ascertaining the actuation point of the switch.

(3) Refer to the appropriate MODEL 99 AIRLINER Wiring Diagram Manual and connect an ohmmeter or continuity checker between pins “4” and “5” of the nose wheel well terminal board.

NOTE: Excessive compression or bottoming out of the switch plunger may damage the switch.

(4) Loosen the locking nuts that hold the switch in position. Adjust the switch until the ohmmeter or continuity checker indicates continuity. Continue to adjust the switch until the switch plunger compresses 0.050 to 0.080 inch beyond the switch actuation point.

(5) Secure the switch with the locking nuts and safety wire.

(6) Disconnect the ohmmeter or continuity checker from the wheel well terminal board.

(7) Operate the gear through at least one complete cycle and check for proper operation of the GEAR DOWN light and the landing gear warning horn.

(8) Connect the landing gear doors to the actuating cams.

(9) Ensure all three gear are down and locked.

(10) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

B. Adjustment (U-50, U-152 and After)

Before adjusting the downlock switch, the landing gear must be rigged correctly as instructed in Chapter 32-30-01 or 32-31-01. With the landing gear down and the actuator locked, adjust the switch as follows:

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disable the landing gear doors by disconnecting from the actuating cams.

(3) Position the adjustable stop with approximately 0.25 inch of threads exposed (Ref. Figure 201).

(4) Install the switch so that the plunger contacts the head of the adjusting bolt but does not compress the plunger.

(5) Secure the switch with the locking nuts and safety wire.

NOTE: Determining the actuation point of the downlock switch by the presence of continuity is the only approved method of ascertaining the actuation point of the switch.

(6) Refer to the appropriate MODEL 99 AIRLINER Wiring Diagram Manual and connect an ohmmeter or continuity checker between pins “8” and “11” (U-50, U-165 and AFTER) or pins “4” and “5” (U-152 thru U-164) of the nose wheel well terminal board.

(7) Turn the adjusting bolt until the ohmmeter or continuity checker indicates continuity.

CAUTION: *Excessive compression or bottoming out of the switch plunger may damage the switch.*

(8) Continue to adjust the stop until the switch plunger compresses 0.050 to 0.080 inch beyond the switch actuation point.

(9) Disconnect the ohmmeter or continuity checker from the wheel well terminal board.

(10) Operate the gear through at least one complete cycle and check for proper operation of the GEAR DOWN light and the landing gear warning horn.

(11) Connect the landing gear doors to the actuating cams.

(12) Ensure all three landing gears are down and locked.

(13) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

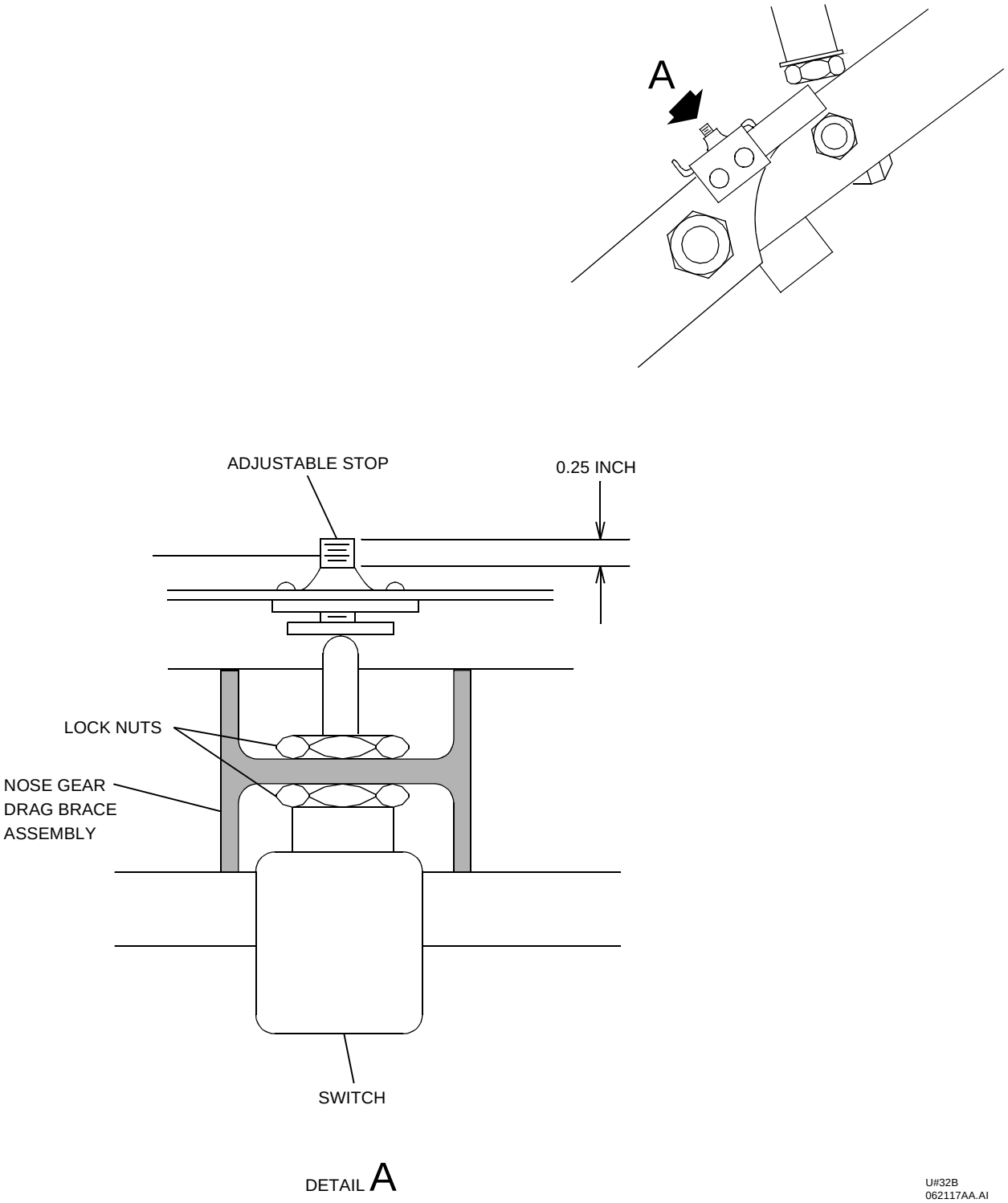

Figure 201
Nose Gear Drag Brace Downlock Switch Adjustment

1. HOOK
2. SWITCH
3. LOCK PLATE

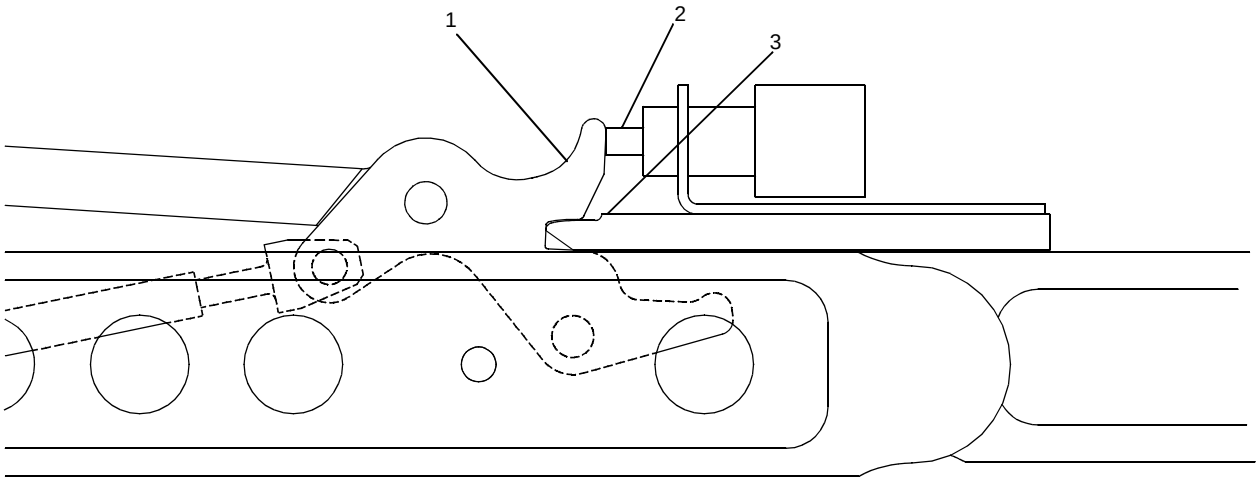

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Figure 202
Main Landing Gear Drag Brace Downlock Switch

3. MAIN LANDING GEAR DRAG BRACE DOWNLOCK SWITCH

A. Adjustment (U-1 thru U-164, except U-50, With Mechanical Landing Gear System)

Before adjusting the downlock switch, the landing gear must be rigged correctly as instructed in Chapter 32-30-01. With the landing gear down and the downlock fully engaged, adjust either main gear drag brace downlock switch as follows:

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disable the landing gear doors by disconnecting from the actuating cams.

(3) Loosen the locknut on the switch, and back the switch out to the point that the plunger is contacted, but not compressed.

NOTE: Determining the actuation point of the downlock switches by the presence of continuity is the only approved method of ascertaining the actuation point of these switches.

(4) Disconnect the electrical connector to the downlock switch being adjusted. Refer to the appropriate MODEL 99 AIRLINER Wiring Diagram Manual and connect an ohmmeter or continuity checker between pins “D” and “E” (LH switch) or pins “C” and “F” (RH switch).

(5) Adjust the switch until it actuates as indicated by continuity on the ohmmeter or continuity checker.

CAUTION: *Excessive compression or bottoming out of the switch plunger may damage the switch.*

(6) Adjust the switch (in) an additional 0.050 to 0.080 inch beyond the actuation point and tighten the locking nuts to hold the switch in position and safety wire.

(7) Disconnect the ohmmeter or continuity checker from the downlock switch connector.

(8) Operate the gear through at least one complete cycle, and check for proper operation of the GEAR DOWN light and the landing gear warning horn.

(9) Connect the landing gear doors to the actuating cams.

(10) Ensure all three gear are down and locked.

(11) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

B. Adjustment (U-1 thru U-164, except U-50, With Hydraulic Landing Gear System)

Before adjusting the downlock switch, the landing gear must be rigged correctly as instructed in Chapter 32-31-01. Adjust either main gear drag brace downlock switch as follows:

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disable the landing gear doors by disconnecting from the actuating cams.

(3) Loosen the locknut on the switch and back the switch out to the point that the plunger is contacted, but not compressed.

NOTE: Determining the actuation point of the downlock switches by the presence of continuity is the only approved method of ascertaining the actuation point of these switches.

(4) Disconnect the electrical connector to the downlock switch being adjusted. Refer to the appropriate MODEL 99 AIRLINER Wiring Diagram Manual and connect an ohmmeter or continuity checker between pins “D” and “E” (LH switch) or pins “C” and “F” (RH switch).

CAUTION: *Ensure that the lines which are to be used in connecting the pressure gage to the airplane's hydraulic system are free of dirt and other contaminants. Purge all air from the pressure gage lines prior to connection to the hydraulic system.*

(5) Install a pressure gage with a scale up to at least 1000 psi in parallel with the extension side of the landing gear hydraulic system.

(6) Extend the actuator by applying 500 to 700 psig hydraulic pressure to the actuator extend port to ensure that the actuator Is fully extended and that the downlock hook is positively engaged.

(7) Adjust the switch until it actuates as indicated by continuity on the ohmmeter or continuity checker.

CAUTION: *Excessive compression or bottoming out of the switch plunger may damage the switch.*

(8) Adjust the switch (in) an additional 0.050 to 0.080 inch beyond the actuation point and tighten the locking nuts to hold the switch in position and safety wire.

(9) Check the switch with the drag brace down and locked, and no hydraulic pressure in the actuator. Pressure is relieved by performing the following:

(a) Retract the landing gear at least enough to unlock the landing gear.

(b) Using the airplane hydraulic hand pump, extend the landing gear to the down and locked position.

(c) Stow the hand pump handle. The hydraulic pressure should relieve.

(d) Notice how the downlock switch (2) contacts the hook. Examine the position of the drag brace hook on the lock plate. The hook (1) must be in the full locked down position on the lock plate (3) (Ref. Figure 202).

(10) Disconnect the ohmmeter or continuity checker from the downlock switch connector.

NOTE: If the exposed length of the plunger measures less than 0.22 inch, inspect the switch bracket to ensure that it is not bent. If the switch bracket is not bent, inspect the landing gear structure to ensure that it is not bent or worn. If the landing gear structure is not bent or worn, replace the switch.

(11) Measure the exposed plunger length to ensure that the exposed length of the plunger does NOT measure less than 0.22 inch.

(12) Operate the gear through at least one complete cycle and check for proper operation of the GEAR DOWN light and the landing gear warning horn.

(13) Connect the landing gear doors to the actuating cams.

(14) Ensure all three landing gear are down and locked.

(15) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

C. Adjustment (U-50, U-165 and After)

Before adjusting the downlock switch, the landing gear must be rigged correctly as instructed in Chapter 32-31-01. Adjust either main gear drag brace downlock switch as follows:

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

(2) Disable the landing gear doors by disconnecting from the actuating cams.

(3) Loosen the locknut on the switch and back the switch out to the point that the plunger is contacted, but not compressed.

NOTE: Determining the actuation point of the downlock switches by the presence of continuity is the only approved method of ascertaining the actuation point of these switches.

(4) Disconnect the electrical connector to the downlock switch being adjusted. Refer to the appropriate MODEL 99 AIRLINER Wiring Diagram Manual and connect an ohmmeter or continuity checker between pins “E” and “F” (LH and RH switches).

CAUTION: *Ensure that the lines which are to be used in connecting the pressure gage to the airplane's hydraulic system are free of dirt and other contaminants. Purge all air from the pressure gage lines prior to connection to the hydraulic system.*

(5) Install a pressure gage (scaled up to at least 1000 psi) in parallel with the extension side of the landing gear hydraulic system.

(6) Extend the actuator by applying 500 to 700 psig hydraulic pressure to the actuator extend port to insure that the actuator is fully extended and that the downlock hook is positively engaged.

(7) Adjust the switch until it actuates as indicated by continuity on the ohmmeter or continuity checker.

CAUTION: *Excessive compression or bottoming out of the switch plunger may damage the switch.*

(8) Adjust the switch (in) an additional 0.050 to 0.080 inch beyond the actuation point and tighten the locking nuts to hold the switch in position and safety wire.

(9) Check the switch with the drag brace down and locked, and no hydraulic pressure in the actuator. Pressure is relieved by performing the following:

(a) Retract the landing gear at least enough to unlock the landing gear.

(b) Using the airplane hydraulic hand pump, extend the landing gear to the down and locked position.

(c) Stow the hand pump handle. The hydraulic pressure should relieve.

(d) Notice how the downlock switch (2) contacts the hook. Examine the position of the drag brace hook on the lock plate. The hook (1) must be in the full locked down position on the lock plate (3) (Ref. Figure 202).

(10) Disconnect the ohmmeter or continuity checker from the downlock switch connector.

(11) Operate the gear through at least one complete cycle, and check for proper operation of the GEAR DOWN light and the landing gear warning horn.

(12) Connect the landing gear doors to the actuating cams.

(13) Ensure all three gear are down and locked.

(14) Remove the airplane from the jacks (Ref. Chapter 07-10-00, LOWERING THE AIRPLANE AFTER THREE POINT JACKING).

4. NOSE GEAR ACTUATOR DOWNLOCK SWITCH

A. Replacement (Hydraulic Landing Gear System)

The actuator downlock switch is preadjusted and should not require further attention. If the switch must be replaced for any reason, proceed as follows:

(1) Remove the actuator from the airplane as described in Chapter 32-31-00.

(2) Carefully remove the old switch so as not to remove or drop the switch actuator ball that contacts the switch plunger.

(3) Apply 500 to 700 psig of hydraulic pressure to the extend port of the actuator to lock the actuator in the extended position. Disconnect the hydraulic pump from the actuator.

(4) Apply a 500-pound column load to the actuator piston rod to ensure that the actuator is down and locked.

(5) Connect an ohmmeter or continuity checker between pins “D” and “E” of the switch connector (U-50, U-165 and after) or switch wires 1-20 and 3-20 (U-1 thru U-164, except U-50, with hydraulic landing gear).

NOTE: Determining the actuation point of the downlock switch by the presence of continuity is the only approved method of ascertaining the actuation point of this switch.

(6) Turn the switch in until the ohmmeter or continuity checker indicates continuity. The switch may be turned in a maximum of one additional turn to align the electrical wiring toward the top of the actuator.

(7) Apply 400 psig to the retract port of the actuator to unlock the actuator. The ohmmeter or continuity checker should indicate an open circuit.

(8) Tighten the jam nut and safety wire.

(9) Cycle the actuator three times, monitoring the ohmmeter or continuity checker to ensure that the switch is actuating properly.

(10) Disconnect the ohmmeter or continuity checker from the actuator and install the actuator in the airplane as instructed in Chapter 32-31-00.

5. LANDING GEAR SAFETY SWITCH

A. Adjustment

This procedure is applicable to the RH and LH safety switches. Refer to the safety switch illustration in Figure 203.

WARNING: Make all adjustments with the power off at the master switch and no external power connected.

(1) Place the airplane on jacks (Ref. Chapter 07-10-00, THREE POINT JACKING). All tires must be clear of the floor.

NOTE: Use a gear type puller to remove the switch arm from the switch shaft to prevent damage to the internal mechanism of the switch.

(2) Remove the safety switch actuator rod (1) from the attaching bracket on the upper torque knee, then remove the retaining nut (2) and the switch arm (3) from the switch shaft.

(3) Jack the landing gear so the shock strut is compressed from 0.50 ± 0.12 inch from the fully extended position.

(4) On airplanes prior to U-165, except U-50, connect wire leads from a 28 vdc test light box to pins “g” and “e” (RH installation) or pins “f” and “c” (LH installation] of the receptacle plug located in the upper rear of each main landing gear wheel well.

(5) On airplanes U-50, U-165 and after, connect the wire leads from a 28 vdc test light box to pins “p” and “n” (LH and RH installations) of the receptacle plug located in the upper rear of each main landing gear wheel well.

(6) Rotate the switch shaft counterclockwise until the test light illuminates.

(7) Remove the safety wire from the locking screw (4) on the switch arm (3) and back off the locking screw.

(8) Install the switch arm (3) on the switch shaft in a position parallel to the upper torque knee, and adjust the actuating rod (1) to align with the attaching bracket on the torque knee. Install the actuation rod connecting bolt.

(9) Position the shock strut to 0.75 to 2.5 inches from the fully extended position and adjust the switch shaft clockwise at the adjusting screw (5) until the test light extinguishes. When satisfactory adjustment is reached, tighten the locking screw (4) and retaining nut (2).

NOTE: Actuation of the switch during compression and extension of the shock strut differs because of tolerances in the switch and its attendant linkage on the landing gear.

(10) Before safety wiring the locking screw (4) to the switch arm, check that the switch opens (test light extinguished) when the shock strut is compressed 0.75 to 2.5 inches from the fully extended position. Check that the switch closes (test light illuminated) when the shock strut is extended to within 0.50 $\pm$ 0.12 inch from the fully extended position. The switch will remain closed when the shock strut is fully extended.

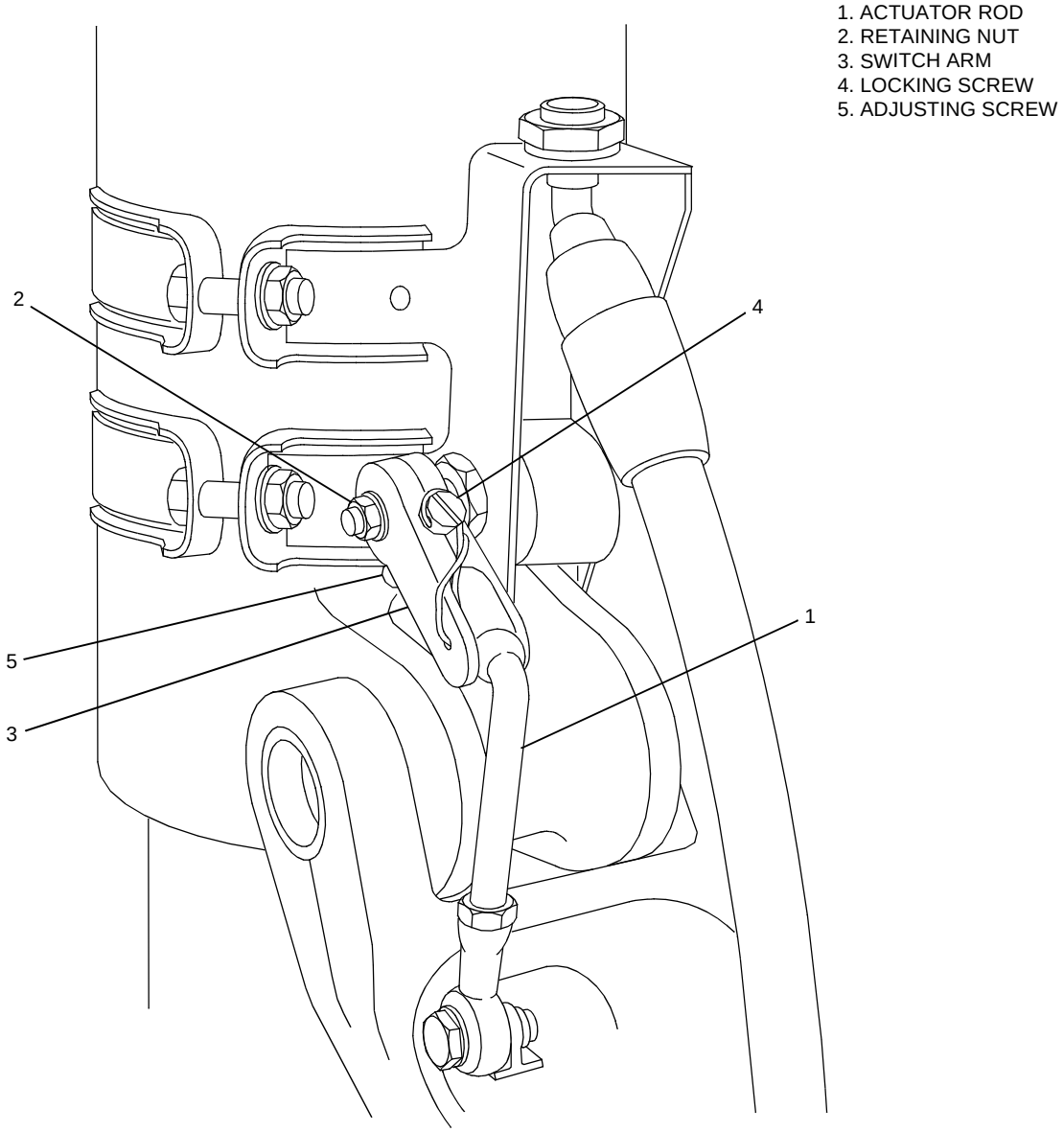

Figure 203
Main Landing Gear Safety Switch Adjustment

6. LANDING GEAR LIMIT SWITCH

A. Adjustment (Mechanical Landing Gear System Only)

CAUTION: *If it should become necessary to adjust the landing gear limit switches, extreme care should be taken to avoid adjustment which would allow overtravel of the landing gear and resultant damage to the system.*

To adjust the switches on U-1 through U-151, except U-149, and airplanes with Kit No. 99-8007-1 installed, position the jam nuts on the threaded shaft of the switch actuator in order to move the switch actuator in or out in relation to the actuator attach bracket. Extending the up (forward) limit switch actuator further through the bracket reduces up travel and backing the actuator out of the bracket increases up travel. A comparable situation exists on the down (aft) limit switch.

To adjust the switches on U-149, U-152 thru U-164 and prior airplanes which have Kit No. 99-8007-1 installed, loosen the upper bolt which secures the switch bracket to the gearbox casting, and adjust the switch in or out with the horizontal adjustment screw at the top of the bracket. Moving the up (forward) limit switch closer to the actuation cam reduces up travel and moving the switch away from the cam increases up travel of the landing gear. A comparable situation exists on the down limit (aft) switch.

7. WARNING HORN

A. Adjustment

A cam operated switch is connected to the linkage of each power lever in the pedestal. The switches are wired so that closing either or both levers beyond the safe flight setting with the gear retracted will sound the warning horn intermittently. The airplane must be in flight when determining the position of the power levers, which correspond to 79% N_1 as the ram air effect will alter the 79% N_1 position indicated during ground operation.

(1) With the airplane in flight, advance the power levers until 79% $\pm$ 2% N_1 (gas generator speed) is attained on each engine.

(2) Mark the position of the power levers on the pedestal with a piece of tape. Land the airplane.

(3) Remove the lower upholstery panels from both sides of the pedestal.

(4) Place the power levers in alignment with the tape on the pedestal.

(5) Adjust the landing gear warning horn switches by moving them in their slotted mounting brackets to actuate in this position (Ref. Figure 201, Chapter 61-21-00).

(6) Move the power levers and listen for an audible “click” when the power levers are moved aft of the tape.

(7) Remove the tape and install the pedestal upholstery panels.

(8) Flight test the airplane with the gear retracted and the power levers retarded until the warning horn sounds at an N_1 speed of 79% $\pm$ 2%.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 33 - LIGHTS

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LIGHTS
STROBE LIGHT SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

Pulsating strobe lights mounted adjacent to wing tip and tail lights greatly increase visibility of the airplane during night flight. A toggle switch on the lower LH instrument subpanel actuates the strobe light system. The strobe lights are powered by a master power assembly installed adjacent to the fuselage immediately forward of the main spar in the LH center section. This assembly consists of three power supply modules (one for each strobe light) and a timing circuit module mounted on a negatively ground subchassis. The timing circuit module prior to U-100 contains a small DC motor that rotates a notched cam to actuate two switches mounted 180° apart. Each time one of the switches is actuated, a relay in the affected power supply module doses to trigger its respective strobe light. A transistorized unit functions as a timing circuit on airplane with serials of U-100 and After. A transistorized circuit in the power supply unit steps up voltage of the airplane electrical system to the level (450 volts) required to operate the strobe light. The stepped-up voltage is stored in a condenser until released to the strobe light by the timer. The current from the power supply assembly is conducted to the flashtube of the strobe light by a specially shielded power cable. A charge of high voltage electricity is momentarily released to a coil in the flashtube assembly. The coil steps up the charge until it ionizes the xenon gas in the flashtube. The high voltage stored in the condenser then surges through the gas to produce the brilliant burst of energy that characterizes the strobe light.

LIGHTS
STROBE LIGHT SYSTEM
TROUBLESHOOTING

1. STROBE LIGHT POWER SUPPLY (U-100 THRU U-151)

A. Module Troubleshooting

CAUTION: *Do not operate the unit without a light attached. Use extreme caution working around the opened unit. Capacitors will remain charged to 600 volts for approximately 15 minutes after power is removed. Do not exceed 5 minutes continuous operation without light flashing.*

(1) Connect a light assembly to the Cannon Connector S-2.

(2) Apply appropriate DC voltage to pin 1 (-) and pin 3 (+) of the Jones connector P-1 located on the bottom of the module. Referring to Figure 101, wire two 10K resistors across pins 1 and 3, tying the junction to pin 3. The light should flash when pin 2 (the center pin) is intermittently shorted to either pin 1 or pin 3. If this test proves OK, the module is good and may be returned to service. If not, proceed to Step (2) in the CASE DISASSEMBLY procedure.

B. Case Disassembly

(1) Place the case on its top, remove the four bottom screws, and lift away the bottom.

(2) Reverse the case, remove the four top screws, and lift away top assembly.

(3) Remove the circuit breaker retainer ring from the front.

(4) Remove the four screws from the MS connector, and pull the connector from the case.

(5) Move the connector wires to one side and slide the case halves apart.

The assembly may be laid out for troubleshooting. Carefully separate all exposed components before operating the unit. Refer to Figure 101 for bench hook-up instructions for troubleshooting.

C. Power Pack Troubleshooting

(1) Connect a voltmeter (1000 volt range) to pins A (+) and C (-) of S-2. This can be done at the red and black wires at the board connection. Apply input power. The meter should read 500 to 700 volts DC. If this reading is present, the power pack is good and the next three Steps may be ignored. If no reading exists, proceed to Step (2).

(2) Unsolder the three pigtails removing the power pack assembly. With the unit removed apply voltage to the green or yellow lead (+) and the black (-) lead. Measure the output between the black (-) and the red (+) wires. The no-load voltage should be approximately 900 volts. If there is no reading, replace or repair the power pack assembly.

(3) The power pack assembly consists of a simple, single transistor, DC to DC converter. The diodes, resistors and transistor can be checked with an ohmmeter. If the large silicone transistor is defective, replace only with a Hoskins PM 61-1015. This is an RCA No. 40363, selected for gain. Substitutes will usually not be satisfactory. If unit does not oscillate, check for shorted diodes SC-3 or SC-4.

(4) If the power pack operated unloaded, but not when connected, the output capacitors C-1 and C-2 are shorted or a short circuit exists on the timer board. Check or replace C-1 and C-2. (Use only Hoskins P/N 61-1000- 401 special duty capacitor). If the short persists, carefully check the printed board and terminals for loose solder, etc.

(5) Once the power pack is operating, check out the timer and trigger circuits by operating the unit as described previously. If the light does not flash, proceed to the trigger circuit troubleshooting procedure.

D. Trigger Circuit Troubleshooting

(1) Measure SCR anode voltage at the junction of R-2 (100K) and C-5 (.22 μ 400V). This point is also the anode of SC-1 (the mounting terminal) and should measure 250 to 300 volts DC with respect to ground (T-2), dropping to zero when a trigger pulse is applied. If this voltage is 0, replace SC-2 and R-2 and repeat the measurement.

NOTE: When SC-1 fails it will often cause R-2 to open.

(2) If the above measurement is correct, but no switching action is observed, measure the control gage of SC-1 (junction of C-4 and R-5) with an oscilloscope. This should appear as a pulse of approximately 2 volts peak. Other components may be checked and replaced.

E. Subchassis/Timer Troubleshooting

(1) The metal subchassis is the power distribution center. The following Steps can be performed to determine proper operation: With the modules removed, apply power to the input wire and measure each pin of the three Jones plugs.

Pin 1 (the pin turned lengthwise) = Gnd

Pin 2 (center) - alternating + 10V (approximately) and OV.

Pin 3 = 28 volts input voltage

If all measurements are correct, except pin 2, replace the timer board.

(2) The timer may be observed for proper operation with a dual trace oscilloscope. Each output should be a square wave (opposite in phase) of approximately 10 volts peak to peak, alternating at a rate of 50 times per minute.

(3) If one or both outputs of the timer are not correct, troubleshoot the timer as follows:

(a) Observe that the unijunction is operating by placing an oscilloscope lead to the junction of C-2 and C-3. If the unijunction does not operate, check for approximately 12 volts across CR-1. Repair the unijunction circuit by replacing TR-1 or related components. Usually, a defective IC will not prevent the unijunction from operating. The correct oscilloscope indication should be a pulse height of 1 to 2 volts at a rate of 100 pulses/ minute.

(b) When the unijunction is operating the IC flip flop can be checked by placing the oscilloscope at the IC end of R-6 and R-10 If no signal exists, try unsoldering R-10 and lifting it intermittently. A shorted driver or predriver will occasionally load down the circuit.

NOTE: The IC flip flop is extremely reliable and rugged. Carefully check out the drivers (TR-4 and TR-5) and predrivers (TR-2 and TR-3) before suspecting the IC itself.

(c) If the driver transistors are suspected, unsolder the base of TR-4 and TR-5 (the junction of TR-4 and R-11 or TR-5 and R-12) and measure the voltage swing at the point removed. This should be several volts. The drivers or predrivers should be replaced as needed.

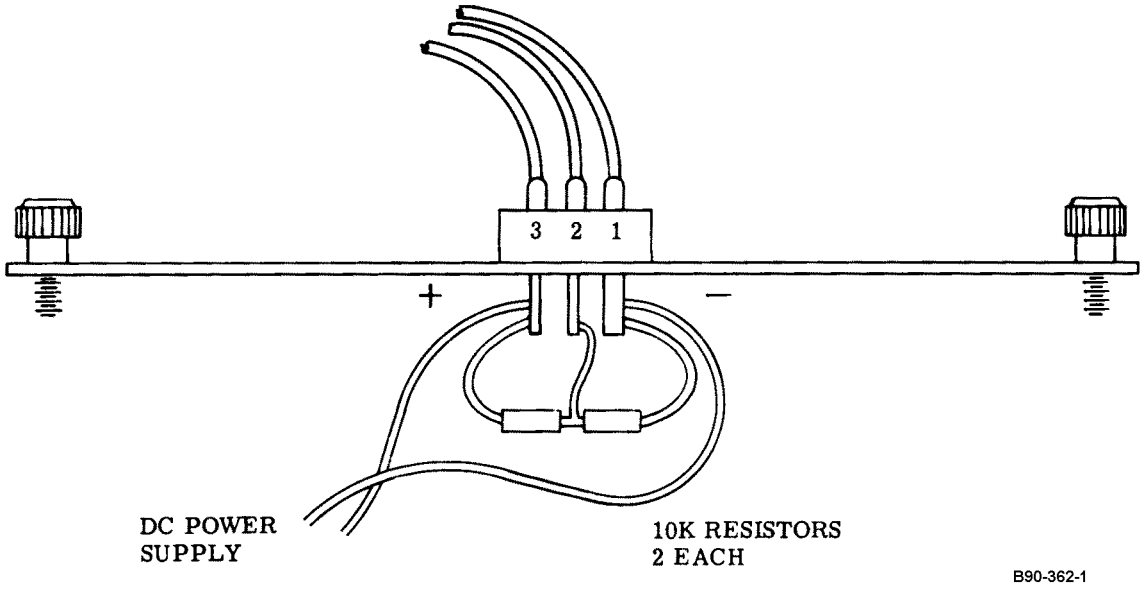


Figure 101
Test for Troubleshooting

2. TROUBLESHOOTING STROBE LIGHT POWER SUPPLY (U-152 AND AFTER)

For troubleshooting of the power supply, refer to Grimes Overhaul Manual with Illustrated Parts List (P/N 33-40-58)

LIGHTS
STROBE LIGHT SYSTEM
MAINTENANCE PRACTICES

1. POWER CABLE

A. Check

(1) Check that the individual conductors of the power cable are soldered to the connector pins as follows: On the Hoskins system the red wire goes to the pin "A", the white wire (or striped wire of red and white) to pin "B", and the black wire (or striped wire of red and black) to the pin "C". On the Grimes system the red wire goes to pin "A", the blue wire to pin "B", and the yellow wire to pin "C". Prior to U-165 the shield wire is grounded to the airframe at the power supply. At U-165 and After the shield wire is grounded to the connector plug at each end.

CAUTION: *Failure to hook up the conductors and pins in this manner will result in extensive damage to both the light units and the power supply modules.*

(2) Using a 500 volt megger, check that the power cable resistance between the connector pins and between all the pins and ground (the cannon plug) is 15 megohms or greater.

(3) Check the condition of the potting for the power supply cable at the terminals and connector clamp cavities. The potting compound used to protect the system against moisture is RTV No. 102 silicon rubber.

(4) Make sure the cable clamps are not tight enough to cause a short in the cable.

2. SYSTEM

A. Check-out

The following check is recommended when the flash tube of a strobe light unit fails to fire:

(1) Check that the flash tube is not broken and that the connectors are tight.

(2) Disconnect the power cable from the inoperative light at the power supply module.

NOTE: To avoid, the chance of shock through contact with the cable at the light fixture or with connector pins "A" and "B" at the power supply module, short out these pins to pin "C" (ground) to dissipate any residual charge left in the condenser after the system has been turned off.

Disconnect the power cable from one of the functioning lights and plug it into the power supply unit of the inoperative light. If the flash tube of the good light then fails to fire, the power supply unit has either failed or has a blown fuse (prior to serials U-100) or tripped a circuit breaker (U-100 and After).

CAUTION: *Never place the power supply unit of the functioning light circuit into the circuit of the inoperative light, for a short in the defective circuit would then damage the functioning power supply unit.*

On airplane prior to U-100, a short in the power cable will normally blow a fuse in the power supply unit. Replace defective fuses only with those that have a three ampere rating.

CAUTION: *Fuses with a higher capacity may permit operation of the power supply unit despite a short in the system, to the ultimate damage of the unit and related components.*

On airplane with serials U-100 and After, a short in the power cable will normally trip the circuit breaker in the power supply unit. The most likely reasons for malfunctions of this unit are shorts in either the power cable or the lamp assembly, shorts caused by contact of the transistor case with a foreign object during operation of the unit, moisture in the connectors, and the buildup of excessive heat within the unit due to inadequate ventilation.

3. POWER SUPPLY UNIT (PRIOR TO U-100)

A. Checks

The most likely reasons for malfunctions of this unit are shorts in either the power cable or the lamp assembly, shorts caused by contact of the transistor case with a foreign object during operation of the unit, moisture in the connectors, and the buildup of excessive heat within the unit due to inadequate ventilation. When the system is operating properly, the action of the relays in each power supply unit can be heard. If the sound of these relays closing at the pulse rate of once per second cannot be heard, perform the timer check to ascertain that the cam is actuating the two switches properly. After 500 hours and 1000 hours of airplane operation, perform the following check:

(1) Open each power supply module and check the relay contacts for pitting. Replace those that are excessively pitted and clean the others.

(2) Perform a capacitance check on the condensers by charging them to 450 volts DC and checking for leakage. Replace condensers whose leakage exceeds 1.5 milliamperes only with condensers obtained from the vendor (Hoskin Inc., 34 N. Bennett, Geneva, Illinois) or his authorized dealers. It should be noted that the foregoing capacitance check can be performed without removing the condensers from the power supply module.

CAUTION: *To prevent short circuiting the system, avoid contact with the exposed transistors on the ends of the power supply modules.*

4. TIMER MODULE (PRIOR TO U-100)

A. Check

Remove the cover from the module and slowly rotate the motor mechanism so that the notched timing wheel between the two switches moves under the switch cam riders. A click should be heard as the rider touches bottom and another click as it reaches the top. If necessary, loosen the top screw on the switch mounting plate and slide it in or out until the switches respond properly.

5. POWER SUPPLY UNIT

A. Removal

Remove the panel from the underside of the wing center section on the left side to gain access to the power supply unit. This unit consists of three power supply modules (one for each light) and a timer mounted in the subchassis. The entire unit may be removed by disconnecting the wiring from the power supply and removing the screws anchoring the subchassis to the support structure.

CAUTION: *If the exposed transistors on the end of each power supply unit are contacted during removal, the resultant damage will cause them to short out.*

B. Installation

Place the unit through the access opening on the under side of the wing center section on the left side. Secure the subchassis to the support structure with the anchoring screws. Connect the wiring from the power supply. Secure the access panel in place.

CAUTION: *If the exposed transistors on the end of each power supply unit are contacted during installation, the resultant damage will cause them to short out. Make sure the unit is hooked up as indicated in the wiring diagram, or it will be permanently damaged by reversed polarity.*

LIGHTS
ANNUNCIATOR SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

The annunciator panel system consists of several single channel circuits which are divided into fault warning and indicating channels. The fault warning circuits (FUEL PRESSURE, GENERATOR etc.) feed into a single control point to operate the fault warning lights as well as the dimming control override The indicating circuits (FUEL CROSSFEED, ALT WARN etc.) feed into a single control blocking diode, the signal controls only the dimming control override.

When a fault warning signal is sent to the annunciator circuit, it is used to illuminate its respective readout in the annunciator panel. It also triggers the fault warning light and shunts out the dimming control circuit, causing the readout to illuminate with full brightness.

Pushing the fault warning light switch will extinguish the fault warning light and return the readout to the brightness set by the dimming control. The indicating readout is also reset to its desired dimming level by pushing the fault warning light switch, even though the warning light is not illuminated. The annunciator system also contains a press-to-test circuit which is used in testing the bulbs in each readout, the fault warning light, and the dimming control shunting circuit.

The dimming control circuit consists of a potentiometer, which varies the conduction of a 5-ampere, PNP-type transistor that provides a common ground to all readout lights. The dimming control is shunted by grounding the base of the transistor, thus causing the transistor to go into full conduction. The maximum probable load of the transistor under actual conditions is 0.32 amperes. The dimming control is designed with a minimum dimming level which effectively reduces glare especially during night operation. The annunciator dimming circuit is inoperative when the bus voltage is at the static battery voltage. After an engine has been started and either or both generators are operating to bring the bus voltage to the regulated voltage (28.25 vdc), the annunciator dimming is operational.

The annunciator system is designed so no single fault will completely render the system inoperative. An open circuit condition to the dimming control circuit will result in complete loss of the readout lights; however, the fault warning light remains operative.

LIGHTS
ANNUNCIATOR SYSTEM
TROUBLESHOOTING

1. ANNUNCIATOR

Troubleshooting - Annunciator System

INDICATION	PROBABLE CAUSE	REMARKS
1. Fault sensor circuit breaker blown. Respective fault sense lost.	a. Short in fault or indicating sensor circuit.	
2. Loss of fault indication readout function upon test.	a. Open in fault or indicating sensor circuit.	
3. Annunciator circuit breaker blows. Loss of fault warning light, dimming control and press-to-test circuit. Readout remains operative.	a. Short between annunciator circuit breaker and annunciator box.	
4. Same as above except circuit breaker does not blow.	a. Open between annunciator circuit breaker and annunciator box.	
5. Inability to extinguish fault warning light or reset dimming control.	a. Reset switch open.	
6. Illumination of readout only. Loss of fault warning light and dimming control shunting circuit. Press-to-test circuit operative.	a. Reset switches shorted (to ground).	
7. Loss of fault warning light during test and fault input.	a. Fault warning light circuit open.	
8. Annunciator circuit breaker blown. Loss of fault warning light dimming control and press-to-test circuit.	a. Fault warning light circuit shorted.	
9. Loss of respective indication during test and signal input.	a. Readout light circuit open.	
10. Fault sensor circuit breaker blown. Loss of respective indication during test and signal input.	a. Readout light circuit shorts.	

Troubleshooting - Annunciator System

INDICATION	PROBABLE CAUSE	REMARKS
11. Annunciator circuit breaker blown. Loss of fault warning light and press-to-test circuit. Readout light illuminates to full bright.	a. Dimming control shorted (to ground).	
12. Loss of all readout lights during press-to-test circuit and fault signal input. Fault warning light operative during press-to-test and dimming control.	a. Dimming control open.	
13. Annunciator circuit breaker blown. Loss of fault warning light and dimming control. Readout light illuminates to full bright.	a. Press-to-test circuit shorted.	
14. Loss of press-to-test circuit only. All other circuits operate properly.	a. Press-to-test circuit open.	

LIGHTS
ANNUNCIATOR SYSTEM
MAINTENANCE PRACTICES

1. ANNUNCIATOR SYSTEM

When it becomes necessary to replace the Korry lamps in the instrument lighting system, observe the following:

(1) Turn off or disconnect electrical power from the system on which lamps are be changed.

(2) Pull out on the cap assembly using fingers only. Damage to the cap can occur if tools, such as pliers or channel locks, are used to remove the cap.

(3) Extract the lamps from the unit using a finger nail or small screwdriver blade. It is recommended that all lamps in a given cap be replaced at the same time, regardless of the number of lamps burned out. This will ensure that all lamps in the cap have approximately the same service life in the future.

NOTE: Be certain that the correct replacement lamp is used. The lamp number can usually be obtained from the base of the lamp itself, or check the service manual for the airplane. Visually inspect the new lamp for quality manufacture. Many lamps are poorly made at the factory, and should be discarded if the glass envelope is distorted or not aligned with the lamp base.

(4) To reinsert the cap assembly into the base of the unit, it is necessary to "cock" the cap prior to insertion into the base assembly. This is done by pulling the two halves of the cap assembly apart until the unit snaps into a locked, extended position. This usually has already occurred when the cap was removed from the base assembly, however, it can snap back to the closed position when the lamps are being replaced.

(5) With the cap ready to install, check for proper orientation by looking into the base assembly for any indexing lugs, and match them to the index grooves intended into the base of the cap assembly. Install the cap with a firm action. The cap should snap into the base and be positively locked into position. Check the unit for proper operation and restore power to the system.

LIGHTS
FLIGHT COMPARTMENT
MAINTENANCE PRACTICES

1. BULBS

Bulb Replacement Guide

LOCATION	BULB NUMBER
Air Conditioner Reset Switch Light (U-50, U-165 and After)	327
Annunciator Panel Light	327
Compass Light	327
Engine Fire Warning Light	327
Fire Extinguisher Pull Handle Light (U-50, U-165 and After)	327
Fuel Panel Circuit Board Light (Red) (U-1 thru U-164, except U-50)	D158-100-4T1
Fuel Panel Circuit Board Light (White) (U-1 thru U-164, except U-50)	D158-100-5T1
Fuel Panel Circuit Board Light (Blue-White) (U-50, U-165 and After)	DS158-100-5
Instrument Indirect Light (Red) (U-1 thru U-164, except U-50)	1864R
Instrument Indirect Light (White)	1864
Instrument Overhead Light	327
Landing Gear Control Knob Light	327
Landing Gear Indicator Light	327
Master Caution Light (U-50, U-165 and After)	387
Master Waming Light (U-50, U-165 and After)	387
Map Light (Pilots and Copilots)	1495
Map Overhead Light (Red) (U-1 thru U-164, except U-50)	303R
Map Overhead Light (White)	303
Outside Air Temperature Light	334
Overhead Floodlight (U-50, U-165 and After)	303
Overhead Light Panel Lights (Red) (U-1 thru U-164, except U-50)	D158-100-4T1
Overhead Light Panel Light (White) (U-1, U-164, except U-50)	D158-100-5T1
Overhead Light Panel Light (Blue-White) (U-50, U-165 and After)	DS158-100-5
Pedestal Edge Light (Red) (U-1 thru U-164, except U-50)	D158-100-4T1
Pedestal Edge Light (White) (U-1 thru U-164, except U-50)	D158-100-5T1
Pedestal Edge Light (Blue-White) (U-50, U-165 and After)	DS158-100-5
Post Light	327
Subpanel Edge Light (Red) (U-1 thru U-164, except U-50)	D158-100-4T1

Bulb Replacement Guide (Continued)

LOCATION	BULB NUMBER
Subpanel Edge Light (White) (U-1 thru U-164, except U-50)	D158-100-5T1
Subpanel Edge Light (Blue-White) (U-50, U-165 and After)	DS158-100-5

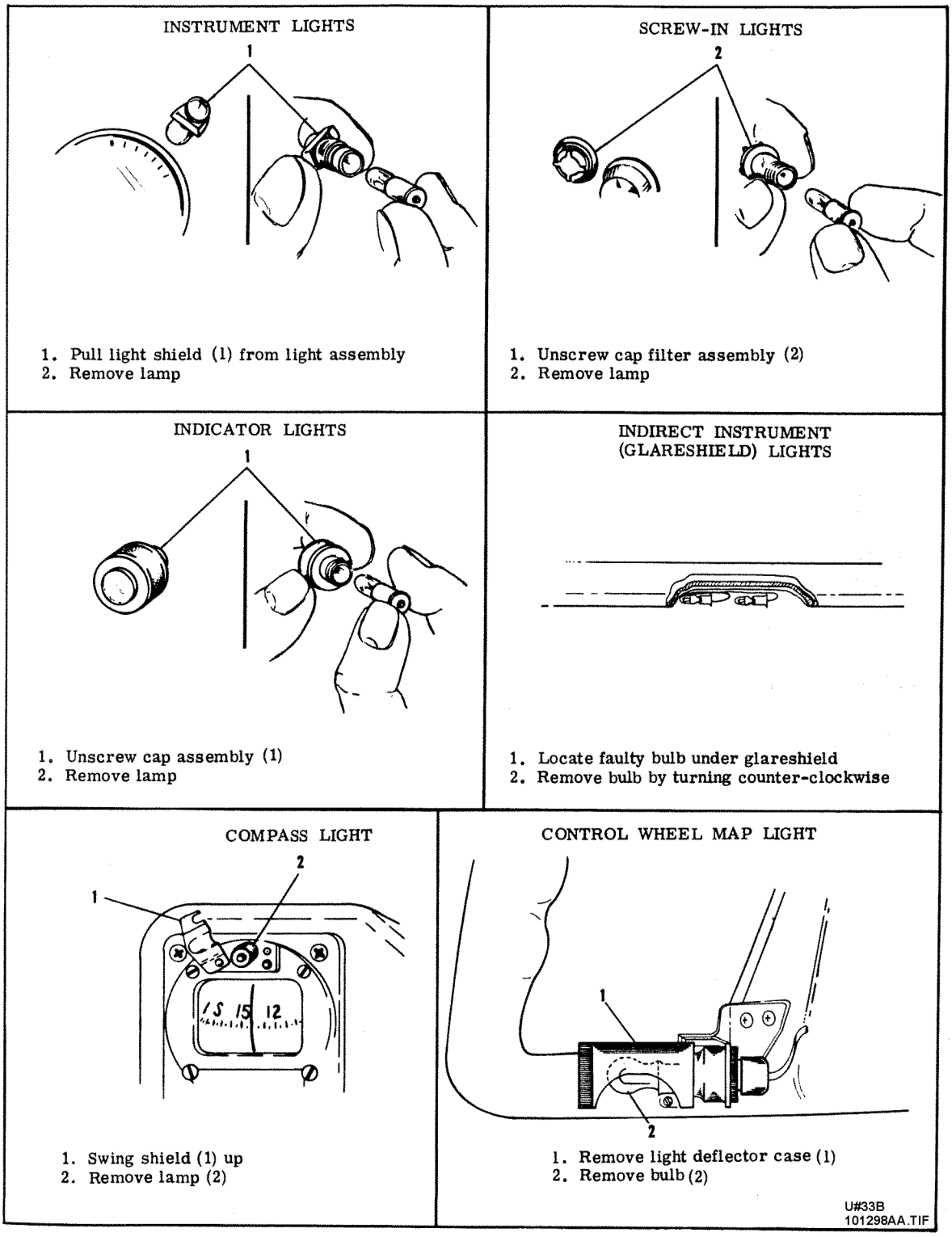


Figure 201 (Sheet 1 of 2)
Bulb Replacement Guide

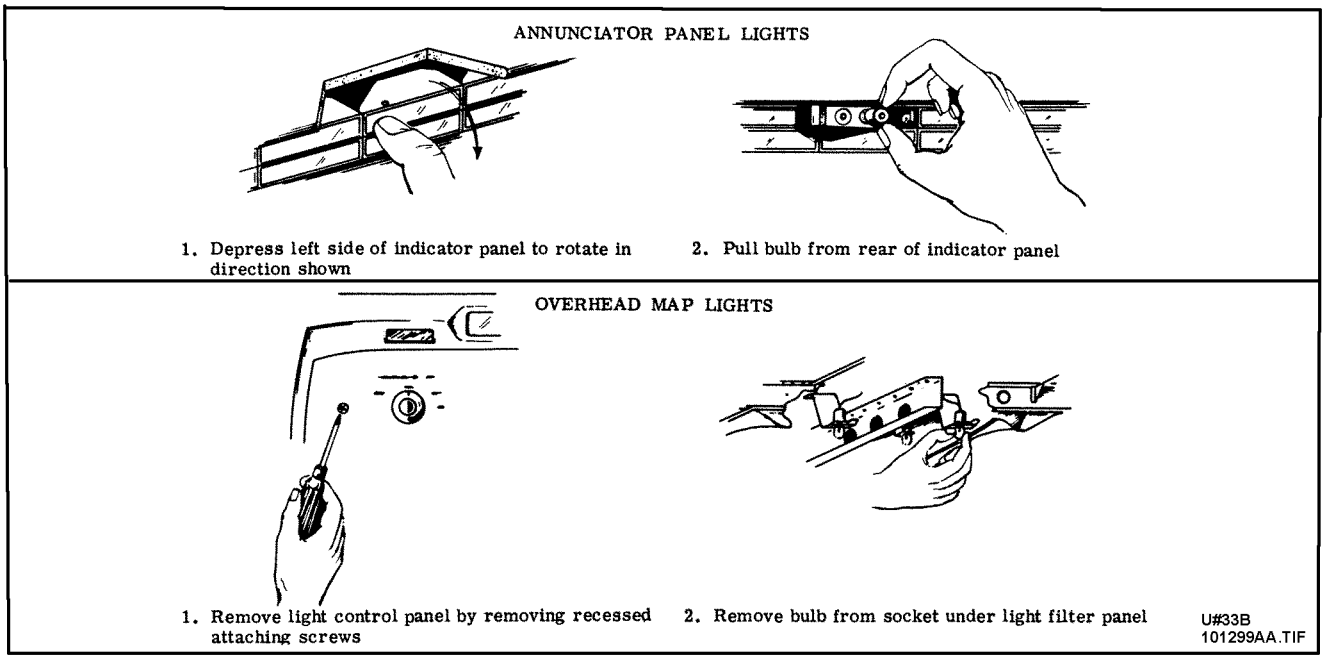


Figure 201 (Sheet 2 of 2)
Bulb Replacement Guide

LIGHTS
LANDING GEAR INDICATOR LIGHTS
MAINTENANCE PRACTICES

1. LANDING GEAR DOWNLOCK LIGHTS LAMP

A. Replacement

WARNING: When replacing lamps, never move the landing gear handle from the extended (down) position.

(1) Remove electrical power from the airplane.

(2) Pull the annunciator module from the annunciator assembly (Ref. Figure 201).

CAUTION: *Any time this annunciator is removed, perform LANDING GEAR DOWNLOCK LIGHT ANNUNCIATOR FUNCTIONAL CHECK.*

(3) Remove and replace the defective lamp (Ref. Table 1, 33-10-00).

(4) Ensure proper orientation of the annunciator module and press it firmly into the annunciator assembly.

WARNING: Verify the landing gear switch handle is in the extended (down) position before restoring electrical power to the airplane.

(5) Apply power to the airplane and verify operation of the annunciator.

2. LANDING GEAR DOWNLOCK LIGHT ANNUNCIATOR

A. FUNCTIONAL CHECK

WARNING: When replacing lamps, never move the landing gear handle from the extended (down) position.

(1) Remove electrical power from the airplane.

(2) Pull the annunciator module from the annunciator assembly (Ref. Figure 201).

(3) Remove one bulb at a time and shine a flashlight from the back of the annunciator forward.

(4) Ensure that the appropriate portion of the annunciator illuminates at the location where the bulb is removed.

NOTE: The annunciator has dividers installed to block the light between the LH and the RH portion of the annunciator. The nose location, either bulb, will light the entire NOSE section.

(5) Continue checking until all of the removed bulb locations are checked.

(6) Install all bulbs and install annunciator assembly.

(7) Restore power to the airplane and verify all bulbs are illuminated.

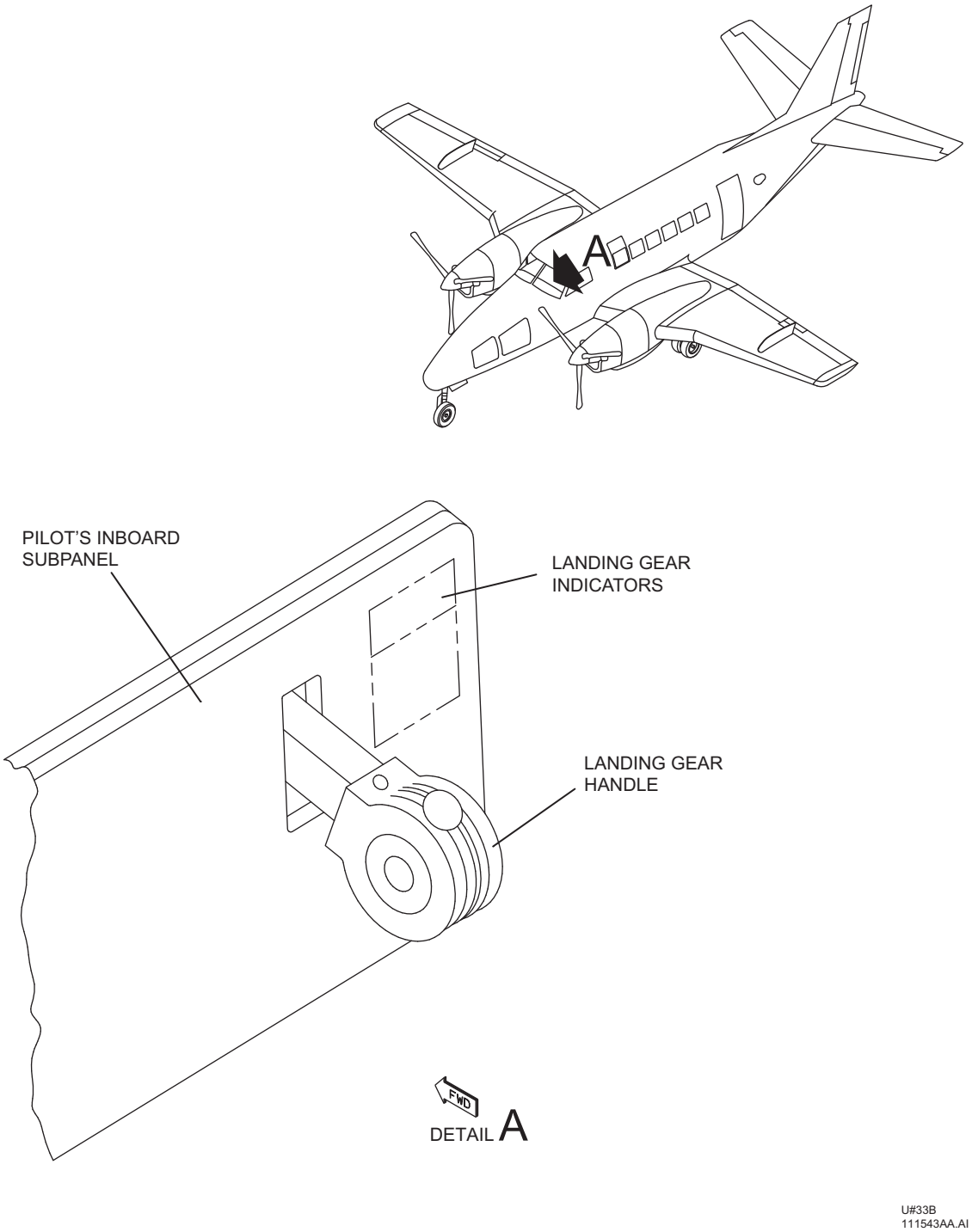


Figure 201
Landing Gear Downlock Annunciator Location

LIGHTS
EDGELIGHTED PANELS
MAINTENANCE PRACTICES

1. EDGELIGHTED PANEL

A. Assemblies

The edgelighted panel assemblies consist of an outer plastic panel and a printed circuit board to which wheat lights are soldered. These panel assemblies are mounted strategically throughout the crew compartment to identify the function and mode af the various switches and systems. The inscriptions on the outer, counter bored, plastic panel of the edgelighted assemblies are illuminated by wheat light bulbs soldered in parallel to a printed circuit board and powered by 28 vdc.

Due to the fact that edgelighted panels can be damaged by inappropriate maintenance practices and contamination, usually moisture resulting from spills or storm windows left open in moist environments, they must be removed and visually inspected after every 1200 hours of operation or every four years (whichever occurs first).

B. Inspection

(1) Disconnect the airplane power.

(2) Remove any switch or rheostat knobs that may interfere with the removal of the panel assemblies.

(3) Remove the attaching screws and lift the edgelighted panel away from the structure

NOTE: Exercise care not to scratch or damage the panel assemblies.

(4) Inspect the outer, plastic panel for cracking, scratches, etc. If damaged, the plastic panel must be replaced. Refer to the appropriate Model 99 Parts Catalog for the component part numbers.

(5) Inspect the printed circuit board for delamination, cracks, abrasions, arcing, and discoloration. If any damage of the type noted in this Step is found, the PCB must be replaced.

(6) Utilizing a regulated 28-vdc power source, check the wheat lights to assure proper illumination of all the bulbs. All bulbs that fail to illuminate properly must be replaced and the PCB cleaned and recoated as outlined in PRINTED CIRCUIT BOARD REPAIR PROCEDURES.

(7) If no damage is found, no further action is required and the panel may be replaced.

2. PRINTED CIRCUIT BOARD

A. Repair Procedure

Because of the bonded type construction of the board, repair of the PCB assembly is limited to replacement or repair of the wheat light bulbs. The following Steps should be observed when replacement or repair of a bulb (or bulbs) becomes necessary.

(1) Utilizing a low wattage soldering iron, heat the solder holding each of the bulb leads to the PCB. As the solder begins to soften, lift the lead away from its mounting pad.

CAUTION: *The bulb mounting pads and lands may lift away from the board if excessive heat is applied to the solder attaching the bulb leads to the mounting pads. If this occurs, the PCB must be replaced.*

(2) Solder the new wheat light bulb to the appropriate pads on the PCB. Do not apply excessive solder to the assembly and ensure that each connection has a good fillet between the pad and the lead. The contour of each bulb lead should be visible in the solder.

(3) After completing repairs check that all bulbs match the counterbores in the outer, plastic panel.

(4) Connect a regulated 28-vdc power source to the PCB and verify that all bulbs illuminate. When normal panel function is attained, the panel must be cleaned and a conformal coating applied before final assembly.

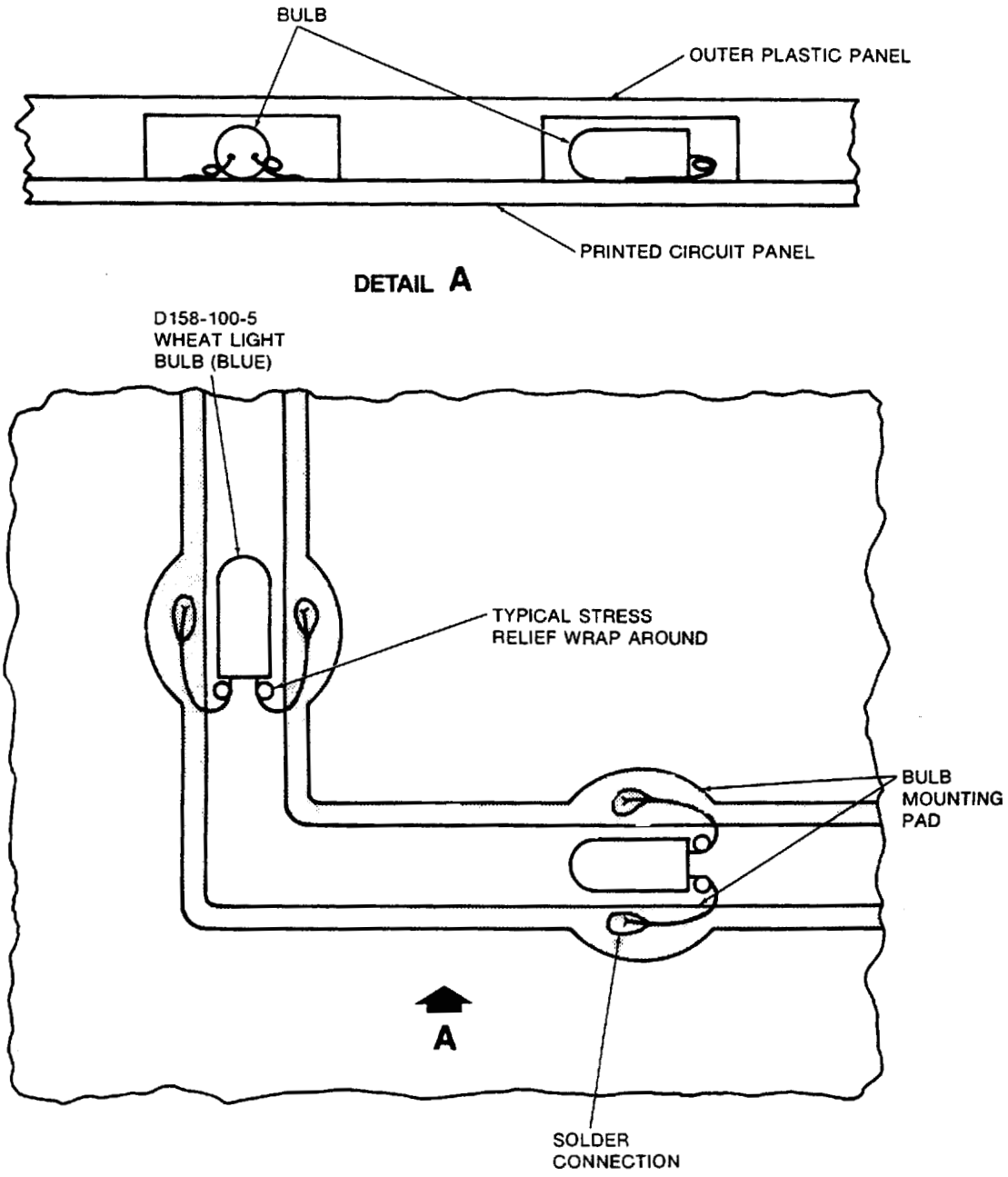


Figure 201
Bulb Replacement Guide

B. Cleaning

(1) Immerse the entire PCB in a container of isopropyl alcohol.

NOTE: If necessary to remove stubborn contaminants, the PCB may be lightly scrubbed with a soft bristled brush.

(2) Assure that all traces of flux and any other contaminants are washed away.

(3) Remove the PCB from the container and rinse it by pouring clean isopropyl alcohol over the panel assembly.

(4) Immediatey after rinsing, place the assembly in a vertical position (to draw off any residual moisture) and allow to dry.

NOTE: After cleaning, care must be exercised to prevent contamination of the PCB through handling or from the environment. All traces of oil, moisture or any other contaminants must be kept away from the PCB. Clean, lint-free, white gloves or clean rubber gloves must be worn at all times during and after the cleaning of a PCB assembly.

C. Conformal Coating

(1) As soon as possible after the PCB is fully dry, the conformal coating should be applied. The coating must completely cover the side of the PCB on which the light bulbs are installed.

NOTE: Hawker Beechcraft Corporation Recommends Conothane CE1155 or CE1164 (117 or 118, Table 6, 91-00-00) thinned 25% with Product S8 (119, Table 6, 91-00-00). CE1155 is a two part product and should be mixed in a ratio of 10 parts of component "A" to seven parts component "B", by weight. CE1155 is the preferred coating agent; however, any coating that conforms to MIL-I-46058 may be used.

(2) The coating may be applied by dipping, brush or spray. Whichever coating method is used, care must be taken to avoid getting any coating in the area of the electrical connector. After application, the coating must be cured as recommended in the manufacturers instructions.

(3) Inspect the coating for complete coverage, fisheyes, bubbles, dry spots, peeling, blisters, wrinkles, and cracks. If any coating defects are found, the panel must be recleaned and recoated.

(4) When the conformal coating has been satisfactorily applied and cured, the edgelighted panel may be reassembled and installed. Replace any knobs that were removed and reconnect the airplane power.

LIGHTS
PASSENGER COMPARTMENT
MAINTENANCE PRACTICES

1. BULBS - CABIN

Bulb Replacement Guide

LOCATION	BULB NUMBER
Aft Dome Light (U-1 thru U-164, except U-50)	MS25232-307
Airstair Door Step Light (U-50, U-165 and After)	1864
Aisle Light	1864
Cabin Door Warning Light	327
Cabin Door Observation Light (U-50, U-165 and After)	1864
Cabin Overhead Light	1864
Cabin Reading Light	1495
Dome Light (U-1 thru U-164, except U-50)	307
Dome Lights (U-50, U-165 and After)	GE1309
No Smoking and Fasten Seat Belt Light (U-1 thru U-164, except U-50)	303
No Smoking and Fasten Seat Belt Light (U-50, U-165 and After)	387

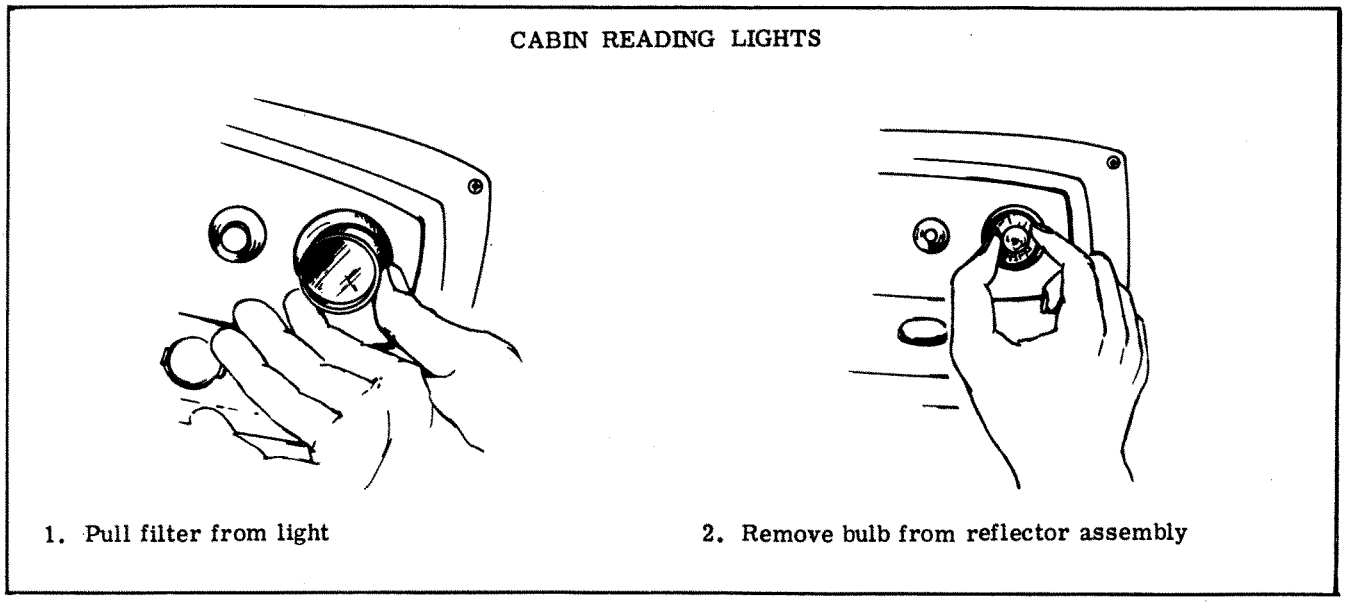


Figure 201
Bulb Replacement Guide

LIGHTS
CARGO AND SERVICE COMPARTMENT
MAINTENANCE PRACTICES

1. BULBS - BAGGAGE

Bulb Replacement Guide

LOCATION	BULB NUMBER
Forward and Aft Baggage Compartment Light	307
Baggage Pod Light (U-1 thru U-164, except U-50)	307
Baggage Pod Light (U-50. U-165 and After)	5108WW

LIGHTS
EXTERIOR
MAINTENANCE PRACTICES

1. BULBS - EXTERIOR

Bulb Replacement Guide

LOCATION	BULB NUMBER
Landing Light	4596
Navigation Light	A7512-24
Rotating Beacon Light	A7079B24
Strobe Light System (Grimes) Tail Light (Flashtube) Wing Light (Flashtube)	 55-0221-1 55-0221-1
Strobe Light System (Hoskins) Tail Light (Flashtube) Wing Light (Flashtube)	 31-0725-1 31-0725-1
Tail Flood Light (U-50, U-165 and After)	1982-SP
Tail Navigation Light	1683
Taxi Light	4587
Wing Ice Light	A7079A24 or A70798B24
Wing Navigation	A7512-24

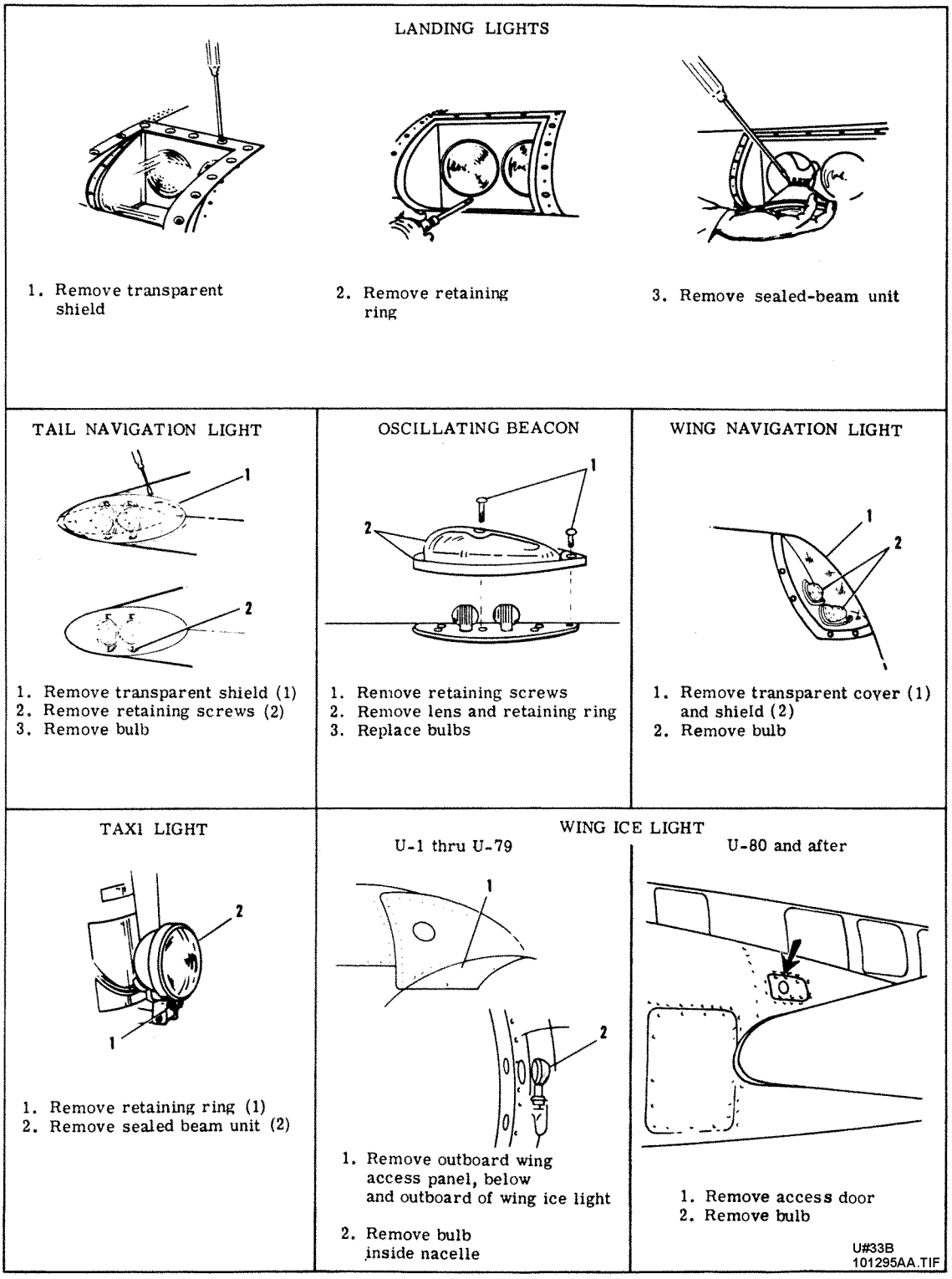
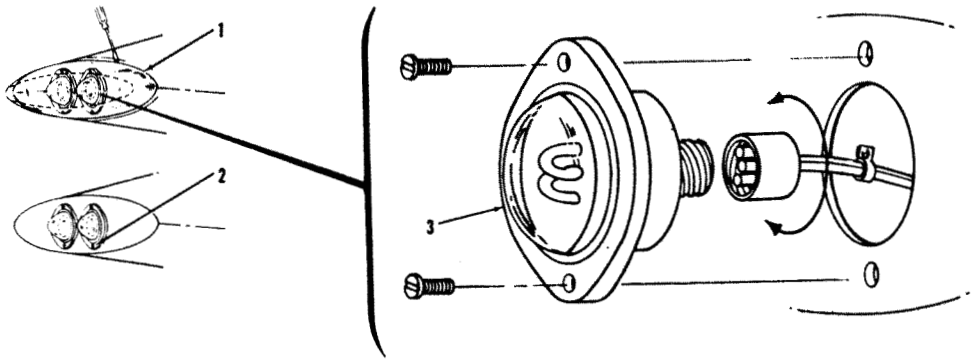


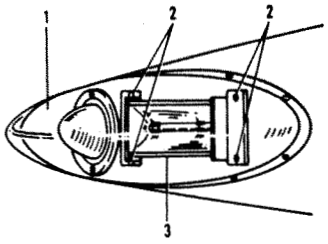
Figure 201 (Sheet 1 of 2)
Bulb Replacement Guide

TAIL STROBE LIGHT (HOSKINS)



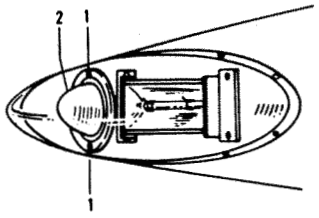
1. Remove transparent shield (1)
2. Remove retaining screws (2)
3. Remove unit (pull out) and disconnect cannon plug

TAIL STROBE LIGHT (GRIMES)



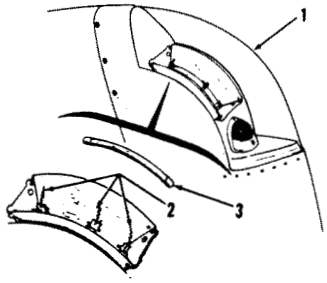
1. Remove transparent shield.
2. Remove four retaining screws and lift out strobe light assembly.
3. Unplug light assembly and replace with new light assembly.

TAIL NAVIGATION LIGHT



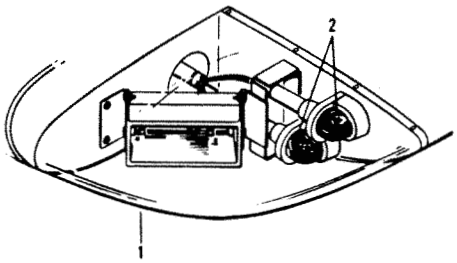
1. Remove retaining screws.
2. Remove bulb and replace with new bulb.

WING STROBE LIGHT (HOSKINS)



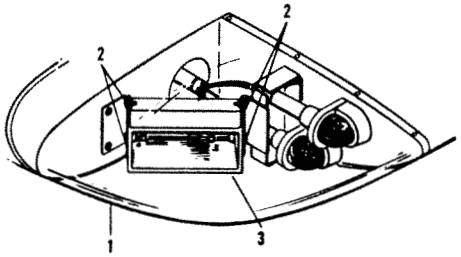
1. Remove transparent cover
2. Remove safety wire from flashtube clips (3)
3. Remove flashtube

WING NAVIGATION LIGHT



1. Remove transparent cover (1) and shield (2).
2. Remove bulb.

WING STROBE LIGHT (GRIMES)



1. Remove transparent shield.
2. Remove four retaining screws and lift out strobe light assembly.
3. Unplug light assembly and replace with new light assembly.

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Figure 201 (Sheet 2 of 2)
Bulb Replacement Guide

LIGHTS
EMERGENCY LIGHTING
MAINTENANCE PRACTICES

1. SELF-ILLUMINATED EXIT SIGNS

A. Brightness Check

The equipment used in this check consists of DB-45-B1 T-light Comparator (Ref. Figure 201). This equipment is available from Self-Powered Lighting Inc., 8 Westchester Plaza, Elmsford, New York 10523.

Self-illuminated exit signs can be inspected on the airplane under all types of lighting conditions without the need for removing the sign. Checking for brightness requires the use of a T-light Comparator. The test is performed as follows:

NOTE: T-light Comparators used for checking illuminated sign brightness must have had a newly calibrated reference source installed during the preceding 12 months for the brightness test to be considered valid.

(1) Place the comparator base plate flat on the surface of the sign and move it gently until the viewing aperture is over the portion of the sign to be inspected.

(2) Compare the brightness of the sign portion seen through the aperture and the calibrated source; if the sign is as bright or brighter than the calibrated source, the sign brightness is equivalent to, or better than the 100
microlamberts required. If the sign brightness is less than that of the comparator, the sign must be replaced.

(3) Move the comparator over all the illuminated portions of the sign to check minimum brightness at all points.

(4) If the sign at any point appears less bright than the comparator, the sign is due for replacement.

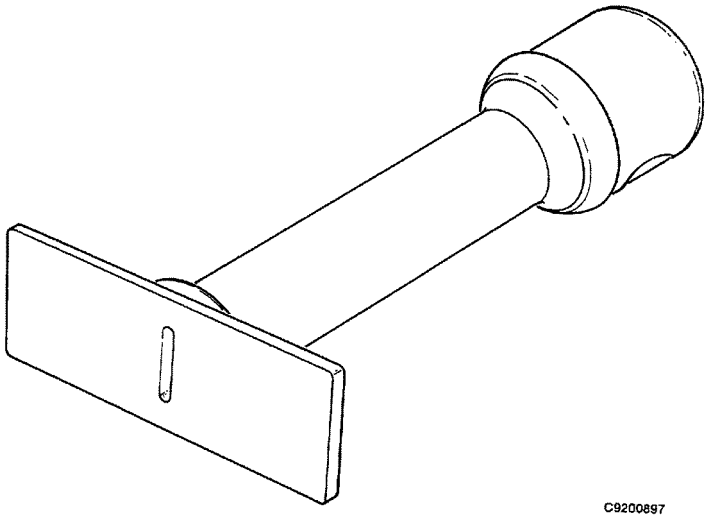

Figure 201
T-Light Comparator

List of Effective Pages

CH-SE-SU	PAGE	DATE
34-LOEP	1	Nov 1/10
34-CONTENTS	1	Nov 1/10
34-10-00	1 thru 6	Nov 1/10
	101	Nov 1/10
	201 thru 206	Nov 1/10

Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 34 - NAVIGATION

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Pitot and Static System 101
Maintenance Practices 201
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NAVIGATION
FLIGHT ENVIRONMENT DATA
DESCRIPTION AND OPERATION

1. GENERAL

A. Pitot and Static Pressure System

The pitot and static pressure system provides a source of impact air pressure and static air for the operation of the instruments. The pitot portion of the system is comprised of two electrically-heated masts located on the upper-fuselage nose. The impact air pressure entering the masts is transmitted through separate tubing to the dual airspeed indicators mounted on the instrument panel. A heating element located in each mast prevents the opening of the mast from becoming clogged with ice. The interconnecting tubing is sloped to permit moisture to settle at the lowest point in each line. The moisture in the lines is drained through a pair of drain pet cocks located directly inside and aft of the nose baggage compartment door on each side. These drain pet cocks should be checked and opened to allow the release of any trapped moisture following each flight in atmosphere containing precipitation or high humidity.

The static portion of the system includes two separate static ports or "buttons"' located on each side of the nose area immediately below and aft (below and forward on U-146, U-148 and After) of the nose baggage compartment door.

Airplane serials U-50, U-165 and After are equipped with an additional set of static buttons on each side of the nose area at FS 21.25.

NOTE: On airplane serials U-50, U-165 and after static air tubing is connected to the buttons at FS 21.25 when the baggage pod is installed. When the airplane is flown without the baggage pod, the static air tubing must be connected to the buttons at FS 33.25.

Each static button is located at the lowest point in the interconnecting lines; therefore, drain pet cocks are not required as each line has a built-in gravity drain through the button opening. Should the static air lines become blocked, an alternate static air source is provided by an alternate static air switch located on the lower left side of the cockpit at FS 107. On U-50, U-165 and After, the alternate static air switch is located on the instrument panel on the pilot's side, just to the left of the Airspeed Indicator and the Electric Turn and Slip Indicator. A spring clip holds the static air switch in the NORMAL (primary) air source position. On U-2 thru U-49, U-51 thru U-103, U-106 thru U-133 and U-135, the emergency static air line is connected to the main static air line at FS 84.00 by a tee. On later serials. the ALTERNATE (emergency) static air source is an open port on the back of the switch, vented to cabin air. Static air is used for the operation of the Rate-of-Climb Indicator, Altimeter, and the Airspeed Indicator.

B. Outside Air Temperature (OAT) Indicator

The outside air temperature indicator is installed in the overhead panel in the pilot's compartment. The thermometer dial is on the inside of the compartment with the stem of the instrument protruding through the skin of the airplane to the outside air. The unit is hermetically sealed against dust and moisture.

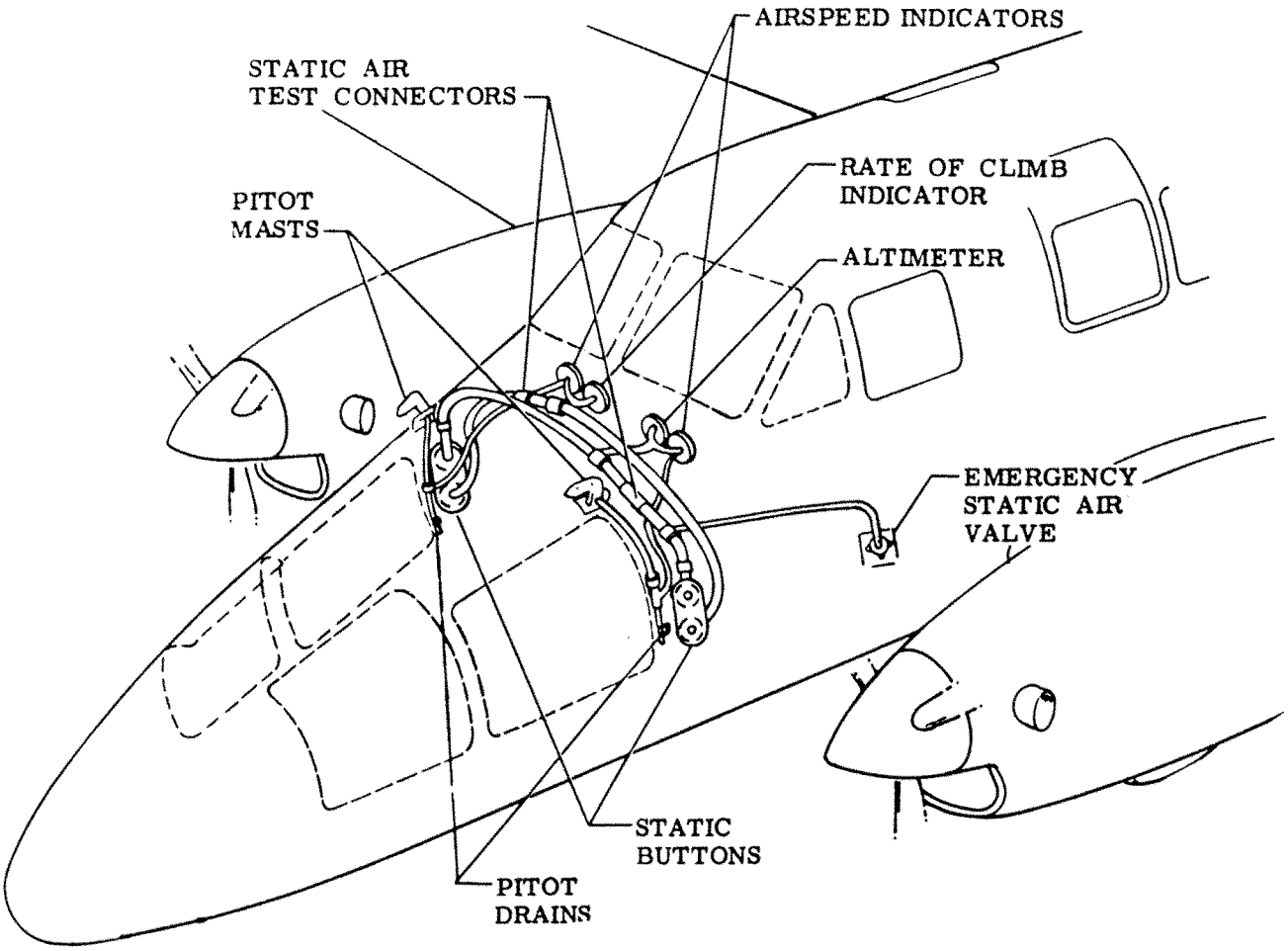


Figure 1
Pitot and Static System (U-2 thru U-49, U-51 thru U-103, U-106 thru U-133 and U-135)

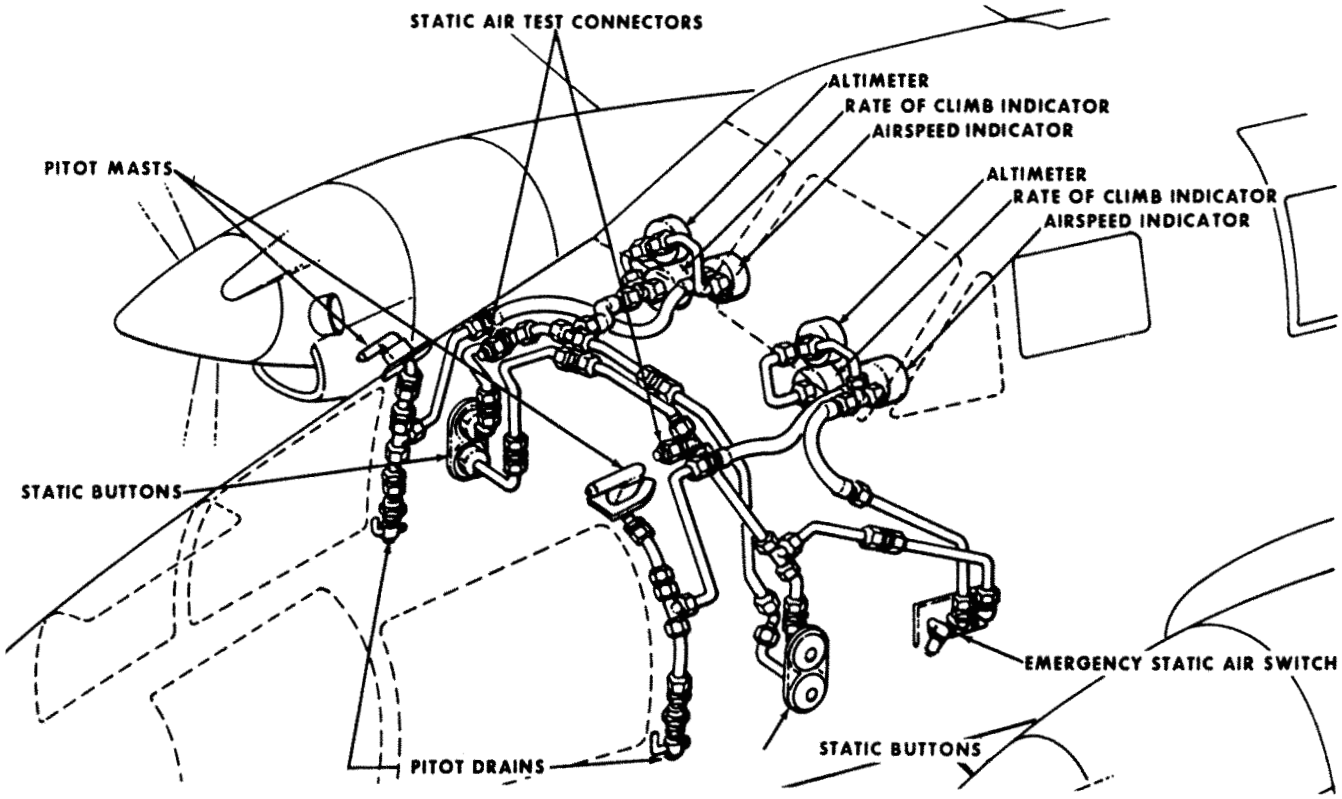


Figure 2
Pitot and Static System (U-1, U-104, U-105, U134, U-136 thru U-145, U-147 and Prior Airplanes Installing Kit No. 99-5005)

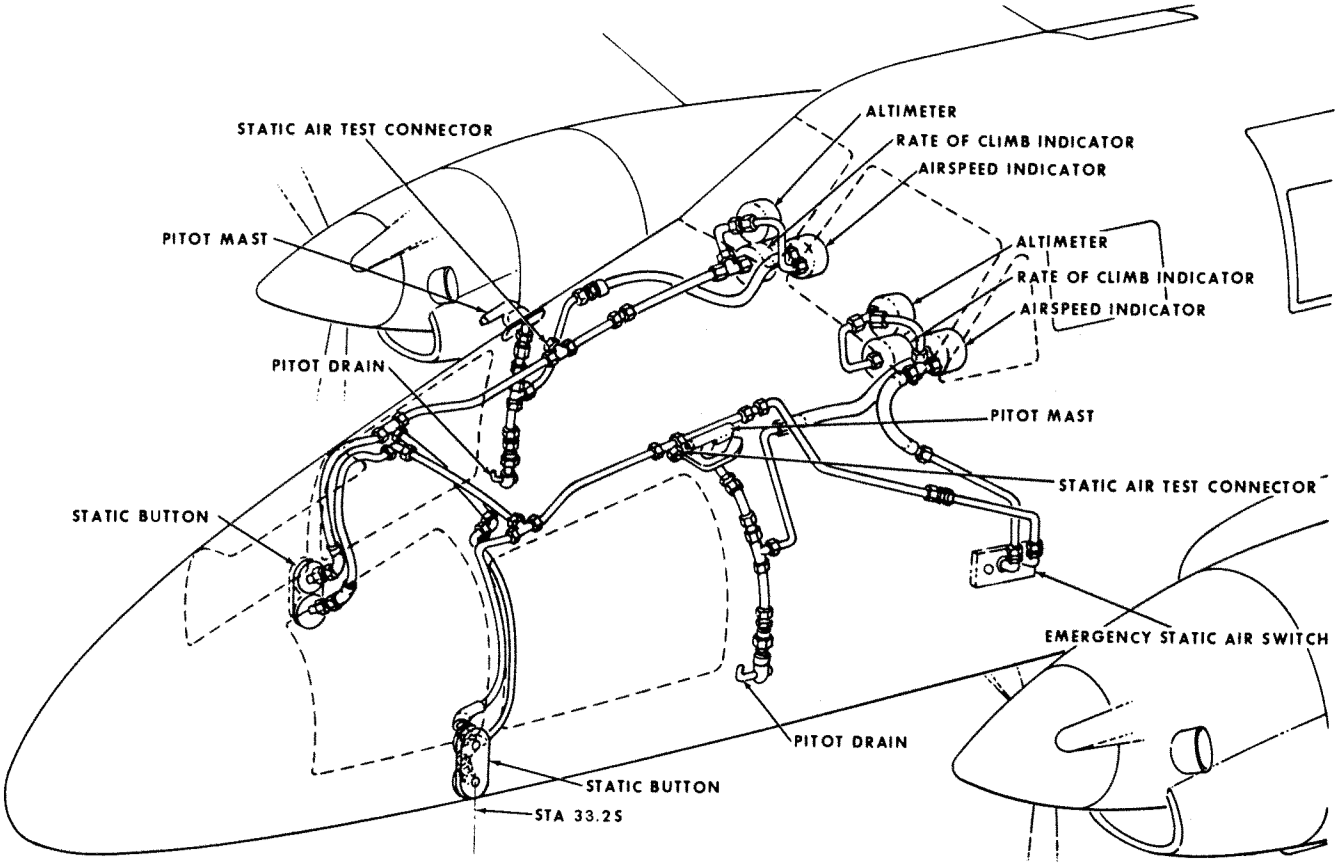


Figure 3
Pitot and Static System (U-146 and U-148 thru U-151)

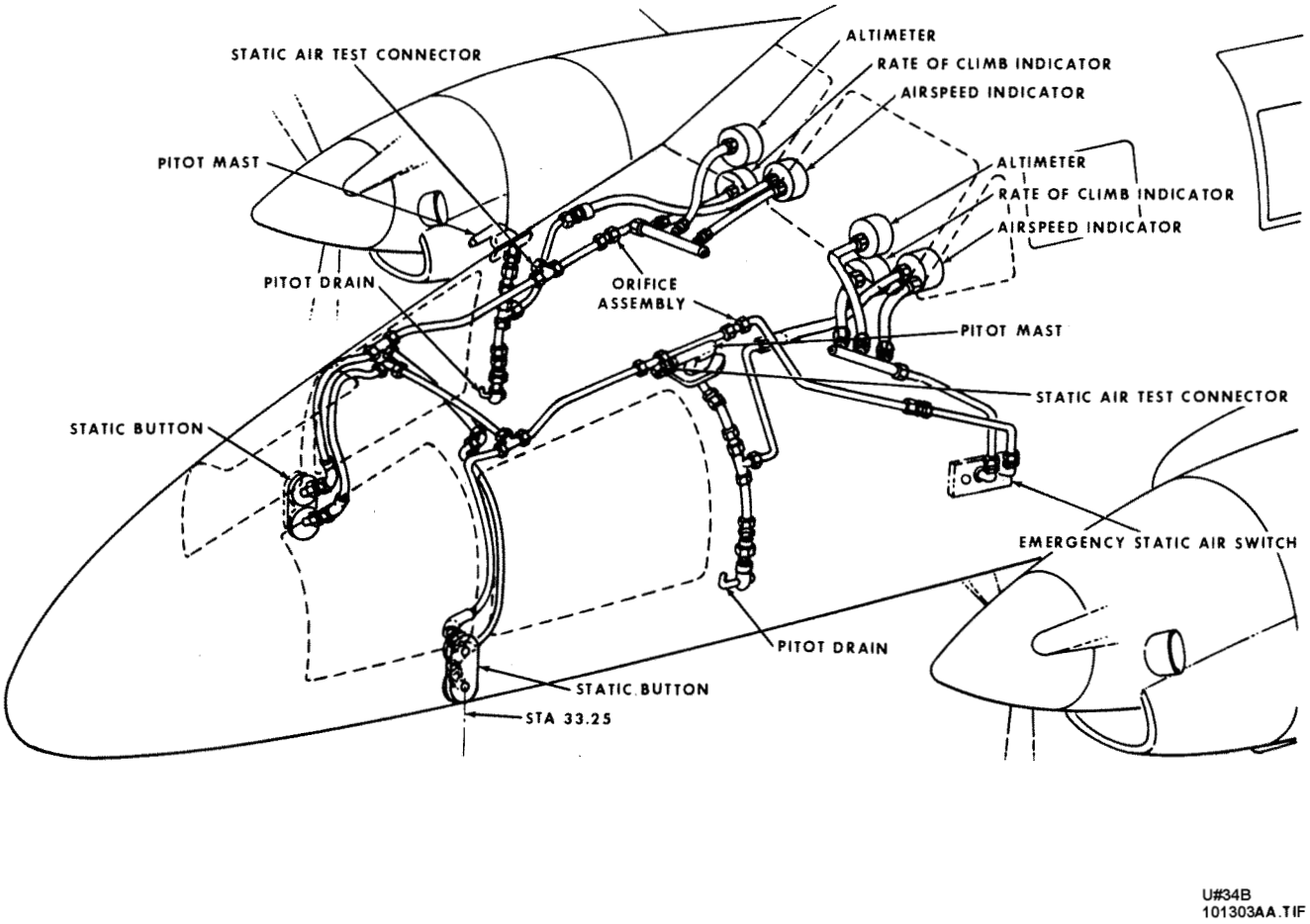
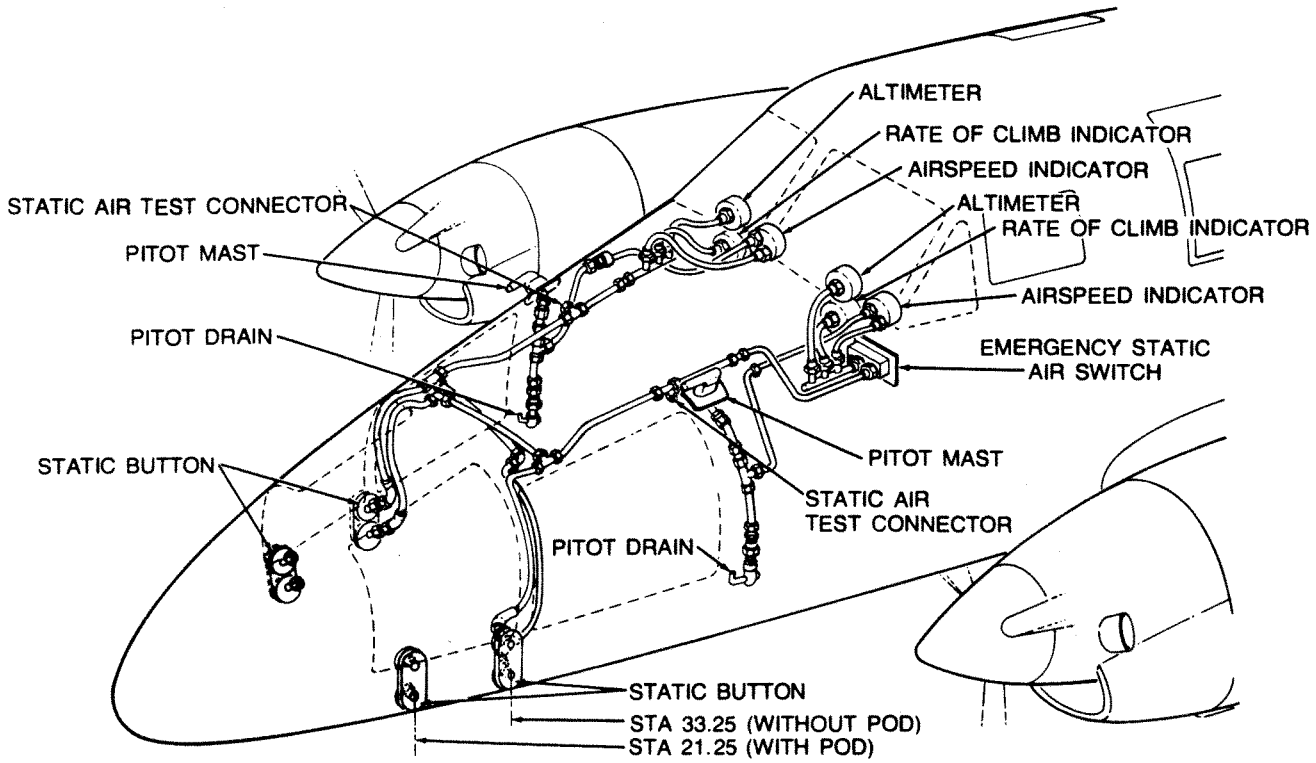


Figure 4
Pitot and Static System (U-152 thru U-164)



On airplane serials U-50, U-165 and after, static air tubing is connected to the buttons at fuselage station 21.25 when the baggage pod is installed. When the airplane is flown without the baggage pod, the static air tubing must be connected to the buttons at fuselage station 33.25.

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Figure 5
Pitot and Static System (U-50, U-165 and After)

NAVIGATION
FLIGHT ENVIRONMENT DATA
TROUBLESHOOTING

1. PITOT AND STATIC SYSTEM

Troubleshooting - Pitot and Static System

INDICATION	PROBABLE CAUSE	REMARKS
1. Heating element inoperative.	a. Defective switch.	a. Replace the switch.
	b. Grounded or open circuit.	b. Check for continuity and repair.
	c. Defective heating element in pitot head.	c. Replace heating element.
2. Circuit breaker keeps tripping.	a. Grounded wire.	a. Check for continuity and repair.
3. Instruments inoperative or erratic in operation.	a. Lines clogged.	a. Drain lines at drain cocks. Disconnect lines at instruments and Mow out with low pressure air.
	b. Line leaks.	b. Check lines for loose connections at all connection points.

NAVIGATION
FLIGHT ENVIRONMENT DATA
MAINTENANCE PRACTICES

1. PITOT SYSTEM

A. Pressure Test

(1) Connect a low pressure air source, equipped with an on-off valve, to the pitot mast.

(2) Apply pressure SLOWLY until the airspeed indicator registers 90% of its maximum reading, then turn the valve off.

NOTE: As an alternate method, should the equipment noted above not be available, clamp a rubber tube over the pitot head inlet so that the connection is airtight. Crimp the end of the tube and roll it up SLOWLY until the airspeed indicator registers 90% of its maximum reading.

CAUTION: *To avoid rupturing the diaphragm of the airspeed indicator, apply pressure slowly and do not build up excessive pressure in the line.*

(3) If the system retains the pressure required to maintain the reading just obtained for a period of five minutes, there are no leaks in the pitot lines. If airspeed indicator reading declines, check the system for leaky hoses and loose connections.

CAUTION: *Release pressure SLOWLY to avoid damaging the airspeed indicator.*

B. Hose Inspection

After the pitot system is checked for leaks, inspect the hoses for signs of deterioration, particularly at bends and at the connection points to the pitot mast and airspeed indicator. Hoses that are cracked or hardened should be replaced with a rubber hose conforming to Military Specification MIL-H-5593. Any time a hose is replaced, repeat the preceding pressure check.

2. EVA (PLASTIC TUBING)

A. Precautions

To improve the service life of EVA (plastic tubing), used for pitot and static line plumbing, the following practices should be observed:

CAUTION: *Clear EVA (plastic tubing) used in instrument plumbing is susceptible to deterioration when exposed to infrared light, such as sunlight. During maintenance, replace all clear EVA tubing with colored tubing (blue) to greatly improve tubing service life.*

The use of unapproved lubricants to install EVA tubing over fitting beads contributes to tubing deterioration. The only approved lubricant is warm water, less than 160°F.

3. STATIC SYSTEM

A. Checks

NOTE: Any time maintenance is performed on the static system, i.e. the system is opened, the system must be tested for leaks per the instructions contained in Federal Aviation Regulation 91.170.

Before proceeding with the static pressure system test, perform the following:

(1) Connect the test equipment to the airplane in preparation for the complete static pressure system test.

(2) Open the alternate static source switch and slowly apply vacuum to the system.

(3) If no altitude increase is noted on the altimeter, the alternate static source is clear and is operating. Proceed with the static system test.

(4) If the altimeter indicates an increase in altitude, the alternate static source is blocked and the following steps should be taken:

(a) Remove the lower LH side upholstery panel in the cockpit to gain access to the alternate static plumbing. On U-50, U-165 and After, the alternate static port is located behind the pilot's side of the instrument panel next to the Airspeed and Turn and Slip indicators.

(b) Check the open end of the alternate static line or port at the back side of the alternate static source switch to be certain the opening is not blocked.

(c) If an obstruction is noted at this point, remove the obstruction. Apply vacuum lightly to the line, if applicable, to determine if the line is clear.

(d) If the alternate static source is now open, proceed with the static system test.

(e) If an obstruction still exists, the altimeters must be disconnected and the lines checked internally for the obstruction and the obstruction removed.

(5) Connect the altimeters and reapply vacuum lightly to determine if the alternate source is operating.

(6) Proceed with the static system test.

B. Testing for Leaks

The static system should be checked for leaks in accordance with the instructions contained in Federal Aviation Regulation 91.170.

CAUTION: *To avoid damaging the airspeed indicators, they should be removed from the system with the lines capped, or an equal pressure should be applied to the pitot side of the indicators while testing the system for leakage.*

4. OUTSIDE AIR TEMPERATURE (OAT) INDICATOR

A. Removal

CAUTION: *Place an assistant in the cockpit to hold the indicator or tape it in place to prevent the indicator from turning or falling while the sunshield is being removed.*

(1) Remove the hexagonal sunshield on the upper portion of the airplane (Ref. Figure 201).

(2) Carefully pull the indicator from the escutcheon.

(3) Remove the exterior spacer and rubber washer from the airplane.

(4) Remove the escutcheon which surrounds the indicator.

(5) Remove the interior spacer and rubber washer.

B. Inspection

Inspect the sunshield for dents, plugged openings and any misalignment that would allow contact with the stem. Inspect the nut or sunshield and stem base for stripped or damaged threads. Inspect the washers or spacers for deep scoring and distortion. Check the rubber mounting washer for peeling, cracking, and resiliency. Inspect the index markings and numerals for legibility. Inspect the pointer for chipped or peeling paint. Replace all defective parts.

C. Installation

(1) Clean the contact area around the indicator mounting hole with a white lint-free cloth dampened with aliphatic naphtha (33, Table 6, 91-00-00). The cloth should not have enough solvent to drip.

CAUTION: *Cleaning solvents should never be poured or sprayed on the structure because of the possibility of the fluid running back between the layers of the structure then returning to contaminate the sealing surface.*

Under no circumstances are red shop rags to be used for cleaning. All cloths and gloves shall contain less than 0.75% oil, and shall be free of silicones.

(2) Immediately dry the area with a white lint-free cloth to absorb the solvent before it evaporates. Repeat the above procedure until there is no discoloration on the cloth.

(3) Apply sealer (93 or 94, Table 6, 91-00-00) to the contact area on both sides of the skin.

(4) Position the exterior and interior rubber washers in their place (Ref Figure 201).

(5) Apply the same sealer to the exposed edges of the washers, then install the interior and exterior spacers in their respective positions.

(6) Install the escutcheon in the overhead panel.

(7) Install the indicator through the skin, ensuring that the probe does not become damaged.

CAUTION: *Tape the indicator in place or have an assistant hold the indicator to prevent it from turning or failing.*

(8) Install the hexagonal sunshield on the portion of the sunshield which protrudes through the airplane skin and exterior washer.

CAUTION: *To avoid damage to the probe, do not overtighten.*

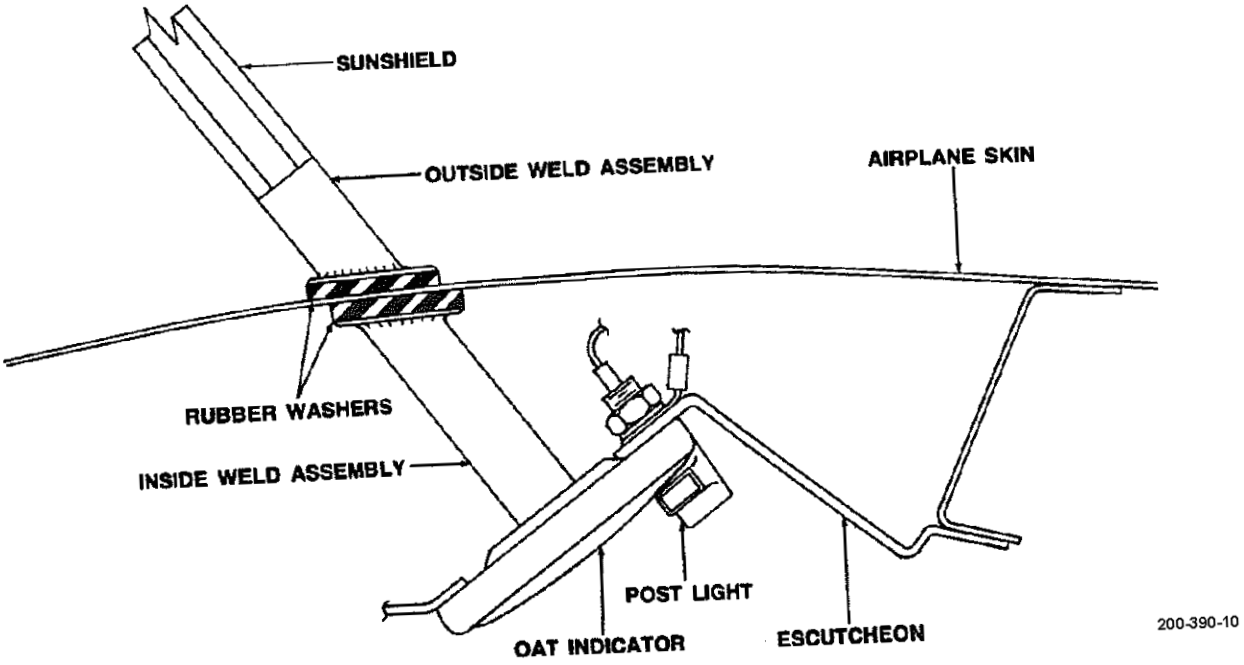

Figure 201
Outside Air Temperature Indicator

5. STANDBY COMPASS

A. Calibration

Correction of the standby compass card for magnetic error is accomplished by removing the correction card plate and adjusting the compensator with a brass or other non-magnetic screwdriver. Calibrate the standby compass to a compass rose or by use of a calibrated precision sight compass. Compass calibration is performed under the following conditions.

(1) Start both engines.

(2) Turn all normal electrical and avionics equipment ON.

(3) Turn Left and Right Generators ON.

(4) Verify windshield electric heat is OFF.

(5) Verify windshield wipers are OFF.

(6) Turn Left and Right Pitot Heat OFF.

(7) Turn stall warning heat OFF.

(8) Turn air conditioner and blower OFF.

(9) Set the adjustment screws of the compensator on zero. Zero position of the adjustment screws are obtained by lining up the dot on the compensator frame.

CAUTION: *The N-S and E-W adjustment screws make approximately 3/4 turn and hit a stop. DO NOT FORCE BEYOND THE STOPS. This can cause a misalignment of the magnets and make the compass unable to be compensated.*

(10) Align the aircraft to a magnetic NORTH heading. Adjust the N-S adjustment screw until the compass reads exactly NORTH.

(11) Align the aircraft to a magnetic EAST heading. Adjust the E-W adjustment screw until the compass reads exactly EAST.

(12) Align the aircraft to a magnetic SOUTH heading. Note the resulting SOUTH error. Adjust the N-S adjustment screw until one half of this error is removed.

(13) Align the aircraft to a magnetic WEST heading. Note the resulting WEST error. Adjust the E-W adjustment screw until one half of this error is removed.

(14) Align the aircraft in successive magnetic 30° headings and record all errors on the deviation card of the compass. In each position the standby magnetic compass should read within ± 10° of the actual aircraft magnetic heading (Ref. Figure 202).

NOTE: Steps (10) thru (14) may be repeated as necessary to obtain an accuracy of ± 10° at each 30° heading.

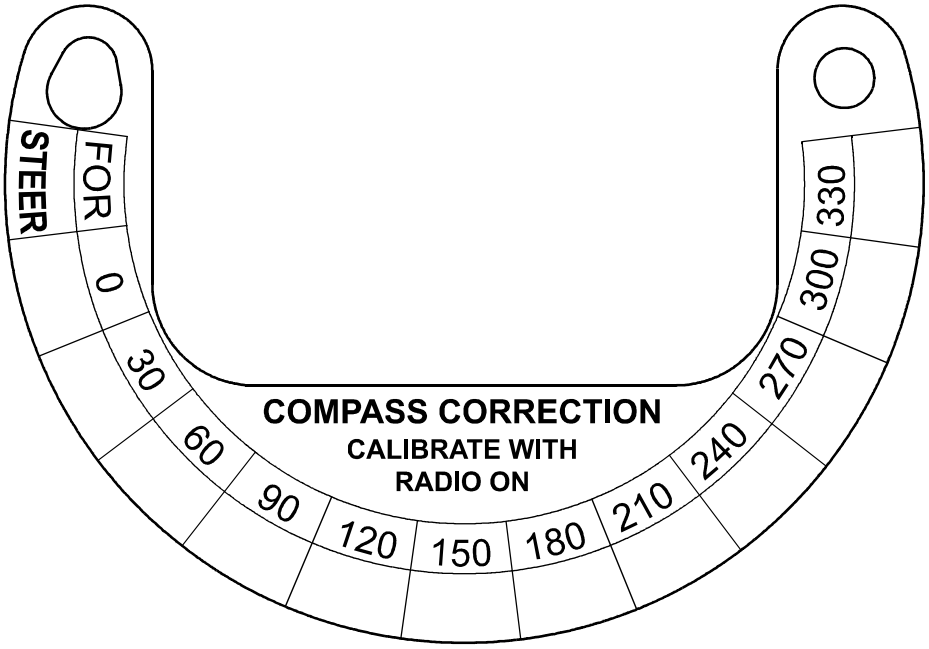

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Figure 202
Magnetic Compass Correction Card Plate

List of Effective Pages

CH-SE-SU	PAGE	DATE
35-LOEP	1	Nov 1/10
35-CONTENTS	1	Nov 1/10
35-00-00	1 thru 3	Nov 1/10
	201 thru 205	Nov 1/10

Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 35 - OXYGEN

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OXYGEN
OXYGEN SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

A. U-1 thru U-164, except U-50

Oxygen for flights at high altitude is supplied by a cylinder (or cylinders) mounted under the avionics compartment floor in the nose section. A standard 22 cubic foot cylinder is mounted along the right outboard sidewall of the nose wheel well. Optional 49.2, 65.6, 98.4 and 115.8 cubic foot cylinder installations are available for mounting along either outboard side of the nose wheel well. The system is serviced by a filler valve accessible through the nose wheel well (Ref. Figure 1). A gage is mounted adjacent to the filler valve for checking system pressure during filling. Oxygen flows from the cylinder through a line to the regulator, pressure gage and shutoff valve in the oxygen panel. The oxygen panel is located on the sidewall to the pilot's left. The regulator is a pressure demand altitude compensating constant flow type which reduces system pressure 50 to 55 psi at the passenger outlets.

B. U-50, U-165 and After

Oxygen for these airplanes is provided by a standard 22 cubic foot cylinder or one of four options. Optional 49, 66, 98, or 115-cubic foot cylinder options are available. If the 22, 49, or 66-cubic foot systems are provided, one oxygen bottle is used and is mounted underneath the nose avionics and nose baggage compartment along the right outboard sidewall. If the 98 or 115-cubic foot system is provided, two oxygen bottles are used, and the second bottle is located under the nose avionics compartment mounted on the left outboard sidewall. When the 98-cubic foot oxygen system is provided, two 49-cubic foot cylinders are used. When the 115-cubic foot oxygen system is provided, a 49 and a 66-cubic foot oxygen cylinder are used. The oxygen system is serviced by a filler valve accessible through the nose wheel well. A high pressure gage is mounted adjacent to the filler valve for checking system pressure during filling. A high pressure gage is also located on the lower right side of the copilot's subpanel (Ref. Figure 2).

A shutoff valve and regulator, located on the right side cylinder, is actuated by a push-pull type shutoff control located on the upper left side of the copilot's subpanel. The regulator is a constant flow type with a reduced controlled outlet pressure of 70 psi at all outlets.

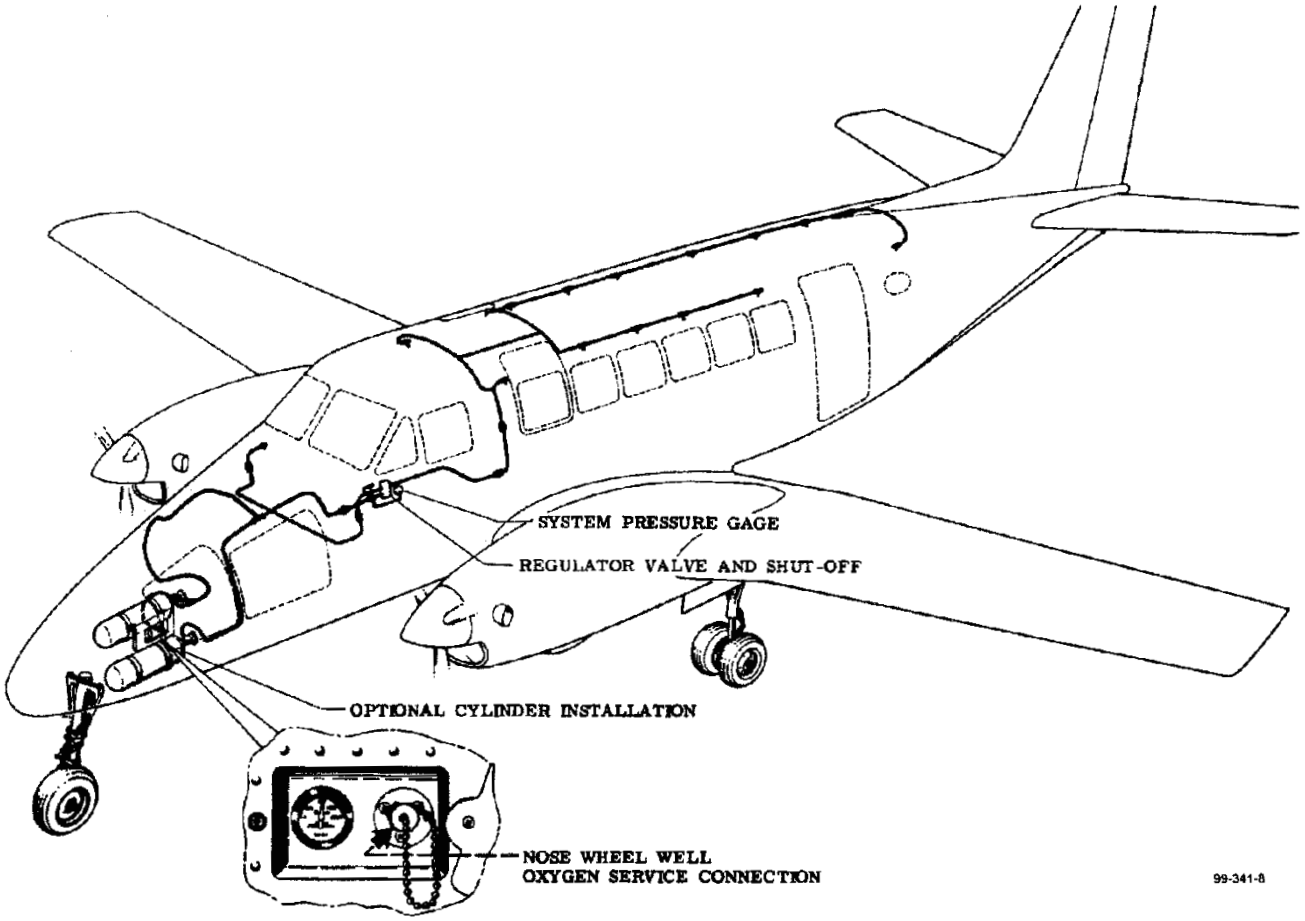

Figure 1
Oxygen System (U-1 thru U164, except U-50)

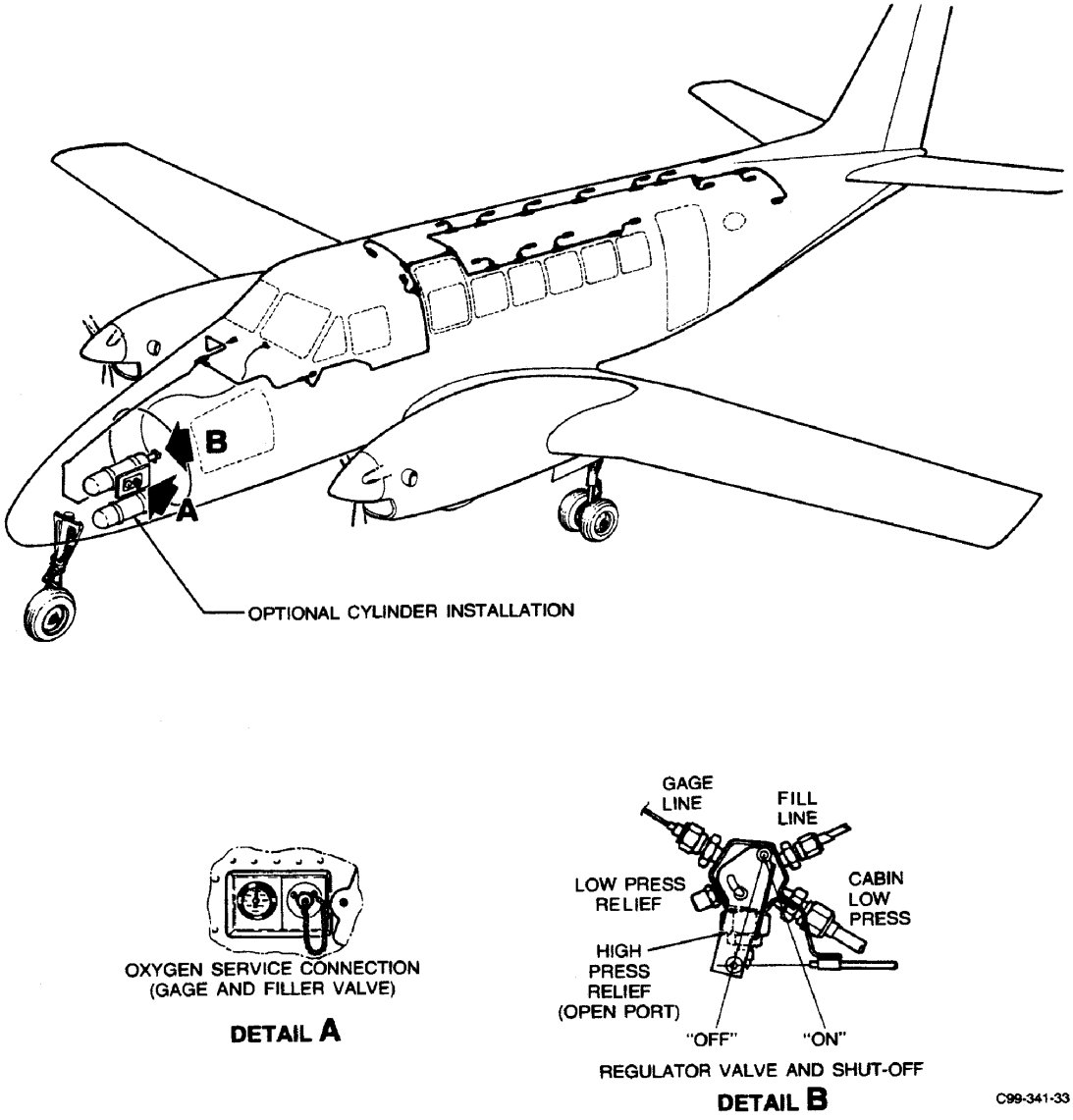

Figure 2
Oxygen System (U-50, U-165 and After)

OXYGEN
OXYGEN SYSTEM
MAINTENANCE PRACTICES

1. OXYGEN SYSTEM

A. Servicing

NOTE: Oxygen used for recharging must be MIL-O-27210 aviator's breathing oxygen (22, Table 6, 91-00-00).

WARNING: Do not use oxygen intended for medical purposes, or such industrial uses as welding. Such oxygen may contain excessive moisture that could freeze up the valves and lines of the oxygen system.

Access to the pressure indicator and filler valve of the oxygen system may be gained through an access door labeled OXYGEN RECHARGE located on the right side of the nose wheel well. To recharge the oxygen system, remove the protective filler cap from the filler valve and attach the hose from an oxygen recharging cart to the filler valve.

WARNING: Avoid making sparks and keep all burning cigarettes or fire away from the vicinity of the airplane. Make sure that the oxygen shutoff valve in the cockpit is in the closed position. Inspect the filler connection for cleanliness before attaching it to the filler valve. Make sure that your hands, tools, and clothing are clean, particularly of grease or oil, for these contaminants will ignite upon contact with pure oxygen under pressure. As a further precaution against fire, open and close all oxygen valves slowly.

To prevent overheating, fill the oxygen system slowly by adjusting the recharging rate with the pressure regulating valve on the cart. Fill the cylinder until a pressure of 1800 ± 50 psig is attained at a temperature of 70°F. On U-50, U-165 and After, the standard 22-cubic foot bottle should be filled to 1800 psi. System pressure should be 1850 psi for optional oxygen installations requiring 49 and/or 66 cubic foot bottles. (This is a steady state condition, after the cylinder has cooled from the recharging heat build-up.) This pressure may be increased an additional 3.5 psi for each degree of increase in temperature; similarly, for each degree of drop in temperature, reduce the pressure for the cylinder by 3.5 psi. When the oxygen system is properly charged, disconnect the filler hose from the filler valve and replace the protective cap on the filler valve. If at any time, in the process of servicing and purging the system or replacing the oxygen cylinder, it becomes necessary to disconnect a fitting, the threads should be sealed with tape (24, Table 6, 91-00-00). Apply to the first three threads of the male fitting. A swage-lock fitting should be tightened one and a quarter turns from the finger tight position.

NOTE: Refer to Advisory Circular 43.13-1A for additional servicing precautions recommended by the FAA for the oxygen system.

B. Purging

Offensive odors may be removed from the oxygen system by purging. The system should also be purged any time system pressure drops below 50 psi or the lines are left open. Purging is accomplished simply by connecting a recharging cart into the system and permitting oxygen to flow through the lines and outlets until any offensive odors have been carried away. The following Steps outline the procedures recommended for purging the oxygen system.

WARNING: Avoid making sparks and keep all burning cigarettes or fire away from the vicinity of the airplane when the outlets are in use. Inspect the filler connection for cleanliness before attaching it to the filler valve. Make sure that your hands, tools, and clothing

are clean, particularly of grease or oil stains, for these contaminants will ignite upon contact with pure oxygen. As a further precaution against fire, open and close all oxygen valves slowly during filling.

(1) Remove the access panel on the left side of the nose area just forward of FS 84.00, then turn the oxygen cylinder valve until it is just cracked open. On U-50, U-165 and After, the primary oxygen tank has a regulator assembly, not a standard valve, therefore it will not be cracked open.

(2) Gain access to the filler valve in the nose wheel well and attach the hose from an oxygen recharging cart to the filler valve.

(3) Plug in an oxygen mask at each outlet in the cabin and cockpit.

(4) Open the cabin door and place the oxygen shutoff valve in the open or ON position.

(5) Set the recharging cart pressure regulator to deliver 200 psi of pressure to the system.

(6) Allow the system to purge for one hour.

If any offensive odor still lingers, continue purging the system for an additional hour. If such odors still remain, replace the supply cylinder. After the system has been adequately purged, service the system.

C. Test Procedure (U-1 thru U-164, except U-50)

Plug a pressure gage (0 to 100 psi range) into the pilot's or copilot's outlet. Note that the regulator is shut off and that no pressure is indicated on the test gage. Charge the high pressure system to 1850 psi. Shut off the cylinder valve and observe the aircraft gage for evidence of leakage. The pressure loss in five minutes shall not exceed 400 psi. Check that there is no leakage past the regulator by observing the low pressure test gage. Turn on the high pressure supply and the regulator. Allow two minutes for the pressure to stabilize in the low pressure system, then turn off the regulator and note the pressure on the test gage. After I5 minutes, the drop in pressure shall not exceed 5 psi. In case leakage is excessive in either system, apply leak test solution (25, Table 6, 91-00-00) sparingly to suspected areas. Make necessary repairs and retest.

NOTE: A small quantity of oxygen under pressure is trapped in the control chamber of the regulator when the regulator is turned to OFF. This oxygen (50 to 70 cc) will continue to bleed overboard until the control chamber pressure equalizes with ambient pressure.

D. Test Procedure (U-50, U-165 and After)

NOTE: Oxygen used for testing or recharging must be Aviator's Breathing Oxygen (22, Table 6, 91-00-00).

(1) Place the oxygen shutoff valve in the CLOSED position and disconnect the low pressure line from the regulator. Plug the line with an AN806-5D (or equivalent) plug.

(2) Connect a test gage to mask outlet.

(3) Plug a low pressure oxygen supply into another mask outlet.

(4) Charge the low pressure system to 70 $\pm$ 10 psi. Allow two minutes for the pressure to stabilize, then remove or shut off the pressure source.

(5) If leakage exceeds 5 psi in 15 minutes, apply leak test solution (25, Table 6, 91-00-00) sparingly to suspected areas. After locating the leak, make the necessary repairs and retest the system.

NOTE: Wipe the area clean and dry after using the leak test solution.

(6) Remove the plug from the low pressure line and connect the line to the regulator.

(7) Place the oxygen shutoff valve in the ON position and observe the test gage. System pressure should be 70 ± 10 psi.

(8) Place the oxygen shutoff valve in the CLOSED position and note that the system pressure bleeds off.

NOTE: If the conditions stated in Steps (7) and (8) are not obtained, the regulator assembly is not functioning properly. No attempt should be made to repair the regulator. The complete cylinder and regulator assembly should be replaced.

(9) Disconnect the test gage from the mask outlet.

(10) With a system pressure of 1500 to 1850 psi (1500 to 1800 psi with 22 cubic foot cylinder), there should not be any pressure drop in 30 minutes as noted on the supply pressure gage.

(11) If any leakage is noted in the high pressure system in 30 minutes, apply leak test solution (25, Table 6, 91-00-00) to suspected areas. After locating the leak, make the necessary repairs and retest the system.

NOTE: Wipe the area clean and dry after using the leak test solution.

E. Retesting

Oxygen cylinders used in this airplane are of two types:

Light weight cylinders (49 and 66-cubic feet), stamped 3HT on the plate on the side, must be hydrostatically tested every three years and have the test date stamped on the cylinder. These bottles have a service life of 4,380 pressurizations or 24 years, whichever occurs first. After the full service life has been reached the bottles must be discarded.

Standard weighs cylinders (22-cubic feet), stamped 3AA, must be hydrostatically tested every five years and stamped with the retest date. The service life on these cylinders is unlimited.

2. OXYGEN CYLINDER (U-1 THRU U-164, EXCEPT U-50)

A. Removal

The oxygen cylinder(s) may be located on either the right outboard side of the nose wheel well or along both outboard sides of the nose wheel well. Access is gained by removing the nose baggage compartment floor.

WARNING: Avoid making sparks and keep all burning cigarettes or fire away from the vicinity of the oxygen cylinder. Make sure that your hands, tools, and clothing are clean, particularly with respect to oil or grease spots, for these contaminants will ignite upon contact with pure oxygen under pressure.

(1) Slowly turn the oxygen supply cylinder valve until fully closed.

(2) Disconnect the line from the supply cylinder.

(3) Cap the open line immediately with a clean fitting.

(4) Support the cylinder and then loosen the two bracket clamp wing nuts.

NOTE: Observe the special handling precautions on the tag attached to the oxygen cylinder.

(5) Remove the old cylinder from the clamps.

NOTE: On systems having a cylinder installed on the left outboard side of the nose wheel well, untape a portion of the ram air duct and move it to one side before removing the cylinder.

B. Installation

(1) Position the cylinder in the mounting brackets and install the bracket clamp wing nuts. If the left cylinder is being installed, properly position the ram air duct and retape the duct.

(2) Tighten the wing nuts and safety.

(3) Carefully inspect the fittings on both the cylinder and the line for cleanliness and the presence of foreign matter, since such matter may contaminate the oxygen until it is unfit for breathing.

(4) Apply tape (24, Table 6, 91-00-00) to the first three threads on the male fittings.

(5) Connect the line fitting to the cylinder being installed.

(6) Open the cylinder shutoff valve.

(7) Test the connections for leaks as described in OXYGEN SYSTEM TEST PROCEDURE.

3. OXYGEN CYLINDER (U-50, U-165 AND AFTER)

A. Removal

The oxygen cylinder(s) may be located on either the right outboard side of the nose wheel well or along both outboard sides of the nose wheel well. Access is gained by removing the nose baggage compartment floor. The forward clamp on the left side bottle may be reached from an access panel in the nose avionics compartment floor.

WARNING: Avoid making sparks and keep all burning cigarettes or fire away from the vicinity of the oxygen cylinder. Make sure that your hands, tools, and clothing are clean, particularly with respect to oil or grease spots, for these contaminants will ignite upon contact with pure oxygen under pressure.

(1) Place the oxygen control in the OFF position. Turn the oxygen supply cylinder valve on the left cylinder (if applicable), until fully closed.

(2) Disconnect the push-pull control from the shutoff valve on the oxygen cylinder.

NOTE: After disconnecting the push-pull control from the shutoff valve, make certain that the valve remains in the OFF position.

(3) Disconnect the lines from the regulator if removing the right cylinder. Disconnect the line at the shutoff valve if removing the left cylinder.

NOTE: Cap or plug open lines and ports to prevent contamination of the system.

(4) Support the cylinder and loosen the two bracket clamp wing nuts for the cylinder being removed.

NOTE: Observe the special handling precautions on the tag attached to the oxygen cylinder.

(5) Remove the cylinder you wish to remove from the two mounting brackets.

B. Installation

(1) Position the respective cylinder in the mounting brackets and install the bracket clamp wing nuts. The cylinder with the regulator always installs on the right side.

(2) Tighten the wing nuts and safety.

(3) Carefully inspect the fittings on the cylinder(s) and the lines for cleanliness and the presence of foreign matter, since such matter may contaminate the oxygen until it is unfit for breathing.

(4) Apply tape (24, Table 6, 91-00-00) to the first three threads on the male fittings.

(5) Attach the lines to the regulator on the right cylinder and tighten.

(6) Attach the line to the left oxygen bottle (if applicable).

(7) Attach the push-pull control to the shutoff valve (right bottle), and rig the cable so the bottle is in the fully closed position with the control knob fully forward.

(8) Open the cylinder shutoff valve on the left oxygen bottle (if applicable).

(9) Test the connections for leaks as described in OXYGEN SYSTEM TEST PROCEDURE.

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CHAPTER 36 - PNEUMATIC

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PNEUMATIC SYSTEM

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PNEUMATIC
PNEUMATIC SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

A. U-1 thru U-151, except U-50

The engine bleed air pneumatic system is operated with pressure obtained by bleeding air from the engine compressors. The bleed air from either engine, as controlled by its respective regulator, provides sufficient pressure to operate the system should the other engine fail. The regulators are mounted in each nacelle immediately forward of the firewall. A check valve incorporated in the line directly behind each regulator prevents air from escaping back through the regulator into the chamber of an inoperative engine compressor. The regulators are set to maintain incoming bleed air at a required pressure of 18 ± 1 psi. As a safety feature, both regulators incorporate a relief valve that regulates pressure approximately 21 psi. The relief valve is set higher than the regulator pressure as a safety feature to protect against regulator failure.

CAUTION: *Do not exceed the recommended 18 ± 1 psi regulator setting because higher settings will result in excessive wear on the relief valve.*

Ordinarily, the regulator will not require adjustment since each unit is preset at the factory; however, each regulator is equipped with an adjusting screw should it become necessary to reset them in the field. The bleed air is routed through an ejector-distributor valve in order to produce a vacuum when the deice system is installed. On U-1 thru U-151, except U-50, the instruments are operated by air pressure, and the schematics for this system are shown in Figure 1.

B. U-50, U-152 and After

Air at a temperature of approximately 650°F (depending on the power setting and ambient air temperature), is bled from the third stage (P_3 station) of each engine compressor at a flow rate sufficient to produce 18 psi of regulated pressure required to operate the bleed air warning system, autopilot, surface deicer, and vacuum system. The line providing the bleed air for these systems tees off the compressor bleed air line and the bleed air line for the cabin heating at a point just aft of the first fireseal forward of the firewall. On U-50, U-165 and After, the bleed air is routed aft from each engine through a check valve to a single pressure regulator valve located adjacent to the check valves under the RH seat deck immediately forward of the rear spar. On U-152 thru U-164, the regulator is located on the aft side of the rear spar just to the right of the airplane centerline. The check valves for the regulator are approximately 20 inches forward of the regulator on U-152 thru U-164. The loss of heat in the pneumatic plumbing will reduce the temperature of the bleed air from approximately 650°F (depending on power setting and ambient air temperature) to 70°F above the temperature of the ambient air by the time it reaches the pressure regulator valve. The regulator valve is set at approximately 18 psi of pressure and incorporates a safety valve that is set at approximately 21 psi. The relief valve is set higher than the regulator pressure as a safety feature to protect against regulator failure.

CAUTION: *Do not exceed the recommended 18 ± 1 psi regulator setting because higher settings will result in excessive wear on the relief valve.*

From the pressure regulator valve, lines are routed to the various airplane systems that utilize pneumatic pressure. On U-50, U-152 and After, the instruments are operated by vacuum (suction) air. Information for the vacuum system is covered in Chapter 37, VACUUM.

2. BLEED AIR WARNING

On U-50, U-165 and After, the airplanes are equipped with a dual bleed air warning system. Refer to Chapter 26, FIRE PROTECTION, for specific information regarding this warning system.

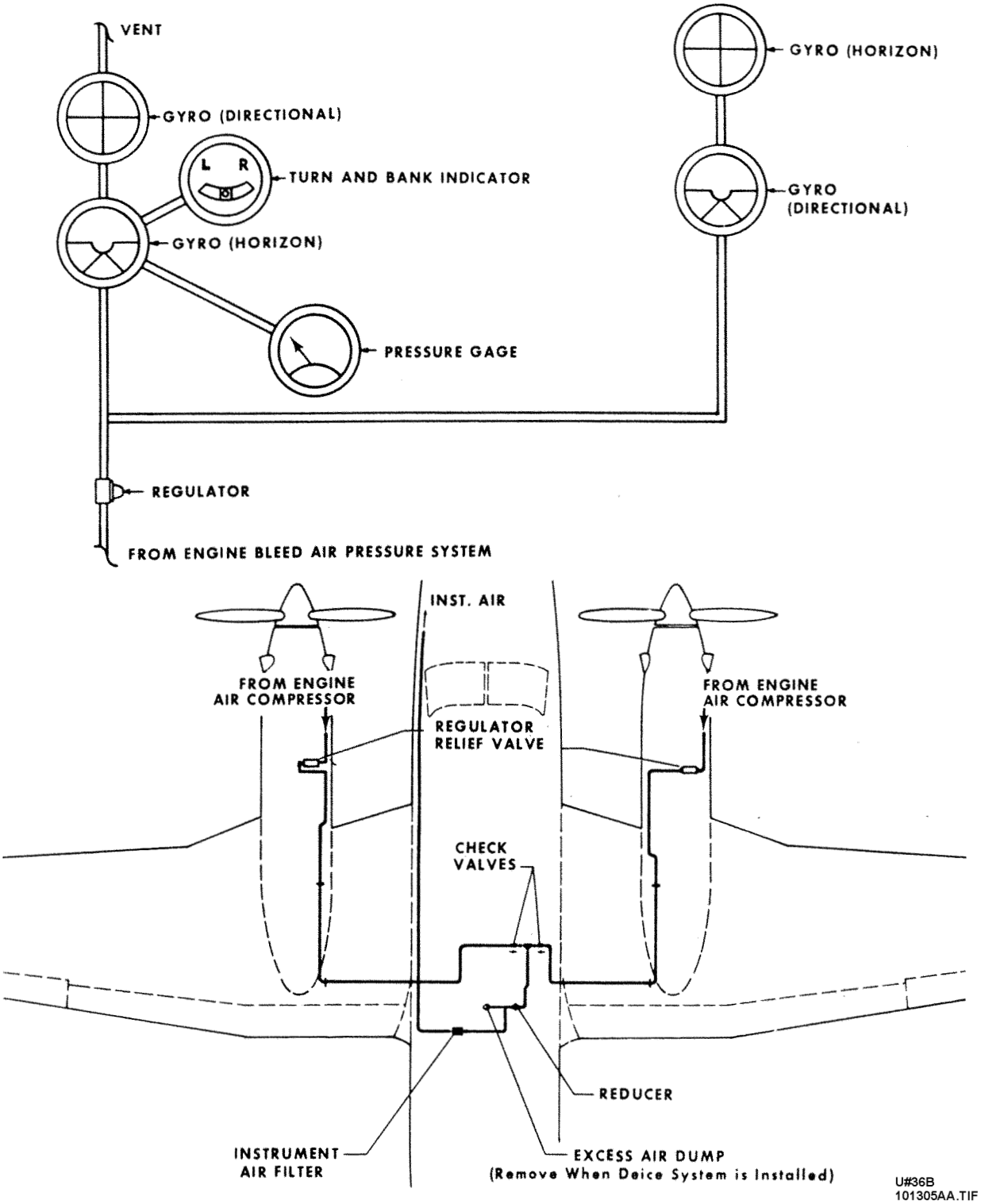

Figure 1
Engine Bleed and Instrument Pressure Air (U-1 thru U-151, except U-50)

PNEUMATIC
PNEUMATIC SYSTEM
TROUBLESHOOTING

1. PRESSURE INSTRUMENT AIR SYSTEM

Troubleshooting - Pressure Instrument Air System (U-1 thru U-151, Except U-50)

INDICATION	PROBABLE CAUSE	REMARKS
1. Lack of adequate pressure to operate instruments properly.	a. Clogged or leaking pressure lines.	
	b. Dirty filters in instruments or dirty relief valve.	
	c. Pressure relief valve needs adjustment or if defective.	
2. Pressure drop registered on gage.	a. Gage defective.	
	b. Leakage in lines.	
	c. Pressure regulator valve needs adjustment or is defective.	
3. Three psi increase in system pressure.	a. Pressure regulator valve has failed and system is operating on relief valve pressure setting.	

PNEUMATIC
PNEUMATIC SYSTEM
MAINTENANCE PRACTICES

1. PRESSURE REGULATOR RELIEF VALVE

A. Adjustment (U-1 thru U-151, Except U-50)

Access to either regulator is gamed by removing the oil cooler air intake scoop at the underneath side of the nacelles. Regulator adjustment can be accomplished b) applying the following procedure:

(1) Remove the center aisle floorboard panel behind the main spar and on airplanes which are equipped with an hour meter, remove the hour meter pressure switch from the pneumatic pressure intake line and connect a calibrated test gage to the line. On airplanes which are not equipped with an hour meter, remove the cap from the hour meter pressure switch connecting point and attach a calibrated test gage. Taps have been provided in the LH and RH bleed air lines to actuate L and R pressure (source) failure indicators on the system pressure gage.

(2) Run up the engine until a reading of between 70 and 80% is registered on the tachometer.

(3) Check pressure for 18 $\pm$ 1 psi. Adjust the regulator valve by unsafetying the lock retaining washer and loosening the adjusting screw locknut located at the top of the regulator until a reading of 18 $\pm$ 1 psi on the test gage in the fuselage is obtained.

CAUTION: *Do not exceed the recommended 18 $\pm$ 1 psi regulator setting because higher settings will result in excessive wear on the relief valve.*

(4) Repeat the above sequence for the other engine.

(5) Run up both engines until a reading of between 70 and 80% is registered on the tachometers and check the "instrument air" system pressure gage in the instrument panel for a reading of 5.5 in. Hg. If the 5.5 in Hg. is not obtained refer to INSTRUMENT AIR.

NOTE: Step (5) pertains to airplanes equipped with surface deicer equipment. On airplanes not equipped with surface deicer equipment, the system pressure may be read directly from the test gage. If the airplane is equipped with a DE-ICING PRESSURE gage, check that the needle is within the shaded area.

B. Adjustment (U-50, U-152 and After)

On U-50, U-152 and After, there is only one pressure regulator valve. On U-50, U-165 and After, the pressure regulator valve is located under the center aisle floorboard just forward of the rear spar on the right side. On U-152 thru U-164, the pressure regulator valve is located just behind the rear spar, just to the right of the airplane centerline.

(1) Gain access to the single pressure regulator relief valve.

(2) Run up the engine until a reading between 70 to 80% is registered by the gas generator tachometer.

(3) Check the deicer pressure gage for a reading of 18 $\pm$ 1 psi. If necessary, unsafety the lock retaining washer and loosen the adjusting screw locknut located on top of the valve.

(4) Turn the adjusting screw clockwise to increase and counterclockwise to decrease the pressure setting until the pressure gage indicates 18 ± 1 psi.

CAUTION: *Do not exceed the recommended 18 ±1 psi regulator setting because higher settings will result in excessive wear on the relief valve.*

(5) Tighten and safety the locknut, then shut down the engine.

CAUTION: *On these serials vacuum (suction) air is used for instrument operation. Check the system vacuum gage for a reading of 5.9 In. Hg., for adjustment of the pressure regulator valve may affect system suction enough to necessitate readjusting the vacuum regulator valve. Refer to Chapter 37, VACUUM for the vacuum regulator valve adjustment.*

2. INSTRUMENT AIR (U-1 THRU U-151, EXCEPT U-50)

NOTE: Electric gyros are installed in either the pilot's or copilot's instrument panel, serials U-134 and After. On prior airplanes, electric gyros were installed by Kit 99-3014S or Kit 99-3016S by Service Instructions No. 0296-391.

Pneumatic air pressure to operate the directional and horizontal gyros, turn and bank indicator and pressure gage is bled off of the engine bleed air system and routed through an instrument air filter located aft of the rear spar and through an instrument air regulator mounted behind the pilot's subpanel. The instrument air filter should be checked every 100 hours and changed each 400 hours or sooner if conditions (operation in heavy smoke or dust) warrant. The instrument air regulator mounted behind the pilot's subpanel is set at the factory to maintain 5.5 inches Hg pressure reading on the instrument air pressure gage with both engines running at 70% N_1 speed. The primary function of the instrument air regulator is to provide instrument air regulation and adequate pressure at low engine rpm. If it is determined that the instrument air regulator requires adjustment, there is a set screw located at the top of the regulator housing. Turning the set screw clockwise increases pressure while turning the set screw counterclockwise decreases pressure.

NOTE: Both engines should be running at 70% N_1 speed when checking the instrument air regulator setting in order that the proper reading will register on the instrument pressure gage.

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CHAPTER 37 - VACUUM

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VACUUM
VACUUM SYSTEM
DESCRIPTION AND OPERATION

1. GENERAL

A. U-50, U-152 and After

Bleed air from the engines is routed through the venturi of an ejector to provide the vacuum necessary for operation of the instruments and deflation of the surface deicer boots. The vacuum is regulated by a suction relief valve designed to admit into the system the amount of air required to maintain sufficient vacuum (5.9 inches Hg.) for proper operation of the vacuum operated instruments. As part of the vacuum system, the turn and slip indicator, operates on a reduced vacuum setting. The vacuum turn and slip indicator includes a regulator with a needle valve that is set to provide 2.0 $\pm$ 0.2 inches Hg. across the instrument.

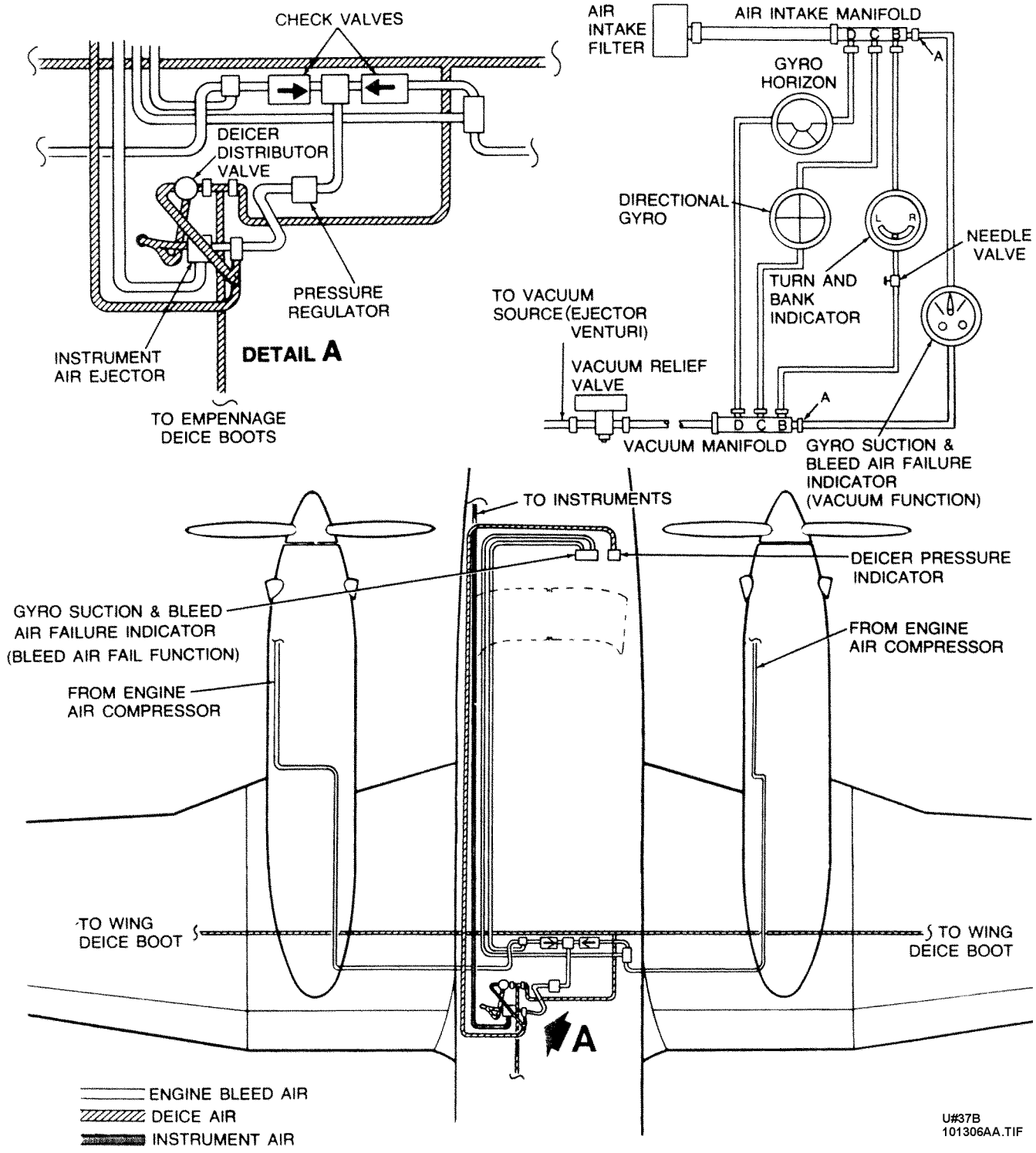

Figure 1
Engine Bleed, Instrument and Deice Air (U-152 thru U-164)

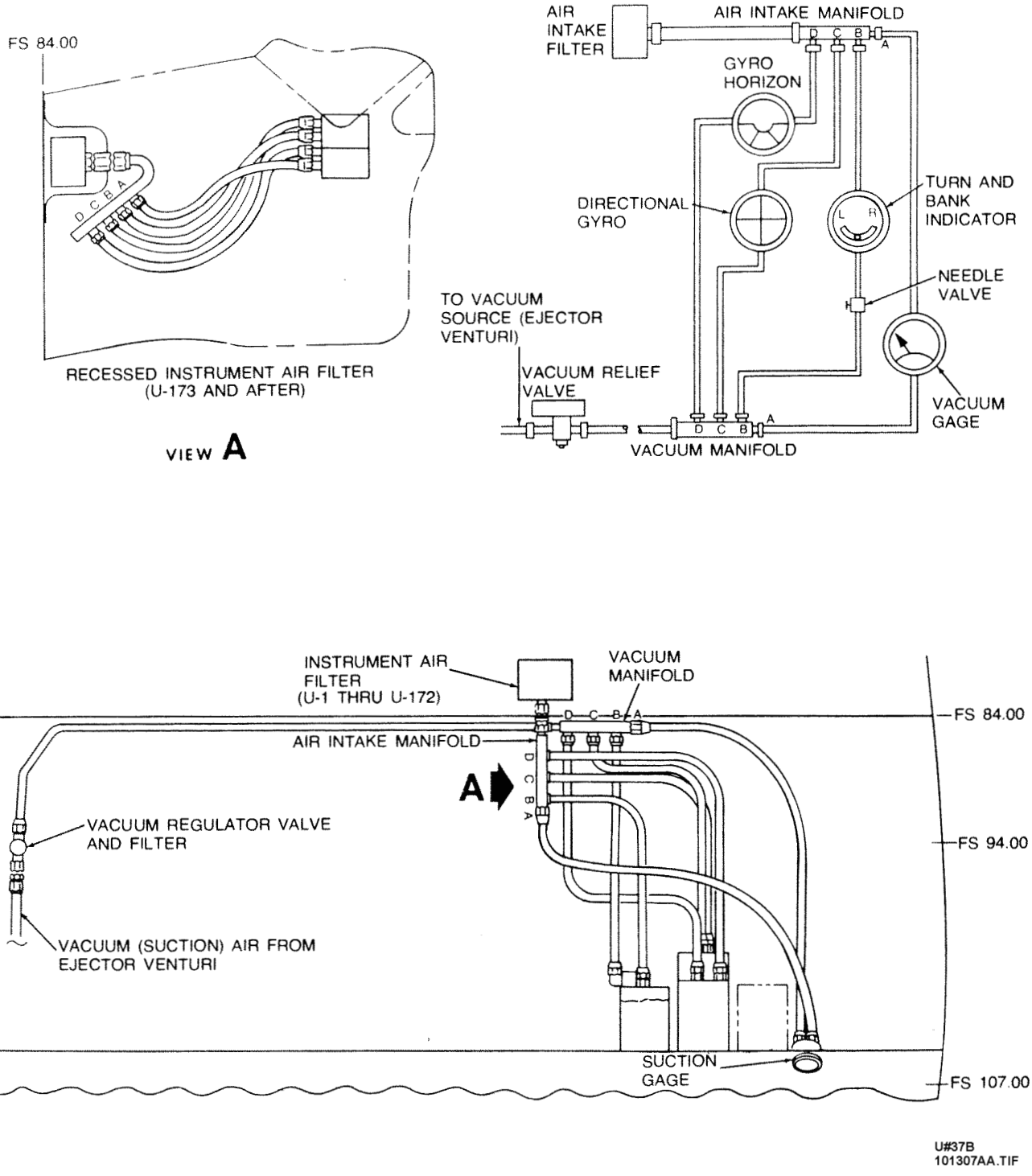

Figure 2
Instrument Vacuum System (U-50, U-165 and After)

VACUUM
VACUUM SYSTEM
TROUBLESHOOTING

1. INSTRUMENT VACUUM SYSTEM (U-50, U-152 AND AFTER)

Troubleshooting - Instrument Vacuum System (U-50, U-152 and After)

INDICATION	PROBABLE CAUSE	REMARKS
1. Lack of adequate vacuum to operate instruments properly.	a. Clogged or leaking vacuum lines.	
	b. Dirty filters in instruments or dirty vacuum regulator.	
	c. Vacuum regulator needs adjustment or is defective.	
	d. Insufficient bleed air pressure through ejector.	
2. Vacuum drop registered on gage.	a. Gage defective.	
	b. Leakage in lines.	
	c. Vacuum regulator needs adjustment or is defective.	
	d. Insufficient bleed air pressure through ejector.	
3 Excessive vacuum registered on gage.	a. Gage defective.	
	b. Vacuum regulator needs adjustment or cleaning or is defective.	

VACUUM
VACUUM SYSTEM
MAINTENANCE PRACTICES

1. VACUUM SYSTEM

A. Plumbing

The vacuum ports of the vacuum operated instruments are plumbed to a vacuum manifold which is located just aft of the cockpit forward bulkhead and to the right of the airplane centerline. The instrument air inlet ports are plumbed to an air intake manifold, above and to the left of the vacuum manifold Filtered air is supplied to the intake manifold through the instrument air filter Port "A" of each manifold is plumbed to the vacuum gage; port "B" of each manifold is plumbed to the turn and slip indicator; port "C" of each manifold is plumbed to the directional gyro; port "D" of each manifold is plumbed to the gyro horizon. Refer to Figures 1 and 2 for the plumbing schematics.

CAUTION: *Permit no oil, grease, pipe compound, or any foreign material to enter the gyros. Make sure that all air lines are clean and free of foreign particles and/or residue before connecting lines to the gyros.*

Do not use thread lube on gyros with plastic housings and threads. Do not exceed 20 inch-pounds torque on the fittings.

2. INSTRUMENT AIR

A. Filter

On U-1 thru U-172, the air filter is located just forward of the intake manifold on the forward side of the cockpit forward bulkhead, on the right side of the airplane centerline. On U-173 and After, the instrument air filter is recessed just aft of the cockpit forward bulkhead and is protected by a filter cover. A bent type manifold is used on these installations but the plumbing remains unchanged. The instrument air filter should be checked every 100 hours and changed every 400 hours or sooner if conditions (operations in heavy smoke or dust) warrant. Refer to Chapter 12 for the procedure required to change the instrument air filter.

3. VACUUM REGULATOR VALVE

A. Filter

The vacuum regulator valve filter is mounted on top of the vacuum regulator. The regulator valve is protected by this foam type filter that should be cleaned with solvent (38, Table 6, 91-00-00) at 100-hour intervals. If the vacuum in the system exceeds 5 9 in. Hg., clean the filter and recheck vacuum pressure before attempting to adjust the regulator valve.

B. Adjustment

Before adjusting the vacuum (suction) air at the vacuum regulator to 5.9 inches Hg., the pressure regulator and relief valve (pressure) air should be adjusted to 18 $\pm$ 1 psi according to the instructions in Chapter 36, PNEUMATIC.

On U-152 through U-164 the vacuum regulator is located on the aft side of the cockpit forward bulkhead at FS 84.00, approximately 20 inches forward of the instrument panel, on the pilot's side. On U-50, U-165 and After, the vacuum regulator valve is located just forward of the bulkhead at FS 94.00 near the left side of the fuselage. Adjust the regulator as follows:

NOTE: Prior to making any adjustments to the vacuum regulator valve, make certain that the instrument air and vacuum regulator filters are clean and free of lint or other foreign materials.

(1) Start one engine and run it up until a reading of 70 to 80% is registered by the gas generator tachometer.

(2) Check the vacuum gage for a reading of approximately 5.9 inches Hg. If the indication is incorrect, adjust the vacuum regulator valve by unsafetying the lock retaining washer and loosening the adjusting screw locknut.

(3) Turn the adjusting screw IN (clockwise) to increase or OUT (counterclockwise) to decrease the vacuum setting.

(4) When the vacuum gage registers a reading of 5.9 inches Hg with the engine operating within the range designated in Step (1), tighten the locknut and safety it, then shut down the engine.

4. TURN AND SLIP INDICATOR

A. Adjustment

The turn and slip indicator has a needle valve regulator at the back of the vacuum instrument. This valve is set to maintain 2.0 ± 0.2 inches Hg. across the instrument with the vacuum regulator properly adjusted. The turn and slip indicator is preadjusted and normally needs no adjustment; however, if need be, the indicator may be adjusted as follows:

(1) Attach a vacuum gage to the normally plugged port of the turn and slip indicator needle valve, located at the back of the vacuum turn and slip indicator.

(2) With the engines running and the vacuum regulator properly adjusted, an indication of 2.0 ± 0.2 inches Hg should be read on the attached vacuum gage.

(3) If this reading is not obtained, loosen the lock nut on the needle valve and adjust the needle valve to obtain the above reading.

(4) After this reading is obtained, tighten the lock nut, remove the gage, and replace the plug in the open port.

B. Flight Adjustment

The turn and slip indicator is also flight adjustable as follows:

(1) In flight, the airplane should be entered into a coordinated standard rate turn as indicated on the vacuum turn and slip indicator.

(2) While the turn is being executed by the pilot, a mechanic should make the adjustment in Step (3).

(3) The needle valve at the back of the turn and slip indicator should be adjusted to provide 3° of turn per second (180° per minute) rate of turn as indicated on the directional gyro with the turn and slip indicator indicating a standard rate turn.

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CHAPTER 38 - WATER/WASTE

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38-30-00

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WATER/WASTE
WASTE DISPOSAL
MAINTENANCE PRACTICES

1. CHEMICAL TOILET

The chemical toilet installation incorporates a dividing partition which separates the cabin from the toilet area. The toilet area is located at the rear of the passenger section of the cabin. Both the partition and the toilet are removable to provide additional cargo space. Space for toilet accessories is provided by a storage compartment below the seating area.

A. Removal

(1) Remove the partition first by removing the connecting bolts at the bulkhead side-mounting seat connection.

(2) Remove the dividing partition from the airplane.

(3) Remove the chemical toilet unit by removing the four corner mounting bolts connecting it to the floor.

(4) Remove the toilet unit from the airplane.

B. Installation

(1) Install toilet and connect to floor with the four corner mounting bolts.

(2) Install dividing partition and connect bolts at bulkhead side-mounting seat connection.

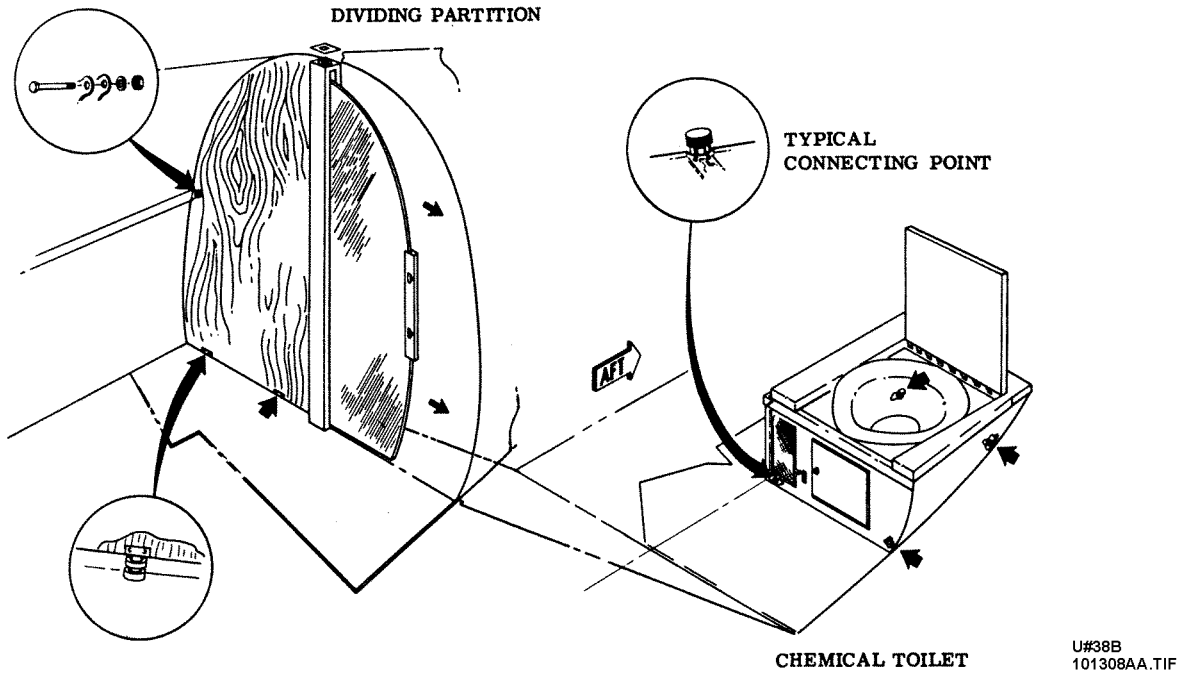

Figure 201
Chemical Toilet and Dividing Partition

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CHAPTER 39 - ELECTRICAL PANELS AND COMPONENTS

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ELECTRICAL PANELS AND COMPONENTS

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ELECTRICAL PANELS AND COMPONENTS
ELECTRICAL PANELS AND COMPONENTS
DESCRIPTION AND OPERATION

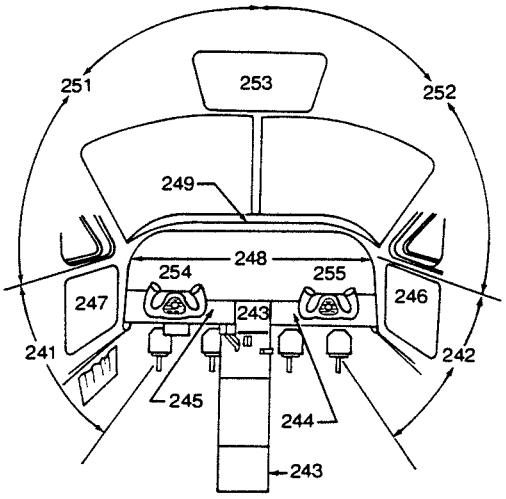
1. GENERAL

The information contained in this chapter is effective for the Model C99 airplanes (serials U-50, U-165 and After) only. The wiring diagrams pertinent to the C99 airplanes are found in the C99 wiring diagram manual P/N 99-590030-5. Refer to the MODEL 99 AIRLINER SERIES WIRING DIAGRAM AND ELECTRICAL SYSTEMS MANUAL, P/N 99-590028A, for maintenance and wiring diagram Information for prior airplane serials.

This chapter is divided into two sections. 39-10-00 contains information necessary for the location and basic maintenance of panels containing the various indicating instruments and panels containing the primary controls necessary for the switching of various systems. Airplane system performance is monitored by reference to these indicating instruments and total airplane system control is affected through the use of the various panels; therefore, all the panels contained in this section are within easy access of the flight crew and are located m the flight compartment.

Section 39-20-00 information deals with the location and maintenance of panels upon which most of the secondary controls and electrical circuitry components are mounted. Access to these panels is necessary only during maintenance; therefore, they are located remotely from the involved systems, but located centrally within the airplane structure. Access to these panels is gained through access openings located throughout the airplane. The secondary controls located on these panels will be found as various forms of relays, resistors, transistors, diodes, limiters and circuit breakers. No electromotive equipment such as motors, actuators and etc. will be found on these panels.

Panels and equipment found within this chapter can be located In the airplane by referring to the ZONE DIAGRAMS (Ref. Figures 1 and 2) which follow.

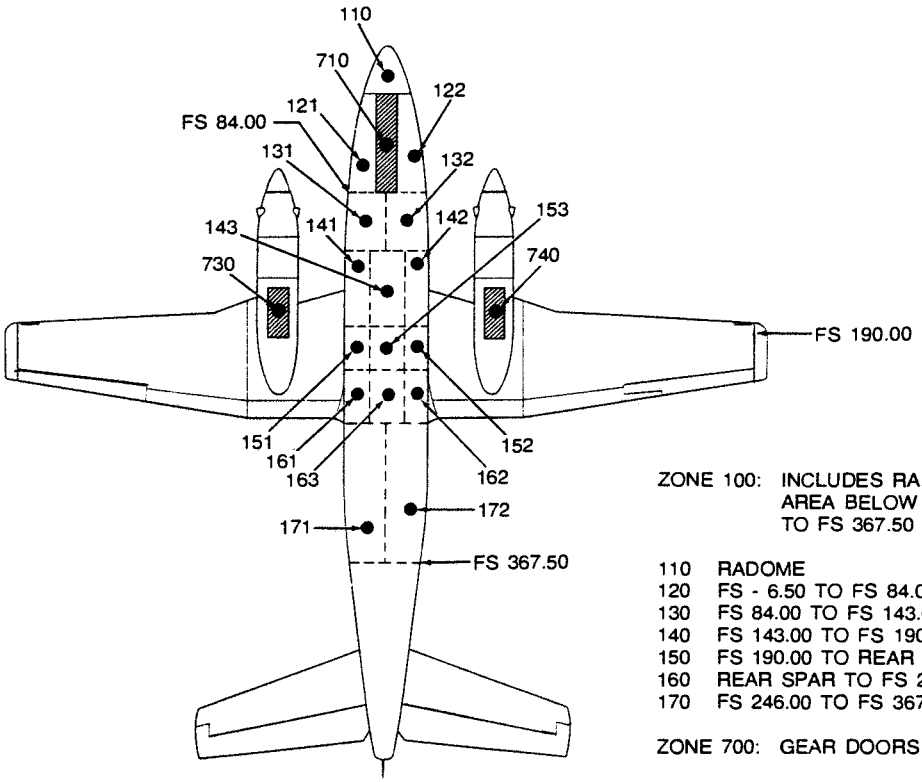


ZONE BREAKDOWN OF CREW COMPARTMENT
ZONE 240: BELOW LOWER LONGERON, FS 104.75 TO FS 143.00 INCLUSIVE

241 LEFT FUSELAGE WALL
242 RIGHT FUSELAGE WALL
243 PEDESTAL, CONSOLE AND CENTER SUBPANEL
244 RIGHT SUBPANEL
245 LEFT SUBPANEL
246 RIGHT SIDEPANEL
247 LEFT SIDEPANEL
248 INSTRUMENT PANEL
249 GLARESHIELD

ZONE 250: ABOVE LOWER LONGERON, FS 104.75 TO FS 143.00 INCLUSIVE

251 LEFT CABIN WALL
252 RIGHT CABIN WALL
253 OVERHEAD PANEL
254 PILOTS CONTROL WHEEL
255 COPILOTS CONTROL WHEEL



ZONE 100: INCLUDES RADOME AND THE AREA BELOW THE FLOOR FROM THE RADOME TO FS 367.50 INCLUSIVE

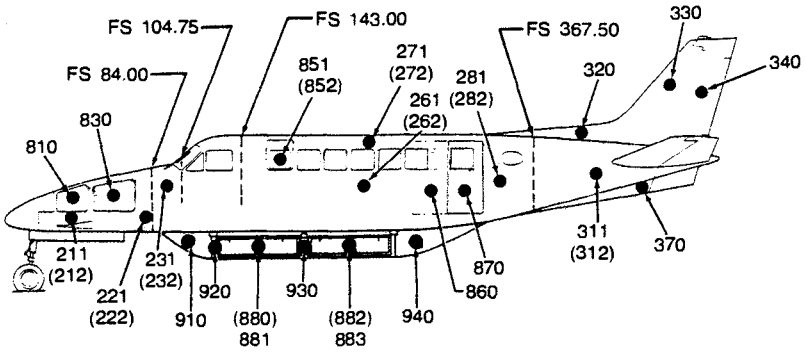
110 RADOME
120 FS - 6.50 TO FS 84.00
130 FS 84.00 TO FS 143.00
140 FS 143.00 TO FS 190.00
150 FS 190.00 TO REAR SPAR
160 REAR SPAR TO FS 246.00
170 FS 246.00 TO FS 367.50

ZONE 700: GEAR DOORS AND WHEEL WELLS

710 NOSE LANDING GEAR
730 LEFT MAIN LANDING GEAR
740 RIGHT MAIN LANDING GEAR

C99-603-473

Figure 1
Zone Diagram



ZONE 200: ABOVE FLOOR RADOME TO FS 367.50 INCLUSIVE

210 RADOME TO FS 84.00
220 FS 84.00 (FORWARD SIDE OF BULKHEAD)
230 FS 84.00 TO FS 104.75 (INSTRUMENT PANEL)
240 SEE SHEET 2
250 SEE SHEET 2
260 FS 143.00 TO CABIN DOOR (BELOW LOWER LONGERON)
270 FS 143.00 TO CABIN DOOR (ABOVE LOWER LONGERON)
280 CABIN DOOR TO FS 367.50

ZONE 300: EMPENNAGE

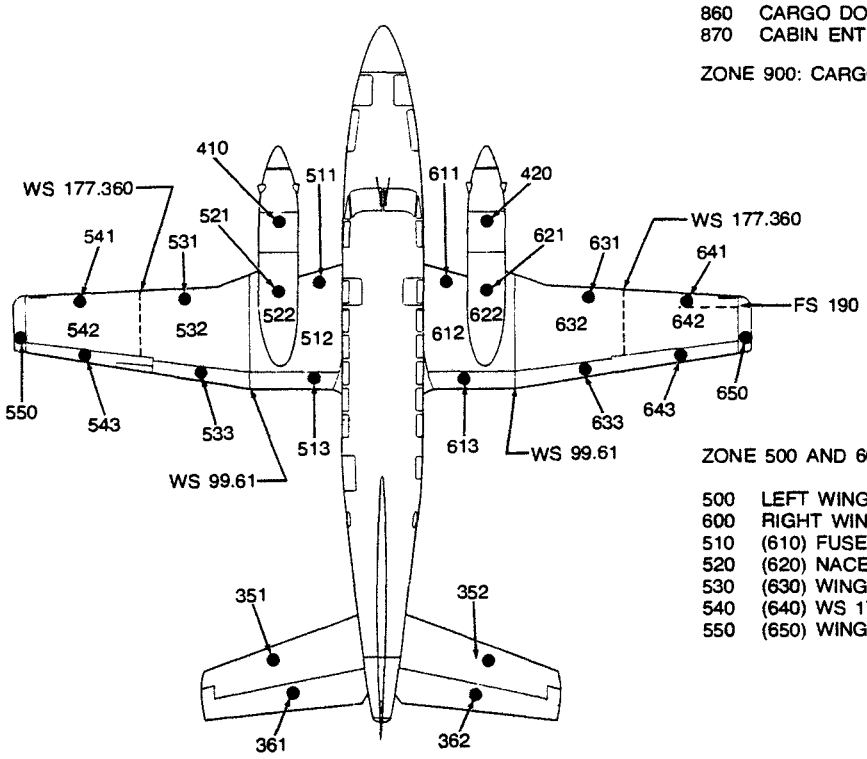
310 FUSELAGE AFT OF FS 367.50
320 VENTRAL FIN
330 VERTICAL STABILIZER
340 RUDDER
350 HORIZONTAL STABILIZER
360 ELEVATORS
370 DORSAL FIN

ZONE 400: POWER PLANTS

ZONE 800: DOORS

810 AVIONICS COMPARTMENT
830 BAGGAGE DOOR
850 EMERGENCY EXIT
860 CARGO DOOR
870 CABIN ENTRANCE DOOR

ZONE 900: CARGO POD



ZONE 500 AND 600: WINGS

500 LEFT WING
600 RIGHT WING
510 (610) FUSELAGE TO WING BREAK
520 (620) NACELLE AFT OF FIREWALL
530 (630) WING BREAK TO WS 177.360
540 (640) WS 177.360 TO WING TIP
550 (650) WING TIP

C99-603-474

Figure 2
Zone Diagram

ELECTRICAL PANELS AND COMPONENTS
INSTRUMENT AND CONTROL PANELS
MAINTENANCE PRACTICES

1. INSTRUMENT PANEL

The standard instrument panel in the C99 Airliner Series Airplane installation is a three part panel. Left hand and right hand panel insets provide the space in which the pilot's and copilot's flight instruments are mounted. The shock mounted stationary portion of the instrument panel provides the mounting frame for the panel insets and the panel center section where the engine indicating instruments and avionic equipment are installed (Ref. Figure 201 Zone 248).

All instruments and avionic equipment mount from the front of the instrument panel. Sufficient excess plumbing lengths and slack in wiring harnesses have been left so that working behind the panel is unnecessary.

Instrument panel components may be removed for service or replacement by the following procedure.

2. INSTRUMENT PANEL COMPONENTS

A. Removal

(1) Free the component from the front of the instrument panel by releasing or removing the fasteners or retainers on the face of the component.

(2) Carefully pull the component out of the instrument panel opening and remove any plumbing or connectors from the back of the instrument.

NOTE: Modular type avionic components have quick disconnect connectors which are released when the retaining allen screw on the face of the component is loosened.

B. Installation

(1) Position the component in front of the instrument panel and attach the plumbing or connectors which were removed previously.

(2) Carefully slide the component back into its opening and install the fasteners or tighten the retainers on the face of the component.

3. PILOT’S SUBPANEL

The pilots subpanel, located beneath the pilot's control column, contains various system control switches and the generator voltmeter and loadmeters (Ref. Figure 202 Zone 245).

Access to the components mounted on the pilots subpanel can be gained by removing the section of the subpanel affected. Removing the edgelit placard panel will reveal the location of the subpanel section retaining screws.

4. COPILOT'S SUBPANEL

The copilot's subpanel, located beneath the copilot's control column, contains the environmental system control switches and several system indicating instruments (Ref. Figure 203 Zone 244).

Components mounted on the copilot's subpanel are accessible by removing the subpanel section affected. The edgelit placard panel must be removed first in order to expose the retaining screws of the subpanel installation.

5. RIGHT HAND CIRCUIT BREAKER PANEL

The circuit breaker panel, located on the copilot's side panel, contains the system control circuit breakers and most of the primary power circuit breakers for the various systems in the airplane (Ref. Figure 204 Zone 246).

6. CIRCUIT BREAKER PANEL

A. Removal

Removal of one or more of the circuit breakers may be effected by the following procedure.

(1) Remove the screws which fasten the edgelit placard panel to the circuit breaker panel and remove the placard panel, being careful not to mar the finish of the panel or surrounding structure.

(2) Remove the screws at the top of the circuit breaker panel.

(3) Lean the panel away from the sidewall at the top carefully to avoid damaging the wires or electrical components behind the panel.

(4) Remove the wires from the circuit breaker which is to be removed.

(5) Remove the jam nut on the front side of the circuit breaker panel which fastens the circuit breaker to the panel. Remove the circuit breaker from the back side of the panel.

B. Installation

(1) Position the circuit breaker in the panel from the back side and install the jam nut on the threaded portion of the circuit breaker from the front side of the panel.

(2) Connect any wires removed previously.

(3) Carefully reposition the panel against the sidewall and install the retaining screws at the top of the panel.

(4) Install the edgelit placard panel.

7. FUEL CONTROL PANEL

The fuel control panel, located on the pilot's subpanel, contains the control switches for the fuel system, circuit breakers, and fuel quantity gages (Ref. Figure 205 Zone 247).

Component replacement is accomplished by removing the panel from the sidewall, using the following procedure.

A. Removal

(1) Remove the edgelit placard panel from the panel section affected.

(2) Release the panel retaining screws and tilt the top of the panel outward exposing the panel backside. Care must be exercised to prevent damage to the wiring and components on the backside of the panel.

(3) Remove and repair or replace the defective component and reinstall the fuel control panel.

B. Installation

(1) Connect any wires to the components which may have been removed earlier.

(2) Reposition the panel on the sidewall and install the retaining screws.

(3) Install the edgelit placard panel.

8. OVERHEAD LIGHT CONTROL PANEL

Dimming rheostats, resistors, transistors and switches for controlling interior lighting are to be found on the overhead light control panel which is located in the center of the flight compartment headliner (Ref. Figure 206 Zone 253).

To facilitate replacement of components mounted on the light control panel, the panel may be completely removed from the airplane using the following procedure:

A. Removal

(1) Remove the knobs from the rheostats and release the retaining screws from the edgelit placard panel and remove the placard panel.

(2) Release the control panel retaining screws. This will allow the control panel to drop down from the headliner.

(3) Disconnect the connectors from the panel and remove the panel for servicing.

B. Installation

(1) Connect the electrical connectors which were disconnected earlier.

(2) Position the panel on the headliner and install the retaining screws.

(3) Install the edgelit placard panel.

(4) Install the knobs which were removed earlier.

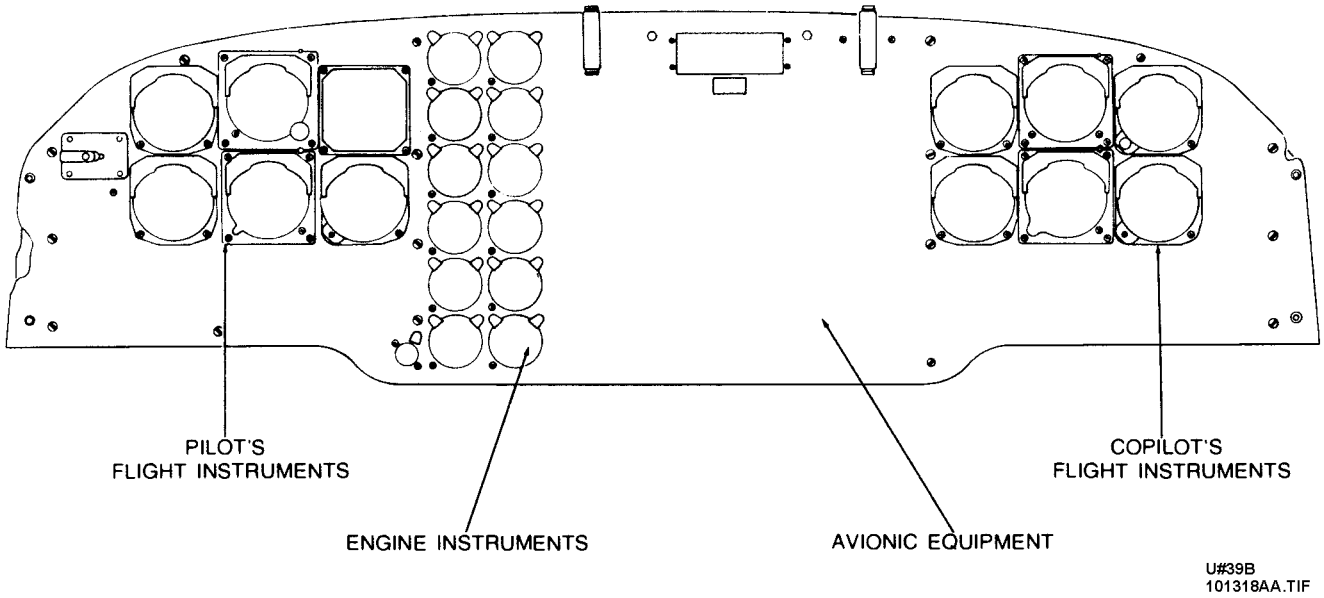


Figure 201
Instrument Panel

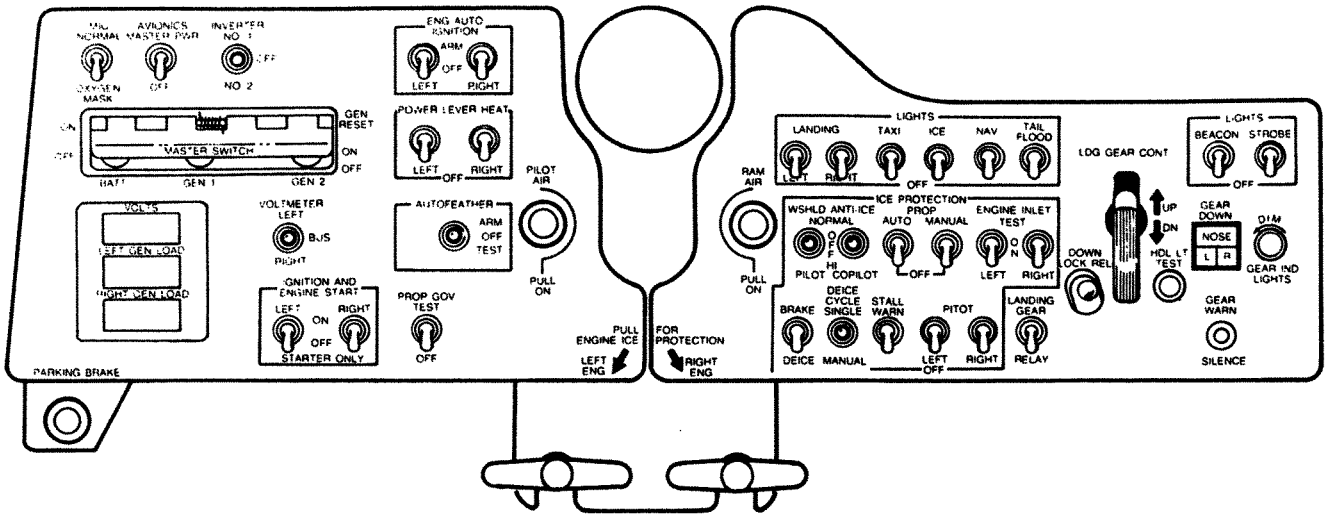


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Figure 202
Pilot's Subpanel

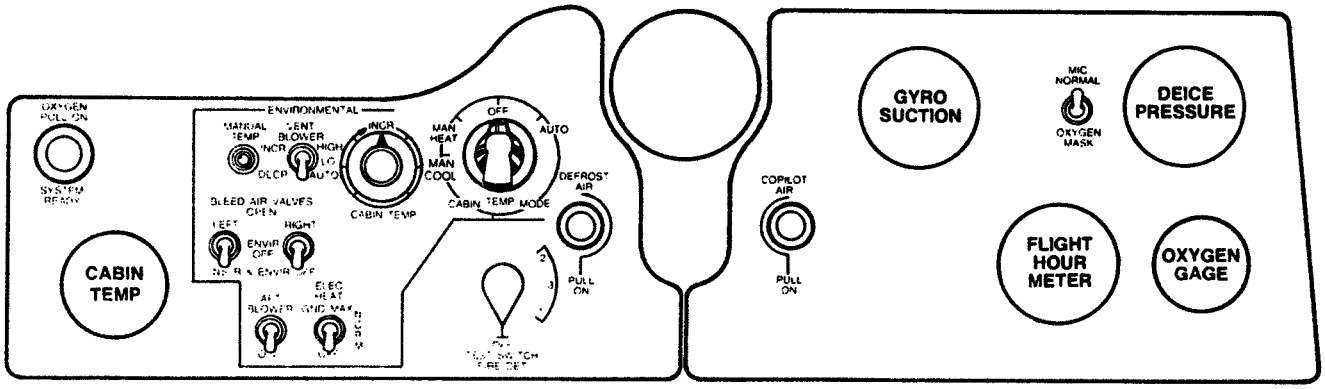

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Figure 203
Copilot's Subpanel

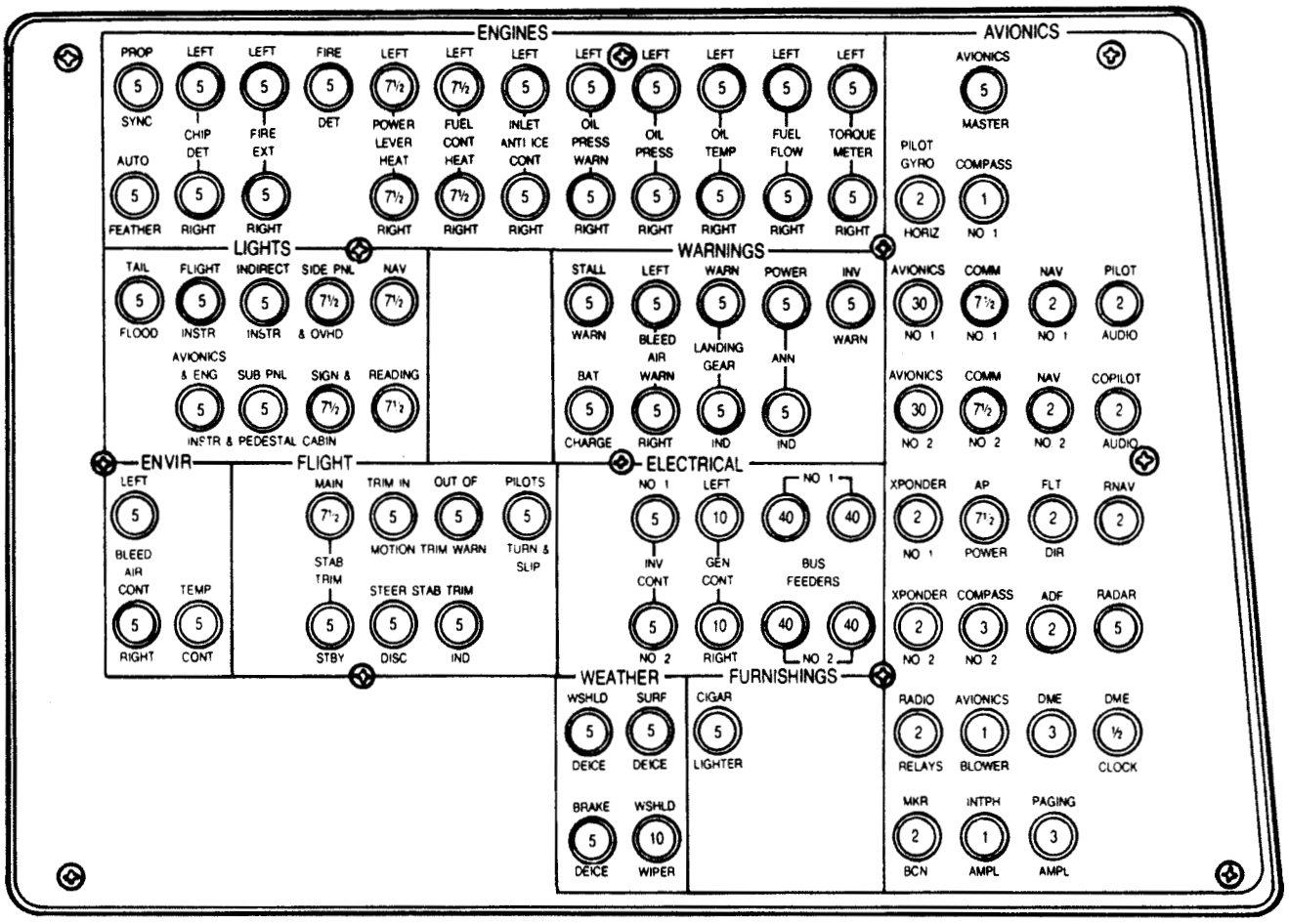

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Figure 204
Circuit Breaker Panel

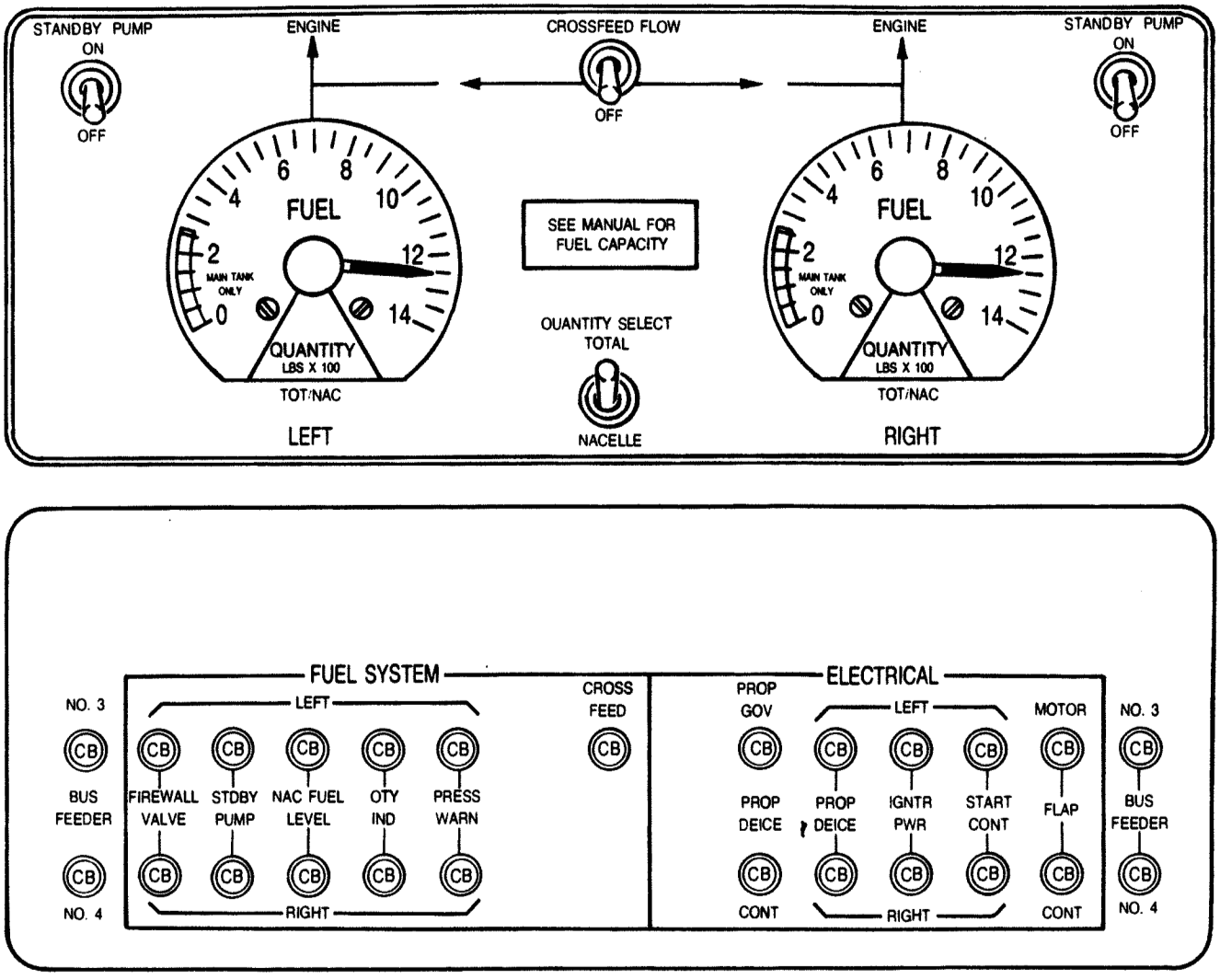

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Figure 205
Fuel Control Panel

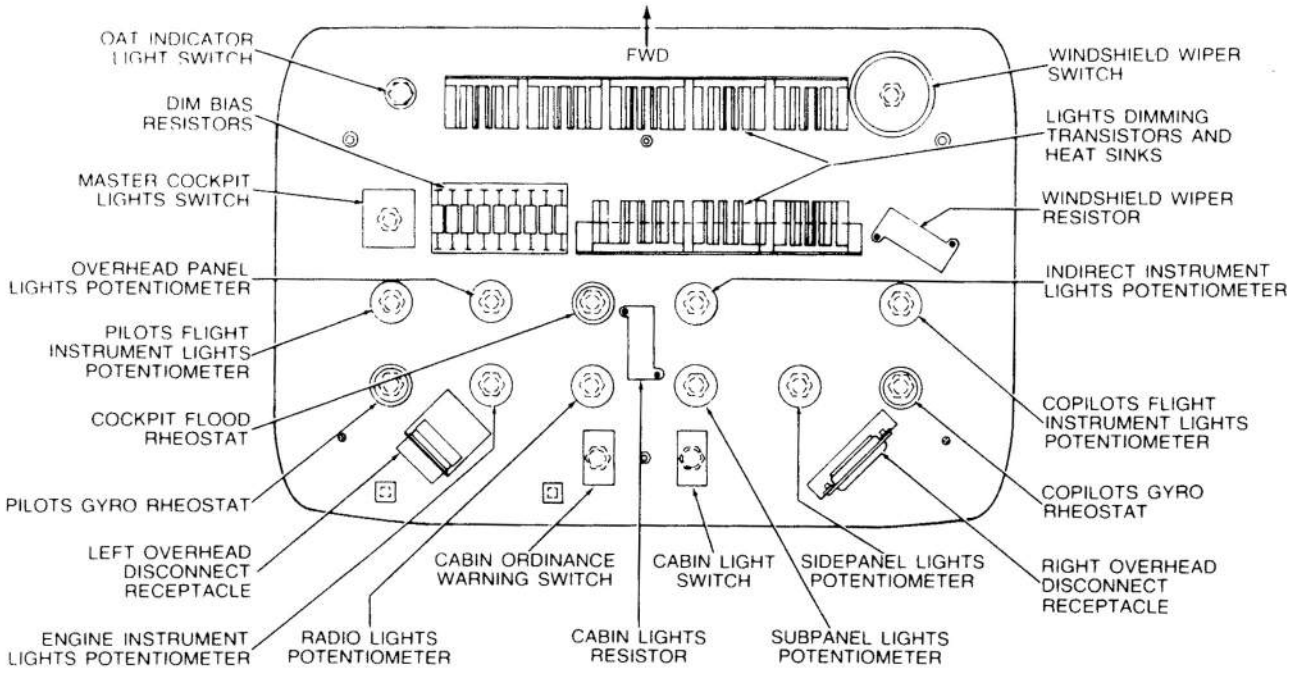


Figure 206
Flight Compartment Center Overhead

ELECTRICAL PANELS AND COMPONENTS
ELECTRICAL AND ELECTRONIC EQUIPMENT RACKS
MAINTENANCE PRACTICES

1. PEDESTAL

A. Forward Face

The following replaceable components are mounted on the forward face of the pedestal (Ref. Figure 201 Zone 243).

Stabilizer position indicator adjusting rheostat

Stabilizer position indicator voltage regulator diode

Standby trim control relays (four each)

Stabilizer trim position indicator resistors (two each)

B. Forward Floor

The forward floor of the pedestal contains the following components which are easily replaceable (Ref. Figure 202 Zone 243):

Main trim "A" up relay

Main trim "A down relay

Main trim "B" up relay

Main trim "B" down relay

2. DUAL FED BUS DIODE PANELS

The dual fed bus diode panels located in the subfloor beneath the pilot's and copilot's Seats are the points of installation of the following equipment (Ref. Figure 203 Zone 131 and 132):

Heat sink (two each)

Current blocking diodes (eight each)

3. PCB CARD RACK ASSEMBLY

Numerous PCB cards are installed in the rack assembly. These PCB cards contain relays and circuitry necessary to control the various systems in the airplane (Ref. Figure 204 Zone 143). The equipment located at this point follows:

Card rack (two each)

Left engine fire detection control amplifier

Right engine fire detection control amplifier

R ANN CONTROL (Caution and Advisory Control) PCB

R ANN WARNING PCB

R ANN CAUTION PCB

R ANN ADVISORY PCB

FLAP CONTROL PCB

R TIME DELAY PCB

BATTERY CHARGE PCB

RELAY CARD NO. 2

RELAY CARD NO. 4

L ANN CONTROL (Warning Control) PCB

L ANN WARNING PCB

L ANN CAUTION PCB

L ANN ADVISORY PCB

COND CLUTCH CONTROL PCB

L TIME DELAY PCB

RELAY CARD NO. 3

NOTE: R and L time delay PCB's each provide time delays for the following systems: Inverters, Nacelle not full, Surface deice, Fire extinguishers.

4. DC POWER DISTRIBUTION CONTROL PANEL

Various electrical system components and secondary controls for the environmental system are mounted on the DC Power Distribution Control Panel which is located in the forward cabin center aisle subfloor (Ref. Figure 205 Zone 143). A list of these components follows:

Condenser blower relay

Left power isolation limiter

Max heat power limiter

Right power isolation limiter

Normal heat power limiter

Fwd vent blower motor relay

Left generator line contactor

Aft evaporator blower relay

Fwd evaporator blower speed control relay

Right Generator line contactor

No. 2 avionic bus relay

No. 1 avionic bus relay

5. ELECTRICAL EQUIPMENT PANEL

The electrical equipment panel, immediately aft of the DC Power Distribution Control Panel in the forward cabin center aisle subfloor, contains landing gear system secondary controls and DC generation system controls (Ref. Figure 206 Zone 143). These controls are listed below:

Left generator control panel

Right generator control panel

Landing gear limiter

Landing gear time delay assembly

External power bus

Landing gear power relay

6. DEICE EQUIPMENT PANEL

The deice equipment panel, as it's name implies, contains most of the secondary controls for the various deice systems in the airplane (Ref. Figure 207 Zone 143). This panel can be found immediately aft of the electrical equipment panel in the forward cabin center aisle subfloor. Equipment mounted in this panel is as follows:

Left propeller deice manual override relay

Right propeller deice manual override relay

Left propeller deice CB

PCB relay assembly

Propeller deice timer

Left lip deice control relay

Right lip deice control relay

Propeller deice shunt

Left lip deice shunt

Left lip deice test relay

Right lip deice test relay

Deice ammeter positive CB

Deice ammeter negative CB

Right propeller deice CB

Right lip deice shunt

7. BATTERY

A. Installation

The battery installation will be found in the forward upper right wing center section (Ref. Figure 208 Zone 611). An access panel retained by screws which are removable from the top of the wing must be removed to expose the installation. Equipment found beneath this access panel is listed below:

Battery

Battery power relay

Current blocking diodes (two each)

Battery charge current shunt

Battery bus circuit breaker

Battery relay circuit breaker

8. VENT BLOWER RESISTORS

Three resistors which control the speed of the forward ventilation blower through the speed control relay are located in the subfloor of the forward baggage compartment. These resistors are easily accessible through the forward baggage compartment and replaceable, should they become damaged (Ref. Figure 209 Zone 121).

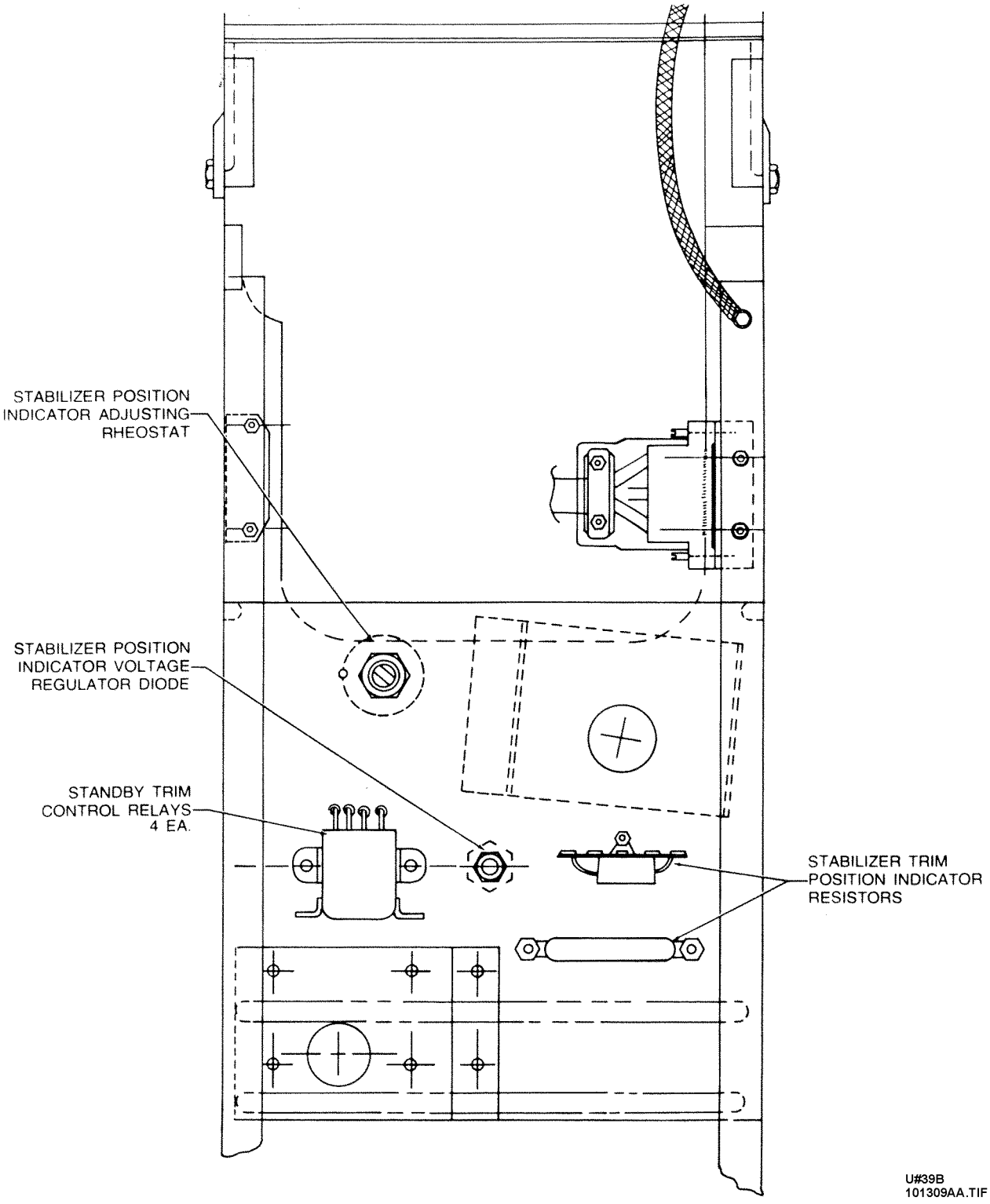


Figure 201
Forward Face of Pedestal

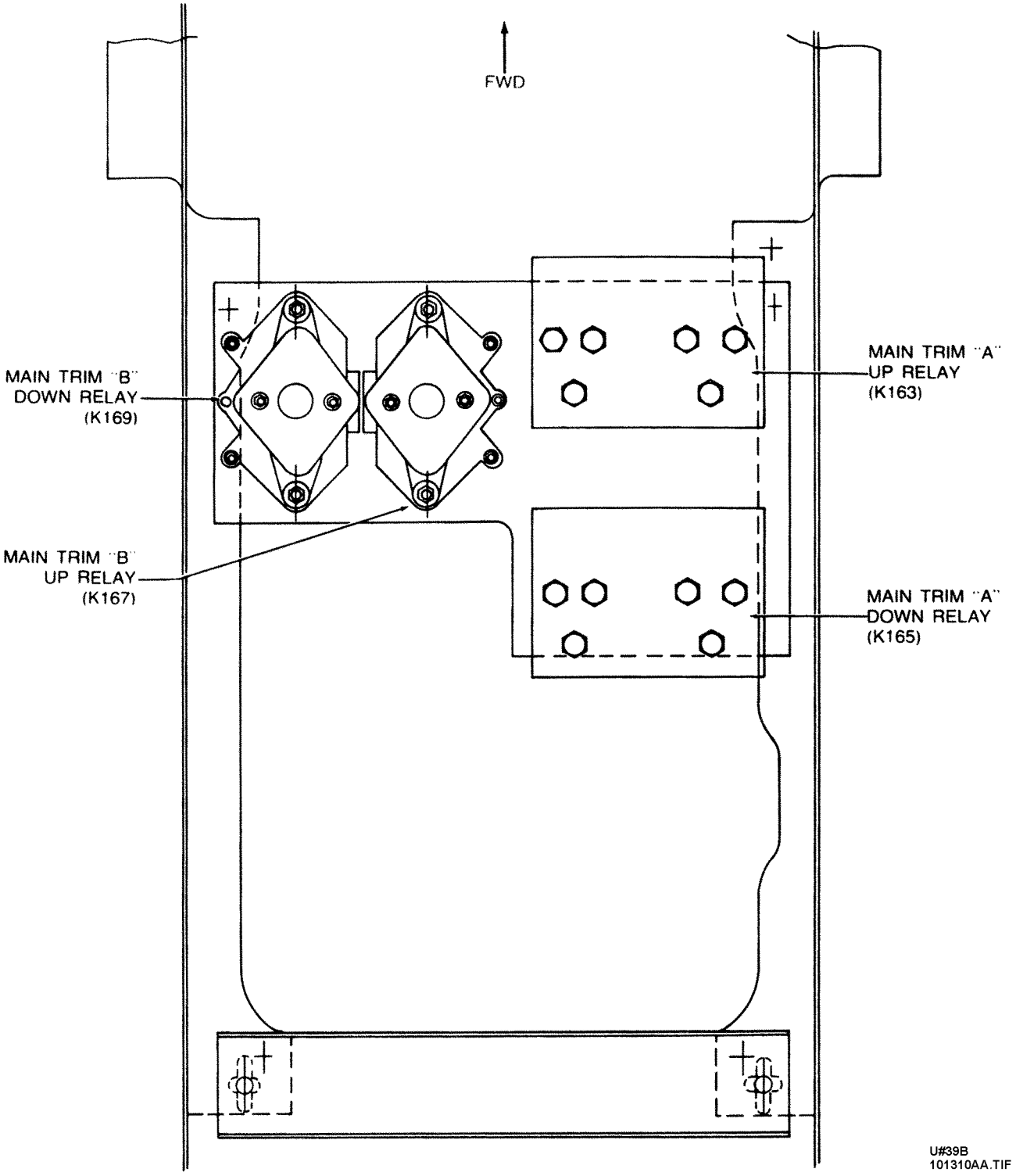


Figure 202
Forward Floor of Pedestal

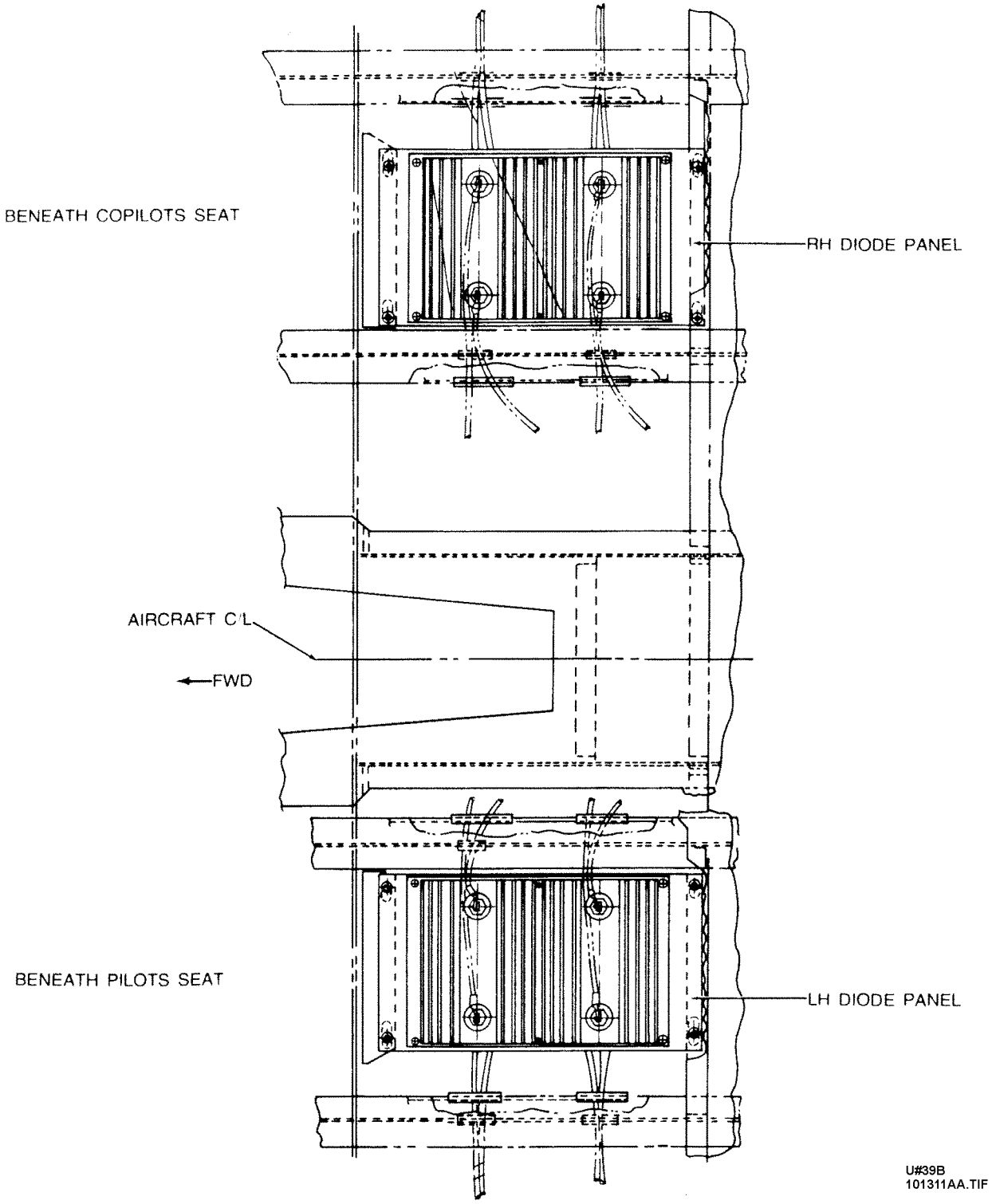
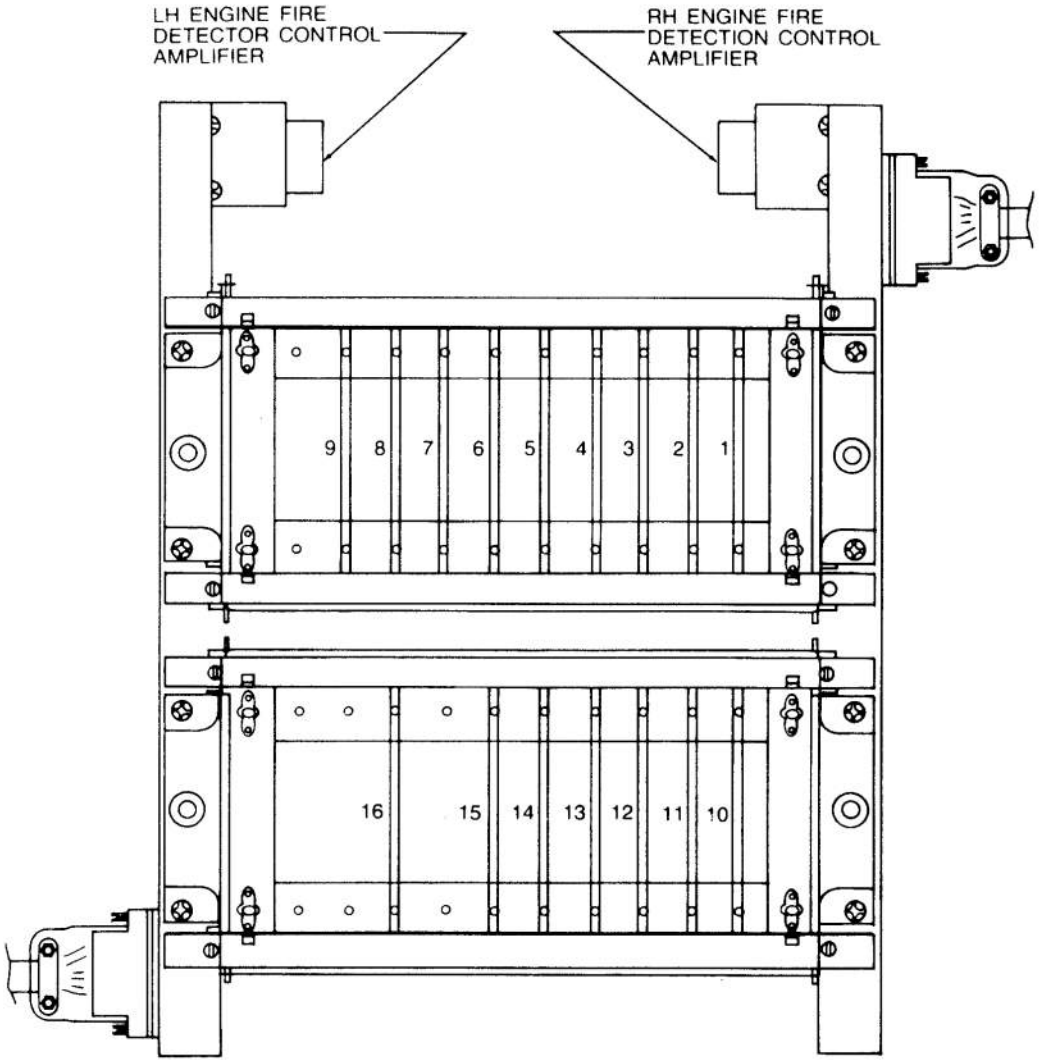


Figure 203
Dual Fed Bus Diode Panel's

1. R ANN CONTROL (CAUTION AND ADVISORY CONTROL) PCB
2. R ANN WARNING PCB
3. R ANN CAUTION PCB
4. R ANN ADVISORY PCB
5. FLAP CONTROL PCB
6. R TIME DELAY PCB
7. BATTERY CHARGE PCB
8. RELAY CARD NO. 2
9. RELAY CARD NO 4
10. L ANN CONTROL (WARNING CONTROL) PCB
11. L ANN WARNING PCB
12. L ANN CAUTION PCB
13. L ANN ADVISORY PCB
14. COND CLUTCH CONTROL PCB
15. L TIME DELAY PCB
16. RELAY CARD NO. 3



NOTE: R AND L TIME DELAY PCB'S EACH PROVIDE TIME DELAYS FOR THE FOLLOWING SYSTEMS: INVERTERS, NACELLE NOT FULL, SURFACE DEICE AND FIRE EXTINGUISHERS.

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Figure 204
PCB Card Rack Assembly Forward Center Aisle Subfloor

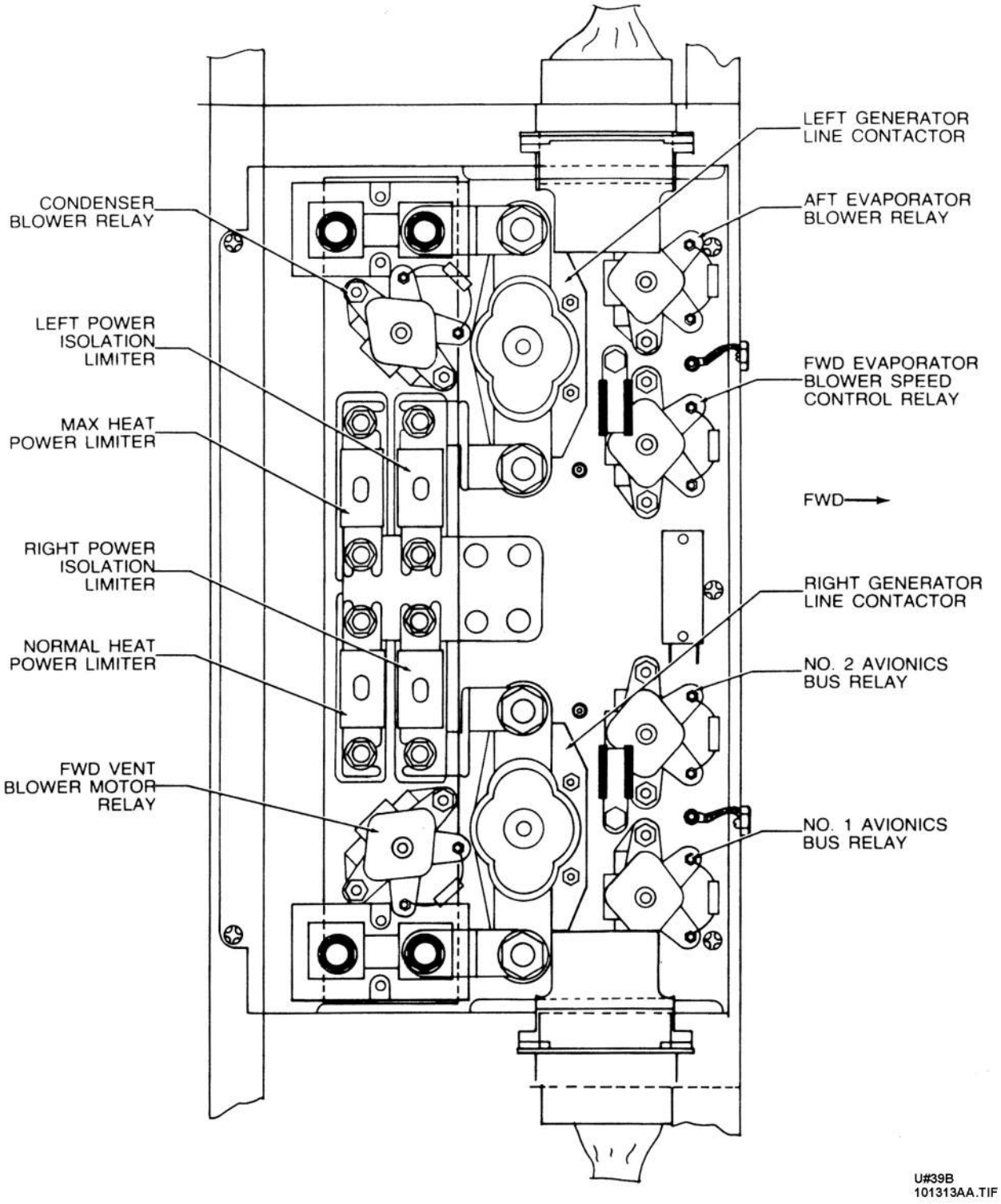

Figure 205
DC Power Distribution Control Panel

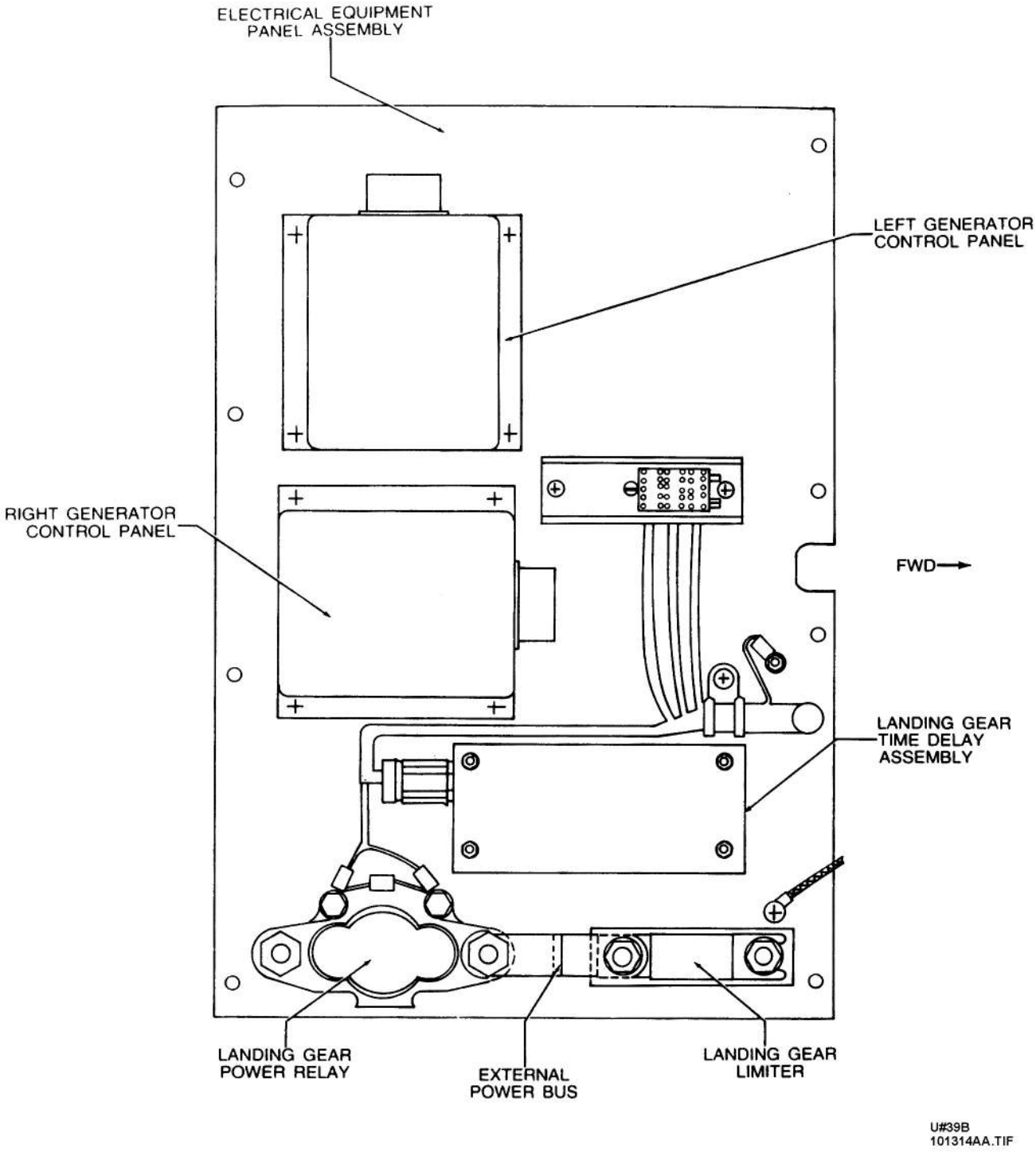

Figure 206
Electrical Equipment Panel Forward Center Aisle Subfloor

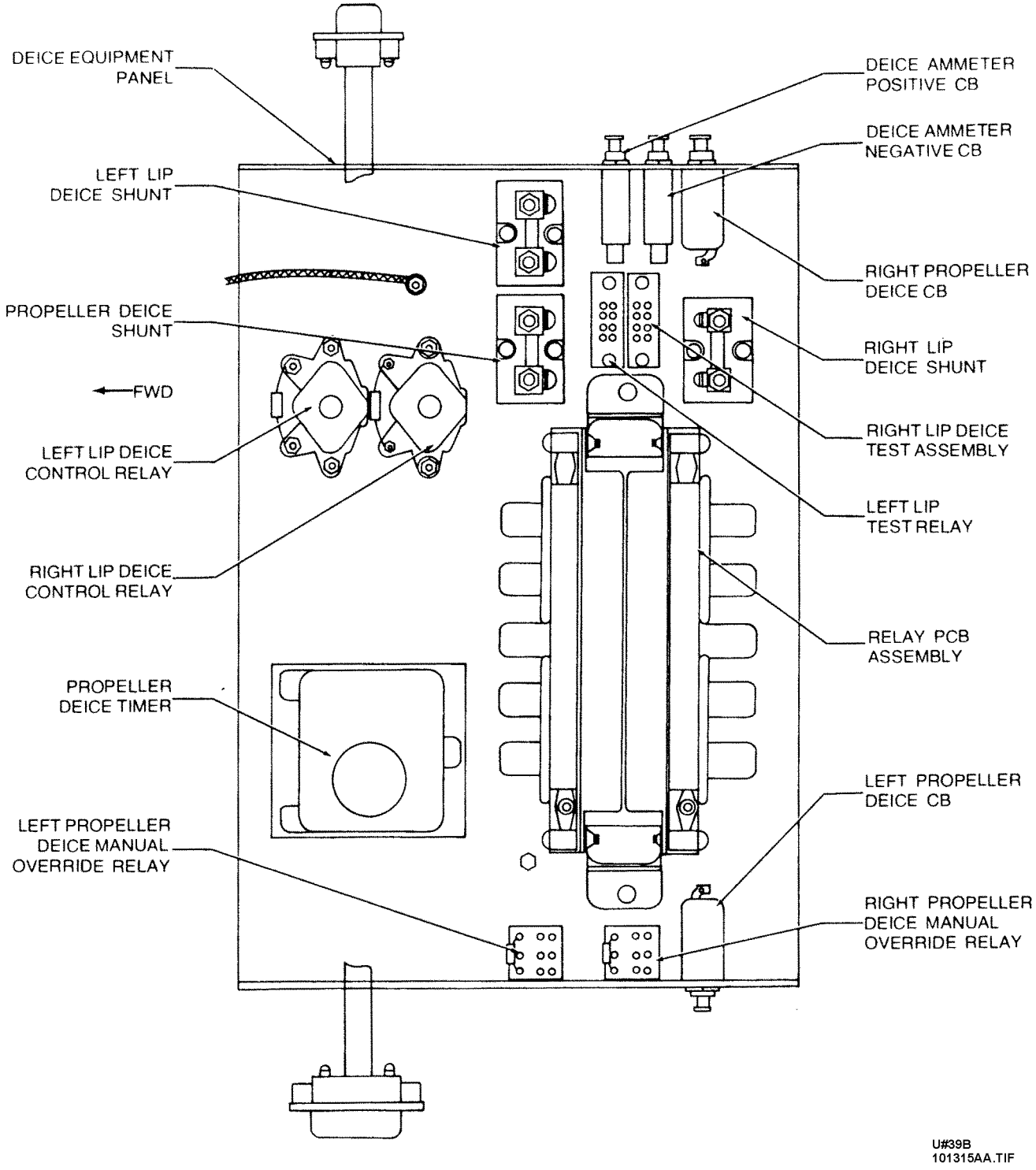

Figure 207
Deice Equipment Panel Forward Center Aisle Subfloor

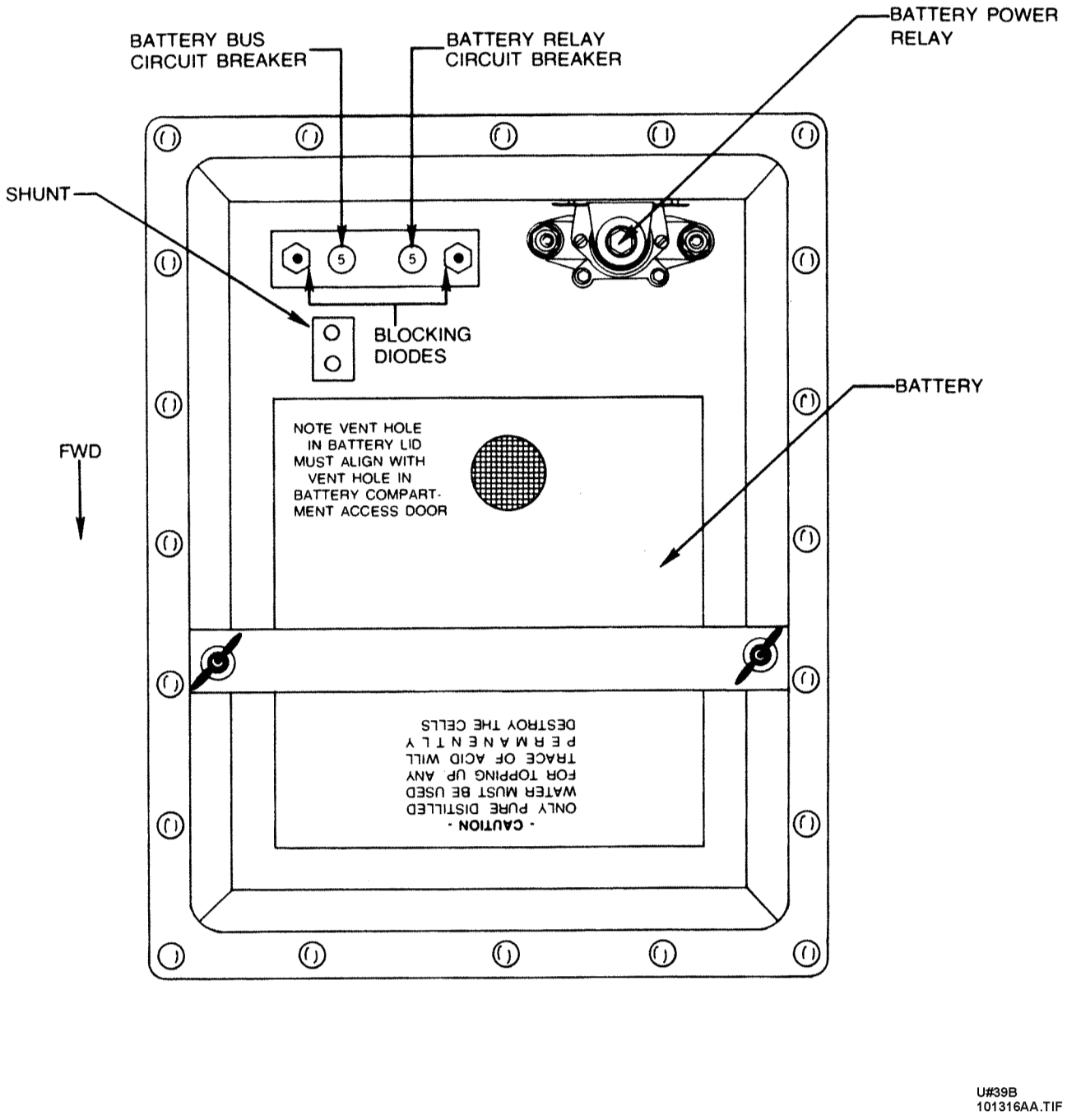

Figure 208
Battery Installation

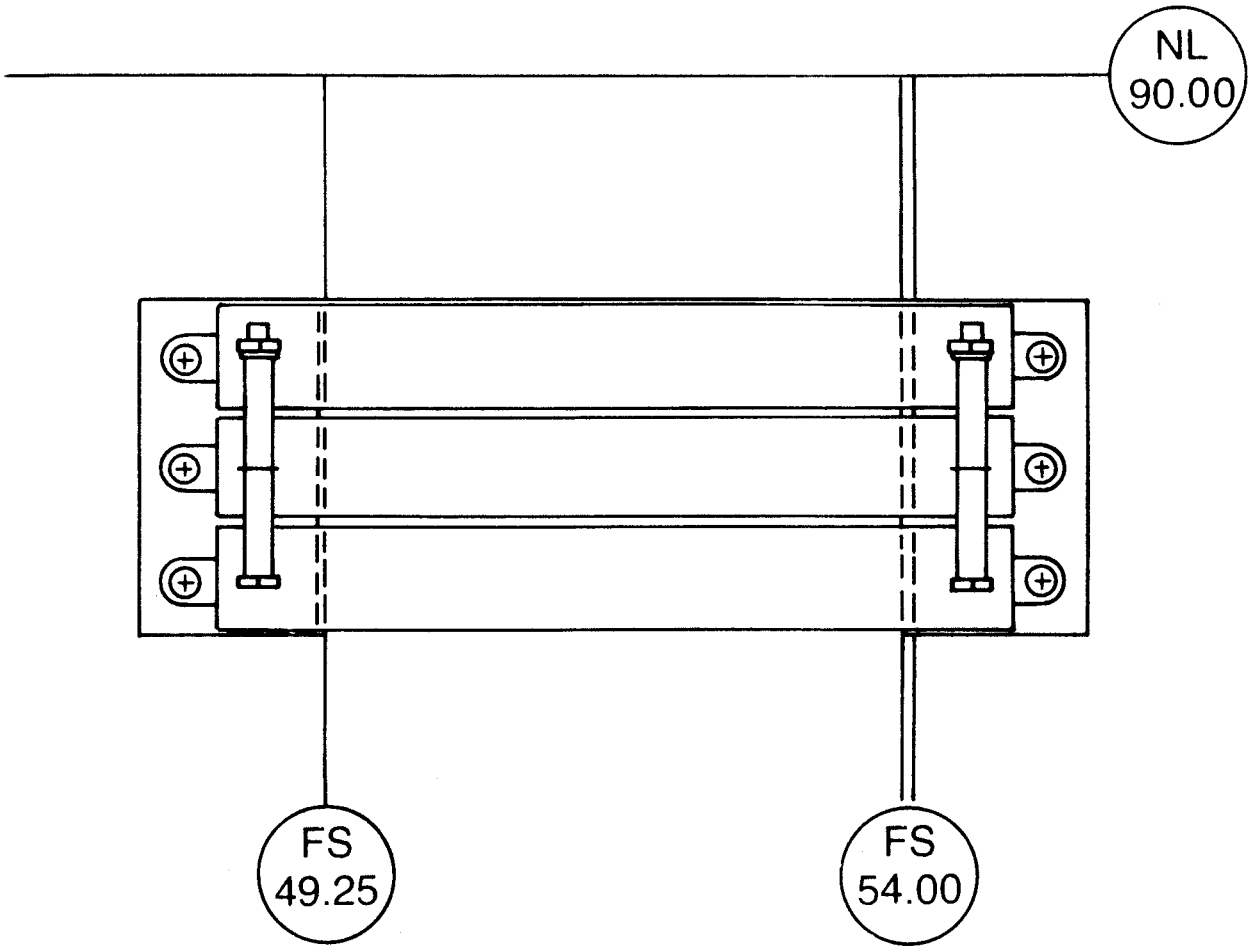

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Figure 209
Vent Blower Resistors

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 50 - CARGO AND ACCESSORY COMPARTMENTS

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General 1

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Baggage Pod (U-1 thru U-164, except U-50) 201
Removal 201
Installation 202
Baggage Pod (U-50, U-165 and After) 203
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CARGO AND ACCESSORY COMPARTMENTS
GENERAL
DESCRIPTION AND OPERATION

1. GENERAL

Those compartments for storage of cargo and various components and accessories. Includes those systems used to load/unload cargo and other cargo related systems.

CARGO AND ACCESSORY COMPARTMENTS
CARGO LOADING SYSTEM
MAINTENANCE PRACTICES

1. BAGGAGE POD

The optional baggage pod provides loading up to 800 pounds which may be distributed into compartments of the pod without disturbing the airplane center of gravity. The compartments are accessible through forward and aft compartment doors. Three flush-mounted snap-lock latches secure each door. While in the upright position, the fold down partition(s) prevents shifting of baggage while the airplane is in motion or may be laid flat to accommodate larger baggage. Interior lighting for the pod is provided automatically when either compartment door is opened. In addition, electrical switches, mounted adjacent to the forward and aft compartment doors, will illuminate a warning light on the annunciator panel whenever a compartment door is open.

2. BAGGAGE POD (U-1 THRU U-164, EXCEPT U-50)

A. Removal

(1) Provide jacks or a suitable support under the baggage pod to prevent damage to the pod when it is lowered.

(2) Disconnect the clamp and connector of the coaxial leads for the DME and transponder antennas (if installed) at the point the leads enter the fuselage.

(3) Remove the carpeting and center aisle floorboard aft of the front spar and locate the terminal strip to which the pod compartment lights and rotating beacon light wiring is attached. This terminal strip is located forward of the plate which covers the mounting point for the rotating beacon on the airplane fuselage. Disconnect all the wiring from the baggage pod to this terminal strip (Refer to the applicable wiring diagram).

(4) Remove the screws securing the pod cover support cable to the fuselage.

(5) Remove the screws and bolts securing the pod to the airplane.

(6) Lower the pod and remove the seal strip. If the airplane is to be put in service with the baggage pod removed, the following Steps should be accomplished before the airplane is flown.

(7) Install the pod attachment screws and bolts in the fuselage.

(8) Locate and remove the plate (on the bottom of the fuselage) covering the rotating beacon light mounting position and store with the baggage pod. Remove the rotating beacon light from the pod and mount it in position on the fuselage. Refer to the applicable wiring diagram and attach the red lead on the beacon light to terminal number 5 of the terminal strip just forward of the beacon light mounting position. The white lead should be connected to terminal number 1 of the terminal strip. Replace the floor covering.

(9) Locate the mounting position for the DME and transponder antennas (if installed) and remove any plug buttons which may be in the mounting holes.

(10) Remove the antennas from the baggage pod and connect the appropriate coaxial leads before securing the antennas to the fuselage. Store the coaxial splices and 90° connector(s) with the baggage pod.

(11) Install a button plug in the hole where the baggage pod wiring is routed into the fuselage and replace the pod cover support cable screws. The above procedures must be reversed when preparing the airplane for baggage pod installation.

B. Installation

(1) Position the baggage pod under the airplane.

(2) Route the wiring for the pod lights into the fuselage and connect to the terminal strip (Refer to the applicable wiring diagram).

(3) Install the seal strip using adhesive (43, Table 6, 91-00-00) or equivalent between the seal strip and the mating surfaces.

(4) Install the attaching screws and bolts.

(5) Connect the pod wiring to the terminal strip. Install the floor covering.

(6) Connect the connector and clamp for the DME and transponder antennas coaxial leads.

NOTE: All airplanes U-152 and after are equipped with attaching components to facilitate field installation of the baggage pod by the user.

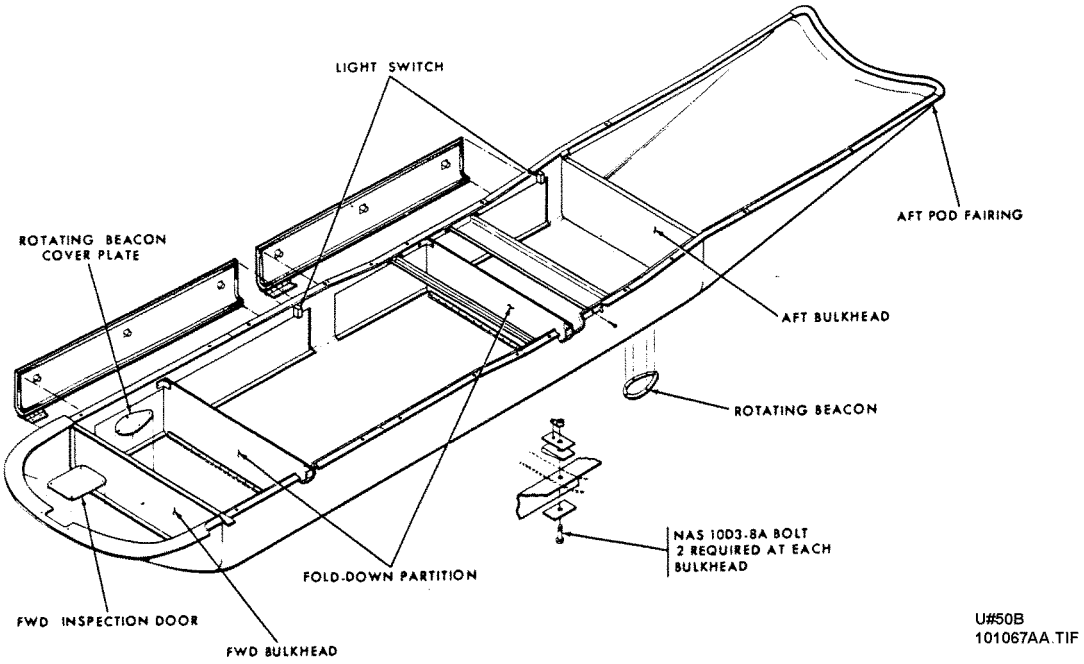

Figure 201
Baggage Pod (U-1 thru U-164, except U-50)

3. BAGGAGE POD (U-50, U-165 AND AFTER)

A. Removal

(1) Provide jacks or a suitable support under the baggage pod to prevent damage to the pod when it is lowered.

(2) At the aft baggage pod fairing, remove the eight attaching screws securing the rotating beacon and disconnect the beacon electrical connector.

(3) Remove the clamps securing the drain tubing at the forward and aft end of the main body of the baggage pod.

(4) Remove the attaching screws securing the forward and aft baggage pod fairings and remove the fairings.

(5) Disconnect the coaxial leads for the DME and transponder antennas (if installed).

(6) Disconnect the baggage pod electrical connector from the fuselage wiring connector.

(7) With the baggage pod supported, remove the eight bolts which attach the baggage pod to the fuselage.

(8) Lower the pod and remove the seal strip. If the airplane is to be put into service with the baggage pod removed, the following Steps must be accomplished before the airplane is flown:

(a) Install the access cover on the fuselage, for the fuselage wiring to baggage pod wiring connector plug.

(b) Install the baggage pod attachment screws and bolts on the fuselage.

(c) Remove the screws and access cover for the rotating beacon from the fuselage and store with the baggage pod.

(d) At the rotating beacon access opening in the fuselage, connect the fuselage electrical beacon connector plug to the rotating beacon electrical connector plug. Position the rotating beacon and install the attaching screws.

(e) Locate the fuselage mounting positions for the DME and transponder antennas (if installed) and remove any plug buttons which may be in the mounting holes.

(f) Remove the antennas from the baggage pod and connect the appropriate coaxial leads before securing the antennas to the fuselage.

(g) Disconnect the static air tubing from FS 21.25 and connect to the static air buttons at FS 33.25.

B. Installation

(1) Remove the rotating beacon attaching screws, disconnect the electrical connector plug, and remove the rotating beacon from the belly of the airplane. Apply sealer (93, Table 6, 91-00-00) between the rotating beacon cover and the edge of the fuselage opening. Position the cover and install the eight attaching screws.

(2) Provide jacks or a suitable support under the baggage pod and align the pod with the attaching bolt holes in the bottom of the fuselage.

(3) Hold the pod firmly in position and bend the seal flange as required to obtain a proper fit with the fuselage.

(4) Install the eight bolts which attach the baggage pod to the fuselage.

(5) Remove the access cover for the fuselage electrical connector from the bottom of the fuselage and connect the baggage pod electrical harness connector plug to the fuselage wiring connector plug.

(6) Install the coaxial leads connector, clamps and antennas for the DME and transponder (if installed).

(7) Position the forward and aft baggage pod fairings and install the attaching screws.

(8) Install the drain tubing and secure with clamps at the tube attach points at the forward and aft ends of the main body of the baggage pod.

(9) Connect the electrical wiring connector plug to the rotating beacon wiring connector plug at the aft pod fairing.

(10) Position the rotating beacon and install the eight attaching screws.

(11) Be certain the instruction plates are installed on the inside of the forward and aft compartment doors.

(12) Disconnect the static air tubing from FS 33.25 and connect to the static air buttons at FS 25.25.

C. Leak Test

Upon completion of the baggage pod installation, perform a leak test as follows:

(1) Using a hose with the nozzle adjusted to a medium firm spray under normal city water pressure, apply a stream of water at a distance of approximately five feet from the baggage pod to simulate in-flight rain conditions.

(2) Inspect the baggage pod for evidence of leakage.

(3) In the event leakage does occur, modify the seal between the fuselage and pod or apply sealer (94, Table 6, 91-00-00) as required to seal the area of the leak.

(4) Repeat Steps (1), (2), and (3) until all leaks are repaired.

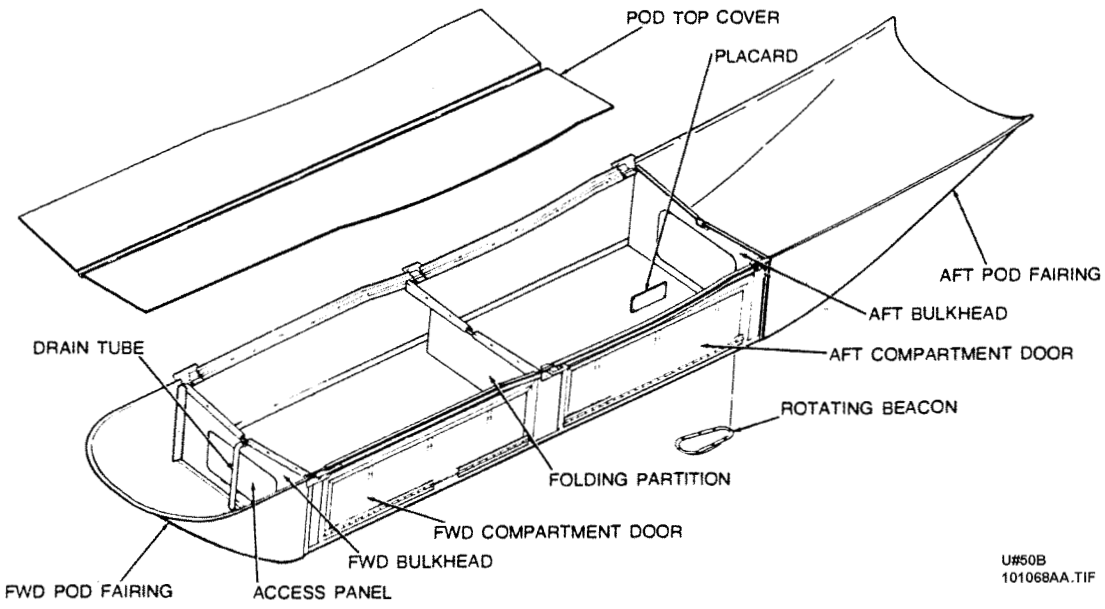

Figure 202
Baggage Pod (U-50, U-165 and After)

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 51 - STRUCTURES

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51-00-00

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Secondary Structure Components 1
Structure Repair 2
Fiberglass Components 2
Repair 2

STRUCTURES
AIRFRAME
DESCRIPTION AND OPERATION

1. AIRFRAME

The MODEL 99 Airliner is a non-pressurized airplane of monocoque construction. Although the wing center section is an integral part of the fuselage, the wing panels outboard of the nacelles are removable. To provide a convenient stairway for boarding the airplane, the cabin entrance door on the left hand side of the fuselage swings down. Dual emergency exits are installed in the forward cabin windows on both sides of the fuselage. Individual passenger seats are provided in a variety of optional seating arrangements.

2. PRIMARY STRUCTURE COMPONENTS

The following primary structure components are essential to the proper function of the airplane. Failure occurring to any of the components would seriously endanger the safety of the airplane and/or the passengers.

(1) Control systems

(2) Engine mounts

(3) Fittings

(4) Coverings of the fuselage, wings, tail surfaces and control surfaces

(5) Wing, tail surface and control surfaces spars

(6) Landing gears and support structure

(7) Auxiliary members used to strengthen or support other members carrying direct loads

(8) Seats and seat support structure

3. SECONDARY STRUCTURE COMPONENTS

The following secondary structural components in the event of failure would require immediate attention, but would not necessarily endanger the safety of the airplane and/or the passengers.

(1) Wing tips

(2) Fairings

(3) Nose cones

(4) Non-structural doors and covers

(5) Furnishings and upholstery (excluding seating)

4. STRUCTURE REPAIR

In general, structural repair methods used on the Model 99 Airliner may be in accordance with AC 43.13-1A AIRCRAFT INSPECTION AND REPAIR MANUAL and AC 43.13-2 AIRCRAFT ALTERATIONS MANUAL. Never make a skin replacement or patch from a material thinner than the original skin. Patches should be of the next thicker material. Some of the skin panels have undergone a chemical milling process for weight reduction. No attempt should be made to patch skins that have been chem-milled; the entire skin panel should be replaced with a matched panel ordered from the applicable Model 99 Airliner Parts Catalog.

5. FIBERGLASS COMPONENTS

A. Repair

Field repair of fiberglass should be limited to those areas where there will not be an excess amount of heat in areas where fiberglass is used and the heat may reach a temperature near 300°F (149°C) or above, a special phenolic resin impregnated fiberglass is used. It is recommended that these parts not be repaired unless there is a shop available which is equipped for this type of fiberglass work.

(1) Large holes and cracks require that the damaged area be cut out and trimmed just beyond the area of damage. If the parts are painted, remove the paint and sand that potion of the part extending at least 2 inches beyond the cutout.

(2) On small holes or cracks, sand lightly using No. 400 sandpaper to remove any raised or rough surface.

(3) Clean with solvent (34, Table 6, 91-00-00) and wipe dry.

(4) Prepare resin (57, Table 6, 91-00-00). Mix per manufacturers instructions.

CAUTION: *Always follow the manufacturers instructions carefully, since mixing of the resin may result in a spontaneous fire.*

(5) Cut fiberglass plies equal to or greater than the number of damaged plies. The first ply should be one inch (1.0") wider and one inch (1.0") longer than the damaged area. Each succeeding ply should be one inch (1.0") narrower and one inch (1.0") shorter than the preceding ply, thus achieving a 1/2 inch step on each side.

NOTE: Each fiberglass ply should be saturated in the resin before applying it to the damaged area to ensure proper bonding.

(6) After all plies have been applied, coat the entire area with resin and allow to cure.

NOTE: Resin, MIL-R-7575, will develop satisfactory handling strength after twelve (12) hours at 70°F.

(7) Clean the area by lightly sanding with No. 400 sandpaper.

(8) Refinish the area as necessary.

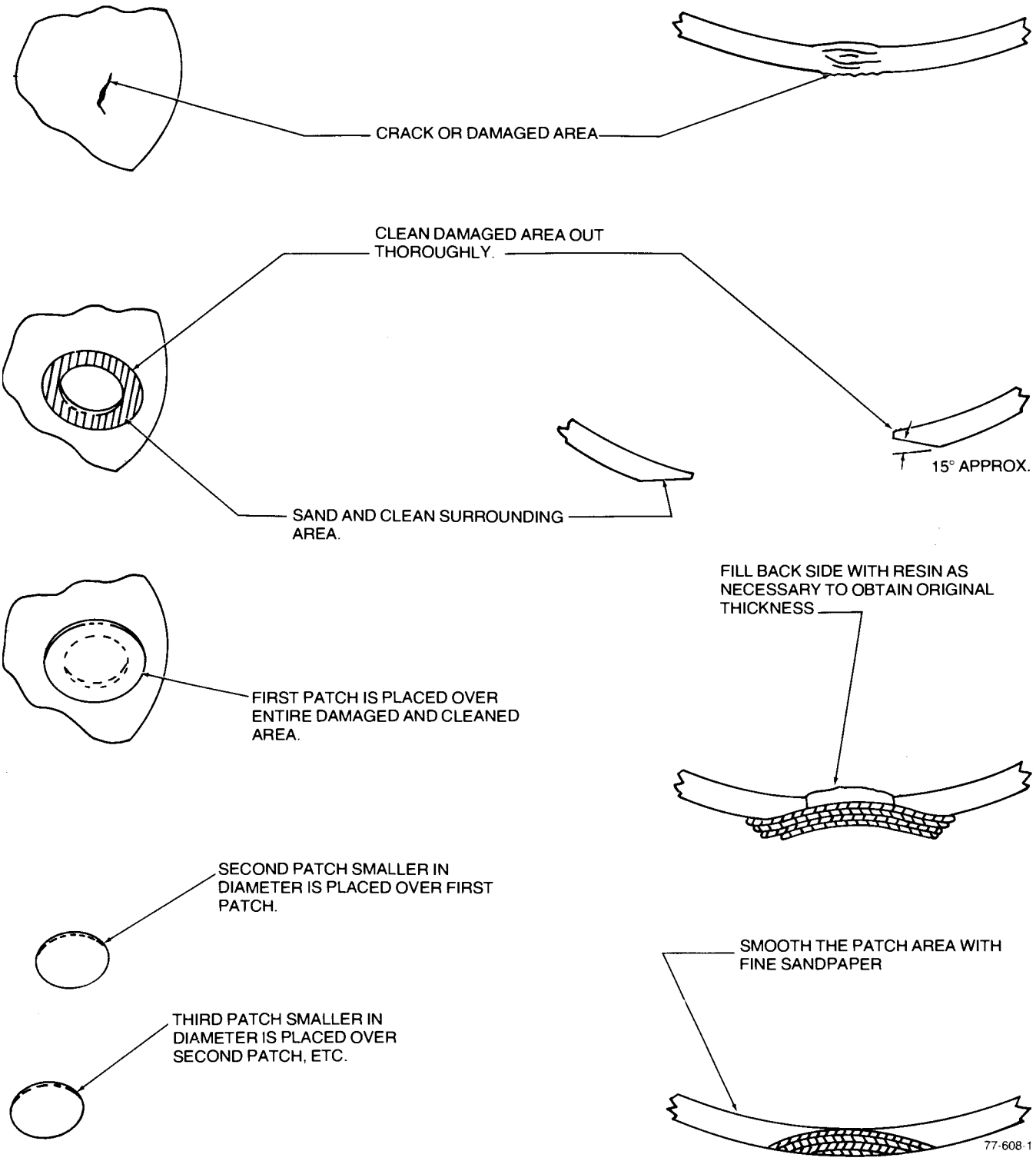

Figure 1
Fiberglass Repair

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 52 - DOORS

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CHAPTER 52 - DOORS

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DOORS
PASSENGER/CREW
DESCRIPTION AND OPERATION

1. DOOR

A swing-down door, hinged at the bottom, provides positive cabin security for flight and a convenient stairway for entry and exit. The standard door is equipped with one manually folding step, located just above the bottom fixed step. On the optional door, two of the three steps are moveable and automatically fold flat against the door in the closed position. A plastic encased cable provides support for the door in the open position, a hand hold for passengers, and a convenience for closing the door from inside the airplane. A hydraulic damper permits the door to lower gradually during opening.

The door locking mechanism is operated by mechanically interconnected inside and outside door handles located in the center of the door. When the handle is rotated, the door latches near the top of the door engage latch plates in the forward and aft door frame. The latches are properly rigged when the indicator pins viewed through the holes in the latching sight gage at the top sides of the door align with the colored bands on the bayonet assembly.

A button adjacent to either handle must be depressed before the handle can be rotated to open the door. This acts as additional safety to aid in preventing accidental opening.

DOORS
PASSENGER/CREW
MAINTENANCE PRACTICES

1. CABIN ENTRANCE DOOR

A. Removal

(1) Remove the hinge seal retaining strip from the door.

(2) Remove the hinge wire from the hinge halves.

(3) Disconnect the snubber assembly.

(4) Disconnect the cable from the fuselage door frame and remove the door.

B. Installation

(1) Position the door and mate the hinge halves.

(2) Install the hinge wire.

(3) Connect the cable to the fuselage door frame.

(4) Connect the snubber assembly.

(5) Install the hinge seal retaining strip with the attaching screws.

C. Fitting and Mounting a New Door

(1) Prepare the door as follows before fitting it to the fuselage.

NOTE: Do not rivet the hinge half to the new door until after the door has been centered in the door frame.

(a) Trim the lower corners of the door skin. To prevent the door from riding on the fuselage upon installation, taper the door skin IN to the first hinge segment.

(b) Loosen the latch rods for the latches in the upper sides of the door to facilitate adjustment of the latches when the door is being fitted to the fuselage.

(c) Clean the seal area of the door and the seal with solvent (34, Table 6, 91-00-00). Apply adhesive (43, Table 6, 91-00-00) to the door and seal and allow to dry. Starting at the top of the door, apply solvent (34, Table 6, 91-00-00) to the adhesive on the door and seal and then press seal into place from top to bottom.

(d) If the new door is not already equipped with an outside door handle, turn the inside door handle to the fully locked position, and lock the outside door handle with the key before installing it. Seal the base of the outside door lock with sealer (103, Table 6, 91-00-00) before installing the attach screws.

(e) Dust the door seal liberally with soapstone before fitting the new door to the fuselage.

(f) Remove all covering from the fuselage hinge half where it is riveted to the fuselage. This material may be folded back inside the fuselage over the door frame.

(g) Remove the upper latch bolts on each side and at the top of the door prior to fitting the door to the fuselage.

(h) After lubricating the door hinge pin with oil (3, Table 6, 91-00-00), work the pin through the fuselage hinge half.

(i) install two 1/4 inch spacers against the forward side of the fuselage door frame. Position one spacer approximately 10 inches below the top of the fuselage door frame and one spacer approximately 10 inches above the bottom of the fuselage door frame.

(j) Install two 0.080 inch spacers on top of the fuselage hinge half. Locate each spacer at a point approximately five inches inboard from each side of the lower door frame. These spacers will shim up the door enough to clear the upholstery material. Trim off all excess door seal so that the door rests on the shims, not the seal.

(2) Secure the hinge half to the new door with a minimum of five sheet metal screws, then position the new door in the fuselage door frame and install the hinge wire.

(3) Close and lock the door. Remove the sheet metal screws from the lower door hinge half.

(4) Working from inside the fuselage, shift the door so that it is squared in the door frame and fits snugly against the 1/4 inch spacers on the forward door frame and the 0.080 inch spacers on the hinge half riveted to the fuselage.

(5) With the door centered and the hinge half on the door mated with the hinge half on the fuselage, install the five sheet metal screws, using the existing pilot holes or drilling new size 40 holes if necessary.

(6) Open the door and trim its upper skin to clear the fuselage skin by 3/32 inch along and below the drop channel above the door. Close the door and recheck for fit.

(7) After the door skin has been properly trimmed and the clearance between the door and forward frame is 1/4 inch and the clearance between the door bottom and fuselage is 0.080 inch, remove the door and drill 24 evenly spaced size 30 holes through the door skin and hinge half, then countersink the holes.

(8) After removing all drill cuttings between the hinge and door skin, coat both sides of the hinge with sealer (103, Table 6, 91-00-00). With the hinge wire in the hinge to maintain alignment and prevent warpage, rivet the hinge to the door.

(9) Install the door on the fuselage but do not cut the hinge wire at this time.

(10) Trim the door as required to prevent the skin from butting against the fuselage (trim the aft door skin first).

(11) Install and adjust the door latches as described in this Chapter under the heading CABIN ENTRANCE DOOR LATCH ADJUSTMENT.

NOTE: Rig the steps as outlined under FOLDING STEP ADJUSTMENT (if folding steps are installed).

(12) After the door has been installed and painted to match the fuselage, lubricate the door seal with a silicone lubricant.

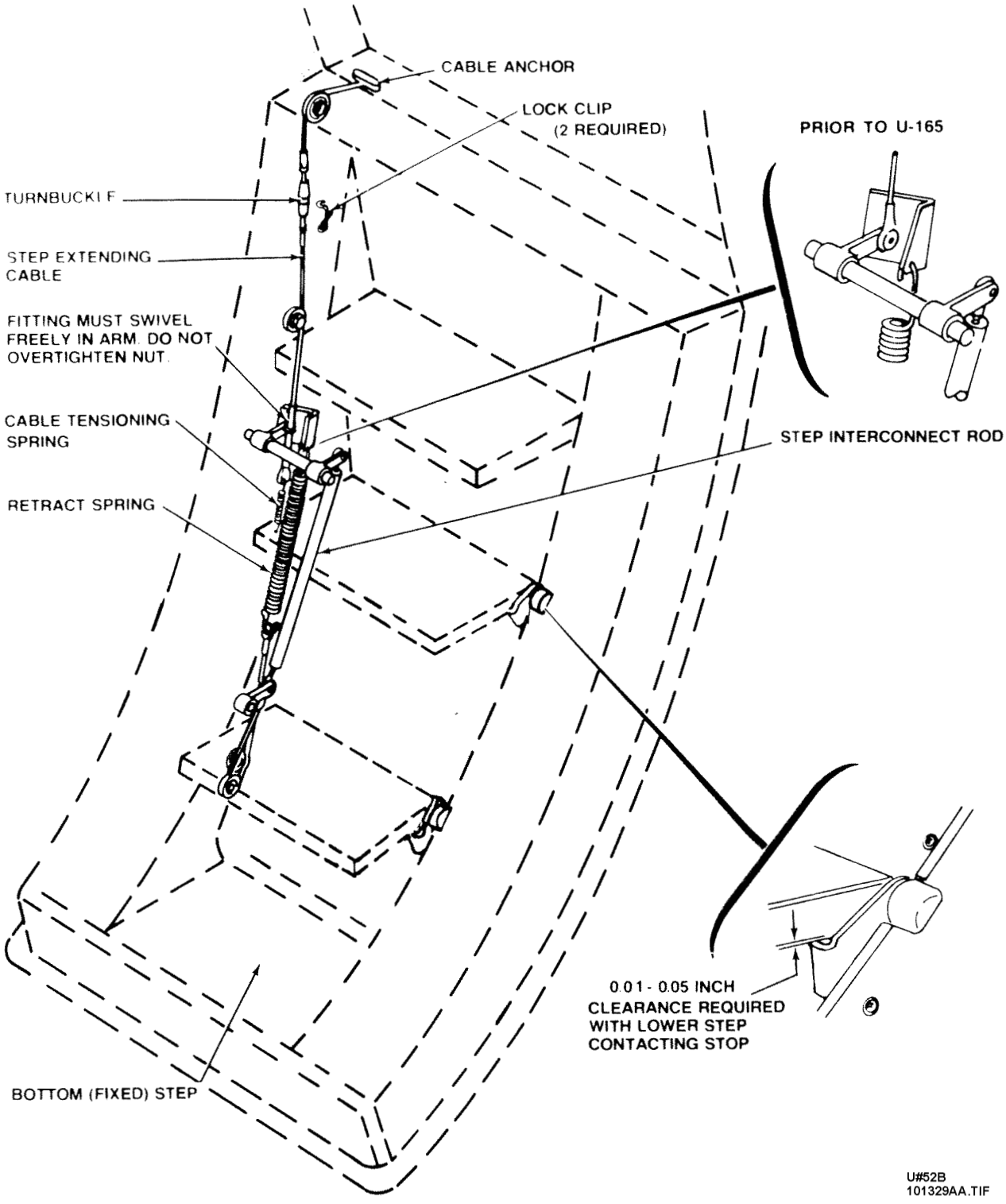


Figure 201
Folding Step Adjustment (Optional)

D. Latch Adjustment

(1) Remove the screws from the door handle.

(2) Remove the screws from the placard and step panel.

(3) Place the outside handle in the LOCK position; disengage the rod from the yoke.

NOTE: The latching mechanism must be in the LOCK position in order to align the clevis pin for removal.

(4) Remove the clevis pin from the yoke and rod.

(5) Place the outside handle in the UNLOCKED position, disengage the rod from the yoke.

(6) Adjust the latches by turning the aft or forward rod to the right (looking from the cam mechanism toward the latch assembly) to decrease the latch protrusion or to the left to increase it. The latch protrusion should measure 29/32 ± 1/32 inch from the door frame facing to the tip of the latch. Place the door latch in the LOCK position when taking the above measurements. After adjusting the door, place the handle in the LOCK position. The latch should be properly rigged when the indicator pins viewed through the holes in the latching sight gage at the top sides of the door align with the colored bands on the bayonet assembly.

2. FOLDING STEP

A. Adjustment

(1) Remove the upholstery panel between the steps.

(2) Raise the door until the turnbuckle in the step interconnect cable is accessible through the lightening holes in the door structure.

(3) Adjust the step interconnect rod until the lower step contacts the stop and the upper step has 0.01 to 0.05 inch gap between the bumper pad and the stop.

(4) Raise the door slightly until the step extending cable is slack.

(5) Remove the turnbuckle locking clips.

(6) With the door open and the handrail installed, adjust the step extending cable as necessary to maintain the gap of 0.01 to 0.05 inch between the upper stop when the lower step contacts its stop.

NOTE: Stand on the bottom (fixed) step while checking the step clearance after adjusting the step extending cable.

(7) Secure the turnbuckle in place with the lock clips.

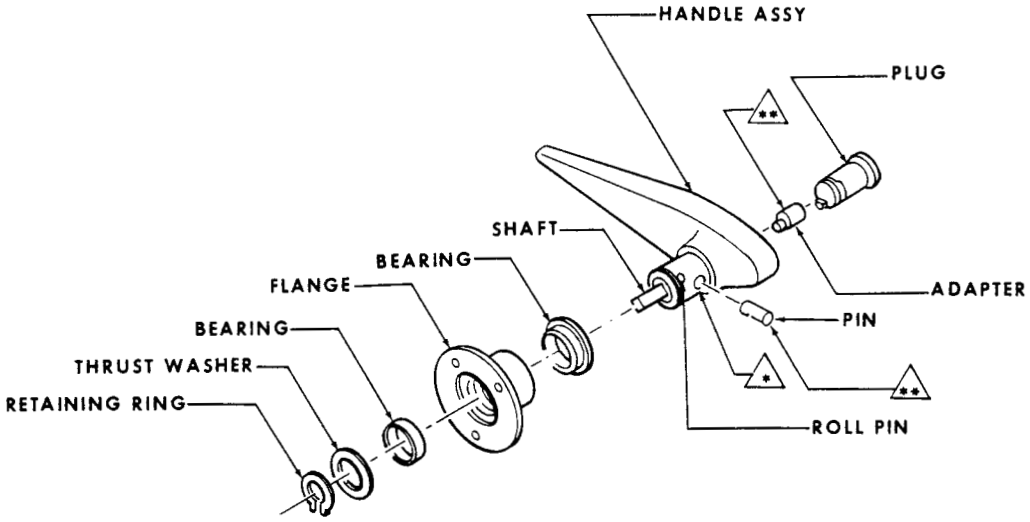
(8) Replace the upholstery panel.

NOTE: Check that the stainless steel bumper pads on each aft corner of each step are securely attached to the step. If the bumper pads are loose or missing, reinstall the bumper pads as follows:

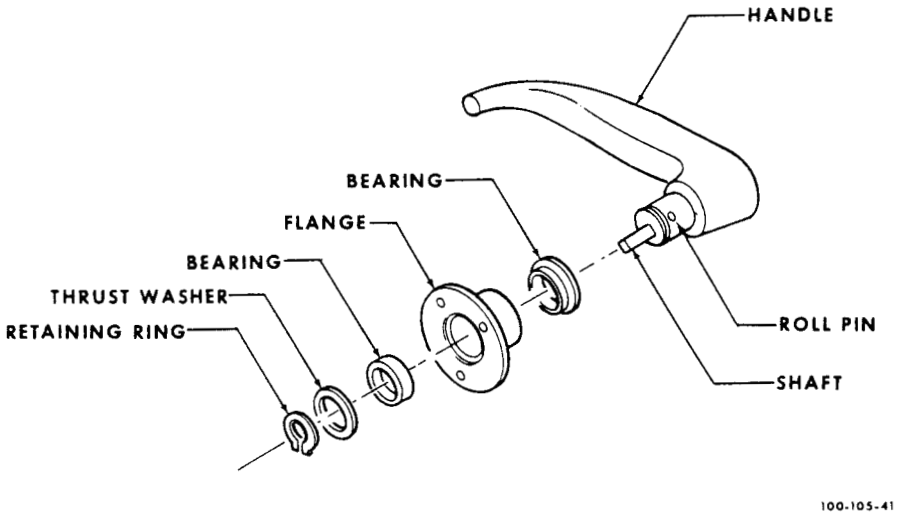
(a) Fabricate a replacement bumper out of 1/6-inch stainless steel if necessary. The replacement should measure 0.56 by 0.76 inch.

(b) Scuff the bonding surface of the bumper and underside of the door with coarse grit sandpaper. Clean the bonding surfaces with solvent (34, Table 6, 91-00-00) or naphtha (33, Table 6, 91-00-00), then wipe the surface clean with a clean dry rag before the solvent can evaporate Apply adhesive (63, Table 6, 91-00-00) to both surfaces. Place a layer of clean gauze or cheesecloth between the step and bumper pad, then clamp the bumper pad to the step until the adhesive has cured.

* To Remove Lock Plug - Insert a wire or pin in hole and pull on key inserted in lock
** Lubricate these parts with MIL-G-21164



Handle Assembly-Airstair Door-Outer



Handle Assembly-Airstair Door-Inner

Figure 202
Handle Assembly - Airstair Door, Outer and Inner

3. CABIN DOOR HANDLE (U-50, U-165 AND AFTER)

NOTE: For information pertaining to prior airplanes, refer to S.I. 0737-105.

A. Removal

(1) Remove the inner and outer door handle assemblies and retain the attaching screws for installation (Ref. Figure 202).

(2) Remove the retaining ring, thrust washer and bearing from the handle shaft.

(3) Remove the flange and bearing from the handle.

(4) Remove the lock plug from the outer handle by inserting a wire or pin in the hole as noted in Figure 202 and pull on the key inserted in the lock. Remove the adapter.

B. Inspection and Lubrication

(1) Inspect all parts for wear. Replace all parts worn beyond serviceable limits.

(2) Clean all parts with lacquer thinner and air dry.

(3) Lubricate the adapter and pin of the outer handle with grease (18, Table 6, 91-00-00) as noted in Figure 202.

C. Installation

(1) Assemble the adapter, lock plug and pin to the outer handle (Ref. Figure 202).

(2) Install the bearing and flange to the handle

(3) Install the bearing, thrust washer and retaining ring to the handle shaft.

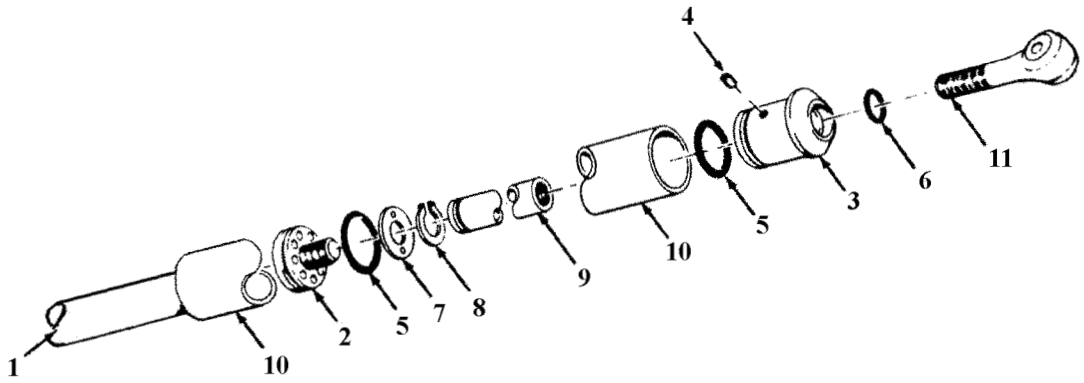
(4) Clean the edge of the exterior handle mounting flange and the door area around it with an oil-free solvent (33, Table 6, 91-00-00). While installing the handles to the door, seal the crack between the door and handle mounting flange by applying sealant (96, Table 6, 91-00-00) using a spatula. putty knife or extrusion gun. The sealer should cure in not more than ten hours with a temperature of 77°F and 50% humidity.

4. CABIN ENTRANCE DOOR DAMPER

A. Hydraulic Snubber - Hawker Beechcraft Model

(1) Attach the barrel rod end to the cabin door supports and the movable piston rod to the supports on the door frame when installing the hydraulic snubber. Shim as necessary with AN960-416 washers to eliminate side play between the rod ends and their supports (Ref. Figure 203).

(2) Service the snubber with hydraulic fluid (9, Table 6, 91-00-00). To service, loosen the cap on the end of the cylinder barrel with the movable rod by removing the three set screws securing the cap to the barrel. Push down on the rod end until the piston is bottomed in the cylinder barrel. Slide the cap up on the rod end and fill the cylinder barrel with hydraulic fluid up to where the cap is to be installed. Secure the cap to the cylinder barrel with the three set screws.



1. Lower Eye
2. Piston Head
3. Cap End
4. Allen-Head Screw
5. Packing
6. Packing
7. Orifice
8. Piston Head Retainer
9. Piston Rod
10. Barrel
11. Upper Rod End

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Figure 203
Cabin Entrance Door Damper (Beech Model)

B. Hydraulic Snubber

Observe the following precautions when installing or servicing the door snubber:

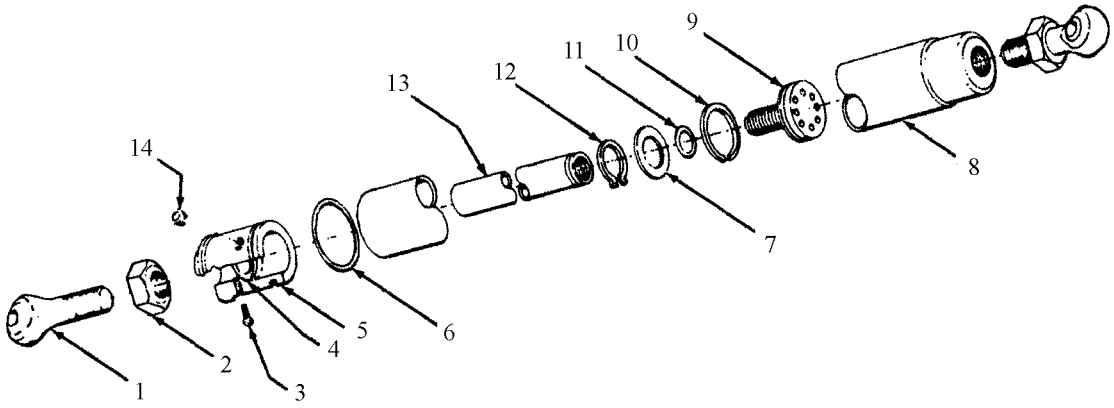
(1) Attach the barrel rod end to the door frame supports and the movable piston rod end to the cabin door supports when installing the hydraulic snubber. Use AN960 washers as shims to eliminate side play between the rod ends and their supports (Ref. Figure 204).

(2) To ensure proper travel of the snubber piston, adjust the rod ends until two threads are visible at each rod end.

(3) Turn the allen-head screw near the end of the barrel clockwise to slow down the action of the snubber or counterclockwise to speed up that action.

(4) Service the snubber with hydraulic fluid (9, Table 6, 91-00-00) as required. Fill the snubber barrel through the hole for the allen-head screw on the end of the barrel nearest the movable piston. Count the number of turns required to remove the screw so that the snubber can be adjusted for the same rate of action upon installation of the screw after servicing.

NOTE: When installing the end cap on the barrel of the snubber, tighten the four retaining screws down until flush with the outside of the snubber barrel. If tightened beyond this point, the screws may bind on the piston rod, rendering it immovable.



1. Rod End
2. Rod End Nut
3. Retaining Screws
4. AN6227-15 "O" Ring
5. Cap End
6. AN6227-23 "O" Ring
7. Orifice
8. Barrel
9. Piston Head
10. Piston Ring
11. AN6290-6 "O" Ring
12. Piston Head Retainer
13. Piston Rod
14. Allen-Head Screw

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Figure 204
Cabin Entrance Door Damper

5. COCKPIT HATCH (OPTIONAL)

An optional cockpit hatch may be installed in the area of the pilot's cockpit window to permit entering and leaving the airplane when the aft cabin area is loaded with cargo in such a manner as to block the cabin entrance door. A detachable ladder, positioned against the fuselage for entering the airplane from the outside at ground level, is normally stored in the nose baggage compartment on the forward side of the aft bulkhead when not in use. The hatch is opened by turning the outside handle at the base of the hatch, or by turning the inside D ring handle in the direction of the arrow. The cockpit hatch is held in the open position by a telescoping rod (Ref. Figure 205).

A. Removal

(1) Open the hatch and disconnect the telescoping hold-open rod, and lower the door to the closed position.

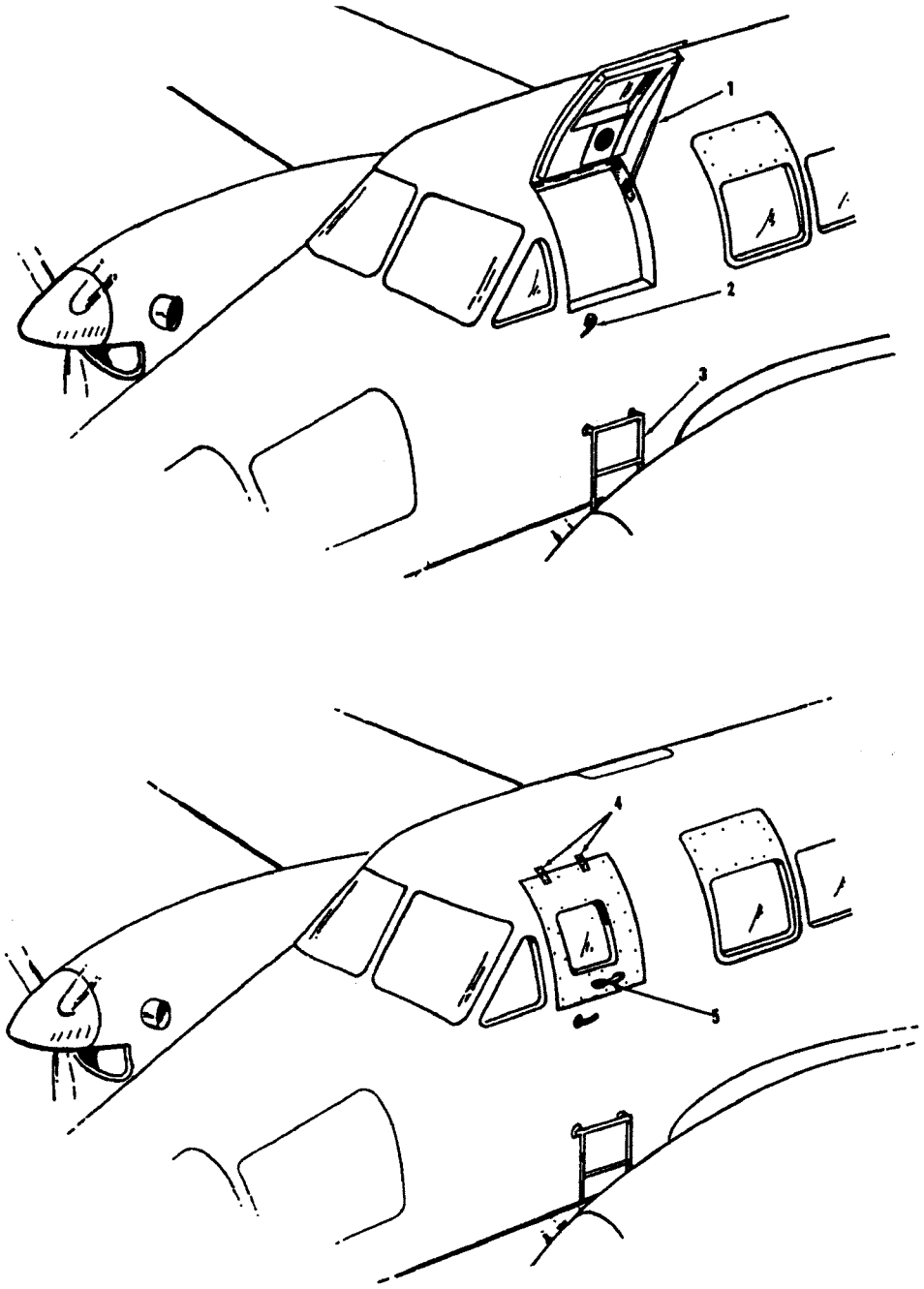
(2) Remove the screws attaching the strap hinges at the top of the hatch and remove the hatch.

B. Installation

(1) Place the door on the airplane in the closed position.

(2) At the top of the hatch, install the strap hinge attaching screws.

(3) Open the hatch and connect the hold-open rod.



1. Telescoping Hold Open Rod
2. Outside Handle
3. Ladder
4. Strap Hinges
5. Hatch Handle

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Figure 205
Cockpit Hatch (If Installed)

6. COCKPIT HATCH LATCH MECHANISM SYSTEM (OPTIONAL)

The latch mechanism system located just beneath the cockpit hatch actuates the latch hooks that engage the fore and aft latch pins mounted on the frame at the bottom of the hatch. The interior handle and the exterior handle of a spring-loaded lockset mesh with the opposing ends of a rotatable tube to which a sprocket is attached. A chain is routed from this sprocket to a smaller sprocket that actuates the forward latch hook. The chain is then routed around the underside of an idler sprocket to another sprocket that actuates the aft latch hook. Each latch hook is mounted on an eccentric cam that forces the hook against the latch pin when the latch is rotated to the closed (locked) position by the handles of the latch mechanism. Chain tension is adjustable by moving the mounting bolt for the idler sprocket up or down in the slotted holes in the channel to which the sprocket is secured. The pressure of the latch hooks against the latch pin can be adjusted by the addition or removal of the shims inserted between the bottom of the latch pin assembly and the hatch frame on which it is mounted.

A. Inspection (If Installed)

(1) Remove the six screws securing the two instruction plates to the upholstery panel below the cockpit hatch. Using a thin edged tool such as a putty knife, pry the ends of the upholstery panel out until it can be snapped free of the two retainers. This will provide access to the latch mechanism.

(2) With the cockpit hatch latched fully closed (locked), check that the latch hooks (14) are over center and apply pressure against the latch pins (20). If necessary, minor adjustments can be made by removing the two attaching screws from the latch pin assembly (20), then adding or removing shims (21) between it and the frame on which it is mounted (Ref. Figure 206).

NOTE: If necessary, shims can be fabricated from shim stock by using an installed shim for a pattern.

(3) Replace latch hooks (14) or latch pins (20) that show extensive wear.

(4) Check bushings (8, 16, and 17) and bearings (3) for wear and replace as required.

(5) Check for the loose play that indicates wear between the hex shaft (11) and its respective sprocket (12) and latch hook and cam assembly (14).

(6) Check that the latch chain (6) is under sufficient tension to properly actuate the latch mechanism (Ref. Figure 206 and 207). Chain tension can be adjusted as follows:

(a) Loosen the nut on the mounting bolt for the idler sprocket (9) (Ref. Figure 206).

(b) Slide the bolt in the slot (7) until chain tension is properly adjusted, then tighten the nut to secure the bolt in place.

NOTE: If proper chain tension cannot be obtained, replace the chain.

(7) Align the upholstery panel between the top and bottom retainers, then bend the panel until it will snap into the retainers when released.

(8) Attach the two instruction plates to the upholstery panel and the underlaying structure with the six retaining screws.

7. INTERIOR LATCH HANDLE (IF INSTALLED)

A. Removal

(1) Remove the three screws securing the handle (1) in place (Ref. Figure 206).

(2) Disengage the handle (1) from the lockset assembly and remove handle with spacer (2).

B. Installation

(1) Insert the handle spindle through the spacer (2) (Ref. Figure 206).

(2) With the latch mechanism in the unlatched position, hold the handle (1) so that the pin hole is visible with the handle positioned to the right.

(3) Hold the handle stationary in this position while rotating the spindle 1-1/4 turns clockwise, then insert the spindle in the hub while maintaining tension against the spring.

(4) To properly latch the hatch, the handle (1) must be rotated counterclockwise to the fully opened position, then rotated into the latched (locked) position. The latch mechanism cannot be properly latched until the handle has been moved to the fully open position first. The latch mechanism should be felt going over center into the latched position and the handle should align with the locked position indicator on the placard behind the handle (Ref. Figure 206, Detail E).

(5) Check that both latch hooks (14) are over center and exerting pressure downward against the latch pins (20).

8. EXTERIOR LATCH HANDLE (IF INSTALLED)

A. Removal

(1) Remove the three screws securing the handle (19) to the fuselage (Ref. Figure 206).

(2) Disengage the handle (19) from the lockset and remove it and the spacer (2).

B. Installation

(1) Insert the handle (19) shaft through the spacer (2) (Ref. Figure 206).

(2) With the latch mechanism and handle in the open position, align the handle and fuselage mounting holes, then insert the handle shaft until it engages the lockset assembly, and install the three mounting screws.

(3) To properly lock the hatch, the latch handle (19) must be rotated counterclockwise to the fully opened position, then rotated into the latched (locked) position. The latch mechanism cannot be properly latched unless the handle has been moved to the fully open position first (the latch mechanism should be felt going over center into the latched/locked position).

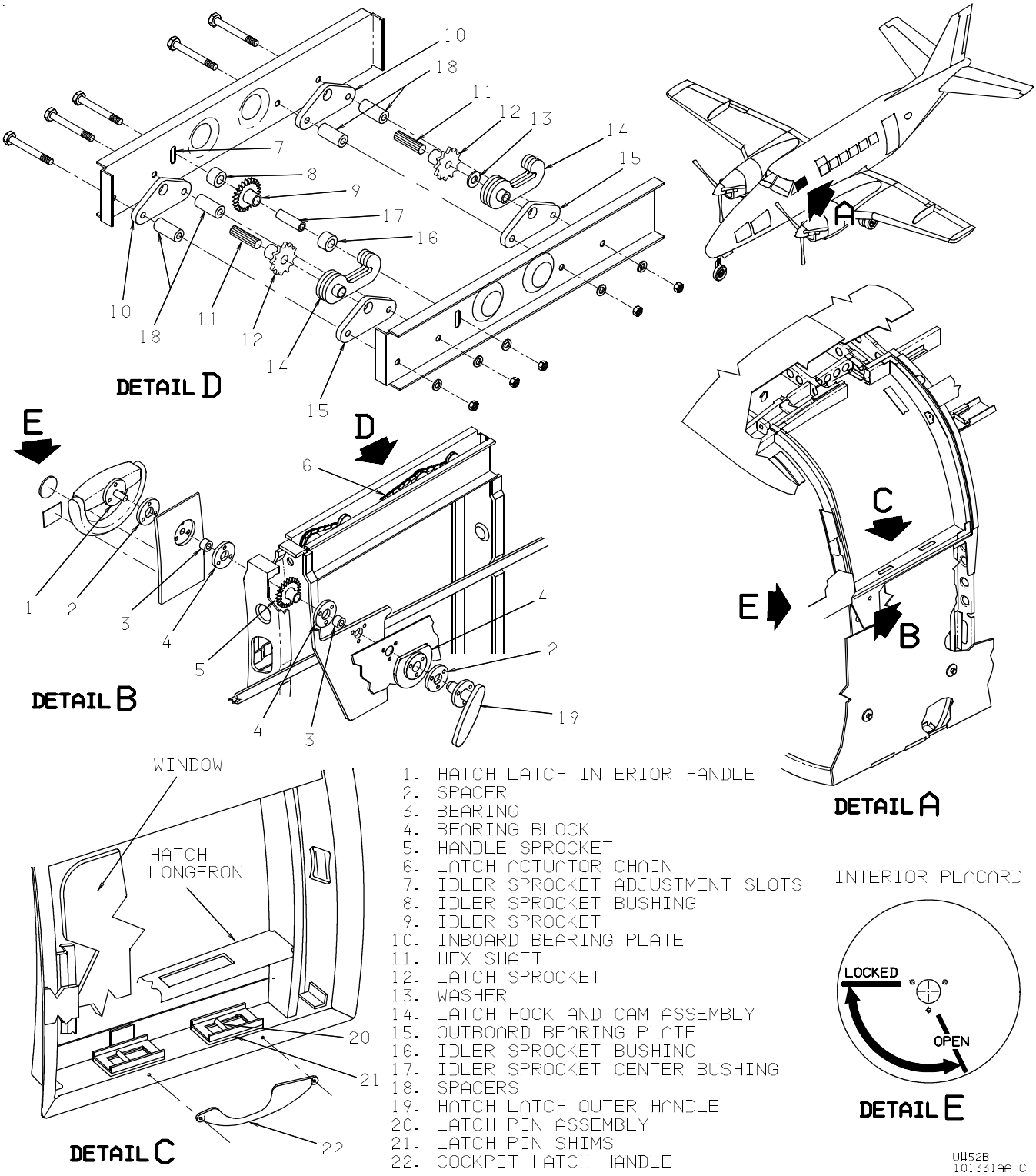

Figure 206
Latch Mechanism (If Installed)

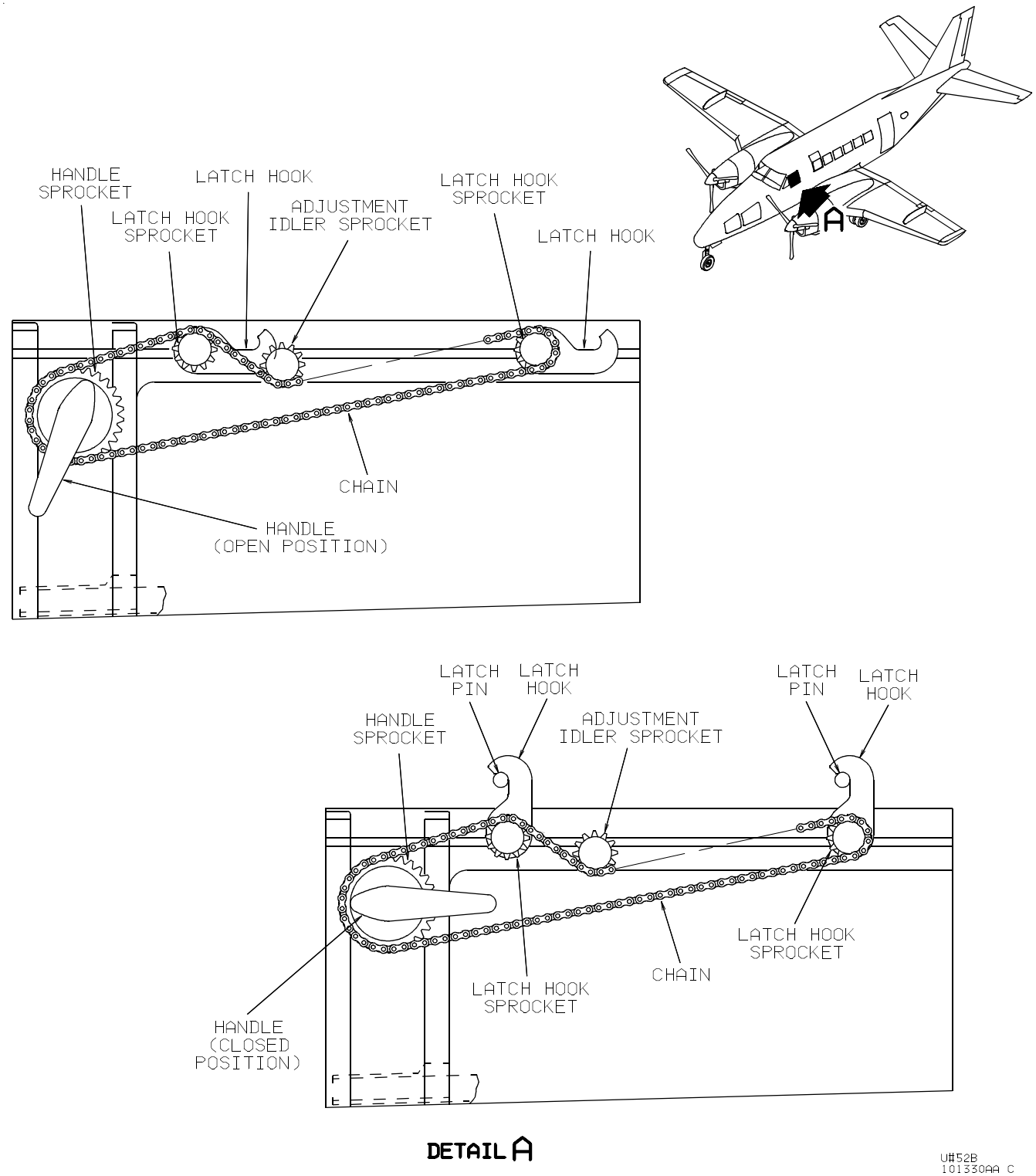

Figure 207
Latch Chain Mechanism (If Installed)

DOORS
EMERGENCY EXIT
MAINTENANCE PRACTICES

1. EMERGENCY EXIT DOOR

The emergency exit doors are located on both sides of the passenger cabin area in the forward cabin windows The doors are released from inside the cabin by a pull down handle, and on the outside by a flush mounted pull-out handle The flush mounted pull-out handles provide access to the cabin area from the outside. The emergency exits can be locked with a key from inside the cabin to prevent opening from the outside In an emergency, the inside actuator handle will override the lock mechanism and allow the doors to open.

A. Removal

(1) Unlock the handle mechanism (Ref. Figure 201).

(2) Pull down the handle to unlatch the door.

(3) Swing the exit door out and lift it away from the fuselage.

B. Installation

(1) Position the door on the fuselage (Ref. Figure 201).

(2) Pull the door flush with the fuselage and lock it into the latched position by pushing up on the actuator handle.

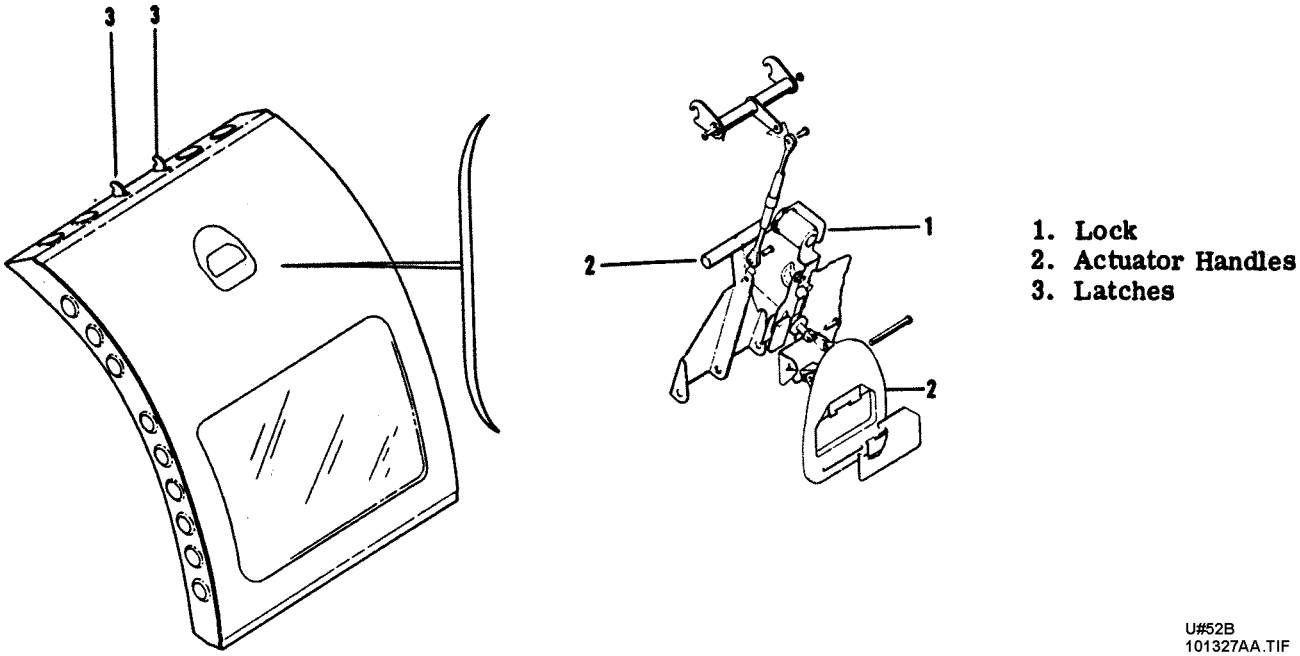

Figure 201
Emergency Exit Doors

DOORS
EMERGENCY EXIT SIGNS
MAINTENANCE PRACTICES

1. EMERGENCY EXIT SIGNS (U-50, U-165 AND AFTER)

Emergency exit signs are attached to the headliner above the airstair door and between the LH and RH cabin emergency exit doors. Each placarded EXIT sign contains two light bulbs, P/N GE354 and two D cell batteries (104, Table 6, 91-00-00) or equivalent. A switch placarded AUTOMATIC-MANUAL is located on the face of each EXIT sign. The signs are impact activated when the AUTOMATIC-MANUAL switch is placed in the AUTOMATIC position. The signs illuminate when the switch is placed in the MANUAL position. The batteries should be tested in this manner monthly and replaced at six month intervals. The emergency exit sign switch should remain in the armed AUTOMATIC position at all other times.

CAUTION: *The emergency exit sign assemblies are NOT interchangeable.*

A. Battery Replacement

The emergency exit signs are attached to the headliner above the airstair door and between the LH and RH cabin emergency doors. The batteries should be replaced every six months as follows:

(1) Remove the four screws attaching the lens to the EXIT sign assembly.

(2) Remove the batteries, two per sign assembly, four per airplane.

(3) Install new D cell batteries (104, Table 6, 91-00-00) as placarded in the battery holder in the sign assembly.

(4) Position the lens on the sign assembly and install the four attaching screws.

(5) Place the AUTOMATIC-MANUAL switch in the MANUAL position to test the sign for proper operation and illumination. Return the switch to the AUTOMATIC position after testing.

DOORS
CARGO
MAINTENANCE PRACTICES

1. CARGO DOOR

For easier handling of cargo, a cargo door may be installed just forward of the cabin entrance door on the left hand side of the fuselage. The door opens forward on two hinges, when the two aft corner locking pins are retracted, providing a large opening for loading and unloading cargo A door hold-open rod, attached to the lower portion of the door, positions in an eye fitting on the lower edge of the door frame to hold the door in the open position. A rubber seal installed around the door on three sides provides a tight seal when the door is in the closed position.

A. Removal

To remove the cargo door, place the door in the open position. Remove the securing bolts from the hinges and remove the door.

B. Installation

Installation of the cargo door is made by placing the door to the hinges in the open position and installing the securing bolts.

2. AFT BAGGAGE COMPARTMENT DOOR

The aft fuselage baggage compartment door is located on the right hand side of the aft fuselage area, just forward of the empennage. The baggage door is opened by turning a D shaped handle and is held in the open position by a folding support brace. The latching mechanism is similar to the cabin entrance door, which incorporates a bayonet type latching mechanism. The door also has a locking mechanism installed which is locked by a key from the outside of the airplane.

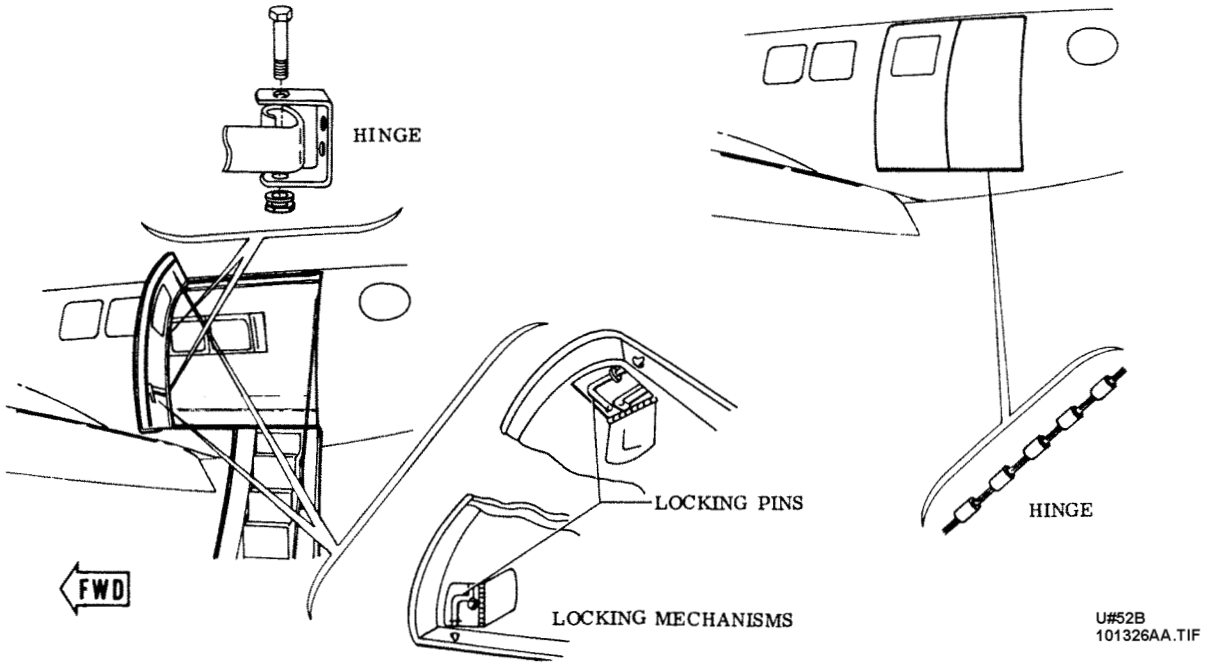

Figure 201
Cargo Door

3. AFT BAGGAGE DOOR

A. Removal

(1) Open the door to the full open position.

(2) Remove the hinge wire from the hinge halves at the top of the door and fuselage.

(3) Lower the door until the support brace can be removed from the nylon retainer.

B. Adjustment

(1) Remove the upholstery panel from the door facing.

(2) Adjust the individual bayonet length at the flat of the eye-bolt as shown in Figure 202.

NOTE: Keep tension on the bayonet actuator springs, and maintain sufficient bayonet length through the door frame to allow the door to close securely.

(3) Install the upholstery panel using the existing clips.

C. Installation

(1) Insert the support brace in the nylon retainer and position the door hinge half to the mating hinge half on the fuselage and install the hinge wire.

(2) Test the door latching mechanism for operational smoothness by opening and closing the door several times, actuating the mechanism each time. If the door does not fit securely in the closed position, remove the door and readjust the bayonets.

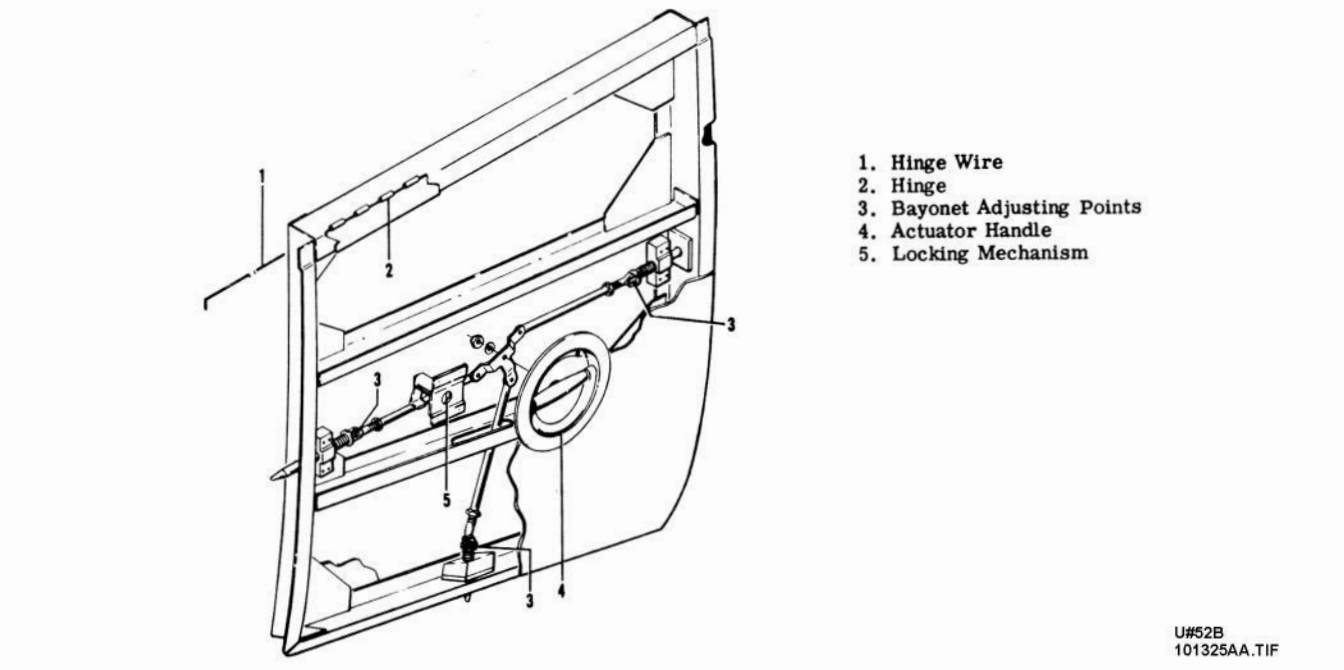

Figure 202
Aft Baggage Door

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MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 55 - STABILIZERS

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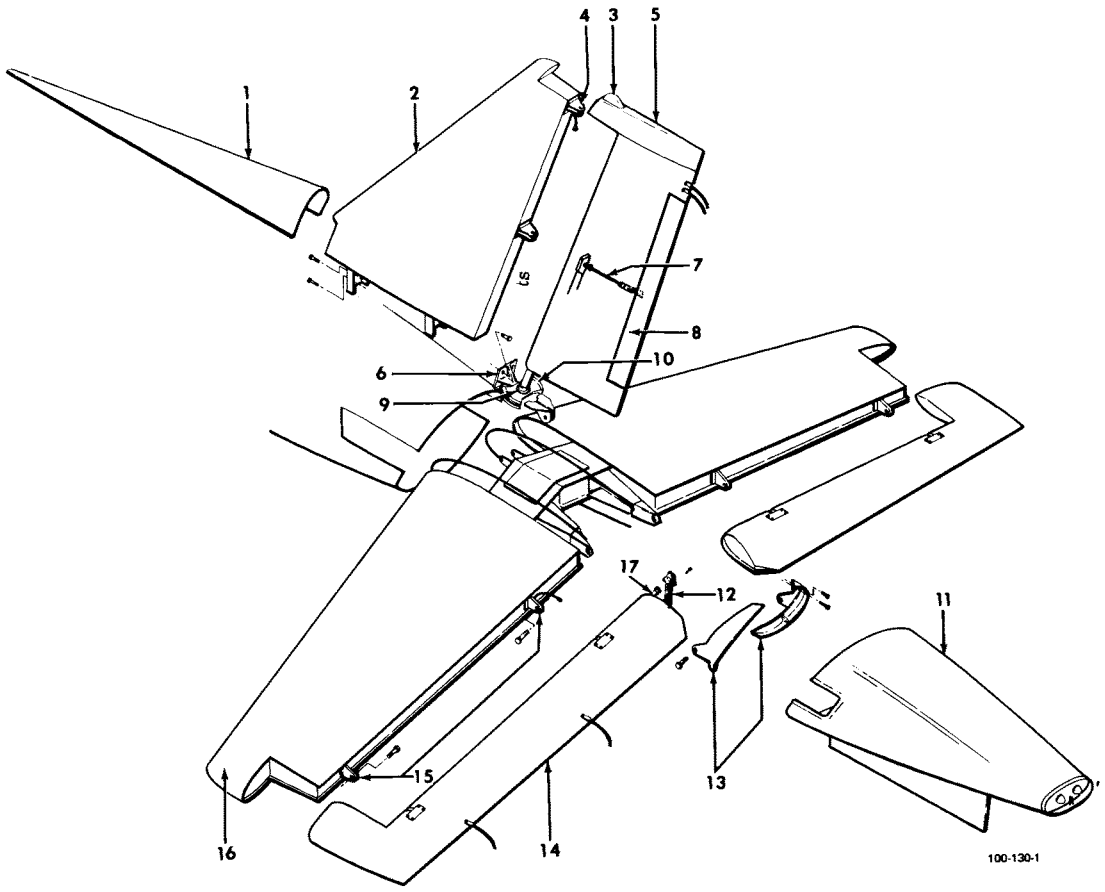
STABILIZERS
EMPENNAGE
DESCRIPTION AND OPERATION

1. GENERAL



1. Dorsal Fin
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12. Elevator Torque Tube
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15. Elevator Hinge Bracket
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17. Elevator Torque Tube

Figure 1
Empennage

STABILIZERS
HORIZONTAL STABILIZER
MAINTENANCE PRACTICES

1. HORIZONTAL STABILIZER

A. Removal

(1) Remove the tail cone.

(2) Remove the elevators as described under the heading ELEVATOR REMOVAL in Chapter 27.

(3) Disconnect the lines to the deicer boots.

CAUTION: *Be certain the stabilizer is properly supported at the aft end during actuator removal. When the attach bolts are removed from the stabilizer actuator, the stabilizer, being heavier at the aft end, is free to swing rapidly to its full up position, possibility incurring damage to the fuselage fairing pan.*

(4) Remove the attach bolts, washers, nuts and cotter pins from the stabilizer actuator.

(5) Remove the bolts, washers, and nuts from the stabilizer hinge points.

(6) Remove the six bolts and washers (four places) from the horizontal stabilizer hinge bracket.

(7) Remove the stabilizer by sliding it aft out of its positioning slot in the fuselage.

B. Installation

(1) Position the horizontal stabilizer in the slot of the aft fuselage and attach the stabilizer hinge brackets to the aft fuselage with the six bolts and washers (four places). Torque the bolts 50 to 70 inch-pounds.

NOTE: On airplane serials U-58 thru U-79, safety wire the drilled bolt heads.

(2) lnstall the two bolts (bolt heads inboard) through the stabilizer pivot support and stabilizer hinge brackets. Install a washer under the bolt head, one on the outboard side of the stabilizer pivot support and washers as required (maximum of three) under the nut.

(3) lnstall the stabilizer actuator as described under the heading HORIZONTAL STABILIZER ACTUATOR INSTALLATION in Chapter 27.

(4) Connect the lines to the deicer boots.

(5) lnstall the elevators as described under the heading ELEVATOR INSTALLATION in Chapter 27. The minimum clearance between the elevator cutout and horizontal stabilizer hinge is 0.06 inch.

(6) lnstall the tail cone.

(7) Check for proper travel as indicated in Chapter 27.

STABILIZERS
ELEVATOR
MAINTENANCE PRACTICES

1. ELEVATOR

A. Balancing

When the elevator control surface is being repainted, suspend it by the trailing edge so that excess paint will drain toward the leading edge. After any repainting or repair, the finished surface should be check balanced to ensure that its static moment about the hinge line is within the manufacturers prescribed limits. The static moment is the total unbalanced weight of the elevator control surface multiplied by the perpendicular distance from its hinge center line to the center of gravity, when the chord line is horizontally level. The weight is measured in pounds and the distance in inches. The static moment of a 100 percent balanced elevator control surface is 0.0 Ib. A tail-heavy surface exhibits static under balance. A nose-heavy surface exhibits static overbalance (Ref. Figure 201).

NOTE: A completed, painted elevator, with the horn and static wicks installed, must have a static balance within the range of 19.0 to 24.6 inch-pounds tail-heavy.

If the aluminum elevators are used to replace the magnesium elevators originally installed on U-1 thru U-146 without Kit No. 99-4018 installed, the old and new sets of elevators must be weighed. The difference in weight between the sets must then be applied to an arm of 468 inches and the resultant moment effect must be added to the last basic weight and moment recorded in the airplane log book.

B. Checking Balance

The elevator balance must be checked in a draft free area with the elevator completely assembled in flying condition. All painting, including strips and touch-up, must be completed. The tab, tab push rod, static wicks, and hinge bolts must be attached. The chord line must be horizontally level and the hinge line must be properly supported when the static moment is measured. Although many different methods of check balancing exist, they can be categorized under the following two headings:

(1) Actual Force Management - Measurement of the force applied by the elevator surface on a single support at a known distance from the center line of the hinge.

(2) Counterbalancing - The application of a known force or weight at a measured distance from the hinge line to counter the unbalance moment of the elevator assembly.

NOTE: Counterbalancing is the simplest method of check balancing.

2. BALANCING PROCEDURE

A. Equipment Required To Perform Check Balancing By Force Measurement Method

(1) A stand with knife edge supports as illustrated in Figure 201. The knife edges should be in the same horizontal plane.

(2) A certified beam balance calibrated in units of 0.01 Ib. or less. The balance should have a flat weighing platform and its capacity should equal tare plus 2.0 Ibs. minimum.

(3) A support spindle similar to the illustration and levelling blocks, as required. (Blocks + spindle = tare).

(4) A straight edge, rule and spirit level.

B. Force Measurement Method

(1) Locate the chord line by placing a straight edge at the inboard end of the elevator so that one end is aligned with the center of the torque tube and the other end is centered on the trailing edge. Mark the chord line by grease pencil or other means on the rib. Remove the straight edge.

(2) Fit correct size bolts in the outboard and center hinge brackets and mount the elevator on the knife edges. Ensure that it is free to rotate about the hinge line.

(3) Place a small platform scale under the trailing edge if the control surface is tail-heavy or under the leading edge if it is nose-heavy.

(4) Place the upper end of the spindle under the trailing edge of tail-heavy surfaces or under the leading edge of the nose-heavy surfaces. The spindle must be vertical throughout the balancing procedure. Hold a spirit level against the marked chord line and level it by extending or contracting the spindle, or by using blocks and shims under the spindle.

(5) Measure the perpendicular distance from the hinge center line to the point supported by the spindle. Ensure that the spirit level and rule are removed from the surface and read the reaction on the beam balance.

(6) Calculate the static under balance moment "M" from the formula:

M = D(R-T) inch-pounds where,

D = Perpendicular distance from the hinge center line to the spindle point (inches).

R = Reaction (Pounds) read from the beam balance.

T = Tare, i.e. spindle plus levelling blocks or shims on the scale platform (Pounds).

EXAMPLE

D is 16.6 inches and T is 1.0 Ib.

M = 16.6 (2.45-1.0); M = 24.07 Ib inches

M is within the range of 19.0 to 24.6 Ib inches tail heavy which is satisfactory.

If M is not within the prescribed range, refer to step (8) under BALANCING PROCEDURE COUNTER-BALANCING METHOD.

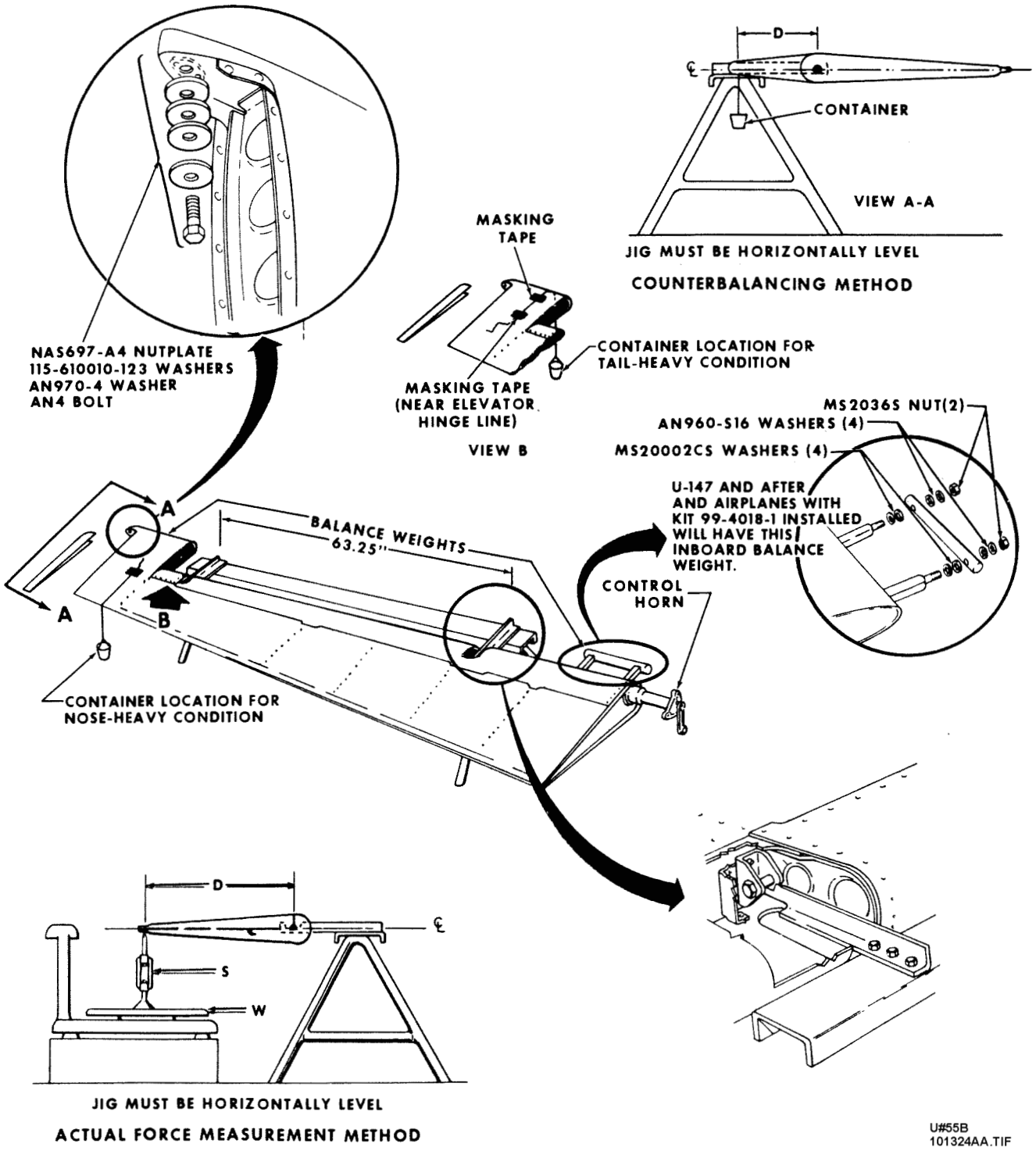

Figure 201
Balancing the Elevator

C. Equipment Required To Perform Check Balancing By Counterbalancing Method

(1) A stand with knife edge supports as illustrated in Figure 201. The knife edges must be in the same horizontal plane.

(2) A paper cup or similar light weight container.

(3) Approximately two pounds of lead shot.

(4) A certified beam balance weighing device calibrated in units of 0.01 pound or less.

(5) Straight edge, ruler, and spirit level.

D. Counterbalancing Method

(1) Locate the chord line by placing a straight edge at the inboard end of the elevator assembly so that one end is on the hinge center line and the other end is centered on the trailing edge. Mark the chord line with a suitable marker, such as a grease pencil, then remove the straight edge.

(2) Fit the correct size bolts in the hinge clevises and mount the elevator on the knife edge supports. Ensure that the elevator is free to rotate about the hinge line.

(3) To determine if adjustment weights should be added or removed, suspend a paper cup from a point on the elevator tailing edge if the balance is nose-down, or near the inboard end of the balance weight assembly on the elevator leading edge if the balance is tail down. Use a short length of small diameter string secured to the surface with a small piece of masking tape as illustrated in Figure 201. The cup must be free to hang vertically.

(4) Add small quantities of lead shot to the cup until the elevator balances with the chord line level. Check this by holding the spirit level aligned with the marked chord line.

(5) Carefully measure the perpendicular distance D within 0.1 inch from the hinge center line to the point of suspension of the cup.

(6) Remove the cup, contents, and string, then weigh them to within 0.005 pound.

NOTE: Since any weighing error is magnified by the distance D, weighing is most important and must be done carefully on scales that are certified for accuracy.

(7) Calculate the static balance as follows:

(a) The weight of the cup and contents is designated by "W".

(b) The distance between the center line of the hinge and the suspension point of the cup is designated by "D".

(c) The over or under balance moment is designated by "M".

(d) M = W x D.

(e) The following is a typical example of a balancing calculation: Assume the elevator is slightly overbalance (nose-heavy) and the paper cup was suspended from the trailing edge. If the elevator balances with the chord line level at "W = 1.78 pound" and "D = 12.8 inches", then...

M = 1.78 x 12.8

M = 22.78 inch-pound. in this instance, "M" is within the required static balance range and is therefore acceptable.

(8) If the static under balance does not fall within the range of 19.0 to 24.6 inch-pounds tail-heavy, adjustment weights must be added or removed and the balance rechecked.

NOTE: Adjustment can be made by relocating the washer from the forward side to aft side of weight assembly, then adding 115-610010-123 trim washers (0.045 pounds each) as required up to a maximum of 10 washers.

STABILIZERS
VERTICAL STABILIZER
MAINTENANCE PRACTICES

1. VERTICAL STABILIZER

A. Removal

(1) Remove the rudder as described under the heading RUDDER REMOVAL in Chapter 27.

(2) Remove both access doors on the left hand side of the fuselage below the horizontal stabilizer.

(3) Disconnect the lines to the deicer boots.

(4) Loosen the rudder cables at the turnbuckles, then remove them from the bellcrank.

(5) Remove the rudder centering springs from the bellcrank.

(6) Remove the rudder bellcrank and bracket.

(7) Remove the screws attaching the dorsal fairing to the vertical stabilizer.

(8) Remove the bolts attaching the vertical stabilizer to the forward and rear attachment points.

B. Installation

(1) Position the vertical stabilizer and install the bolts at the front and rear attachment points.

NOTE: Gaps at bolt holes between the vertical stabilizer front spar and bulkhead frame are permitted up to 0.60 maximum if closed by bolt torque not exceeding one half of the recommended allowable torque for installation of bolts. If the gap and torque tolerances cannot be obtained, use shims P/N 115-600000-19, -21, -23, -25, or -27 as required to close gaps to meet the noted tolerances. Only one shim may be used in each vertical row of four bolt holes. Use the next longer grip length bolt as required.

(2) Tighten the front and rear spar bolts 60 to 80 inch-pounds of torque. After torquing, the bolts shall not rotate in the hole when 70 inch-pounds of torque is applied to the nut, or bolt head (as applicable), and the other end is not held.

(3) Install the screws attaching the dorsal fairing to the vertical stabilizer.

(4) Connect the lines to the deicer boots.

(5) Install the rudder bellcrank and bracket.

(6) Install the rudder and connect the rudder cables to the bellcrank. Connect the rudder trim tab cables at the turnbuckles.

(7) Connect the rudder centering springs to the bellcrank.

(8) Rig the rudder and rudder tab cables.

(9) Connect the rear navigation light wires and install the tailcone and the access doors.

NOTE: If the oscillating beacon light is removed from the rudder the airplane must not be flown until the light is installed or a cover plate installed and the rudder rebalanced.

STABILIZERS
RUDDER
MAINTENANCE PRACTICES

1. RUDDER

A. Balancing

When the rudder surface is being repainted, suspend it by the trailing edge so that excess paint will drain toward the leading edge. After any repainting or repair, the finished surface must be check balanced to ensure that its static moment about the hinge line is within the manufacturers prescribed limits. The static moment of the rudder is determined by multiplying the unbalanced weight of the rudder assembly times the perpendicular distance from the hinge center line to the center of gravity, when the chord line is horizontally level. The weight is measured in pounds and the distance in inches. Tail heaviness indicates static under balance while nose heaviness indicates static overbalance.

NOTE: The maximum static under balance (tail-heavy) for all completed, painted rudder assemblies, which include static wicks, tab chain and cable assembly, bell crank, tab pulley brackets and pulley, shall not exceed 21.0 ± 2.25 inch-pounds.

B. Checking Balance

The rudder balance must be checked in a draft free area with the rudder completely assembled in flying condition. All painting, including stripes and touch-up, must be completed. The tab, tab push rod, static wicks. and hinge bolts must be attached. The chord line must be horizontally level and the hinge line must be properly supported when the static moment is measured. Although many different methods of check balancing exist, they can be categorized under the following two headings:

(1) Counterbalancing - The application of a known force or weight at a measured distance from the hinge line to counter the unbalance moment of the rudder assembly.

(2) Actual Force Measurement - Measurement of the force applied by the rudder surface on a single support at a known distance from the center line of the hinge.

NOTE: Counterbalancing is the simplest method of check balancing.

2. BALANCING PROCEDURE

A. Equipment Required To Perform Check Balancing By Force Measurement Method

(1) A stand with knife edge supports as illustrated in Figure 201. The knife edges should be in the same horizontal plane.

(2) A certified beam balance calibrated in units of 0.01 lb. or less. The balance should have a flat weighing platform and its capacity should equal tare plus 2.0 lbs. minimum.

(3) A support spindle similar to the illustration and levelling blocks, as required. (Blocks + spindle = tare).

(4) A straight edge, rule and spirit level.

B. Force Measurement Method

(1) Locate the chord line by placing a straight edge at the inboard end of the rudder so that one end is aligned with the center of the torque tube and the other end is centered on the trailing edge. Mark the chord line by grease pencil or other means on the rib. Remove the straight edge.

(2) Fit correct size bolts in the outboard and center hinge brackets and mount the rudder on the knife edges. Ensure that it is free to rotate about the hinge line.

(3) Place a small platform scale under the trailing edge if the control surface is tail-heavy or under the leading edge if it is nose-heavy.

(4) Place the upper end of the spindle under the trailing edge of tail-heavy surfaces or under the leading edge of the nose-heavy surfaces. The spindle must be vertical throughout the balancing procedure. Hold a spirit level against the marked chord line and level it by extending or contracting the spindle, or by using blocks and shims under the spindle.

(5) Measure the perpendicular distance from the hinge center line to the point supported by the spindle. Ensure that the spirit level and rule are removed from the surface and read the reaction on the beam balance.

(6) Calculate the static under balance moment "M" from the formula:

M = D(R-T) inch-pounds where, D = Perpendicular distance from the hinge center line to the spindle point (inches).

R = Reaction (Pounds) read from the beam balance.

T = Tare, i.e. spindle plus levelling blocks or shims on the scale platform (Pounds).

EXAMPLE

D is 14.0 inches, R = 2.5 lbs and T is 1.00 lbs.

M = 14.0 (2.5 - 1.00); M = 21.0 inch-pounds.

M is within the prescribed range which is satisfactory.

The complete painted rudder assembly, with static wicks, tab chain and cable assembly, bell crank, tab pulley brackets and pulley shall not exceed 21.0 $\pm$ 2.25 inch-pounds under balance. If the static under balance is not as noted, remove the rudder horn weight and add or remove lead to bring the rudder balance within the required limits.

C. Equipment Required To Perform Check Balancing By Counterbalancing Method

(1) A stand with knife edge supports as illustrated in Figure 201. The knife edges must be in the same horizontal plane.

(2) A paper cup or similar light weight container.

(3) Approximately two pounds of lead shot.

(4) A certified beam balance weighing device calibrated in units of 0.01 pound or less.

(5) Straight edge, ruler, and spirit level.

D. Counterbalancing Method

(1) Locate the chord line by placing a straight edge at the lower closure rib of the rudder so that one end is aligned with the center of the torque tube while the other end is centered on the trailing edge. Mark the chord line with a suitable marker, such as a grease pencil, then remove the straight edge.

(2) Secure the trim tab position in its neutral position with a small piece of masking tape.

(3) Fit the correct size bolts in the hinge brackets and mount the rudder on the knife edge supports. Ascertain that the rudder is free to rotate about the hinge line.

(4) To determine if weight should be added or removed, suspend a paper cup from a point near the center of the rudder trailing edge if the balance is nose-down or near the center of the leading edge if the balance is tail-down. Use a short length of small diameter string secured to the surface with a small piece of masking tape as illustrated in Figure 201. The cup must be free to hang vertically.

(5) Add small quantities of lead shot to the cup until the rudder balances with the chord line level. Check this by holding a spirit level aligned with the marked chord line.

(6) Carefully measure the perpendicular distance "D" within 0.1 inch from the hinge center line to the point of suspension of the cup.

(7) Remove the cup, contents, and string, then weigh them to within 0.005 pounds.

NOTE: Since any weighing error is magnified by the distance D, weighing is most important and must be done carefully on scales that are certified for accuracy.

(8) Calculate the static balance as follows:

(a) The weight of the cup and contents is designated by "W”.

(b) The distance between the center line of the hinge and the suspension point of the cup is designated by "D”.

(c) The over or under balance moment is designated by "M".

(d) M = W x D.

(e) The following is a typical example of a balancing calculation: Assume the rudder is slightly under balance (tail-heavy) and the paper cup was suspended from the leading edge. If the rudder balances with the chord line level at "W = 2.1 pounds" and "D = 10.0 inches", then...

M = 2.1 x 10.0

M = 21.0 inch-pounds.

In this instance, "M" is within the required static balance range and is therefore acceptable.

(9) The complete painted rudder assembly, with static wicks, tab chain and cable assembly, bell crank, tab pulley brackets and pulley, shall not exceed 21.0 ± 2.25 inch-pounds under balance. If the static under balance is not as noted, remove the rudder horn weight and add or remove lead to bring the rudder balance within the required limits.

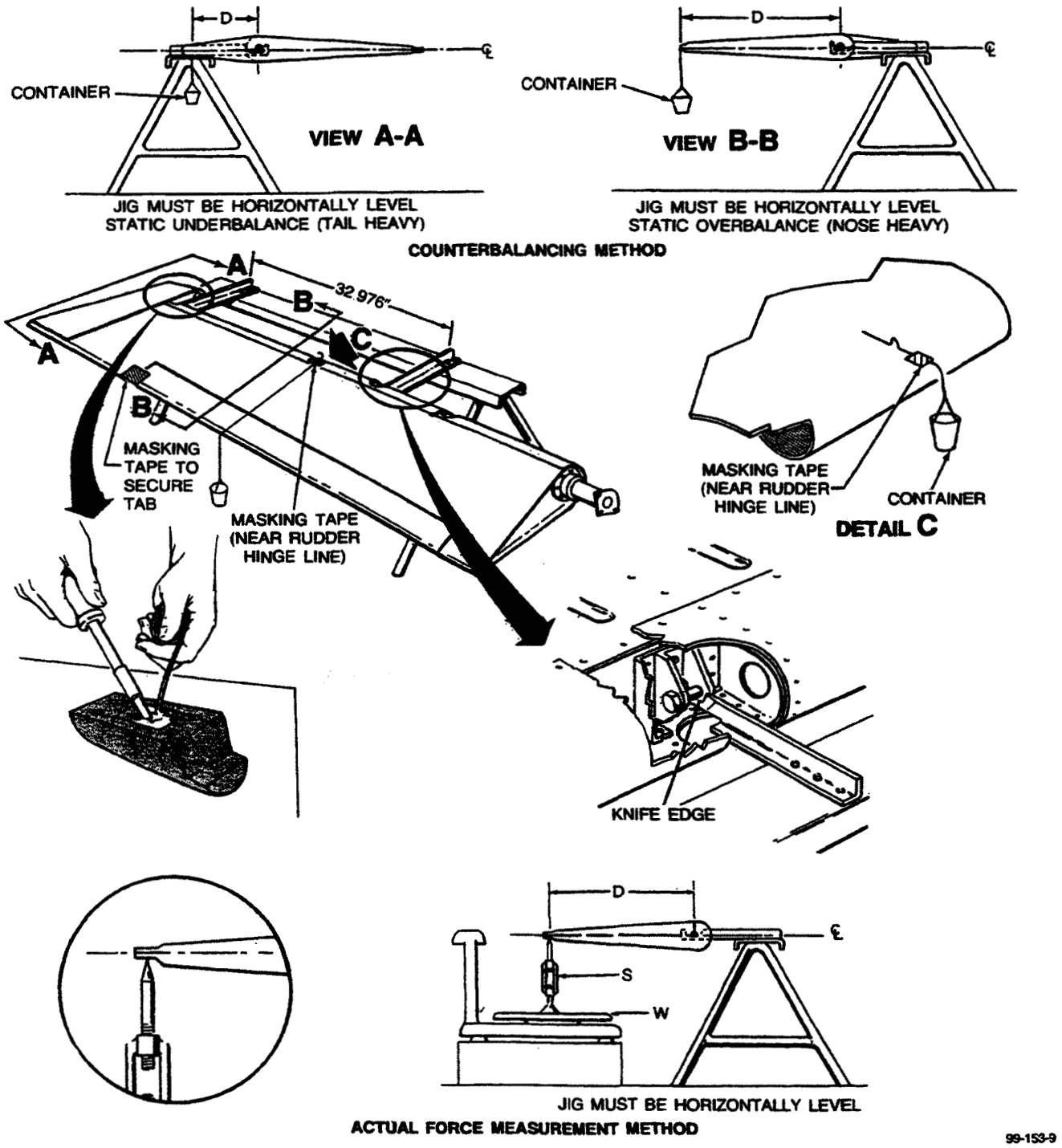

Figure 201
Balancing the Rudder

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CHAPTER 56 - WINDOWS

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WINDOWS
WINDOWS
MAINTENANCE PRACTICES

1. PLASTIC WINDOWS

A. Cleaning

The plastic windows should be kept clean and waxed at all times. Any nonacid, nonabrasive material may be used for this purpose. Automotive cleaners and waxes may be used on plastic windows only if they are nonacrylic. To prevent scratches and crazing, wash the windows carefully with plenty of soap and water, using the palm of the hand to feel and dislodge dirt and mud. A soft cloth, chamois, or sponge may be used, but only to carry water to the surface. Rinse the window thoroughly, then dry it with a clean moist chamois. Rubbing the surface of the plastic with a dry cloth will build up an electrostatic charge that attracts dust.

Remove oil and grease with a cloth moistened by kerosene. Never use gasoline, benzine, alcohol, acetone, carbon tetrachloride, fire extinguisher or anti-ice fluid, lacquer thinner, or glass cleaner, for such materials will soften the plastic and may cause crazing.

After removing dirt and grease from the window, it should be waxed with a good grade of commercial wax. The wax will fill in minor scratches and help prevent further scratches. Apply a thin, even coat of wax and bring it to a high polish by rubbing lightly with a clean, dry, soft flannel cloth. Never use a power buffer, for the heat generated by the buffing pad may soften the plastic.

2. WINDSHIELD

For windshield crack criteria, refer to Chapter 05-10-00.

A. Cleaning

Class windshields with an antistatic coating should be cleaned as follows:

(1) Wash excessive dirt and other substances from the glass with clean water.

(2) Wash the windshield clean with mild soap and water or with a 50/50 solution of isopropyl alcohol (90, Table 6, 91-00-00) and water. Wipe the glass surface in a straight rubbing motion with a soft cloth or sponge. NEVER use any abrasive materials or any strong acids or acid base material to clean the glass.

(3) Rinse the glass thoroughly and dry, but do not apply wax.

NOTE: It is equally essential that the windshield wipers be thoroughly cleaned, for grit trapped by the wipers is a common source of scratches in the windshield when the wipers are operated. Do not attempt to polish such nicks or scratches in the glass surface.

WINDOWS
FLIGHT COMPARTMENT
MAINTENANCE PRACTICES

1. WINDSHIELD

A. Removal

(1) Remove the screws that hold the retainer strips on the front of the windshield.

(2) Cut the sealing compound free from the retainer strips and window frame.

(3) Inside the flight compartment, remove the screw that secures the ground jumper wire and any power and sense leads, if applicable, from the outboard lower comer of the windshield at the terminal block. Remove the power lead from the inboard lower comer of the windshield at the terminal block. Cut the sealant loose from the windshield and push the windshield panel outward to free the windshield.

(4) Scrape away any remnants of sealing compound, filler, or sealer that may have adhered to the window frame.

(5) Wash the window frame and channel clean with solvent (34, Table 6, Chapter 91-00-00).

B. Installation

(1) Mask off the interior windshield frames with 2 inch masking tape and cover the area over the instrument panel with kraft paper or equivalent.

CAUTION: *Keep solvent (34, Table 6, Chapter 91-00-00) away from plexiglass.*

(2) Clean the windshield frames using a cloth dampened with solvent (34, Table 6, Chapter 91-00-00).

(3) Attach spacer blocks in position in the lower frame with adhesive (43, Table 6, Chapter 91-00-00) (Ref. Figure 201).

NOTE: If the Dortite does not stick tight to the frame, apply a coat of adhesive (43, Table 6, Chapter 91-00-00) to the frame members and reactivate with solvent (34, Table 6, Chapter 91-00-00) as Dortite is placed into position.

(4) Install 5/16 inch x 3/8 inch Dortite on the inner edges of the top and bottom frame members and 1/8 inch x 3/8 inch Dortite to the side and center frame members. The edge of the Dortite should be flush with the edge of the frame and the corners must have a tight butt joint.

(5) Place both of the windshields into position and temporarily install retainers with a minimum number of screws so as to establish masking locations on the glass and fuselage. Do not tighten the screws. At this time mark any areas where shims will be required on Dortite. Check for gaps.

NOTE: This should allow later removal of the 2 inch tape and 3/8 inch tape without removing the 3/4 inch tape strip.

(6) Apply a strip of 3/8 inch masking tape on the glass and fuselage at the edge of the retainers. Butt a strip of 3/4 inch masking tape next to the 3/8 inch tape just applied. With a soft cloth, apply a coating of Talc to the 3/4 inch tape. Apply 2 inch masking tape over the 3/8 inch and 3/4 inch strips (Ref. Figure 202).

(7) Remove the retainers and windshields and mask off the outside surface of the retainers. Cutout all countersunk screw holes with a razor blade.

(8) Turn the windshields inside-up and mask off the inside of the glass with 2 inch masking tape to correspond with the mask installed on the outside surface.

(9) Install an additional thickness of Dortite in areas marked for shims as outlined in Step (5).

NOTE: Sealing compound (115, Table 6, Chapter 91-00-00) has a pot life of four hours at 77°F. For every 10°F above 77°F, the pot life is shortened by 50%. For every 10°F below 77°F, the pot life is doubled.

(10) Apply a bead of sealing compound (115, Table 6, Chapter 91-00-00) in the windshield frames between the Dortite and frame radius and fill all voids in the corners.

(11) Apply a bead of sealing compound (115, Table 6, Chapter 91-00-00) to the inside surface of the retainers at all edges.

(12) Center the windshields in the frames. Install temporary retainer clips at the center and both outside edges of the windshields, three at each location. Do not overtighten.

(13) Fill voids between the glass and frames with sealing compound (115, Table 6, Chapter 91-00-00) at the top and bottom of the windshields. This bead of sealer must extend above the glass surface 1/8 inch to 3/16 inch so that the area under the retainers will be completely full of sealer upon completion of this installation.

(14) Place the top and bottom retainers in place and install MS24693BB274 screws. While the screws are being tightened down in sequence, the sealer will be forced out from under the retainers on both sides.

(15) Remove the temporary retainer clips and apply sealing compound (115, Table 6, Chapter 91-00-00) between the glass and frame as outlined in Step (13).

(16) Set the spacers in place and install the side and center retainers with MS24693BB271 screws.

(17) Using a tongue depressor with the ends cut square, remove excess sealer that has been forced out from under the retainers.

(18) Remove the 2 inch masking tape installed in Step (6). The 3/8 inch tape may come off with the 2 inch strip. If not, remove the 3/8 inch strip and leave the 3/4 inch strip of tape in place. There should now be 2 strips of 3/4 inch masking tape positioned 3/8 inch from the edge of the retainers on both the glass and fuselage.

(19) Apply a bead of sealing compound (115, Table 6, Chapter 91-00-00) around the edges of the retainers with a plastic squeegee with square corners. Form a fillet of sealer between the retainers and the edge of the masking tape.

(20) Remove all masking tape and protective covering and perform any cleanup required.

NOTE: If any sealer has been forced out from between the glass and the Dortite on the inside of the windshield, allow the sealer to cure for at least 10 hours and trim off with a razor blade.

(21) Inside the flight compartment, install the screw that secures the ground jumper wire and any power sense leads (if applicable) to the outboard lower corner of the windshield at the terminal block. Install the power lead at the terminal block on the inboard lower corner of the windshield.

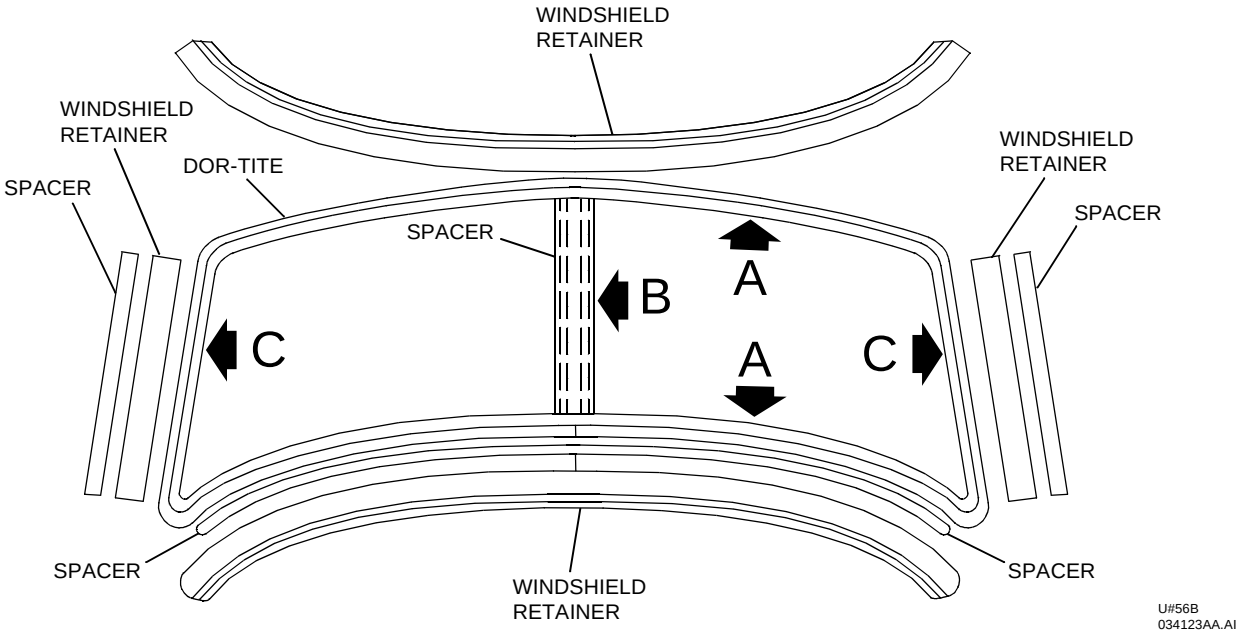


Figure 201 (Sheet 1 of 2)
Windshield Installation

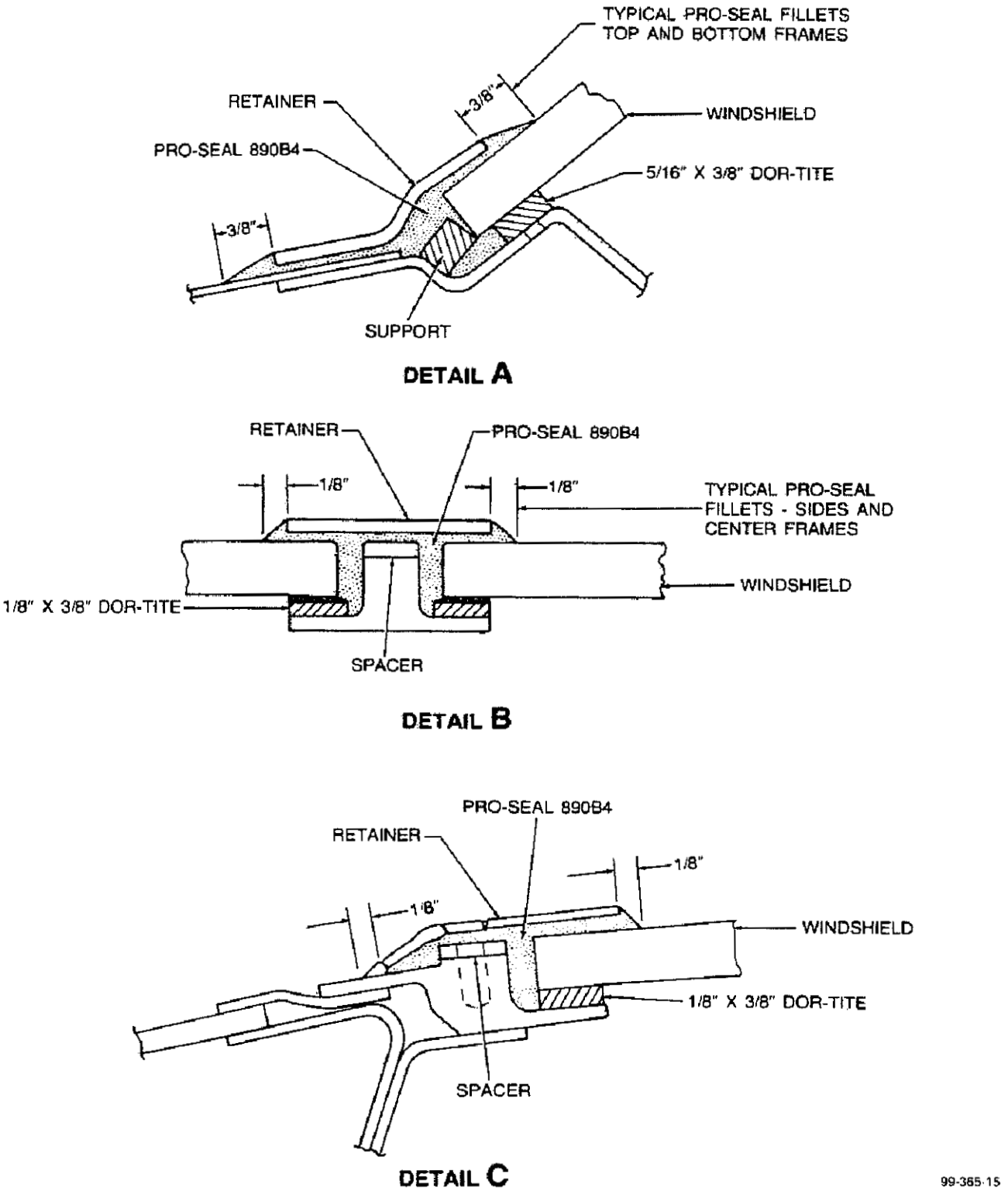


Figure 201 (Sheet 2 of 2)
Windshield Installation

C. Windshield Fillet Seals

Inspect windshield fillet seals for voids and deterioration. Fill voids or renew deteriorated sealant (115, Table 6, Chapter 91-00-00) or current preferred sealant with a plastic squeegee with square corners. If void extends under windshield retainer, remove affected retainer, renew sealant as required and replace retainer. Refer to Figure 201, sheet 2 of 2, details A, B and C for windshield cross section examples. Observe sealant curing instructions.

D. Replacement Conditions

The electrically heated laminated glass windshield is subject to a gradual process of delamination due to the effect of chemical action and differentials of temperature at varying altitudes and under varying weather conditions. This delamination is not detrimental to the structural integrity of the windshield, although it may significantly decrease visibility or the deicing capability of the windshield. Beyond certain limits, either of these effects will require the replacement of the windshield. Observe the following guidelines to determine when a windshield should be replaced:

(1) **Visibility Impairment:** Replace the windshield when areas of delamination are enlarged to the point of impairing vision, whether or not these areas extend to the edge of the glass. Smaller delamination appearing as bubbles is no cause for concern. The rate of growth of delaminated areas should be used as a guide for scheduling windshield replacement.

(2) **Heating Impairment Limits:** Replace the windshield when the lost heating area exceeds 1/4 to 1/3 of the total heated area.

(3) **Cracked Glass:** Both glass plies are chemically strengthened and are structural plies. Depending on the stress level on the glass at the time of fracture, the break pattern will generally be of relatively large size fragments adhered to the interlayer. The windshield must be replaced if either glass ply is fractured.

(4) **Scratches:** Scratches or nicks on either surface as deep as 0.005 inch are considered acceptable as long as vision is not seriously impaired. Scratches deeper than 0.005 inch require the windshield to be replaced.

(5) **Glass Adhesion Chips:** Glass adhesion chips can occur on the inboard surface of the outer ply, or the outboard surface of the inner ply. Chips are caused by the vinyl interlayer pulling glass out of the glass surface and usually form at or near the glass edge. The chip will continue to expand/propagate until the stress that caused the chip is relieved. A chip in a non-structural ply is not cause for windshield replacement unless vision is seriously impaired. A chip in a structural ply causes major glass strength reduction, and is cause for windshield replacement.

(6) **Moisture Damage:** The primary cause of moisture ingress is a deteriorated weather seal. Moisture ingress caused by a deteriorated weather seal will eventually lead to one or all of the following windshield degradation modes:

(a) Delamination between the glass and PVB interlayer (Separation between glass and interlayer characterized by a separation line with a different reflection characteristic on each side of the line).

(b) Corrosion of the bus bar - heating wire junctions (black spots at the bus bar - heating wires interface).

(c) Failure of the electric heating system (lost heating area exceeds 1/4 or 1/3 of the total heated area).

(d) Outer glass ply fracture (see cracked glass).

If the windshield is severely degraded due to any or all of the above conditions, the useful service life is limited and is cause for windshield replacement.

2. WINDSHIELD ANTISTATIC COATING AND TAB

A. Inspection and Replacement

NOTE: This procedure applies to airplane serials U-165 and After and prior airplanes equipped with electrically heated windshields.

The electrically heated windshields are to be inspected at the intervals specified in Chapter 05. This inspection may be accomplished as follows:

(1) Turn OFF all electrical power.

CAUTION: *Use care while cleaning the windshield glass and performing this inspection to avoid damage to the windshield.*

(2) Clean the exterior surface of the windshield glass with isopropyl alcohol (90, Table 6, Chapter 91-00-00) to remove loose dirt and dust.

(3) If necessary clean a spot on the windshield frame until bare metal is exposed.

(4) Using a volt-ohmmeter capable of reading 100 K ohms, perform the following inspection. Set the range dial to 100 K and zero the ohmmeter reading needle.

(5) Attach one ohmmeter test lead to the surface of the windshield frame on the bare metal.

(6) Attach the other ohmmeter test lead to a copper wire mesh pad, such as a Brillo scrub pad.

(7) Contact the windshield glass with the wire mesh pad. Check the ohmmeter reading. If the ohmmeter reading is between 5 K ohms and 100 K ohms in at least ten different locations, no further action is required. Paint the windshield frame if necessary. If the ohmmeter reading is more than 100 K ohms, proceed with the following Steps.

(8) If some of the ohmmeter readings were more than 100 K ohms, draw a picture of the windshield and note the locations where the ohmmeter readings were 100 K ohms or more. Contact Hawker Beechcraft Corporation Technical Support for consultation. Do not remove the windshield before contact is made with Hawker Beechcraft Corporation Technical Support.

(9) If all of the ohmmeter readings were more than 100 K ohms, perform the following anti-static tab inspection Steps:

(a) Remove the ohmmeter test lead from the wire mesh pad.

(b) Place the ohmmeter test lead on the windshield glass within 0.5 inch of the anti-static tab. Do not press down on the anti-static tab during this inspection.

(c) If the ohmmeter reading is 10 K ohms or less as the ohmmeter test lead is drawn around the anti-static tab, the anti-static tab is providing sufficient electrical continuity between the windshield retainer and the anti-static coating on the windshield glass. If the ohmmeter reading was more than 10 K ohms, continue with the following Steps.

NOTE: If more than one anti-static tab is installed, check each tab.

(d) Insuring the hands and body are completely insulated from the airframe, windshield and meter probes, reconnect the ohmmeter test lead to the wire mesh pad. Place the wire mesh pad on the windshield glass. Remove the ohmmeter test lead from the windshield frame and place on the windshield glass within 0.5 inch of the anti-static tab. Using the wire mesh pad, take ten resistance readings in different locations on the windshield glass. If the ten resistance readings are 100 K ohms or less, a new anti-static tab must be installed. Proceed to Step (11) for installation instructions.

Note: Kit No. 90-5048-1 provides the parts to install two new anti-static tabs.

If occasional 100 K ohms or less readings were indicated, repeat Step (9). If all resistance readings were more than 100 K ohms, proceed with the following Steps.

(e) Move the ohmmeter test lead and wire mesh pad approximately four inches away from the edge of the anti-static tab but within 0.25 inch of the windshield retainer and place on the windshield glass. Take ten resistance readings at different locations on the windshield. Place a mark on the windshield retainer where the ohmmeter test lead was located. At the mark, write the number of resistance readings that were 100 K ohms or less. If all ten resistance readings were 100 K ohms or less, a new anti-static tab should be installed centered on the mark. Anti-static tab installation instructions begin with Step (11). Proceed to Step (11).

(f) If ten resistance readings of 100 K ohms or less were not found, move the ohmmeter test lead approximately four inches from the previous test point and take an additional ten resistance readings with the wire mesh pad. The other ohmmeter test lead must remain within 0.25 inch of the windshield retainer. At each test point, mark the windshield retainer and record the number of ohmmeter readings of 100 K ohms or less. If a point has ten resistance readings of 100 K ohms or less, a new anti-static tab should be installed at that mark. Proceed to Step (11) for anti-static tab installation instructions. If only occasional readings of 100 K ohms or less were found, repeat Step (9).

(10) Remove the existing anti-static tabs from windshields that have removable tabs (Ref. Figure 203). Clean the windshield glass with isopropyl alcohol (90, Table 6, Chapter 91-00-00). A wood or plastic scraper may be used to aid in the removal of the sealer. Install the new anti-static tab using Steps below as required. For windshields with anti-static tabs that are not removable (Ref. Figure 203, View A).

(11) Center the new anti-static tab on the mark on the windshield retainer with 0.59 inch of the anti-static tab covering the windshield glass. Mark the anti-static tab top edge at a point 0.5 inch from the outer edge of the windshield retainer (Ref. Figure 203, View B).

(12) Mark the 0.5 inch line at a point 0.09 inch from the side of the anti-static tab. Drill a 0.0980 inch diameter hole through each side of the anti-static tab.

(13) Using the two holes drilled in the anti-static tab as a guide, mark two holes in the windshield retainer.

(14) Drill two 0.0935 inch diameter screw holes in the windshield retainer to the depth required for the anti-static tab attach screws at the mark made in Step (13). Vacuum to remove any burrs.

(15) Tap the two 0.0935 inch diameter holes with a No. 4-48 flat bottom tap.

NOTE: If the windshield frame is buffed, buff the anti-static tab before installation.

(16) Install the anti-static tab on the windshield retainer with the two attach screws. Mask off the anti-static tab with masking tape on each side so that, when the anti-static tab is removed, the masking tape will show the position. Remove the anti-static tab. Lightly sand the bottom of the anti-static tab. Sand the windshield retainer until bare metal is exposed.

(17) Clean inside the masked off area with isopropyl alcohol (90, Table 6, Chapter 91-00-00). Apply a strip of electrical conductive tape approximately 1.125 inches long centered in the masked off area. Locate 0.47 inch of the electrical conductive tape on the windshield glass. Press the electrical conductive tape down with finger pressure only.

(18) Clean the bottom surface of the anti-static tab and the top surface of the electrical conductive tape with isopropyl alcohol (90, Table 6, Chapter 91-00-00). Apply electrical conductive sealant (105, Table 6, Chapter 91-00-00) to the top surface of the electrical conductive tape on the windshield glass.

(19) Install the anti-static tab on the windshield retainer. The electrical conductive sealant should flow from under the anti-static tab on the inboard and outboard sides of the anti-static tab on the windshield glass. If the sealant did not flow from under the anti-static tab, remove the anti-static tab and apply additional sealer.

NOTE: Do not press on the top of the anti-static tab to cause the electrical conductive sealant to flow from under the anti-static tab.

(20) Remove the masking tape from around the anti-static tab and allow the electrical conductive sealant to cure to the manufacturers recommendation.

(21) Mask off the four sides of the anti-static tab with 0.0937 inch clearance between the masking tape and the anti-static tab. Mask off the top surface of the anti-static tab.

(22) Apply adhesive (106, Table 6, Chapter 91-00-00) or equivalent to the four sides of the anti-static tab to provide a weather seal. Remove the masking tape and allow the adhesive to cure to the manufacturers recommendation.

(23) If necessary, paint the anti-static tab to match the windshield frame finish.

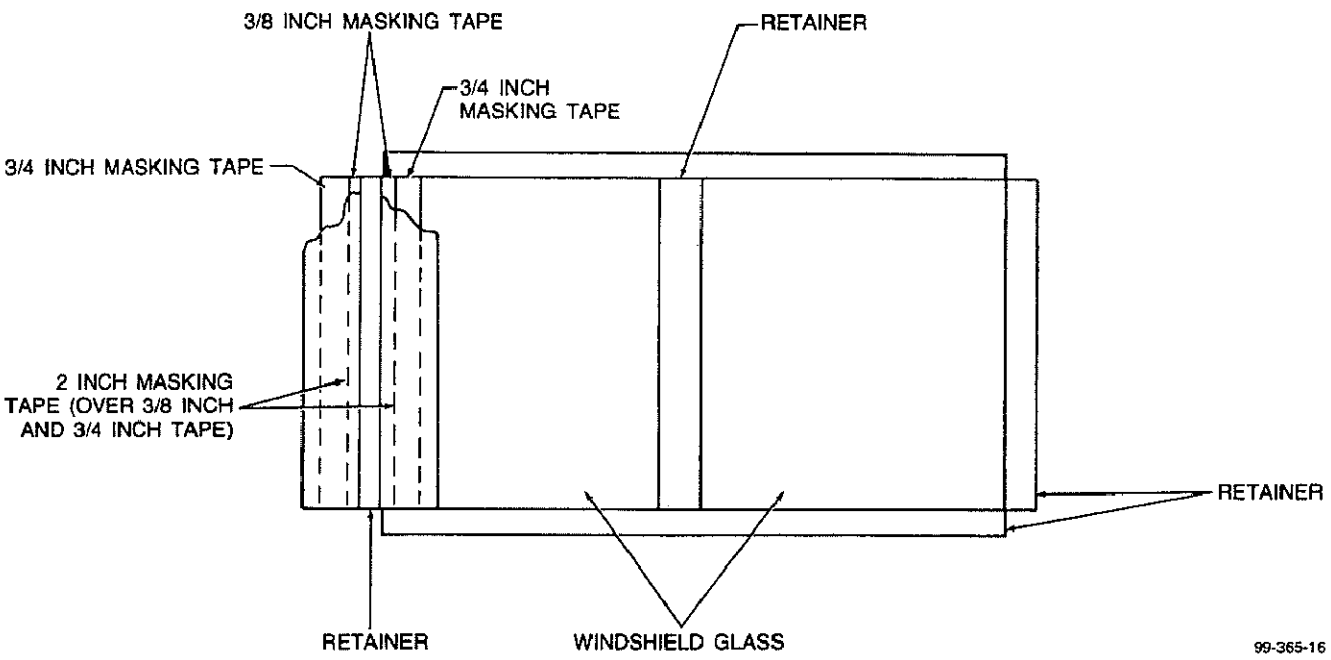


Figure 202
Masking Tape Procedure

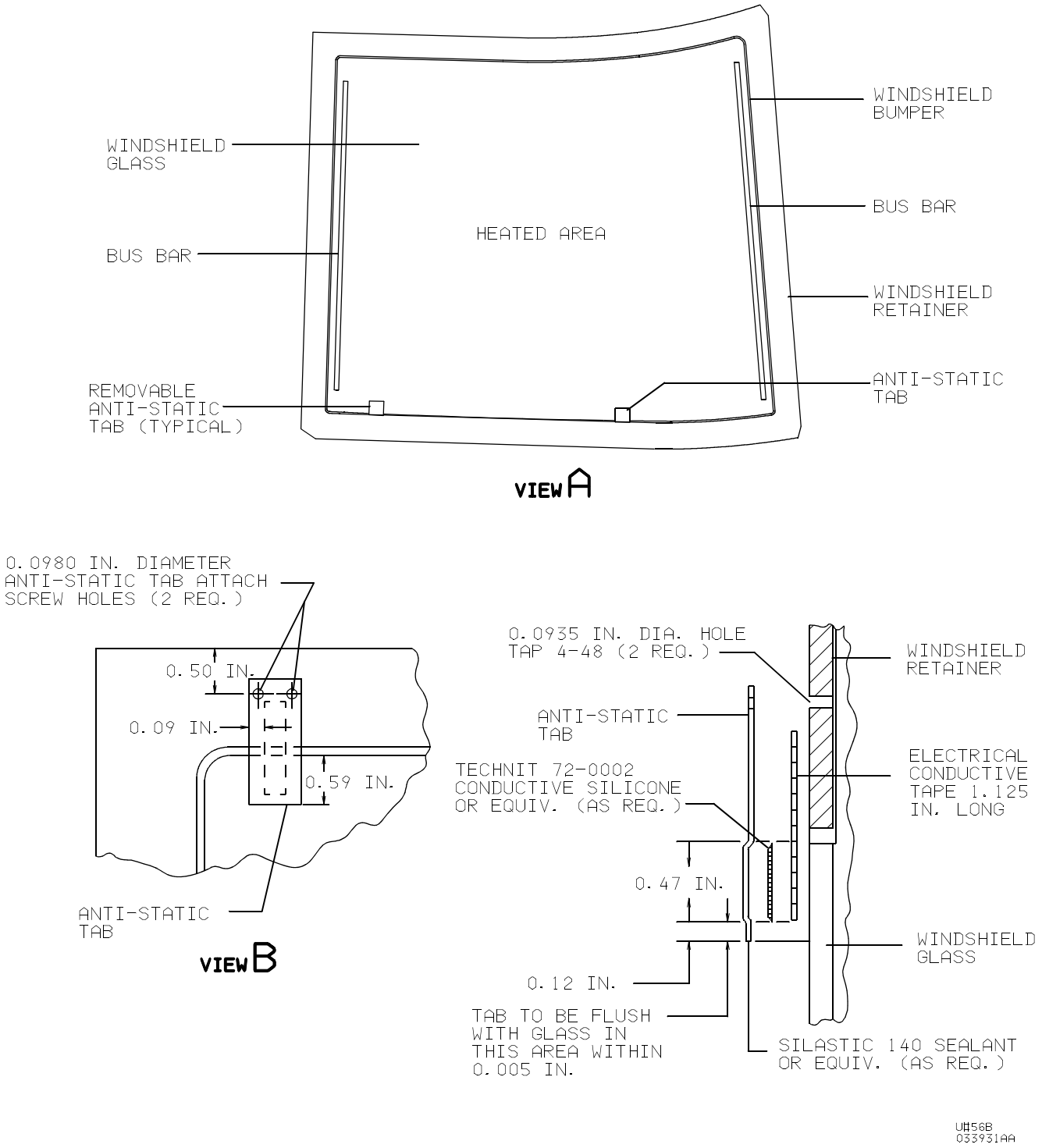


Figure 203
Anti-Static Tab Installation

3. FLIGHT COMPARTMENT SIDE WINDOW

A. Removal

(1) Remove the screws (5) and washers (6) attaching the window molding (4) assembly to the window frame (2) (Ref. Figure 204, Sheets 1 and 2).

(2) Remove the screws (5) and clips securing the inner window (3) (if installed) and remove the inner window (3) assembly.

(3) Drill out the rivets attaching the window (1) and window frame (2) to the airplane structure (9) and skin (8).

(4) Remove the window assembly from the airplane.

B. Installation

(1) Install tape (97, Table 6, Chapter 91-00-00) around the window (1) in such a manner that an equal width of tape extends on each side of the panel (Ref. Figure 204 Sheets 1 and 2).

(2) Position the window (1) and window frame (2) assembly on the airplane, drill and secure with rivets.

NOTE: If the Dortite (10) does not stick tight to the frame, apply a coat of adhesive (43, Table 6, Chapter 91-00-00) to the frame members and reactivate with solvent (34, Table 6, Chapter 91-00-00) as Dortite (10) is placed into position.

(3) Install 3/16 inch x 3/8 inch Dortite (10) along the window frame (2) top longeron. The edge of the Dortite (10) should be flush with the edge of the longeron and centered horizontally on the inner window (3).

(4) Install 5/16 inch x 3/8 inch Dortite (10) along the window frame (2) lower longeron. Center the Dortite (10) horizontally on the inner window (3).

(5) Install 5/16 inch x 3/8 inch Dortite (10) along the forward and aft window frame (2) bulkhead. Center the Dortite (10) vertically on the inner window (3).

(6) Install the inner window (3) and secure using screws (5) and clips.

NOTE: Verify that the support (7) is installed on the lower window frame (2) before installing the window molding (4).

(7) Position the window molding (4) and secure using washers (6) and screws (5).

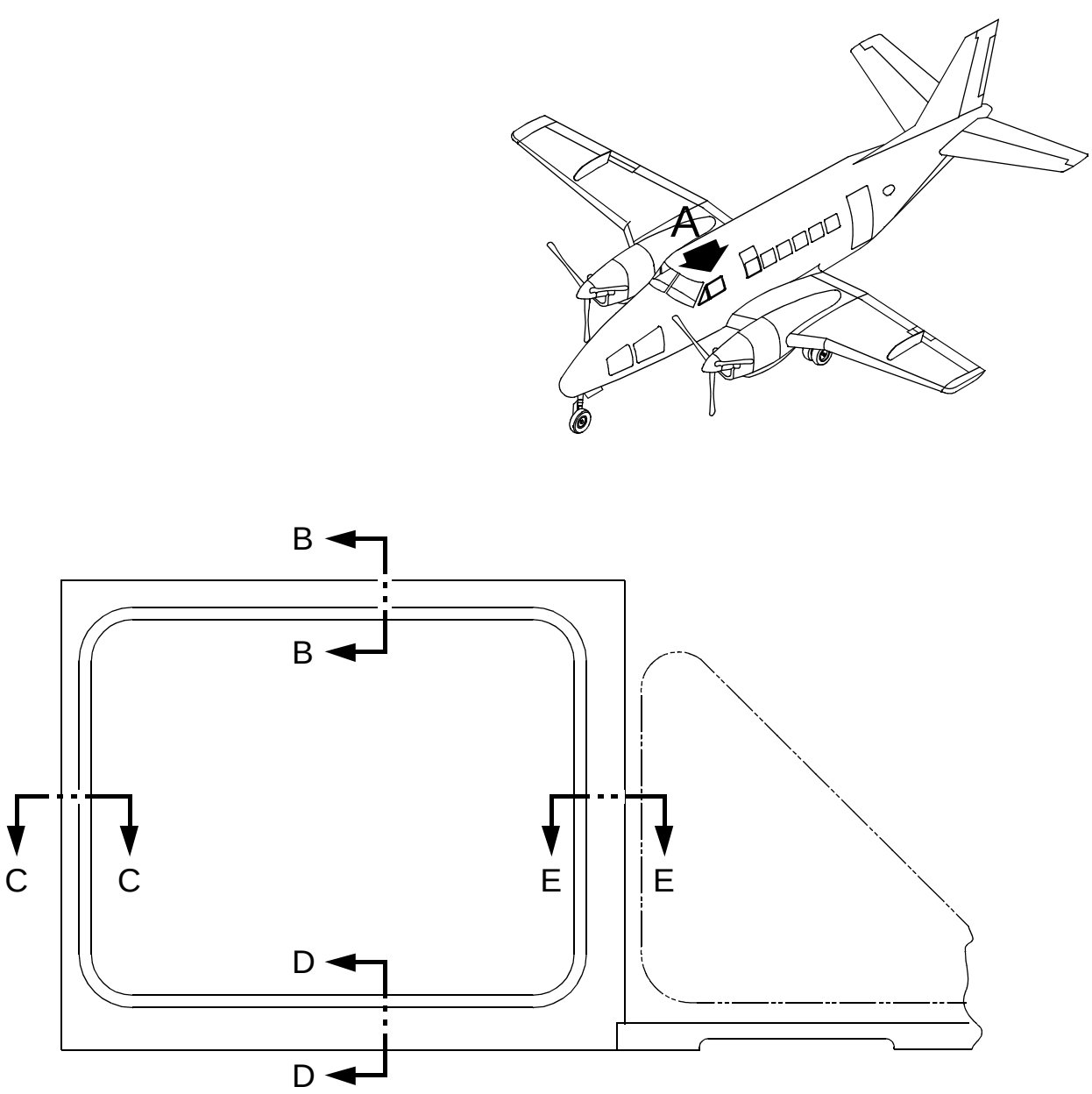
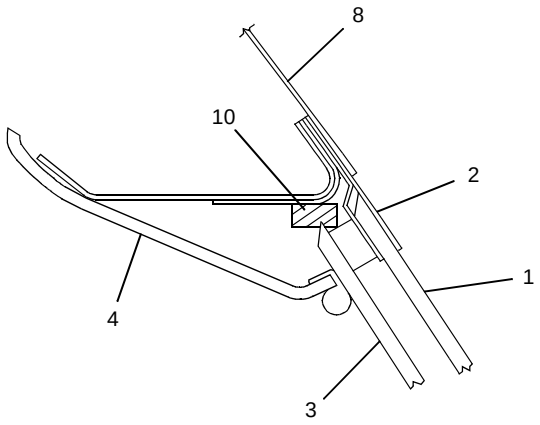
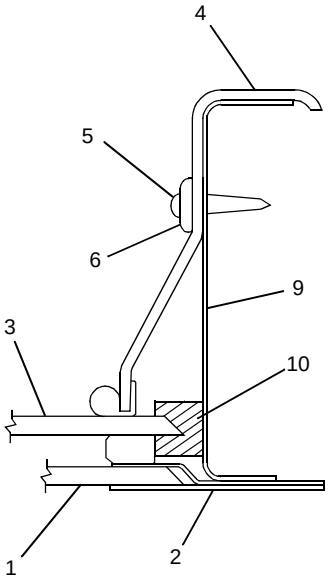


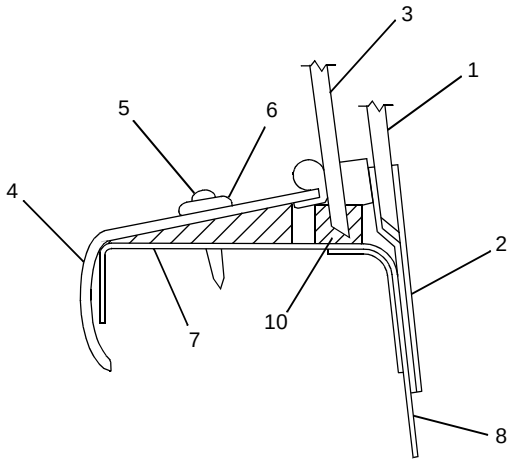
Figure 204 (Sheet 1 of 2)
Flight Compartment Side Window Assembly



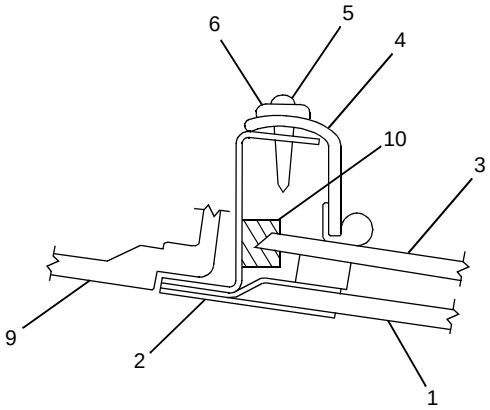
Section B-B



Section C-C



Section D-D



Section E-E

1. WINDOW
2. WINDOW FRAME
3. INNER WINDOW
4. WINDOW MOLDING
5. SCREW
6. WASHER
7. MOLDING SUPPORT
8. AIR PLANE SKIN
9. AIR PLANE STRUCTURE
10. DORTITE

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Figure 204 (Sheet 2 of 2)
Flight Compartment Side Window Assembly

WINDOWS
PASSENGER COMPARTMENT
MAINTENANCE PRACTICES

1. CABIN WINDOW

A. Removal

(1) Remove the screws securing the thermoplastic panel in place around the window.

(2) Remove the screws securing the inner soundproofing glass (if installed) and thermoplastic cover in place on the window frame.

(3) Remove the screws securing the retainer to the window frame and remove the window glass.

B. Installation

(1) Install tape (97, Table 6, 91-00-00) around the window pane in such a manner that an equal width of tape extends on each side of the panel.

(2) Insert the window pane in place and secure it to the window frame with the retainer and attaching screws.

(3) Secure the inner soundproofing glass (if installed) and thermoplastic cover in place on the window frame with the attaching screws.

(4) Secure the thermoplastic panel in place around window with the attaching screws.

2. BAGGAGE COMPARTMENT WINDOW

A. Removal

(1) Remove the screws securing the thermoplastic panel in place around the window.

(2) Drill out the rivets securing the retainer to the window frame and remove the window glass.

(3) Remove the sealing compound from the window frame and channel.

B. Installation

(1) Insert the window pane in place and secure it to the window frame with the retainer and attaching rivets.

(2) Fill any cracks around the window with sealing compound (96, Table 6, 91-00-00).

(3) Secure the thermoplastic panel in place around the window with the attaching screws.

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57-00-00	201 thru 227	Nov 1/10
57-30-00	201 and 202	Nov 1/10
57-50-00	201 thru 205	Nov 1/10

Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 57 - WINGS

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Wing Access Openings 201

TRAILING EDGE/TRAILING EDGE DEVICES

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WINGS
WINGS
MAINTENANCE PRACTICES

1. WING

A. Removal

The following procedure outlines the removal of the outboard section of the wing from the center section.

NOTE: When plumbing lines are disconnected, they must be capped immediately to prevent contamination of the respective system.

(1) Disconnect the battery electrical leads from the battery. Disconnect the battery drain (if installed in the wing).

(2) Drain and purge the fuel cells.

CAUTION: *The fuel must be drained and purged from both wings.*

(3) Install a tail stand.

(4) Disconnect the aileron and (on the left wing only) the aileron tab cables at the turnbuckles inside the wheel well.

(5) Remove the leading edge skin outboard of the nacelle and the upper and lower access doors immediately aft of the main spar.

(6) Disconnect the wing wiring at the terminal strip.

(7) Disconnect the gravity feed and crossvent fuel lines at the outboard wing fuel cells.

(8) Remove the fuel vent line from the leading edge and inboard wing fuel cells.

(9) Disconnect the deicer lines in the wheel well.

(10) Remove the inboard access cover on the lower surface of the outboard wing, just forward of the rear spar. Disconnect the flap actuator drive shaft and remove the grommet in the wing rib.

(11) Remove the strobe light plug, if installed.

(12) Support the wing being removed with a suitable cradle. Support the opposite wing in order to assure airplane stability during and after removal of the wing.

(13) On airplanes with the Wing Spar Bridge Kit No. 90-4090 or 100-4011 installed, accomplish the following:

(a) Remove the fairings covering the spar bridge.

(b) Attach weight hangers to the spar bridge as shown in Figure 201. Attach 125 pounds of weight to each hanger.

(c) Remove the screws and brackets from the lower spar block.

(d) Apply dry ice to the bottom side of the spar bridge immediately below the spacer blocks. Allow the bridge to chill for approximately 10 minutes. Tap the lower block until the adhesive breaks. Remove the lower block and shim (if installed) from the bridge.

(e) Remove the weights and weight hangers from the spar bridge.

(f) Remove the expandable bolts from the outboard wing and center section bridge fitting interface; refer to Figures 201 and 219.

CAUTION: *Before removing any wing attaching bolt, mark the position of the fittings in relation to each other to aid in wing realignment. Use a grease pencil only. DO NOT SCRIBE OR SCRATCH THE FITTING. If (while using hand pressure only) wing attach bolt binding occurs upon disengagement or reinstallation, support the outboard wing panel, loosen the remaining three bolts and reposition the wing to allow the bolt to be removed.*

(14) Remove the wing bolt cover plates and wing attaching bolts.

(15) Move the wing straight away from the airplane until all fuel lines are clear of the center section.

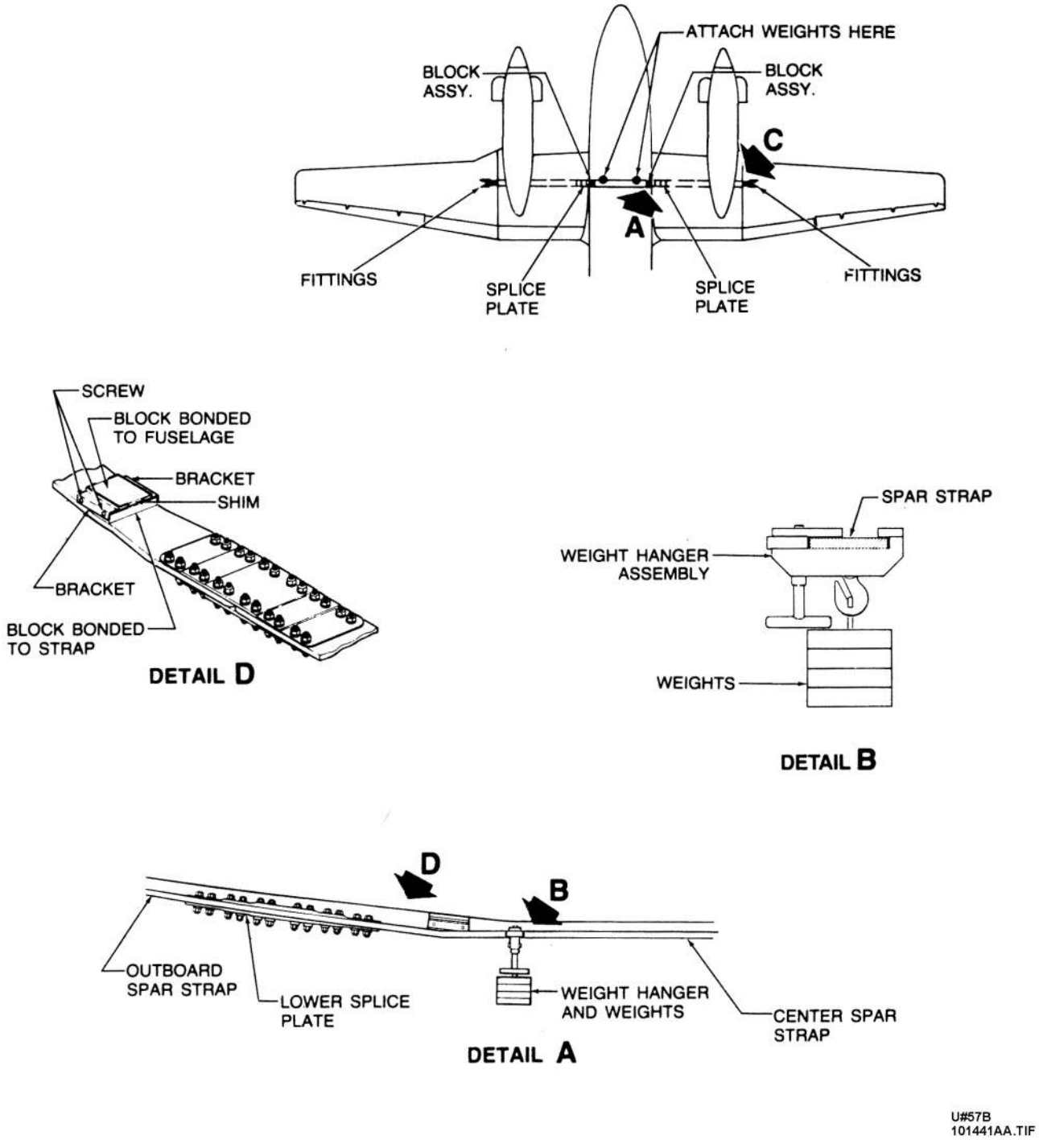


Figure 201 (Sheet 1 of 2)
Center Section And Outboard Wing Spar Bridge Connection

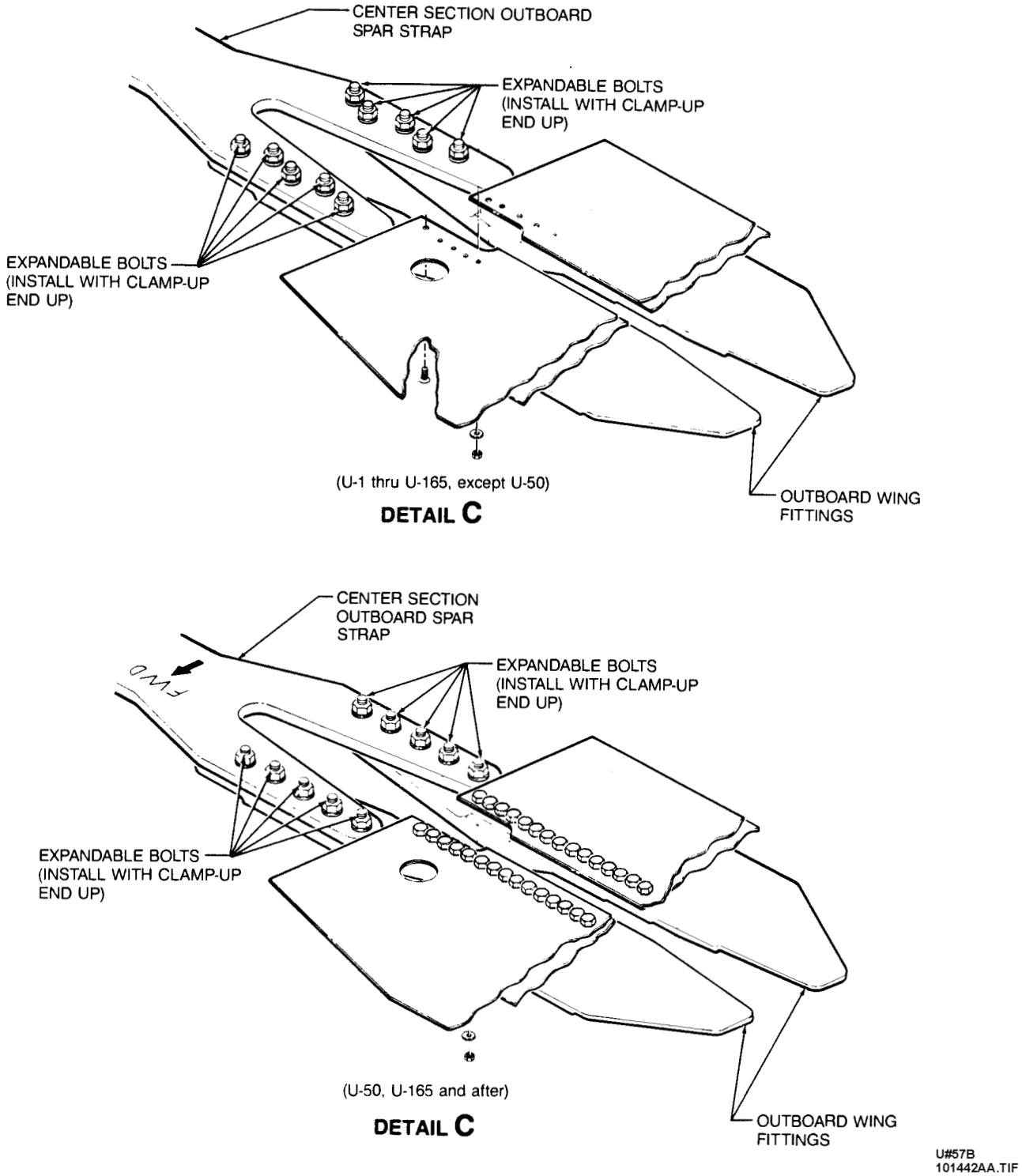


Figure 201 (Sheet 2 of 2)
Center Section And Outboard Wing Spar Bridge Connection

B. Disassembly

(1) Support the wing on a suitable cradle.

(2) Remove the wing tip, aileron, fuel cells, and other equipment as required by the work being done.

(3) Remove the screws around the spar caps at the root ribs.

(4) Free the ends of the hinge pins. Remove each hinge pin by one of the following methods:

CAUTION: *Never use a drill. motor to spin a hinge pin in or out of a hinge. The resulting friction heat will cause the pin to expand and seize in the hinge.*

NOTE: Before removing any single hinge pin, pull all hinge pins out one inch (two hinge lugs) to break any adhesion or binding caused by corrosion or buildup of dust.

CAUTION: *Support the leading edge/spar assembly while the second main wing section hinge pin is being removed to prevent the assembly from falling away from the wing.*

NOTE: Remove the hinge pins from the main wing section first, then from the leading edge.

(a) Method #1 (Figure 202):

1 Clamp locking pliers (or equivalent) tightly on the pin.

2 Place a board in a suitable position on the wing spar to use as a fulcrum for a lever.

CAUTION: *Use slow, steady pressure on the lever to avoid breaking the hinge pin.*

3 Bring the lever to bear against the locking pliers and pry the hinge pin out of the hinge as shown in Figure 202.

4 Use thicker boards and shift the position of the locking pliers as the hinge pin is pulled out of the spar.

(b) Method #2 (Figure 203):

1 Fabricate a support bracket and crank as shown in Figure 203.

2 Attach the bracket to the end stiffener of the spar with two screws as shown in Figure 203.

3 Insert the crank through the bracket. Align the hole in the crank so that tension will be applied to the hinge pin in as straight a line as possible.

4 Insert the free end of the hinge pin through the hole in the crank.

WARNING: Keep clear of the hinge pin end when the hinge pin is nearly removed The sudden release of tension may cause the pin to spring free at high velocity.

CAUTION: *Use slow, steady rotation of the crank to avoid breaking the hinge pin.*

5 Turn the crank to pull the hinge pin out of the hinge.

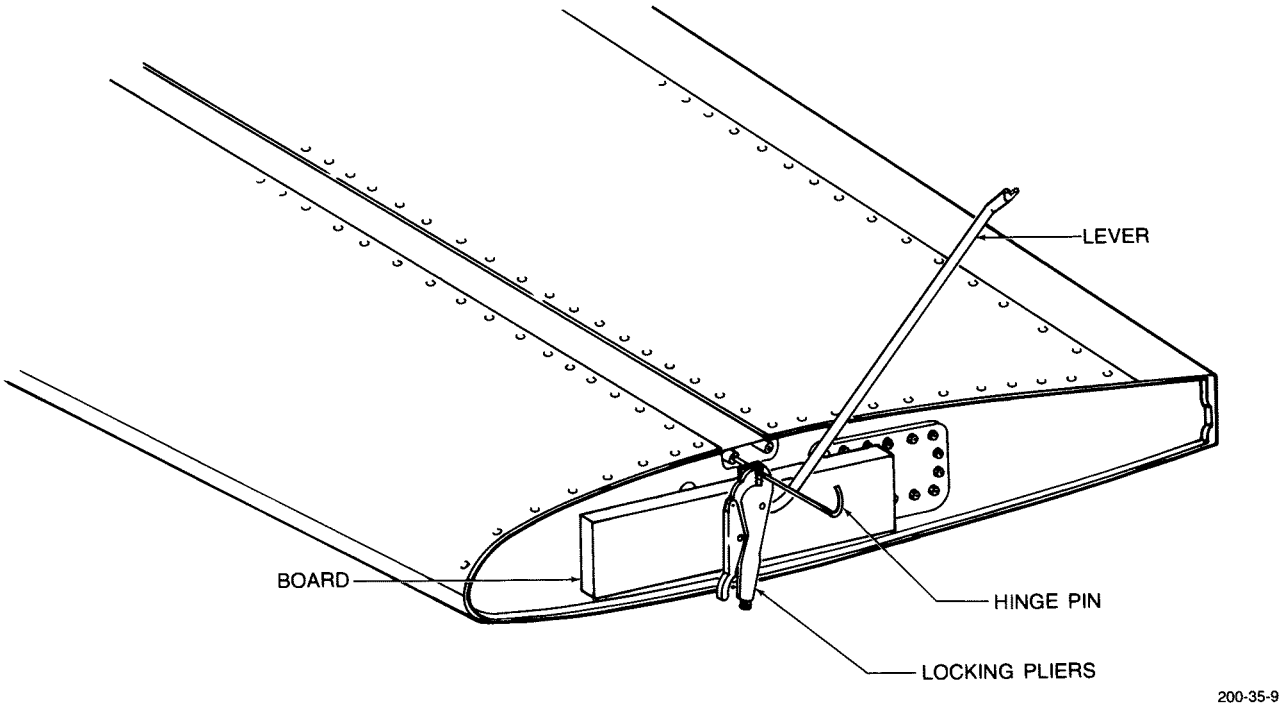


Figure 202
Hinge Pin Removal (Method #1)

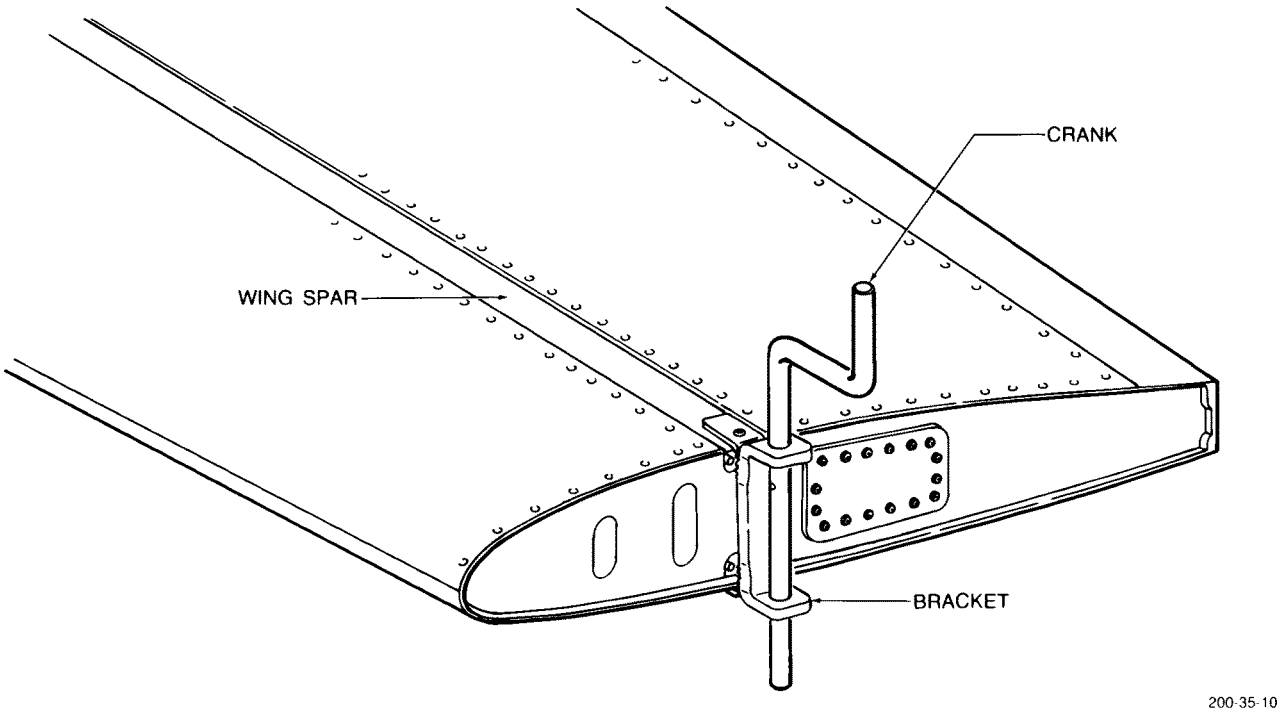


Figure 203
Hinge Pin Removal (Method #2)

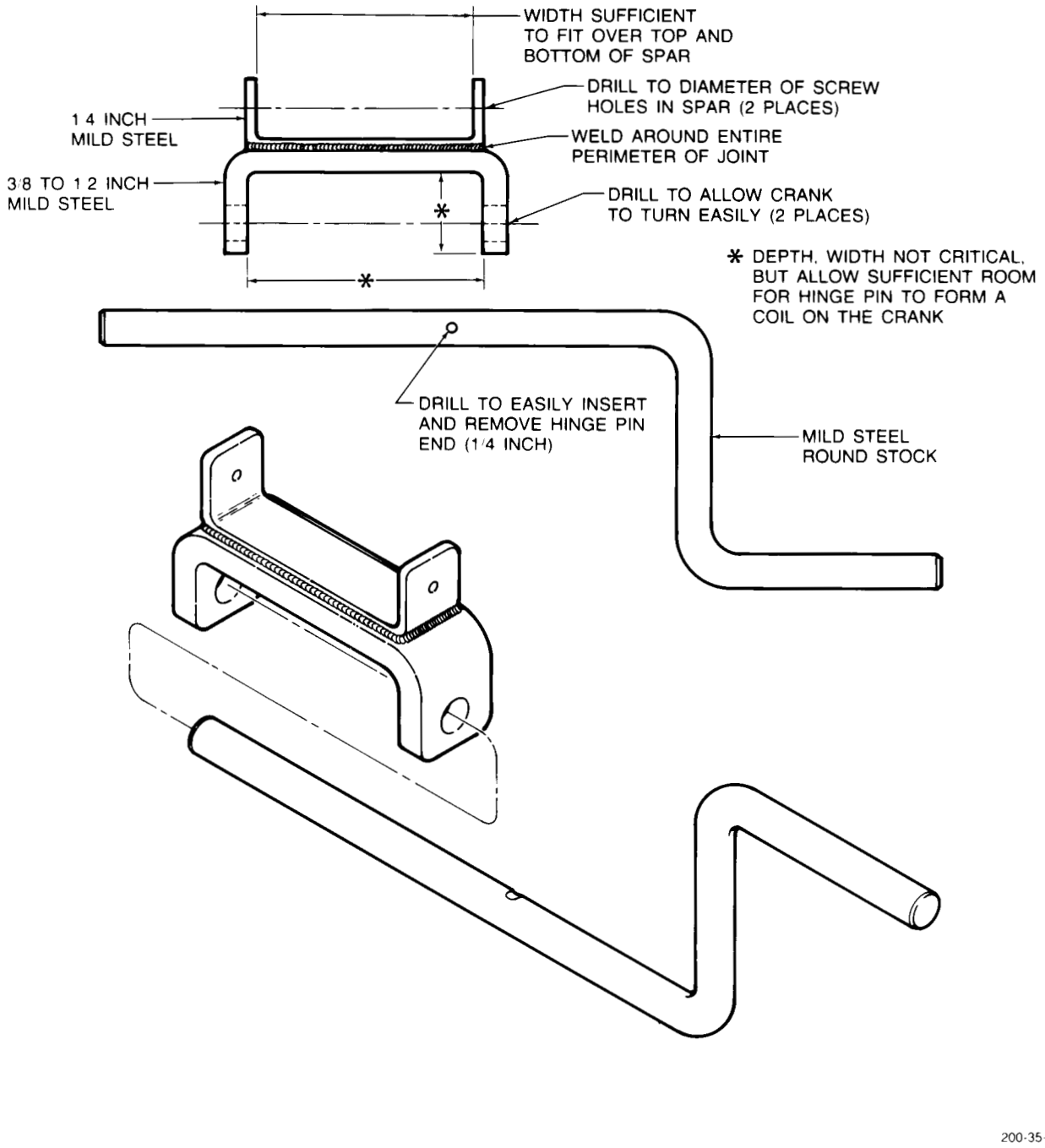


Figure 204
Hinge Pin Removal Tool

(5) Remove the support bracket when all hinge pins have been removed.

(6) After the spar is separated from the leading edge, install a temporary hinge pin into each open hinge to prevent damage, distortion, or blockage of the hinge lugs during handling.

(7) If the spar is to be replaced, remove the fuel vent line from the spar. Retain the line for installation on the new spar.

(8) Remove the outboard leading edge fuel cell from the leading edge.

(9) Inspect the interior of each fuel cell bay for loose, torn, cracked, or missing anti-chafing tape. Apply tape as shown in Figure 205 over any exposed rivets, fasteners, sheet metal edges, or protrusions which scuff the fuel cell.

(10) Pressure check each fuel cell for leaks in accordance with Chapter 28.

(11) Inspect the entire interior and structure of the wing and leading edge for corrosion or damage.

C. Assembly

(1) Place the leading edge in locally fabricated cradles as shown in Figure 206.

(2) Install the outboard leading edge fuel cell into the leading edge bay and secure all retaining buttons to the inner top surface of the leading edge.

NOTE: Ensure that the fuel cell is pressure checked for leaks before installation.

(3) Install the fuel cell as instructed in Chapter 28-10-00.

(4) Apply anti-chafing tape around the lip of the outboard leading edge fuel cell bay as shown in Figure 205 so that installed tape will seal to the spar.

(5) Grind a conical point on one end of each new hinge pin to ease installation.

NOTE: Use only new hinge pins. Provide several extra pins in case any are damaged or bent during installation. Never attempt to install a bent or distorted hinge pin.

(6) Remove the temporary pins from the hinges of the leading edge and the forward hinges of the spar. Leave the temporary pins in the aft hinges of the spar.

(7) Lubricate each open hinge lug and new pin with Brayco 300, Royco 308, or equivalent water displacing lubricant meeting specification VV-L-800 (2, Table 6, 91-00-00).

NOTE: If necessary, gently tap the spar into alignment with a rawhide mallet.

(8) Insert the pointed end of the pin into the spar and slide it into the hinge as far as possible by hand. Slide the telescoping tubes of Repair Kit 35-5885 over the pin.

NOTE: If necessary, apply Door-Ease (58, Table 6, 91-00-00) or an equivalent beeswax preparation to the hinge pins to ease installation.

(9) Place supports at a suitable height to align the telescoping tubes to the hinge. Clamp a stout bungee cord or spring to the larger tube and attach it to the spar as shown in Figure 207 so that the tube is held firmly against the hinge throughout the pin installation. Use the longest tubes at the beginning of the installation and change the tubes as necessary as the pin is driven into the hinge.

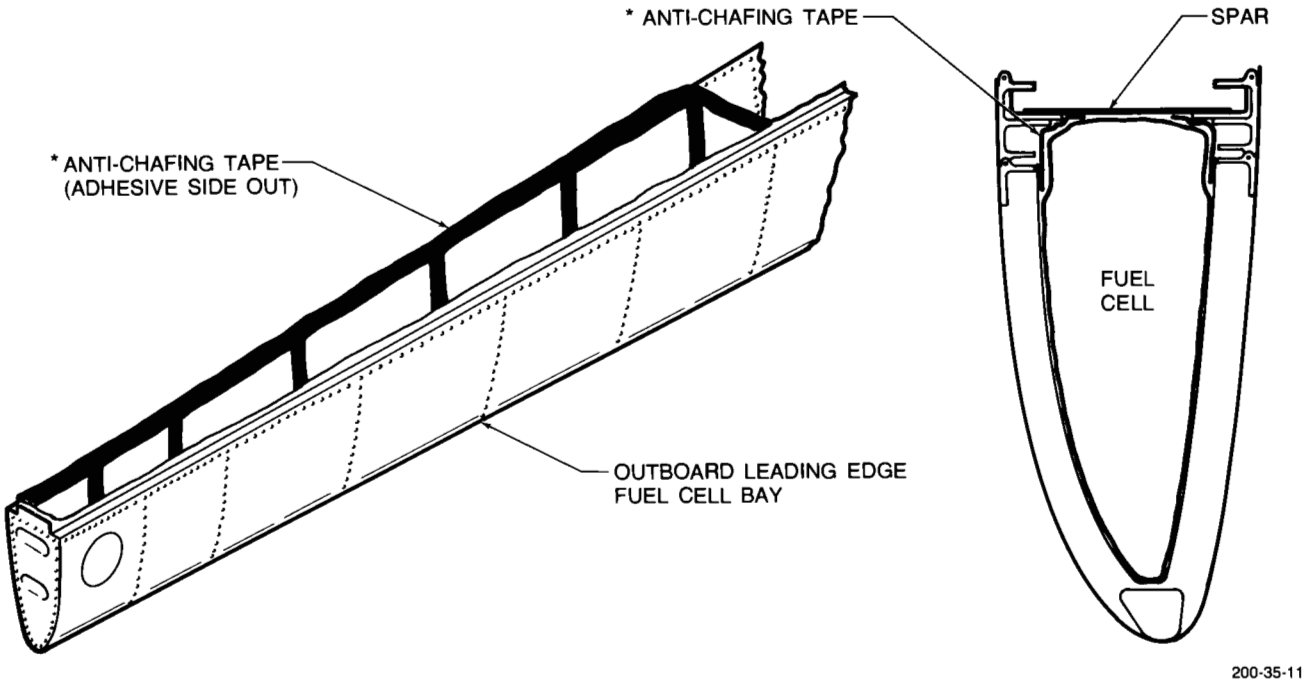


Figure 205
Anti-Chafing Tape Application

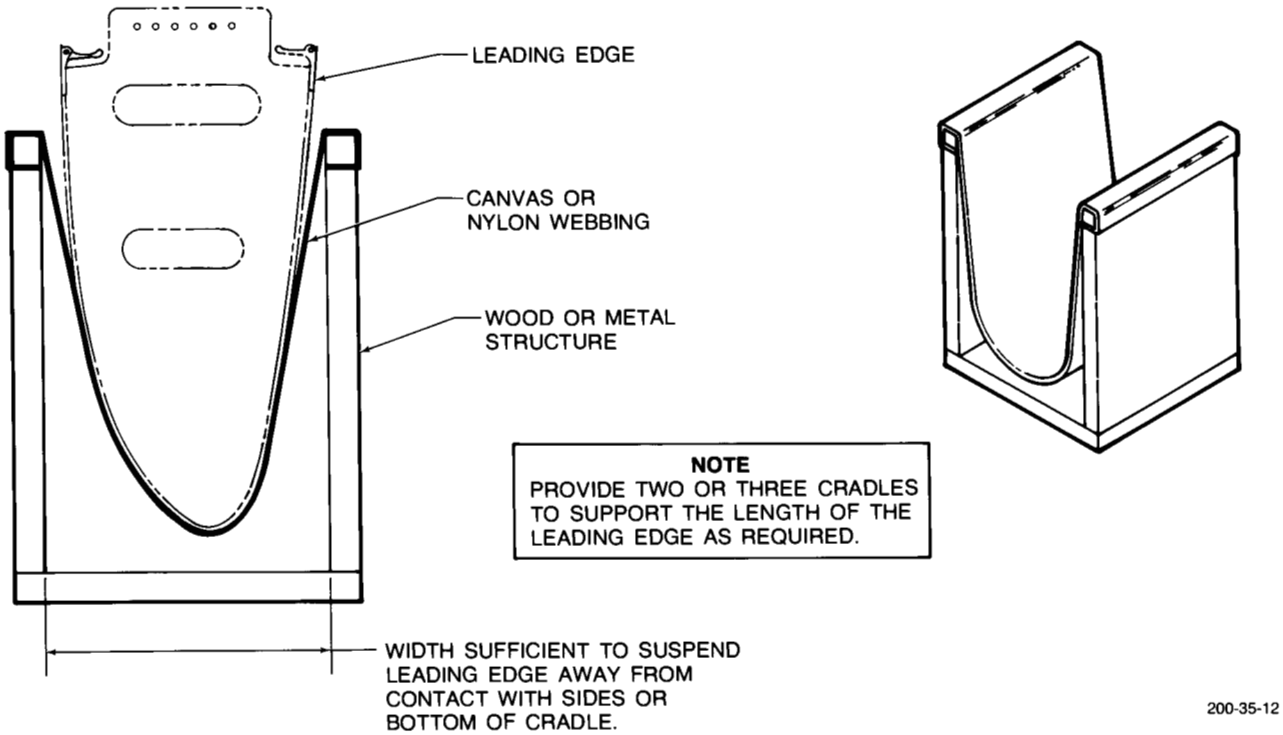


Figure 206
Leading Edge Cradle

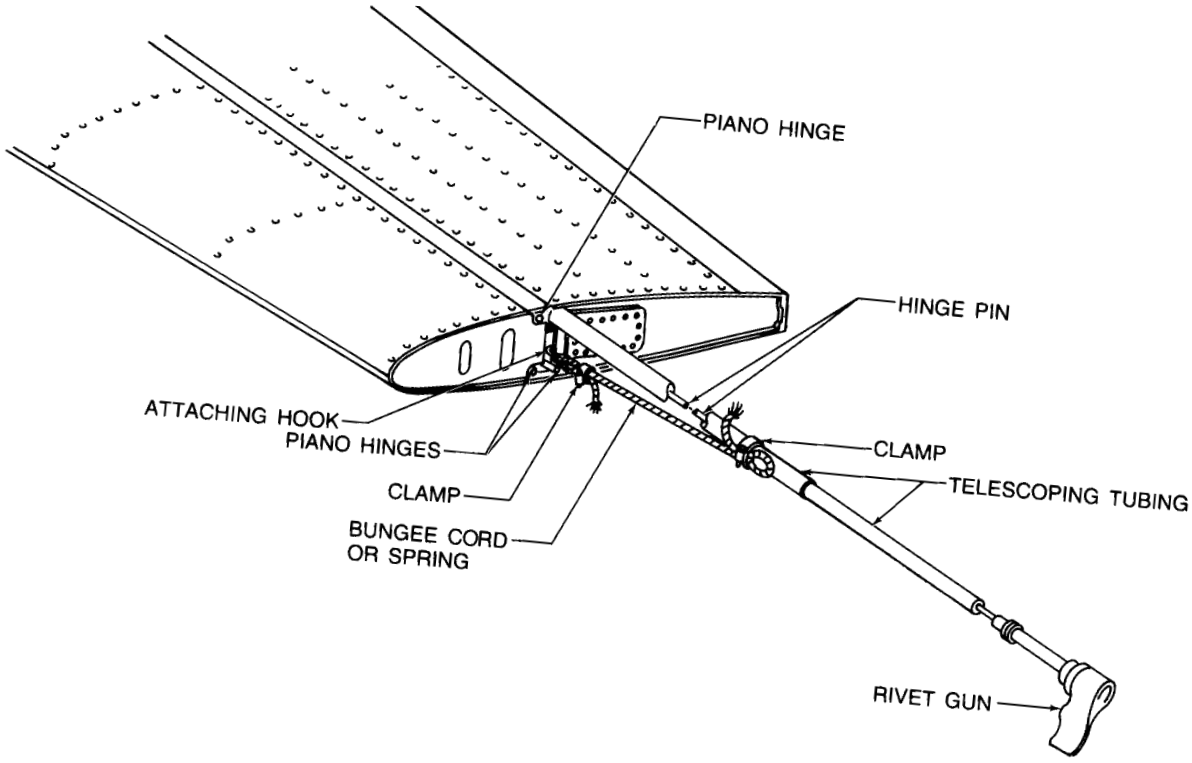


Figure 207
Hinge Pin Installation

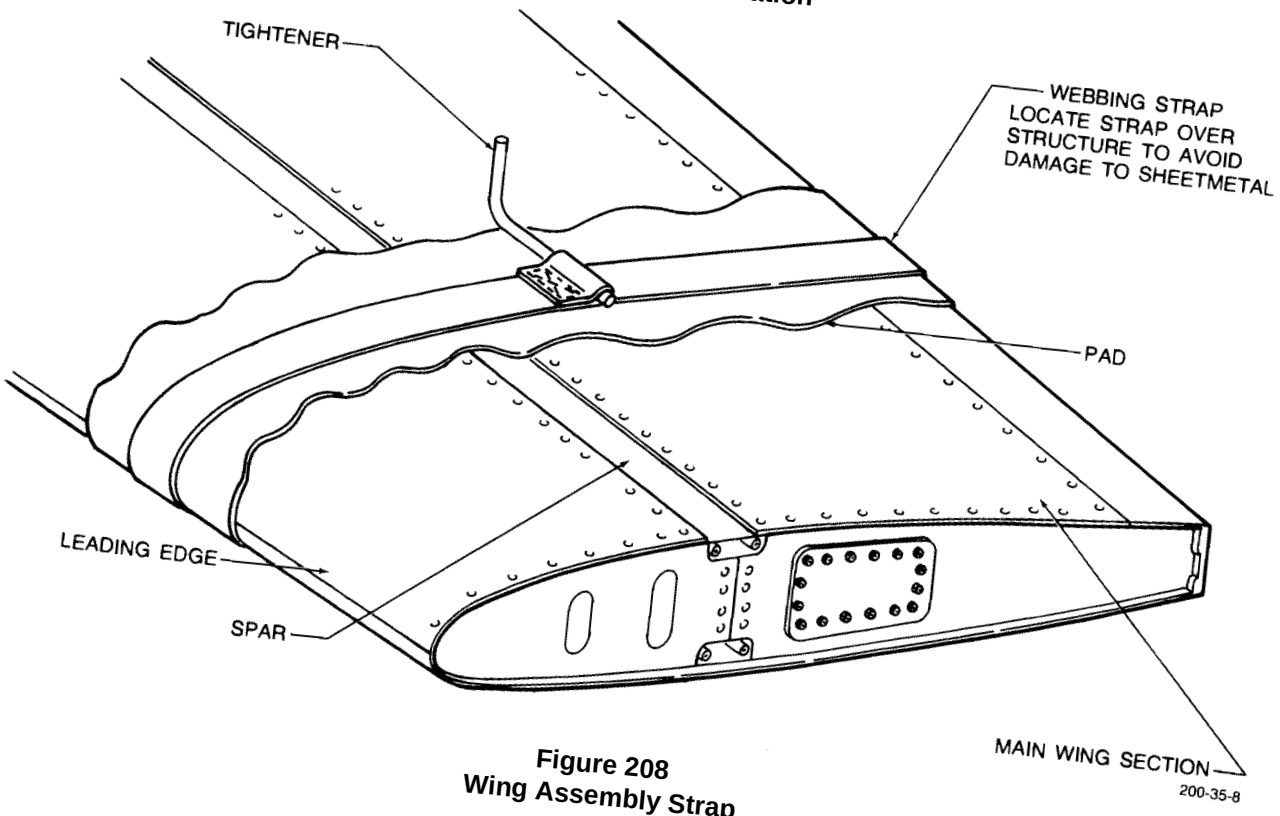


Figure 208
Wing Assembly Strap

NOTE: Always keep the end of the inner (smaller) tube tight against the spar to avoid bending or kinking the pin in the intervening gap. If a bungee cord or spring is not available, station a person at the end of the tube to hold it firmly against the spar with locking pliers.

CAUTION: *Do not allow the hinge pin to strike the wing attach fitting. Measure the hinge and the hinge pin before installation to determine the proper installation distance. Mark the hinge pin to indicate a safe stopping point.*

(10) Drive the pin into the hinge as shown in Figure 207, using a E-4 rivet gun or equivalent and a rivet set with a 3/32-inch hole. Replace the telescoping tubes as necessary to properly support the hinge pin until the hinge pin is completely driven, then remove the tubes. Continue driving the hinge pin until it is completely through the last hinge loop, but not in contact with the wing attach fitting.

NOTE: If necessary, vibrate the spar with a rawhide mallet or another rivet gun to aid installation. If a rivet gun is used, place a phenolic block between the spar and the gun to protect the spar.

(11) Insert the free end of the hinge pin into a convenient hole in the end of the spar, or tape it to the end of the spar, to keep it from interfering during the installation of the remaining pins.

(12) Repeat Steps (9) through (12) for the remaining hinge.

(13) If a new spar is installed, install the fuel vent line on the new spar in the same manner and location as on the old spar. Ensure that the vertical stiffeners have flat, smooth edges near the line and that they provide adequate clearance for the line.

(14) Remove the temporary pins from the hinges of the main wing and the aft hinges of the spar/leading edge assembly.

(15) Lubricate each open hinge lug and new pin with Brayco 300, Royco 308, or equivalent water displacing lubricant meeting specification VV-L-800 (2, Table 6, 91-00-00).

NOTE: If necessary, apply Door-Ease (58, Table 6, 91-00-00) or equivalent beeswax preparation to the hinge pins to ease installation.

(16) Position the assembled spar and leading edge on the main wing section and align the hinge sections.

NOTE: If necessary, a strap or belt of webbing may be used to cinch the spar into place as shown in Figure 208, and a rawhide mallet may be used to gently tap the spar into alignment. If a strap is used, locate it over a stiffener or other structure to avoid damage to sheet metal.

(17) Repeat Steps (9) through (12) for the remaining hinges.

(18) Trim the excess length of the pins to 4.88 $\pm$ 0.12 inches. Loop both pin ends clockwise around the nearest screw on the spar end stiffener and tighten. Allow the washer to retain the loops.

(19) Install the screws around the spar caps at the root and root tip rib.

(20) Paint the wing and spar as required.

D. Installation

NOTE: Airplane serials U-50, U-165 and After are equipped with a barrel nut and grease fitting at both the lower forward and upper forward wing bolt attach points. Airplane serials prior to U-165, except U-50 with Kit No. 99-4023-1 S are equipped with a barrel nut grease fitting at the lower forward wing bolt attach point.

WARNING: Wing bolts and nuts that have reached their life limit (6 years installed time for steel components and 10 years installed time for Inconel components) must be replaced with new components.

(1) Using a nonmetallic brush and solvent (33 or 34, Table 6, 91-00-00), thoroughly clean the wing attach fittings and hardware (bolts, washers, and nuts). Inspect the wing attach components as outlined in the latest revision of the King Air Airliner Structural Inspection and Repair Manual (P/N 98-39006).

(2) Coat the bores and washer/nut bearing surfaces of each fitting, each bolt, washer, nut, and preload indicating washer assembly with corrosion preventive compound (88, Table 6, 91-00-00).

WARNING: The soft aluminum washer between each upper fitting must be replaced any time the fittings are separated or repositioned. Replace the preload indicating washer assembly any time the load on its fitting is released. Ensure that the radiused washers, the countersunk washers and the preload indicating washer assemblies are properly positioned as indicated in the applicable Figure 209 thru 216 when installing the assembly. Similarly, the countersink of the washer must mate with the head of the bolt on the upper forward and the upper and lower aft wing bolts.

(3) With the wing supported in position with respect to the center section, insert the wing bolts into the fittings. Install the bolts, washers, and nuts as indicated in the applicable Figure 209 through 216. On airplane serials U-50, U-165 and After and prior airplanes with Kit No. 99-4023-1S installed, position the barrel nuts so that the lubrication fittings shown in Figure 217 are readily accessible.

CAUTION: *Each bolt must be inserted by hand without binding. If a bolt cannot be easily inserted, reposition the wing until the bolt moves freely.*

(4) Snug the lower forward bolt down sufficiently to provide support while still permitting rotation of the wing until the alignment marks on the fittings match.

NOTE: Torque should be applied to the nut except on installations with barrel nut assemblies installed.

(5) After the fittings are properly aligned, snug the lower forward bolt down to 1200 inch-pounds, but do not load the preload indicating washer until the other bolts have been properly torqued.

CAUTION: *Ensure that the wing bolt wrenches do not bottom out on the wing fittings during the following torquing procedure to prevent damage to the fittings and/or false torque readings.*

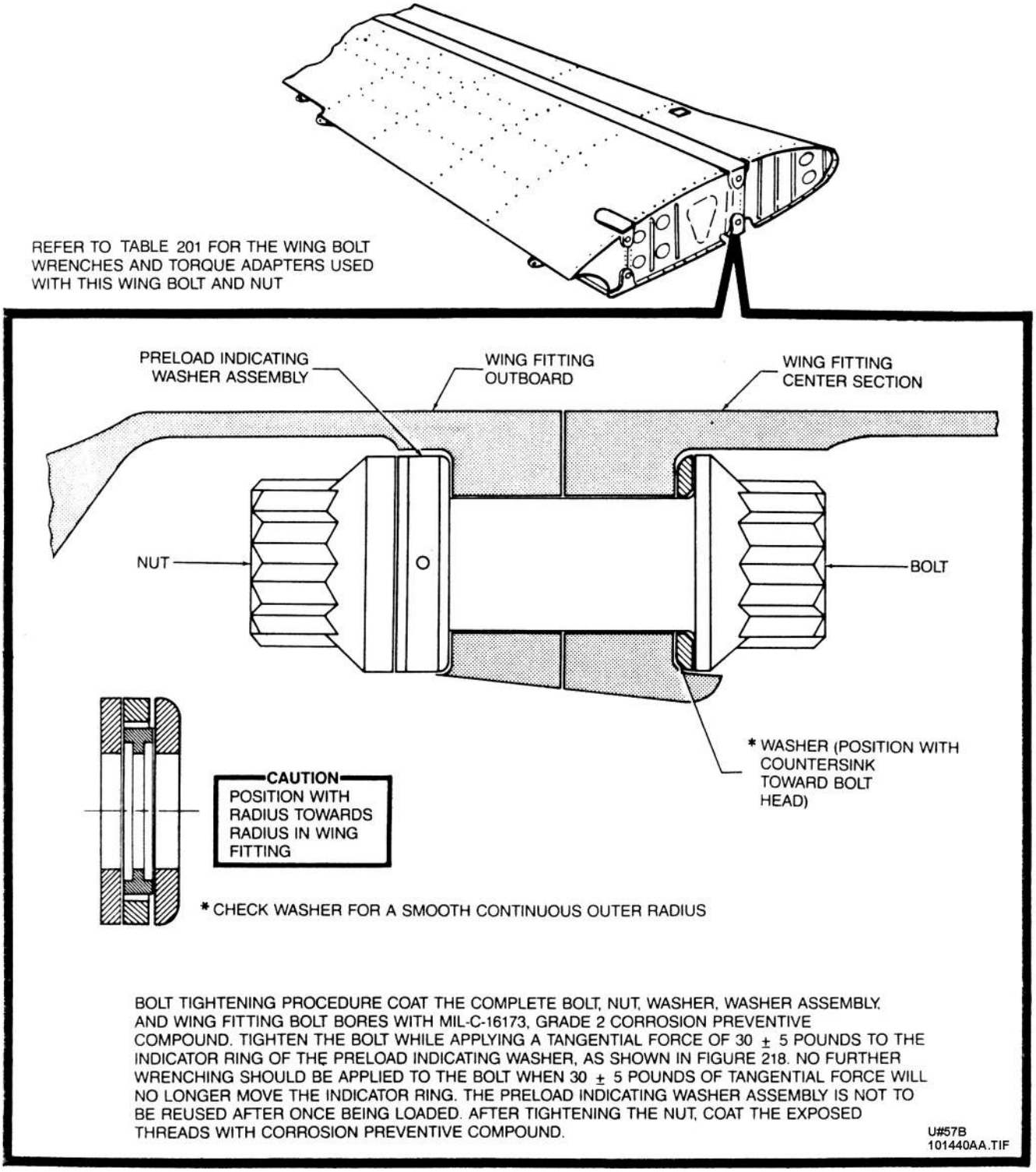


Figure 209
Lower Forward Wing Bolt Installation
(U-1 thru U-164, Except U-50 Without Kit No. 99-4023-1S installed)

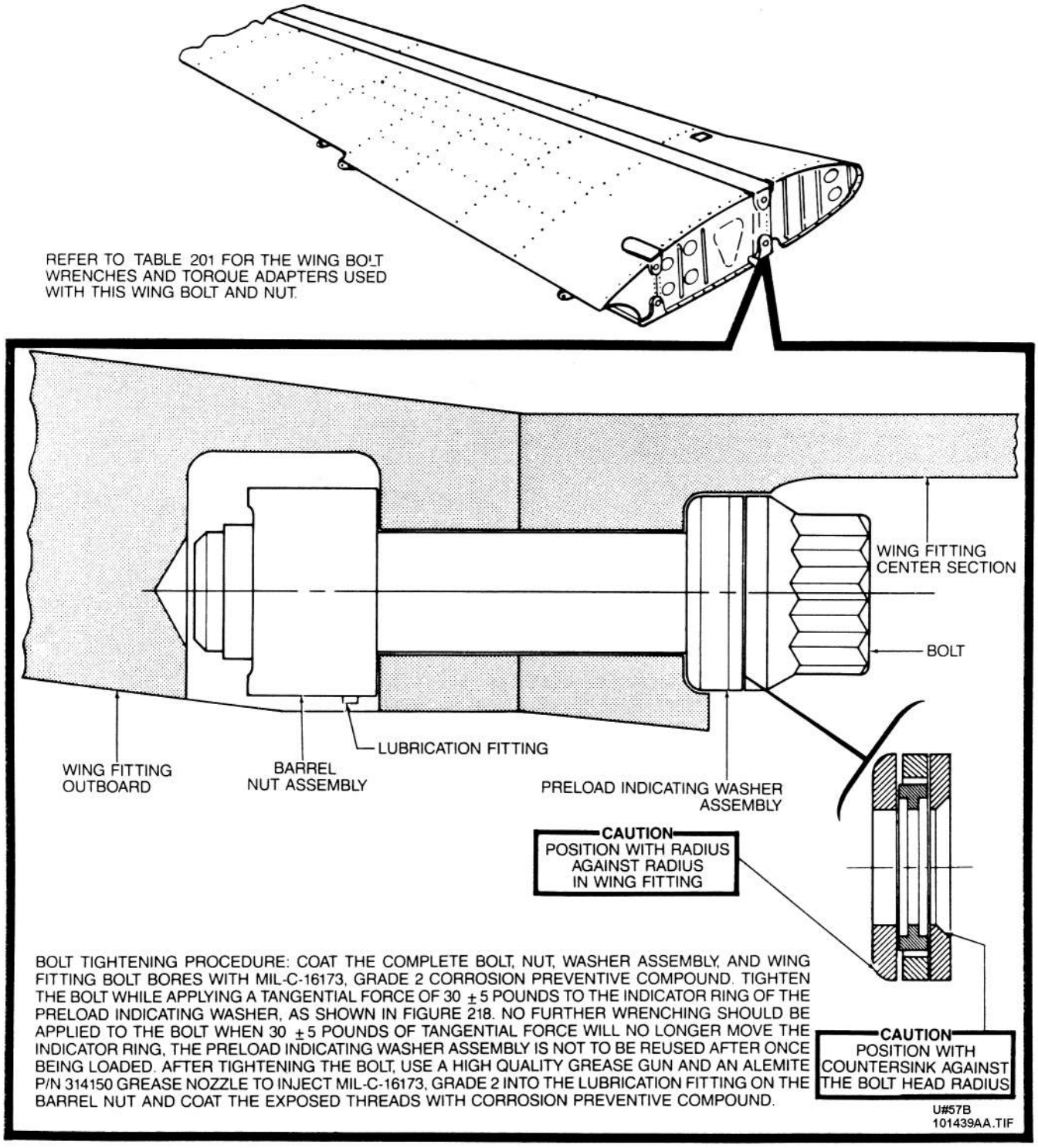


Figure 210
Lower Forward Wing Bolt Installation
(U-50, U-165 And After And Prior Airplanes With Kit No. 99-4023-1S installed)

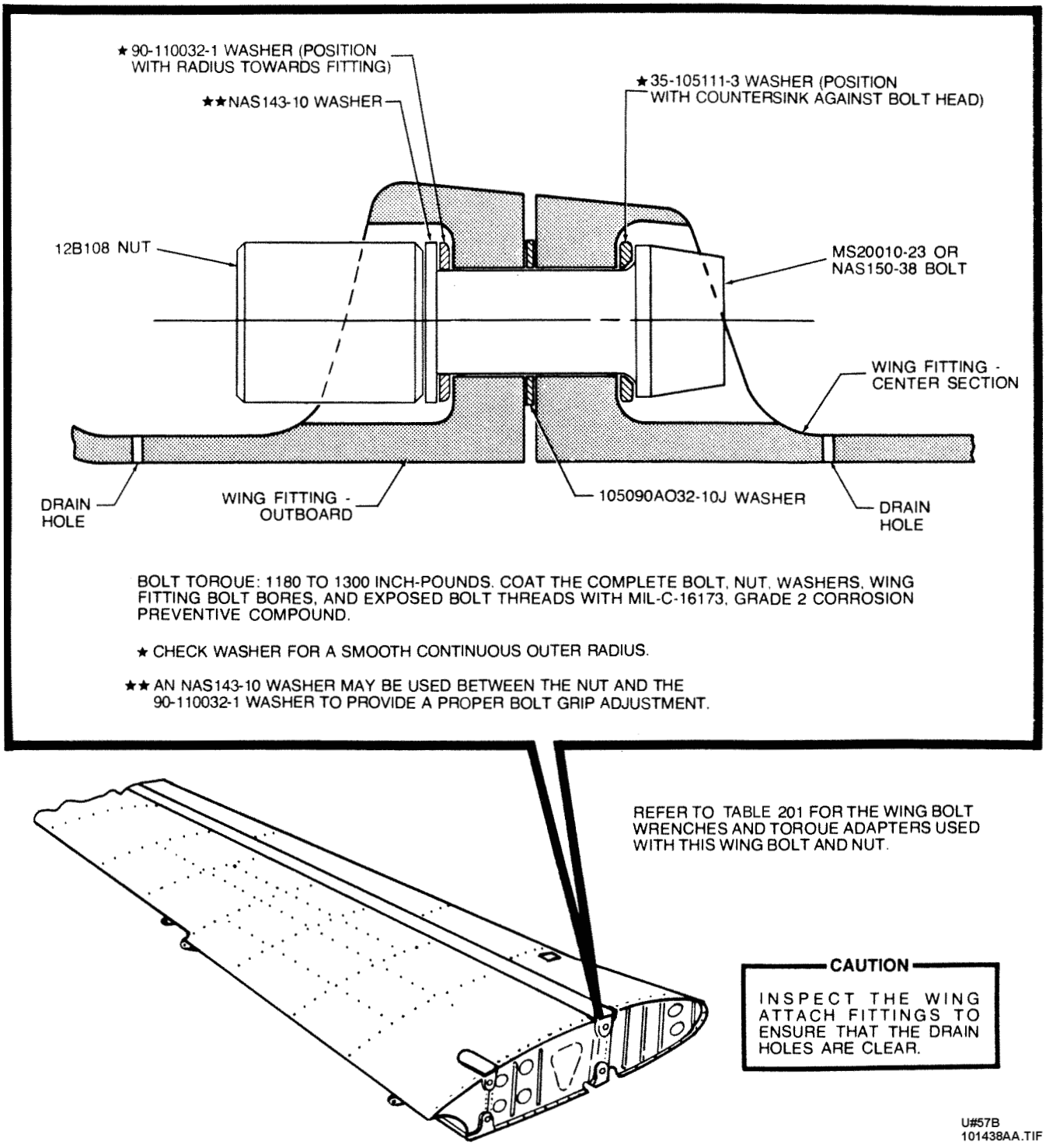


Figure 211
Upper Forward Wing Bolt Installation
(U-1 thru U-150 Except U-50 Without Kit No. 99-4023-1S Installed)

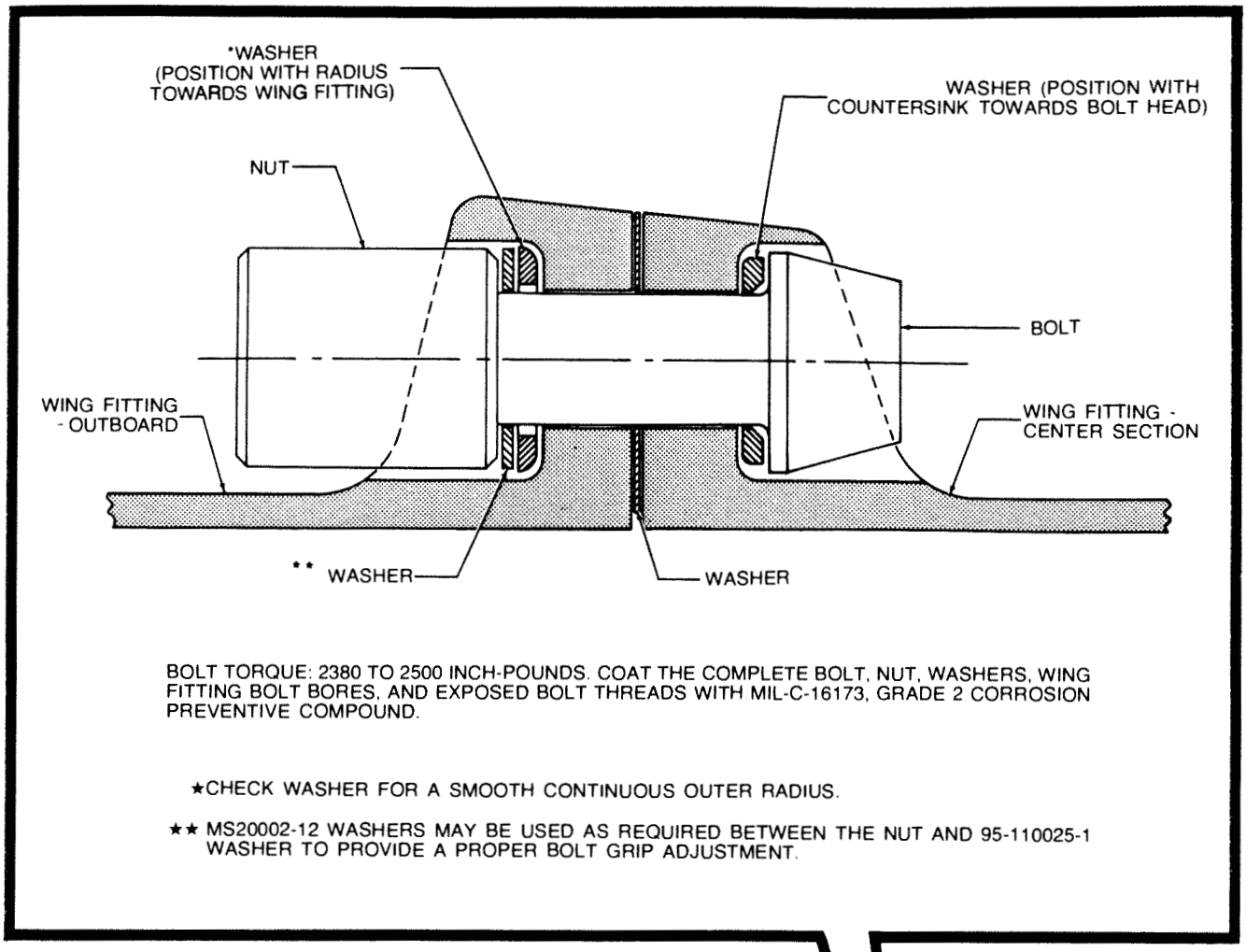
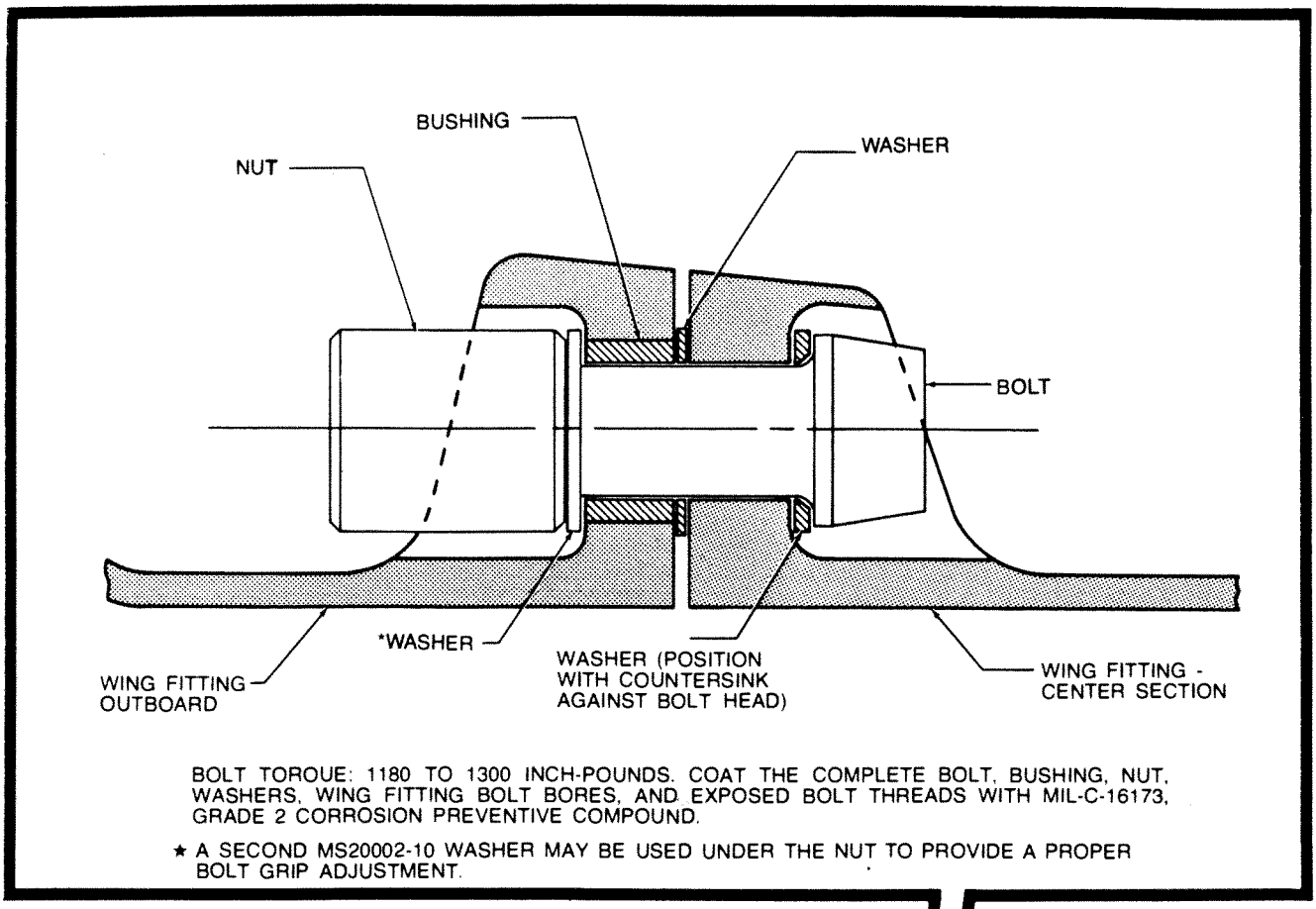
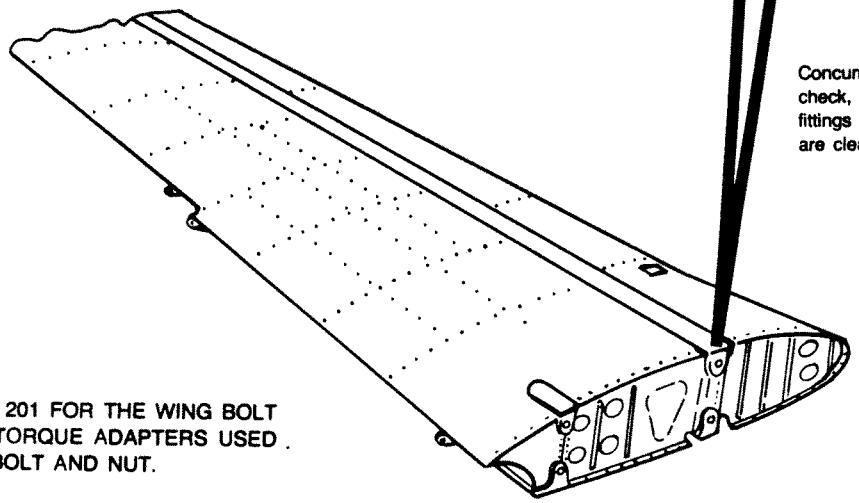


Figure 212
Upper Forward Wing Bolt Installation
(U-151 thru U-164)





NOTE

Concurrent with the scheduled torque check, inspect the upper wing attach fittings to assure that the drain holes are clear.

REFER TO TABLE 201 FOR THE WING BOLT WRENCHES AND TORQUE ADAPTERS USED WITH THIS WING BOLT AND NUT.

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Figure 213
Upper Forward Wing Bolt installation
(U-1 thru U-I50 Except U-50, With Kit No. 99-4023-1S Installed)

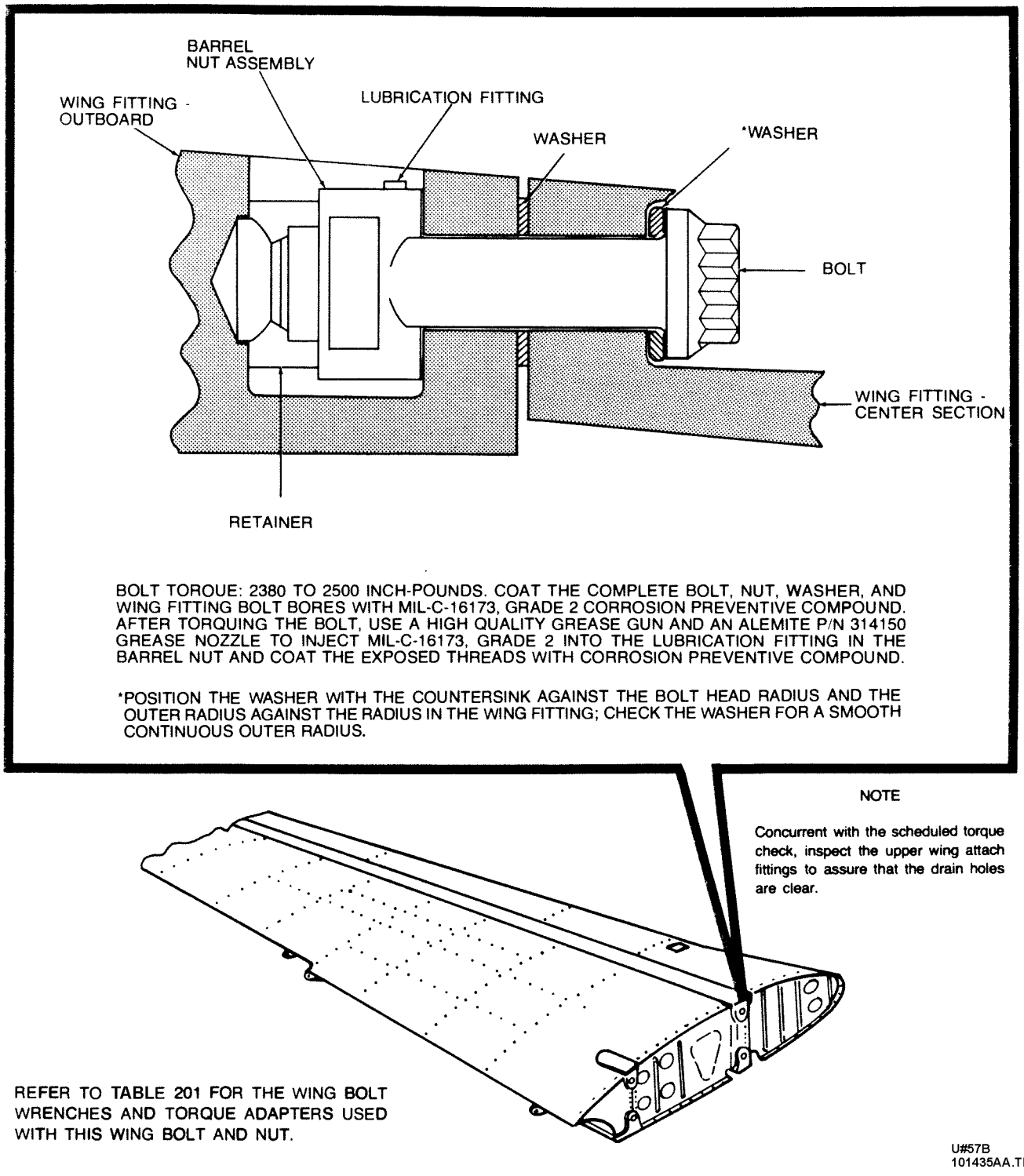


Figure 214
Upper Forward Wing Bolt Installation (U-50, U-165 and After)

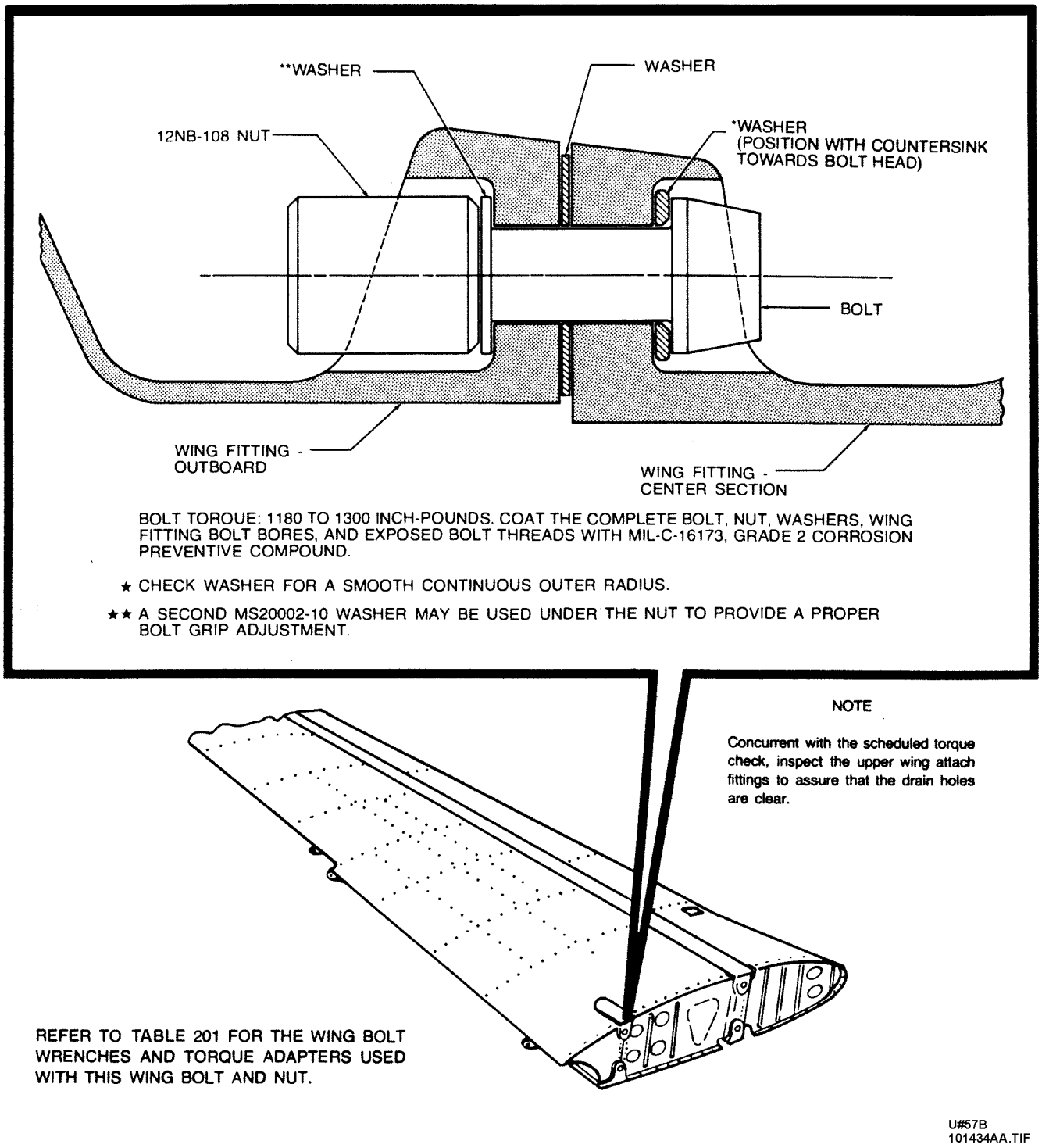


Figure 215
Upper Rear Wing Bolt Installation (U-1 and After)

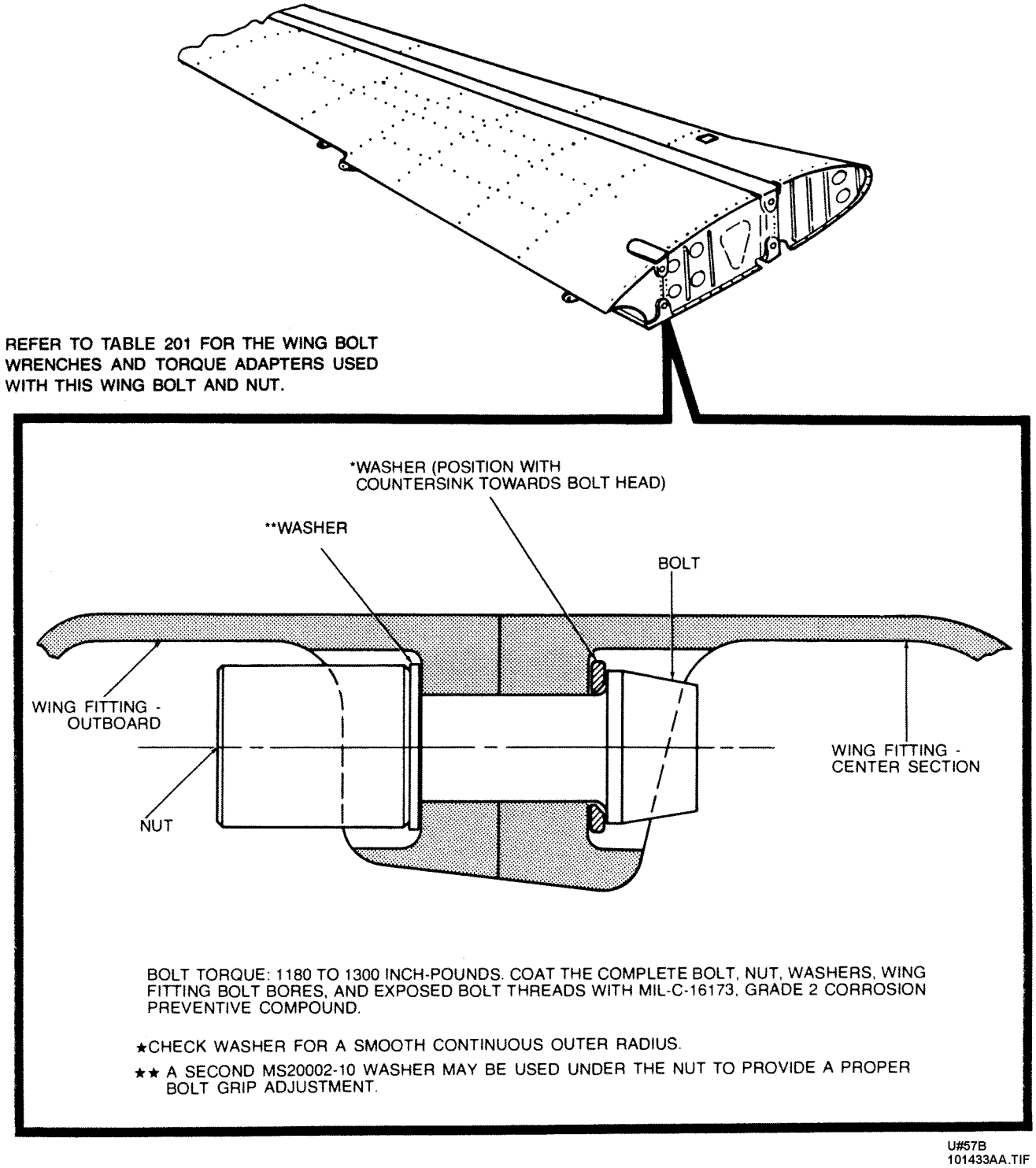


Figure 216
Lower Rear Wing Bolt Installation (U-1 and After)

Table 201
Wing Bolt Wrenches and Nut Adapters

POSITION	WRENCH PART No. (FOR BOLT HEAD)	TORQUE WRENCH ADAPTER (FOR NUT)
LOWER FORWARD	TK1817 922-5 or 15/16-inch box end wrench adapted to accept a 40-inch handle	50-590014 (U-1 thru U-164 except U-50 and airplanes without Kit No. 99-4023-1S installed)
		None Required (U-50, U-165, and earlier serials with Kit No. 99-4023-1S installed)
UPPER FORWARD	TS1222-7 or 1/2-inch Allen Wrench (U-1 thru U-150, except U-50)	TS1176-1, TS1171-1 or 50-590013-1 (U-1 thru U-150 except U-50)
	TS1222-5 or 50-590012 (U-150 thru U-164)	TS1176-2 or TS1171-2 (U-151 thru U-164)
	TS1176-6 Torque Wrench Adapter (U-50, U-165 and After)	None Required (U-50, U-165 and After)
UPPER AND LOWER AFT	TS1222-7 (U-1 thru U-196 except U-180, U-185, U-193)	TS1176-1, TS1171-1 or 50-590013-1 (U-1 and After)
	TS1222-5 or 50-590012 (U-180, U-185, U-193, U-197 and After and earlier serials which have had existing hardware replaced with Inconel bolts and nuts)	

(6) Tighten the other wing bolts to the torque specified for each respective bolt in the applicable illustration (Figure 209 through 216). On airplane serials U-50, U-165 and After, torque the upper forward bolt at the bolt head and the upper and lower aft bolts at the nut. On airplane serials prior to U-165, except U-50, torque the upper forward bolt and the upper and lower aft bolts at the nut. When a torque wrench adapter is used, the length of the adapter must be added to the length of the torque wrench for adjustment of the torque requirements as outlined in Chapter 20-00-00.

(7) On airplane serials U-50, U-165 and after and prior airplanes with Kit No. 99-4023-1S installed, tighten the lower forward wing bolt at the bolt head. On airplane serials prior to U-165 (except U-50) without Kit No 99-4023-1S installed, tighten the lower forward wing bolt at the nut. Tighten the wing bolt until the indicator ring of the preload indicating washer assembly remains Immovable under a tangential force of 30 ± 5 pounds applied as shown in Figure 218.

NOTE: The following Step applies only to airplane serials U-50, U-165 and After and prior airplanes with Kit No 99-4023-1S installed.

(8) When torquing procedures have been completed, inject additional corrosion preventive compound (88, Table 6, 91-00-00) into the lubricating fitting of the barrel nut with a high quality grease gun fitted with an Alemite 314150 nozzle. Sufficient corrosion preventive compound has been added when one of the following criteria has been met.

(a) The corrosion preventive compound emerges from between the two portions of the barrel nut assembly.

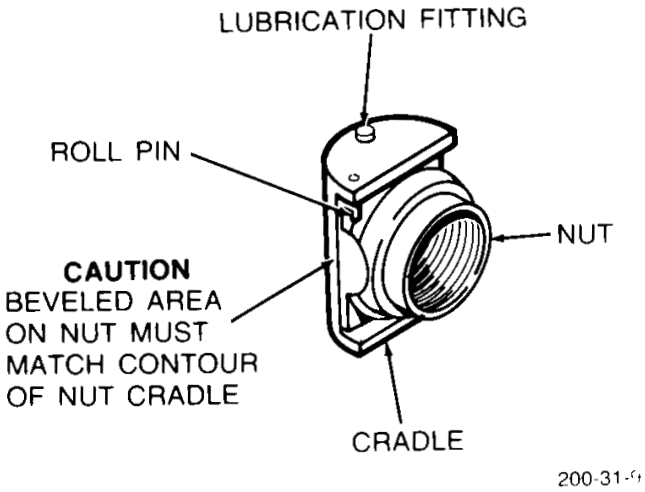
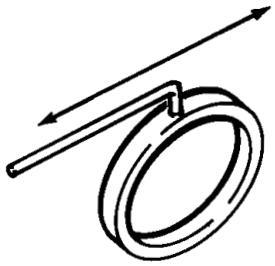


Figure 217
Barrel Nut Assembly



WHEN 30 ± 5 POUNDS OF TANGENTIAL
FORCE WILL NO LONGER MOVE THE
INDICATOR RING ON THE
PRE LOAD INDICATING WASHER,
THE BOLT IS TIGHT. DO NOT TIGHTEN
THE BOLT BEYOND THIS POINT

200-31-8

Figure 218
Lower Forward Wing Bolt Torquing

(b) The compound emerges from the threaded portion of the bolt.

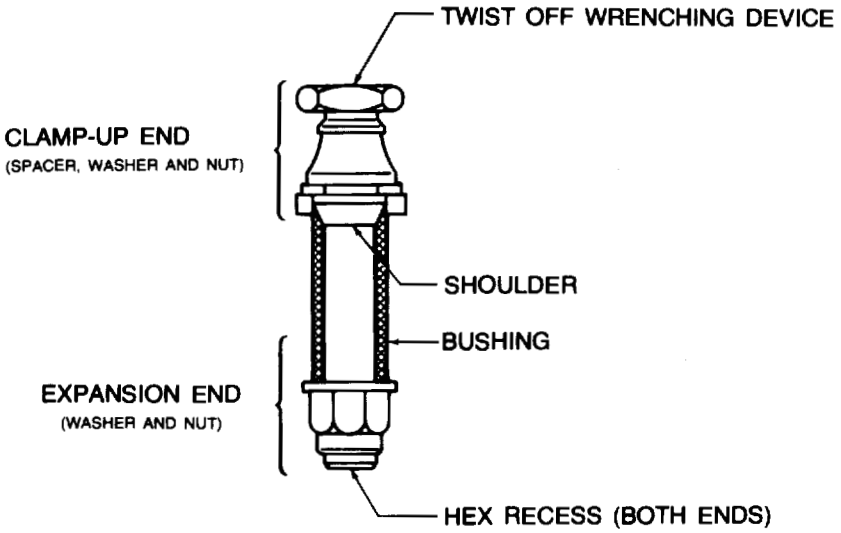
(c) The lubrication fitting fails to accept additional compound when the grease gun is repeatedly pumped.

(9) Coat the exposed threads of the wing bolts with corrosion preventive compound (88, Table 6, 91-00-00).

(10) On airplanes with the Wing Spar Bridge Kit No. 90-4090 or 100-4011 installed, accomplish the following:

(a) Lubricate the expandable bolts and fill the bolt holes In the spar bridge with corrosion preventive compound (88, Table 6, 31-00-00). Install the bolts as instructed in Figures 201 and 219, coat any exposed areas of the bolts with corrosion preventive compound.

(b) Attach the weight hangers and 125 pounds of weight to the spar strap as shown in Figure 201.



FASTENER INSTALLATION: METHOD

1. REMOVE CLAMP-UP NUT, WASHER AND SPACER. BUSHINGS WILL BE HELD IN PLACE BY THE SHOULDER.
2. SNUG THE EXPANSION NUT UP TO THE BUSHINGS.
3. CHECK TO ENSURE THAT SHOULDER PROTRUDES ABOVE PART.
4. INSTALL SPACER ON CLAMP-UP END OF FASTENER AND KEEP IN POSITION WHILE EXPANSION NUT IS BEING TIGHTENED.
5. WITH AN ALLEN WRENCH INSERTED IN HEX RECESS, AND KEEPING THE EXPANSION END WASHER PUSHED UP AGAINST THE PLATE, TIGHTEN THE EXPANSION NUT TO APPROXIMATELY 50% OF THE RECOMMENDED TORQUE VALUE. THIS SQUEEZES THE TAPERED BUSHINGS BETWEEN THE EXPANSION WASHER AND THE SHOULDER, EXPANDING THE BUSHINGS TO FILL THE HOLE.
6. INSTALL CLAMP UP WASHER AND NUT. WITH AN ALLEN WRENCH IN THE HEX RECESS, TIGHTEN CLAMP UP NUT TO APPROXIMATELY 55 INCH-POUNDS.
7. TORQUE THE EXPANSION NUT TO 250 + 50 – 0 INCH-POUNDS. THE BUSHING FRICTION AGAINST THE HOLE SHOULD PREVENT BOLT ROTATION.
8. TIGHTEN THE CLAMP-UP NUT UNTIL THE WRENCHING DEVICE TWISTS OFF. IF BOLTS DO NOT HAVE THE TWIST OFF DEVICE SHOWN IN THIS ILLUSTRATION, TORQUE THE CLAMP-UP NUT TO 200 + 40 – 0 INCH-POUNDS. THE BUSHING FRICTION AGAINST THE HOLE SHOULD PREVENT BOLT ROTATION.

200-31-29
001

Figure 219
Lower Forward Wing Bolt Torquing

(c) Bond the lower spacer block to the spar bridge with adhesive (116, Table 6, 91-00-00). Install the shim (if installed) between the upper and lower spacer blocks. Install the brackets to the lower spacer block.

NOTE: If any rough areas are noted on the mounting surfaces of either the spar bridge or spacer block, smooth the area with fine sandpaper prior to apply the adhesive.

(d) Remove the weights and weight hangers from the spar bridge.

(11) Connect the wing fuel cell interconnect tube and attach the electrical bonding cables.

(12) Connect the fuel vent lines and mate the electrical connectors.

(13) Connect the aileron and trim tab control cables at the turnbuckles in the wheel well.

(14) Connect the flap flex shaft to the outboard flap actuator.

(15) Connect the pneumatic line to the deicer boot hose.

(16) Rig the aileron and trim tab control cables as described under AILERON CONTROL SYSTEM RIGGING and AILERON TAB RIGGING in Chapter 27-10-00.

(17) Rig the flaps as described under FLAP CONTROL SYSTEM RIGGING in Chapter 27-50-00.

(18) Secure the wing-to-nacelle fairing to the center section with the attaching screws.

(19) On airplanes with the Spar Bridge Kit installed, install the spar bridge fairings.

(20) Remove the tail stand and wing supports.

(21) Refuel the airplane as described under FILLING THE TANKS in Chapter 12-20-00.

(22) Check all connections for leakage and correct as necessary.

(23) When it has been certain that there is no fuel leakage, connect the electrical power.

(24) Check for proper operation of the fuel quantity transmitters, strobe lights, and navigation light.

(25) Install all access covers and doors.

(26) Test fly the airplane to ensure proper wing and stall warning vane adjustment.

E. Adjustment

Adjustment of the outer wing panel incidence angle is not ordinarily necessary. Most prevailing wing-heavy conditions can be traced to improperly rigged ailerons or flaps. If, however, the ailerons and flaps are properly rigged, change the angle of incidence of one of the wings slightly. Since it is desirable to have the trailing edges of both wings as high as possible to minimize fuselage drag, when one wing is heavy, the trailing edge of the opposite wing should be raised, decreasing its lift relative to the heavy wing. Only if this is not possible should the trailing edge of the heavy wing be lowered. If this combination does not correct the wing-heavy condition, rig the flaps of the heavy wing slightly down. Do this only as a last resort since it will result in additional drag on the airplane.

Before loosening the wing attaching bolts to adjust the wing incidence angle, mark (but do not scribe or scratch) the position of the wing fittings on the center section fittings for a reference point. Replace both soft aluminum washers between the upper forward and upper rear fittings and the preload indicating washer assembly.

NOTE: Torque should be applied to the nut except installations with barrel nuts.

(1) After the fittings are properly aligned, snug the lower forward bolt down to 1200 inch-pounds, but do not load the preload indicating washer until the other bolts have been properly torqued.

CAUTION: *Ensure that the wing bolt wrenches do not bottom out on the wing fittings during the following torquing procedure to prevent damage to the fittings and/or false torque readings.*

(2) Tighten the other wing bolts to the torque specified for each respective bolt in the applicable illustration (Figure 209 through 216). On airplane serials U-50, U-165 and After, torque the upper forward bolt at the bolt head and the upper and lower aft bolts at the nut. On airplane serials prior to U-165, except U-50, torque the upper forward bolt and the upper and lower aft bolts at the nut. When a torque wrench adapter is used, the length of the adapter must be added to the length of the torque wrench for adjustment of the torque requirements as outlined in Chapter 20-00-00.

(3) On airplane serials U-50, U-165 and After and prior airplanes with Kit No. 99-4023-1S installed, tighten the lower forward wing bolt at the bolt head. On airplane serials prior to U-165 (except U-50) without Kit No. 99-4023-1S installed, tighten the lower forward wing bolt at the nut. Tighten the wing bolt until the indicator ring of the preload indicating washer assembly remains immovable under a tangential force of 30 ± 5 pounds applied as shown in Figure 218.

NOTE: The following Steps (4) through (5) apply only to airplane serials U-50, U-165 and After and prior airplanes with Kit No. 913-4323-1S installed.

(4) When torquing procedures have been completed, inject additional corrosion preventive compound (88, Table 6, 91-00-00) into the lubricating fitting of the barrel nut with a high quality grease gun fitted with an Alemite 314150 nozzle. Sufficient corrosion preventive compound has been added when one of the following criteria has been met:

(a) The corrosion preventive compound emerges from between the two portions of the barrel nut assembly.

(b) The compound emerges from the threaded portion of the bolt.

(c) The lubrication fitting fails to accept additional compound when the grease gun is repeatedly pumped.

(5) Coat the exposed threads of the wing bolts with corrosion preventive compound (88, Table 6, 91-00-00).

After the adjustment is completed, test fly the airplane and check the new wing position. When it is determined that the wing incidence angle is correct, check the stall warning for operation at speeds 5 to 10 mph above stalling speed. Adjustment procedures for the stall warning vane are presented in Chapter 27.

WARNING: When adjusting the wing, always replace both soft aluminum washers and the preload indicating washer. Check all wing bolts for proper torque at the first scheduled inspection.

2. WING BOLT

A. Maintenance

NOTE: Refer to Chapter 05-10-00 for Wing Bolt Inspection Criteria.

(1) At the first scheduled normal airplane inspection (not to exceed 100 hours) after the bolts have been loosened and retorqued or after removal and installation of the wings, check the wing bolts for proper torque as outlined in the applicable illustration, Figure 209 through 216. Also check the preload indicating washer at the lower forward wing bolt (Ref. Figure 218 for proper checking procedure). Concurrent with the bolt torque check, check the upper wing attach fittings to ensure that the drain holes are unobstructed.

NOTE: Airplane serials U-50, U-165 and After are equipped with a barrel nut and grease fitting at both the lower forward and upper forward wing bolt attach points. Airplane serials prior to U-165 (except U-50) with Kit No. 99-4023-1S installed are equipped with a barrel nut and a grease fitting at the lower forward wing bolt attach point. The following Steps apply to these airplanes.

(2) At the first scheduled normal airplane inspection (not to exceed 160 hours) after the bolts have been loosened and retorqued or after removal and installation of the wings, and annually thereafter, apply corrosion preventive compound (88, Table 6, 91-00-00) through each upper forward and lower forward barrel nut assembly. Use a good quality grease gun and an Alemite P/N314150 grease nozzle. Apply the corrosion preventive compound until one of the following conditions is met:

(a) Corrosion preventive compound emerges from between the two portions of the barrel nut assembly.

(b) Corrosion preventive compound emerges from the threaded portion of the wing attach bolt (past the locking feature of the barrel nut assembly).

(c) The fitting will not accept corrosion preventive compound after several actions of the grease gun.

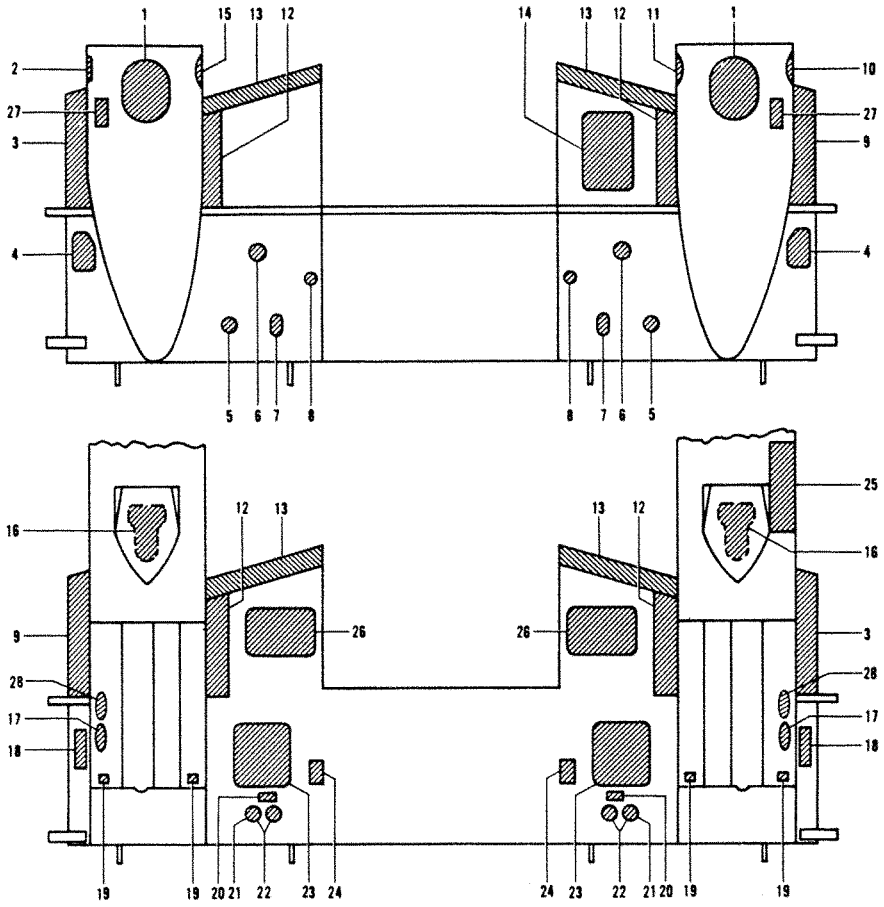
NOTE: Coat the exposed threads of the wing bolts with corrosion preventive compound (88, Table 6, 91-00-00).

3. WING INSPECTION

Inspect the wing attach bolts and nuts and the basic structural components of the center section and outboard wing main spars for corrosion and cracks as detailed in the latest revision of Hawker Beechcraft Structural Inspection and Repair Manual (P/N 98-39006).

WINGS
ACCESS OPENINGS
MAINTENANCE PRACTICES

1. WING ACCESS OPENINGS



1. Nacelle Fuel Cell Filler Neck and Tank Unit (Not on A99A)
2. Fuel Line
3. Fire Extinguisher Lines and Fuel Tank Connector Lines
4. Inverter and Wing Electrical Connections
5. Aileron and Aileron Tab Cable
6. 44 Gallon Fuel Cell Installation
7. Inboard Flap Actuator
8. Fuel Tank Unit
9. External Power Receptacle, (prior to U-89), Fuel Line Check Valve
10. Fuel Line, Fuel Purge Valve, Fire Extinguisher Lines, Engine Control Cables
11. Bleed Air Lines and Firewall Fuel Shut Off Valve
12. Starter Relays, External Power Relay (RH Only), Crossfeed Valve, (LH Only), Fuel Supply and Return Lines Fuel System Drain Tee (LH Only)
13. Electrical Wiring, Hydraulic Lines, Engine Control Cables, ITT Balance Resistor
14. Battery (Optional Location)
15. Deice Lines, Electrical Lines, Firewall Fuel Shut-off Valve, Fuel Purge Valve, Engine Control Cables
16. Primary and Secondary Boost Pumps, Drain Manifold, Fuel Cell Access
17. Rear Nacelle Access Door
18. Inverter
19. Main Landing Gear Points
20. Fuel Line Connection
21. Inboard Flap Actuator
22. Aileron Cables
23. 44 Gallon Fuel Cell Installation
24. Fuel Strainer and Drain Access
25. Engine Access Door
26. Landing Gear Retract Universal Bearing, (RH Only) Emergency Fuel Fuse Panel; (LH Only) Heater Fuel Pump, Fuel Shut-off Valve, Pressure Regulator, Strobe Light Power Unit
27. Wing Ice Light (U-89 and after)
28. Spar Inspection and Inverter Vent

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Figure 201
Wing Center Section Access Openings (U-1 thru U-164, Except U-50)

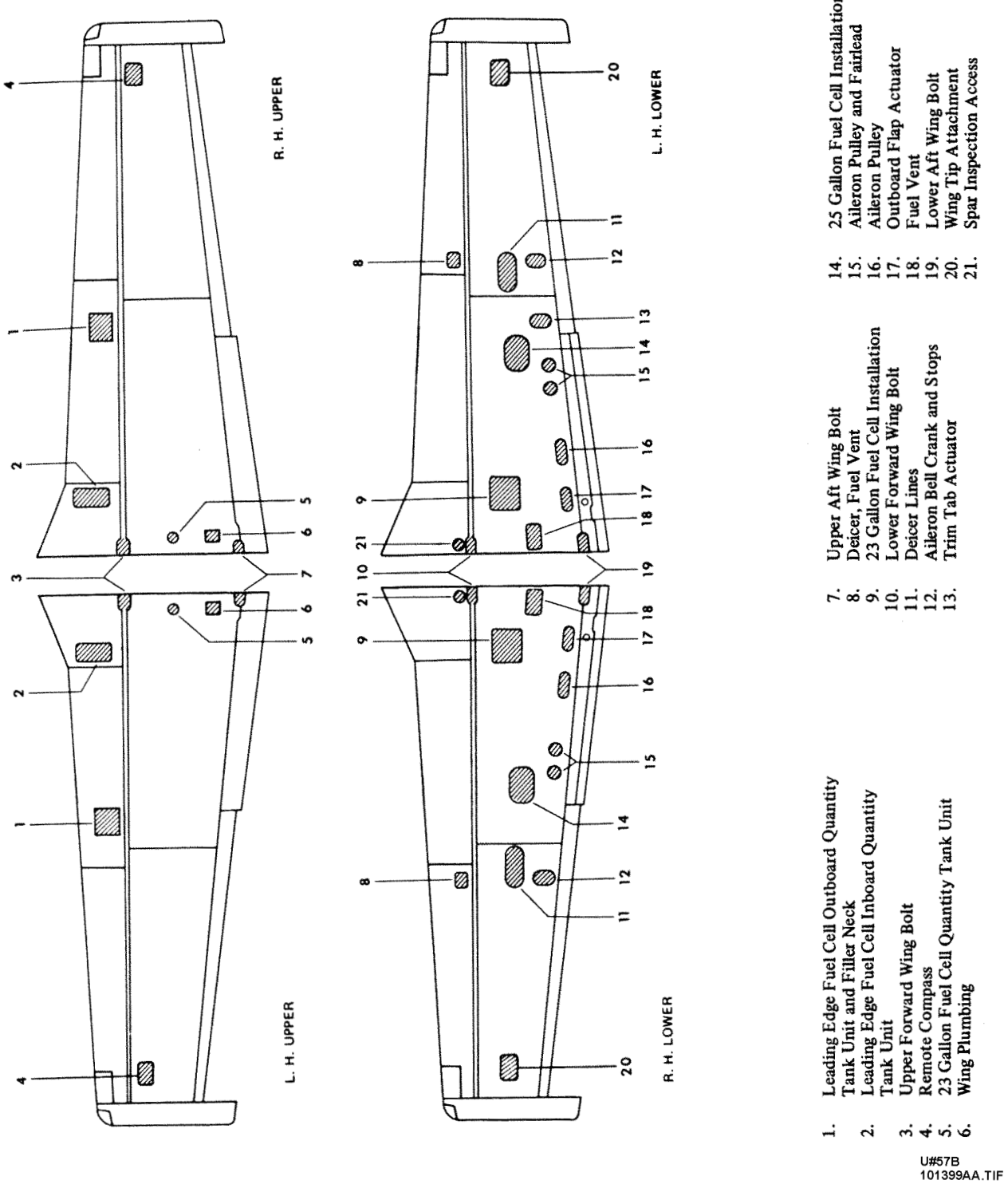

Figure 202
Wing Section Access Openings (U-1 thru U-164, Except U-50)

WINGS
TRAILING EDGE/TRAILING EDGE DEVICES
MAINTENANCE PRACTICES

1. FLIGHT SURFACES

A. Aileron Balancing

When the aileron control surface is being repainted, suspend it by the trailing edge so that excess paint will drain toward the leading edge. After any repainting or repair, the finished surface must be check balanced to ensure that its static moment about the hinge line is within the prescribed limits.

NOTE: The finished aileron assembly, with static discharge wicks (if required) installed, must have a static overbalance of between 0.0 and 3.0 inch-pounds.

The static moment of the aileron is determined by multiplying the unbalanced weight of the aileron assembly times the perpendicular distance from the hinge center line to the center of gravity when the chord line is horizontally level. The weight is measured in pounds and the distance in inches. The static moment of a 100% balanced control surface is 0.0 inch-pounds. A tail-heavy surface exhibits static under balance. A nose-heavy surface exhibits static overbalance.

B. Checking Balance

The aileron balance must be checked in a draft free area with the aileron completely assembled in flying condition. All painting, including stripes and touch-up, must be completed. The tab, tab push rod, static wicks, and hinge bolts must be attached. The chord line must be horizontally level and the hinge line must be properly supported when the static moment is measured. Although many different methods of check balancing exist, they can be categorized under the following two headings:

(1) Counterbalancing - The application of a known force or weight at a measure distance from the hinge line to counter the unbalance moment of the aileron assembly.

(2) Actual Force Measurement - Measurement of the force applied by the aileron surface on a single support at a known distance from the center line of the hinge.

NOTE: Counterbalancing is the simplest method of check balancing.

2. BALANCING PROCEDURES

A. Equipment Required to Perform Check Balancing by Force Measurement Method

(1) A stand with knife edge supports as illustrated in Figure 201. The knife edges should be in the same horizontal plane.

(2) A certified beam balance calibrated in units of 0.01 Ib. or less. The balance should have a flat weighing platform and its capacity should equal tare plus 2.0 Ibs minimum.

(3) A support spindle similar to the illustration and levelling blocks, as required. (Blocks + spindle = tare).

(4) A straight edge, rule and spirit level.

B. Force Measurement Method

(1) Locate the cord line at the inboard end of the surface by placing a straight edge at the inboard end so that one edge bisects the center of the hinge point and the trailing edge.

(2) Mark the chord line.

(3) Place a small platform scale under the trailing edge if the control surface is tail-heavy or under the leading edge if it is nose-heavy.

(4) Place the upper end of the spindle under the trailing edge of the tail-heavy surfaces or under the leading edge of the nose-heavy surfaces. The spindle must be vertical throughout the balancing procedure. Hold a spirit level against the marked chord line and level it by extending or contracting the spindle, or by using blocks and shims under the spindle.

(5) Measure the perpendicular distance from the hinge center line to the point supported by the spindle. Ensure that the spirit level and rule are removed from the surface and read the reaction on the beam balance.

(6) Calculate the static under balance moment "M" from the formula:

M = D (R-T) inch-pounds where,

D = Perpendicular distance from the hinge center line to the spindle point (inches).

R = Reaction (Pounds) read from the beam balance.

T = Tare, i.e. spindle plus levelling blocks or shims on the scale platform (Pounds).

EXAMPLE

D is 10.0 inches, R is 1.28 Ib. and T = 1.13 Ib.

M = 10.0 (1.28 - 1.13); M = 1.50 inch-pounds

M is within the prescribed range which is satisfactory.

If "M" is not within the prescribed range, refer to Step (9) under BALANCING PROCEDURE COUNTER BALANCING METHOD.

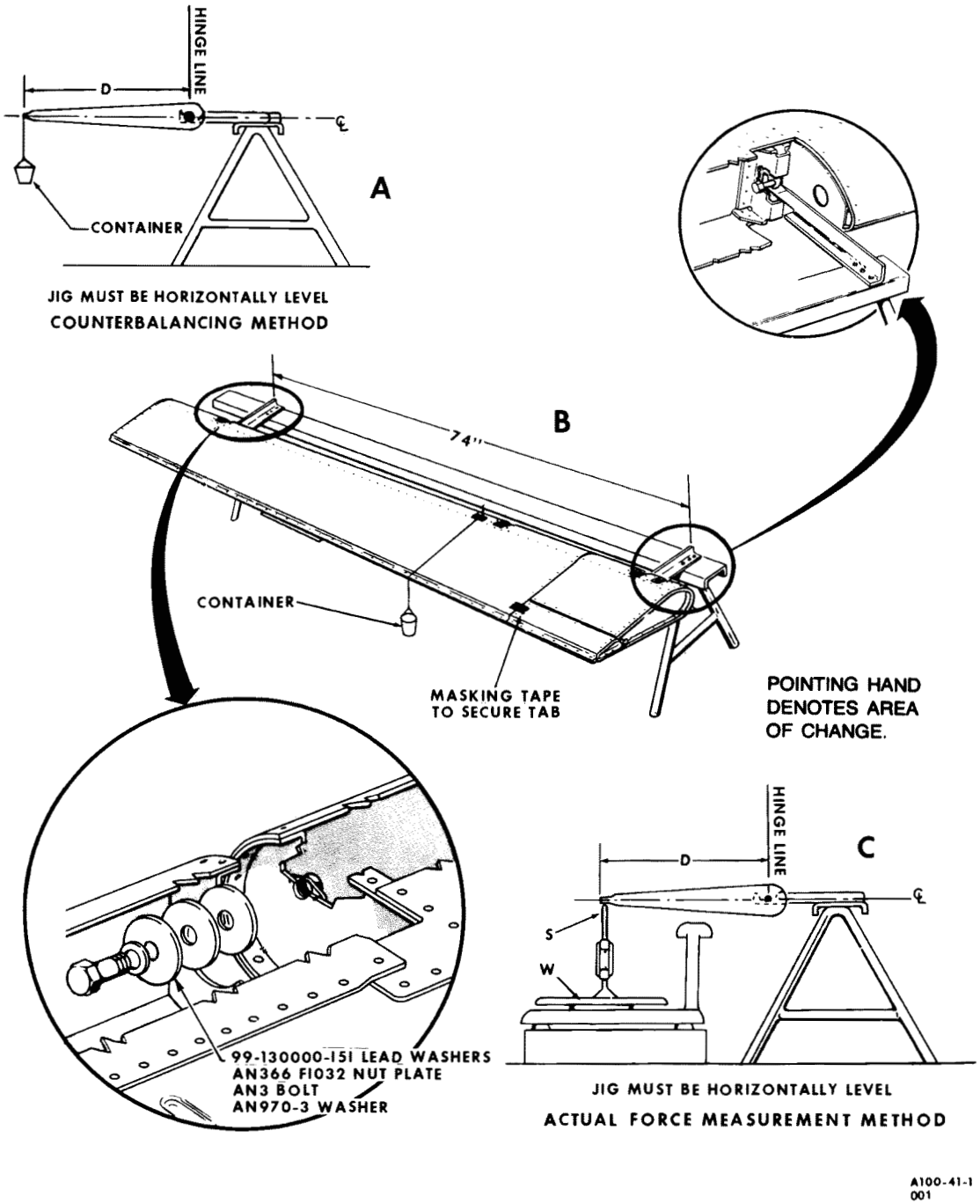

Figure 201
Wing Center Section Access Openings (U-1 thru U-164, Except U-50)

C. Equipment Required to Perform Check Balancing by Counterbalancing Method

(1) A stand with knife edge supports as illustrated in Figure 201. The knife edges must be in the same horizontal plane.

(2) A paper cup or similar light weight container.

(3) Approximately 2 pounds of lead shot.

(4) A certified beam balance weighing device calibrated in units of 0.01 pound or less.

(5) A straight edge, ruler, and spirit level.

D. Counterbalancing Method

(1) Locate the chord line by placing a straight edge at the inboard end of the aileron assembly so that one end is on the trailing edge and the other end is centered on the leading edge. Mark the chord line with a suitable marker such as a grease pencil, then remove the straight edge.

(2) Secure the trim tab (LH only) in its neutral position with a small piece of masking tape.

(3) Install the correct size bolts in the hinge brackets and mount the aileron on the knife edge supports. Ensure that the aileron is free to rotate about the hinge line.

(4) To determine if weight should be added or removed, suspend a paper cup from a point near the center of the aileron trailing edge. Use a short length of small diameter string secured to the surface with a small piece of masking tape as illustrated in Figure 201. The cup must be free to hang vertically.

(5) Add small quantities of lead shot to the cup until the aileron balances with the chord line level. Check this by holding the spirit level aligned with the marked chord line.

(6) The distance "D must be perpendicular to the hinge line. Measure "D" from the hinge line to the suspension point of the cup.

(7) Remove the cup, contents, and string, then weigh them.

NOTE: Since any weighing error is magnified by the distance "D", weighing is most important and must be done carefully on scales that are certified for accuracy.

(8) Calculate the static balance as follows:

(a) The weight of the cup and contents is designated by "W".

(b) The over or under balance moment is designated by "M".

(c) M = W x D.

(d) The following is a typical example of a balancing calculation: Assume the aileron is overbalanced (nose-heavy) and the paper cup was suspended from the trailing edge. Assume that the aileron balances with the chord line level at "W = 0.150 pound and "D = 10.0 inches", then....

M = 0.150 x 10.0

M = 1.50 inch-pounds. The product of "W x D".

In this instance, "M" is within the required static balance range and is therefore acceptable.

(9) If the static balance is not as noted in BALANCING THE AILERON, add or remove the lead washers as needed to attain the desired balance.

NOTE: A maximum of three lead washers (99-130000-151) may be added on the flange of the 99-130000-61, -65. -67. -73, -75 and -33 ribs to provide static overbalance. Attach the washers with AN3 bolts of suitable length, and use one AN970-3 washer between head of bolt and lead washer (see Figure 201).

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 61 - PROPELLERS

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PROPELLERS
PROPELLER ASSEMBLY
DESCRIPTION AND OPERATION

1. REVERSING PROPELLER (PT6A-20 ENGINE)

Each constant speed, full feathering, reversible propeller is controlled by engine oil from a single acting, engine-driven governor backed by an overspeed governor and a power turbine governor. A servo piston mounted on front of the propeller spider hub moves the propeller blades through links connected to the trailing edge. Centrifugal counterweights on each blade, in conjunction with a feathering spring on the servo piston, increase pitch (decrease rpm) to the feathered position as governor oil pressure is relieved. The feathering spring completes the feathering operation when centrifugal twisting moment is lost as the propeller stops rotating. The propeller automatically feathers on engine shutdown, preventing the free turbine from windmilling. To further expedite feathering, the normal feathering mechanism is backed by an automatic feathering system that provides a means of immediately dumping oil from the propeller governor to enable the feathering springs to start feathering the propeller blades as soon as engine torquemeter oil pressure drops below 6.5 psi at power settings of 90% N_1 or greater.

The servo piston of the reversing propeller is connected by three spring loaded sliding rods to a mechanical feedback ring located behind the propeller. A switch brush block mounted in a bracket on the front of the engine rides on the low stop collar to actuate the valve that shuts off the flow of oil to the propeller for the secondary flight idle stop. The movement of the low, stop collar is transmitted by two carbon blocks through a reversing lever and connecting linkage to the constant speed governor pilot valve, which is positioned to maintain the blade angle selected by the controls. The reversing lever is also connected to the power turbine governor to limit propeller rpm in the reverse position. A push-pull cable extends from the reversing lever aft to a control cam connected to the cockpit controls and fuel control unit. Movement of the controls is transmitted through the cam and interconnecting linkage to the fuel control unit and governors to regulate propeller speed and pitch.

NOTE: For the description and operation of the reversing propeller on PT6A-27, PT6A-28 and PT6A-36 engines refer to 61-20-00.

2. PROPELLER AND GOVERNOR

A. Checks and Adjustments

Full Feathered Angle..... $87 \pm 0.5°$ at 30 inch station

Mechanical Reverse Pitch Stop..... $-11 \pm 0.40°$

Angle at which prop yoke just touches the 3 low stop rods (which move the mechanical low stop collar) is $20.2 \pm 0.1°$ when blades are held toward decrease position.

Low RPM..... 1625 to 1675 (at detent)

Maximum RPM..... 2200 (at take-off)

NOTE: When a propeller synchronizer or synchrophaser is installed there is no change in governor setting procedures. The synchronizer or synchrophaser must be turned off while making these settings.

PROPELLERS
PROPELLER ASSEMBLY
MAINTENANCE PRACTICES

1. REVERSING PROPELLER

A. Lubrication

Lubricate the propeller hub and low pitch stop rods at 100-hour intervals as detailed in the Lubrication Chart in Chapter 12.

NOTE: When greasing propeller hubs, care should be taken to avoid blowing out damp gaskets. Stop pumping grease when it starts to ooze from between the clamp and the hub.

B. Cleaning Hardware

As a rust preventive measure, propeller-reversing hardware may be rubbed clean with steel wool and coated with aerosol clear lacquer spray.

2. PROPELLER

A. Removal

(1) Remove the upper forward cowl from the engine.

(2) Remove the two nuts, washers, and screws securing the propeller deicer brush block assembly to the engine bracket.

CAUTION: *To prevent damage to the brushes during and after removal, tape them in place before removing the brush block.*

NOTE: Step (3) should be omitted on airplane serials U165 and After or earlier airplanes with Kit No. 99-3026-1s installed.

(3) Remove the carbon block from the proximity sensor bracket by disconnecting the spring and removing the attaching bolt.

(4) Remove the spinner dome by removing the attaching screws from around the rear circumference.

(5) Disconnect the propeller reversing lever from the propeller linkage.

(6) Install the feedback ring puller (P/N TK 1573-918) as shown in Figure 201 and pull the low pitch stop collar fully forward.

CAUTION: *Make sure that the tool is not cocked to avoid damaging the propeller. Take the precautions necessary to avoid bending or otherwise damaging the three spring-loaded rods and the low pitch stop collar (brass ring).*

(7) Remove the safety wire from the propeller mounting bolts. Using the tools noted in Figure 202, remove the eight bolts securing the propeller in place and remove the propeller from the airplane.

B. Installation

NOTE: If for any reason a propeller blade is replaced, be sure it is of the same category as the remaining blades on the hub. Because of the weight difference between Category I and Category II propeller blades, it is not permissible to mix Category I and Category II blades on the same hub. All blades on each hub must be of the same category. Category I blades have no category markings and Category II blades are identified as such where the usual blade identification markings appear.

(1) Place the new packing seal over the engine shaft.

(2) Pull the low pitch collar fully forward with the puller (P/N TK 1573-918) (Ref. Figure 201).

CAUTION: *Make sure that the tool is not cocked to avoid damaging the propeller. Take the precautions necessary to avoid bending or otherwise damaging the three spring-loaded rods and the low pitch stop collar (brass ring).*

(3) The 83001-2 low pitch stop collar is not installed on new propellers delivered from the factory as a precaution against damage during shipment. Only one of the three jam nuts is installed on one of the 83002-2 rods. This one nut is locked in position on the rod with Loctite for proper positioning of the 83001-2 collar. When a new propeller is being installed, proceed as follows:

(a) Install the two jam nuts on the two 83002-2 rods.

(b) Loosen the nut on the front end of each rod.

(c) Install the 83001-2 collar on the ends of the three 83002-2 rods. The rods can be screwed into the collar by applying a thin wrench to the flats provided on the rods. Do not tighten the jam nuts at this time.

(d) Install the propeller on the engine as indicated in Steps (4) and (5) below.

(e) Tighten the jam nut that is locked in position on the rod to 12 foot-pounds.

(f) Using a dial indicator of the type used by machinists, check the runout of the low pitch stop collar. The total runout should not exceed 0.010 inch. In the event of excessive runout, adjust the two remaining 83002-2 rods and jam nuts as required to obtain the specified runout.

(g) Torque the two remaining jam nuts to 12 foot-pounds.

(h) Torque the nut on the front end of each 83002-2 rod to 12 foot-pounds.

(4) Install the propeller on the engine by inserting the two mounting studs on the propeller into the mounting holes on the drive shaft of the engine.

(5) Install the propeller mounting bolts and washers. Refer to Figure 202 for required tools and torquing procedures.

NOTE: Step (6) should be omitted on airplane serials U-165 and After, and earlier airplanes which incorporate Kit No. 99-3026-1.

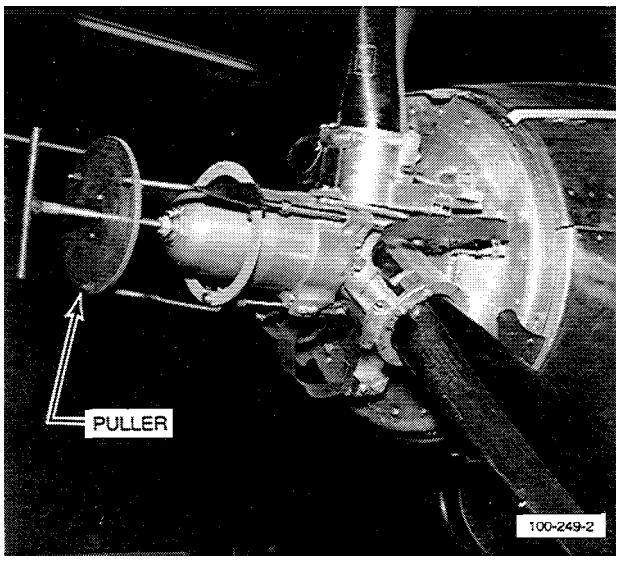


Figure 201
Feedback Ring Puller

(6) Secure the brush block and arm assembly to the mounting bracket for the proximity sensor with the attaching bolt. Connect the spring to the brush block arm.

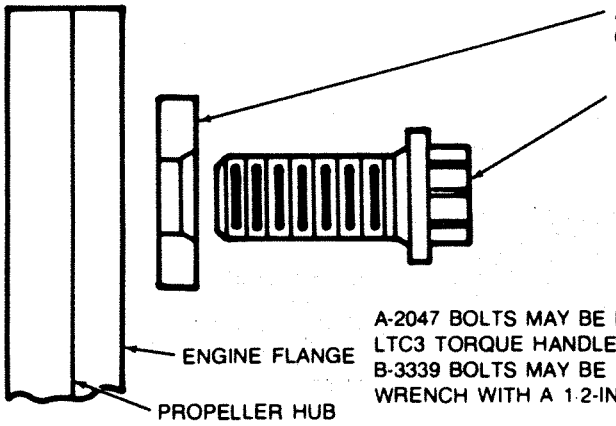
NOTE: To protect the low pitch stop collar (brass ring) against scoring, the clearance between the bottom of the collar and the rivet head of the brush block should be from 0.030 to 0.120 inch. Bend the arm assembly as necessary to obtain this clearance. Check for a clearance of 0.060 to 0.090 inch between the sensor and the target (end of the arm assembly opposite the brush block). Add AN960-10L washers as shown in Figure 201, Chapter 61-40-00 to obtain this clearance. On U-152 thru U-164, it is only necessary that a clearance be maintained between that portion of the arm assembly that passes into the slot of the electro-optic switch and both sides of the slot. (Ref. Figure 203, Chapter 61-40-00).

(7) Remove the feedback puller and connect the propeller reversing lever to the propeller control linkage. (Ref. Figure 202, Chapter 76-00-00 for the PT6A-20 and Figure 213, Chapter 76-00-00 for the PT6A-27 and PT6A- 28, and Figure 218, Chapter 76-00-00 for the PT6A-36).

CAUTION: *With the carbon brush block(s) held against the low pitch stop collar, check the clearance between the low pitch stop collar and the metal retaining clip of the carbon block. If at any point this clearance is 0.005 inch or less, replace the carbon block to prevent damage to the low pitch stop collar by the brush block retaining clip (Ref. Figure 203).*

WARNING

THE A-2047 BOLTS SHOULD NOT BE REUSED AFTER REMOVAL. THE B-3339 BOLT (WITH 5/8-INCH HEAD) SHOULD BE INSTALLED.



A-2048-2 WASHER (INSTALLED WITH CHAMFER TOWARD BOLT HEAD)

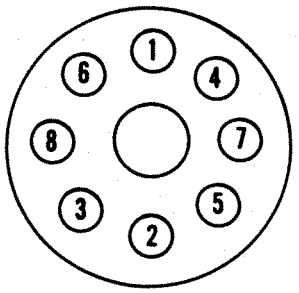
A-2047 BOLT WITH 9/16-INCH HEAD (MODELS U-1 THROUGH U-201 THAT HAVE NOT INCORPORATED KIT NO. 101-9027-1).

B-3339 BOLT WITH 5/8-INCH HEAD (MODELS U-202 AND AFTER, ALSO EARLIER SERIALS THAT HAVE KIT NO. 101-4027-1 INSTALLED).

NOTE

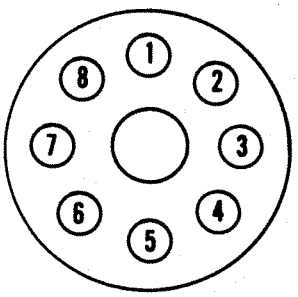
A-2047 BOLTS MAY BE REMOVED OR INSTALLED USING A 9/16-INCH SBH ADAPTER AND A LTC3 TORQUE HANDLE.
B-3339 BOLTS MAY BE INSTALLED OR REMOVED USING A 10113 ADAPTER AND A TORQUE WRENCH WITH A 1/2-INCH DRIVE.

INTALLATION PROCEDURES FOR THE A-2047 BOLTS (WITH 9/16-INCH HEADS); APPLY MIL-T-5544 PETROLATED GRAPHITE TO THE BOLT AND WASHER SURFACES. INSTALL THE A-2047 BOLTS AND A-2048-2 WASHERS (EIGHT EACH). TORQUE ALL BOLTS TO 100 + 25 0 FOOT POUNDS, SAFETY WIRE BOLTS PER FAA AIRCRAFT INSPECTION AND REPAIR MANUAL (AC 43.13-1).



A

USE THIS TORQUING SEQUENCE FOR THE INITIAL AND SECONDARY TOROUING PROCEDURE



B

USE THIS TOROUING SEQUENCE FOR THE FINAL TORQUING PROCEDURE

INSTALLATION PROCEDURES FOR THE B-3339 BOLTS (WITH 5/8-INCH HEADS): APPLY MIL-T-5544 PETROLATED GRAPHITE TO THE BOLT AND WASHER SURFACES. INSTALL THE B-3339 BOLTS AND A-2048-2 WASHERS (EIGHT EACH). TORQUE ALL BOLTS TO 40 FOOT-POUNDS, THEN TO 80 FOOT-POUNDS, USE TORQUE SEQUENCE A FOR THE INITIAL AND SECONDARY TORQUING PROCEDURE, FINAL TOROUE ALL BOLTS TO 100 + 5 0 FOOT-POUNDS, USE TORQUE SEQUENCE B FOR THE FINAL TORQUE SAFETY WIRE ALL BOLTS PER FAA AIRCRAFT INSPECTION AND REPAIR MANUAL (AC 43.13-1).

WHEN USING THE 10113 ADAPTER, THE LENGTH OF THE ADAPTER MUST BE ADDED TO THE LENGTH OF THE TOROUE WRENCH AND THE PROPER WET TORQUE VALUE COMPUTED AS DETAILED IN CHAPTER 20-00-00.

NOTE

ALL TOROUES IN THIS PROCEDURE ARE WET TORQUES.

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Figure 202
Propeller Bolt Torquing Procedures

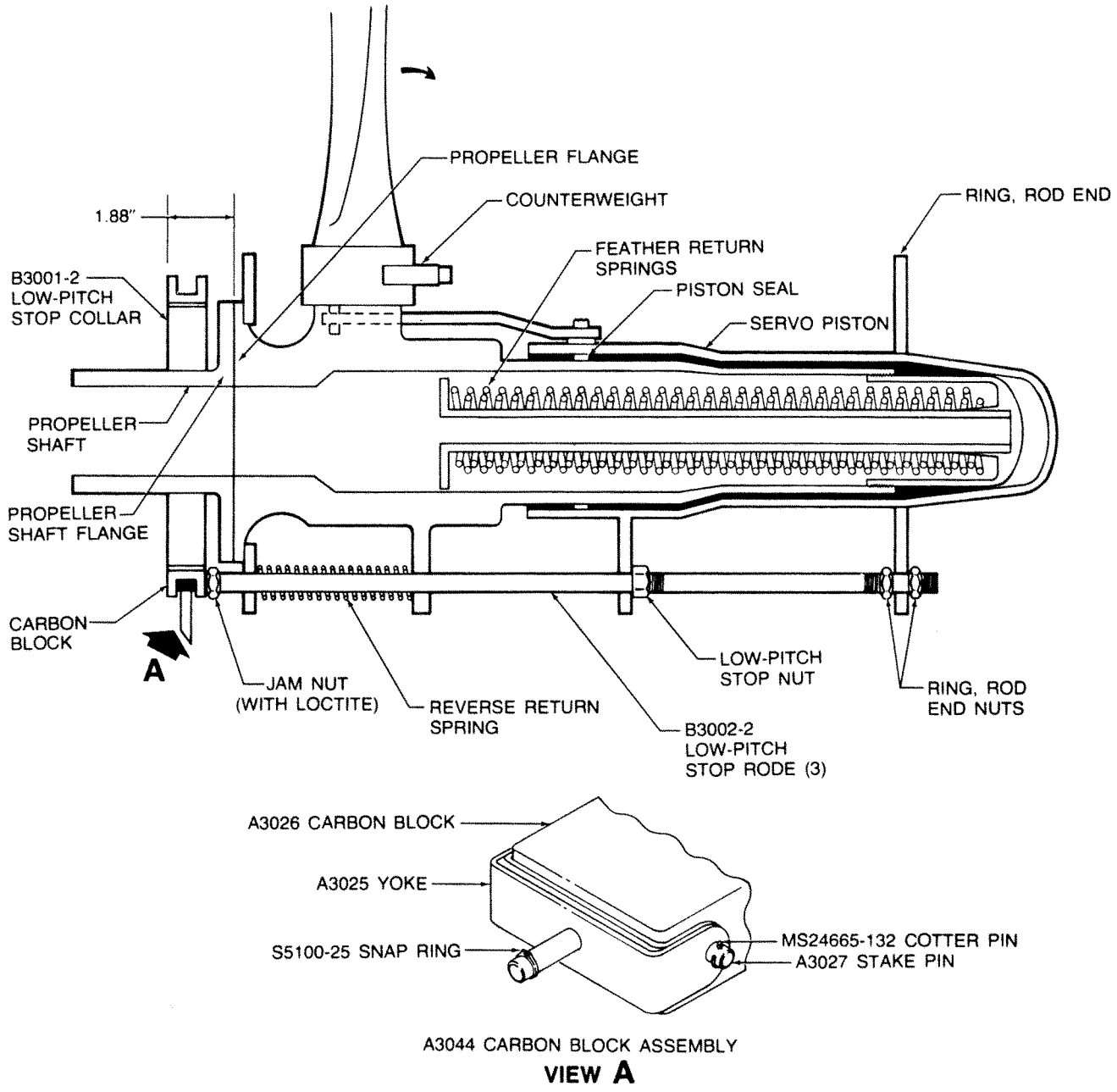


Figure 203
Propeller Assembly

(8) Install the deicer brush block assembly in accordance with the procedures in Chapter 30.

(9) Check the propeller reversing linkage on the front end of the engine for proper rigging.

(10) Install the spinner dome and engine cowling.

(11) Perform the necessary engine runup checks.

3. PROPELLER SHAFT OIL SEAL

A. Replacement

A propeller shaft oil seal of a non-split configuration is installed on all new or overhauled engines. Should it become necessary to replace the propeller shaft oil seal between overhauls, a split-type seal should be installed in place of the nonsplit seal.

The propeller shaft oil seal may be replaced as follows:

(1) Remove the upper and lower forward engine cowlings.

(2) Remove the propeller per PROPELLER REMOVAL procedure.

(3) Remove the bolts securing the seal retaining plate to the thrust bearing cover and slide the seal retaining plate forward against the propeller shaft flange.

(4) Remove the seal from the cavity in the thrust bearing cover.

(5) Remove the garter spring from the back of the seal. Cut and discard the spring.

(6) Remove the seal from the shaft.

(7) Coat the new seal with engine oil.

(8) Gently spreading the seal just enough to encompass the propeller shaft, install it on the propeller shaft with the "V" side of the seal facing aft.

CAUTION: *Avoid applying loads locally to the lip of the leaf springs, for the seal element is soft and subject to damage unless handled carefully at all times.*

(9) With the split ends of the seal placed together at the 9 o'clock position as viewed when looking forward at the propeller, press the seal gently into its cavity.

(10) Check to see that the seal is aligned and pinched evenly in place.

(11) Place the seal retaining plate in position and secure it to the thrust bearing cover with the bolts removed in Step (3). Torque the bolts 20 to 30 inch-pounds and secure them in place with safety wire.

(12) Install the propeller as noted in the PROPELLER INSTALLATION procedure.

(13) Install the upper and lower forward cowlings.

PROPELLERS
CONTROLLING
DESCRIPTION AND OPERATION

1. REVERSING PROPELLER CONTROLS (PT6A-20 ENGINE)

An increase in the flow of oil regulated by the primary governor passes through the oil transfer housing and the hollow center of the propeller shaft to move the propeller blades toward the low pitch (high rpm) hydraulic stop and reverse positions. The primary governor, which is mounted on top of the gear reduction housing, is a constant speed unit that regulates power turbine (N_2) speed by varying propeller blade pitch. The governor consists of a gear type oil pump with a pressure relief valve, a pair of pivoted fly-weights mounted on a rotating head, and a spring-loaded pilot valve that regulates the flow of oil to and from the propeller servo piston. The position of the pilot valve is controlled by the rotating flyweights, in conjunction with the spring load imposed by the externally mounted speed control lever through the control cable from the propeller control lever in the cockpit. When engine rpm drops to an underspeed condition below the control setting, speeder spring force overcomes flyweight force to lower the pilot valve plunger and open the port in the governor drive gear shaft. through which oil flows to the propeller servo piston and decreases blade angle. The decrease in pitch reduces the load on the engine. The resultant increase in engine rpm also increases the centrifugal force of the rotating flyweights, which then lift the pilot valve plunger to cover the port in the governor drive gear shaft and shut off the flow of oil to the propeller. The forces exerted on the pilot valve plunger by the flyweights and speeder spring then balance to initiate the on-speed cycle of the governor. An overspeed condition occurs with a decrease in propeller load or with movement of the propeller control to decrease rpm. Flyweight force then overcomes speeder spring force and raises the pilot valve plunger to open the port through which oil drains from the propeller through the governor to the sump. The load on the engine increases and rpm drops as the counterweights and feathering spring increase propeller pitch. The pilot valve then centers in the governor drivegear shaft to block the flow of oil to and from the propeller as governor flyweight and speeder spring force reach a state of equilibrium.

A propeller overspeed governor. mounted on the left side of the reduction gear housing, acts as a safeguard against propeller overspeed should the primary governor fail. The overspeed governor regulates the flow of oil to the propeller pitch-change mechanism by means of a flyweight and speeder spring arrangement similar to that of the primary governor. The overspeed unit governs at 104% N_2 speed (approximately 2288). Since it has no mechanical controls, the overspeed governor is equipped with a testing solenoid that resets the normal overspeed setting to approximately 2000 $\pm$ 50 rpm for ground testing.

Low pitch propeller position is determined by a mechanically monitored hydraulic stop, corresponding to the fixed low pitch mechanical stop used with non-reversing propellers. This is accomplished by controlling the position of the pilot valve plunger in the primary governor with an override rod linked to the propeller slip ring and control mechanism. The propeller servo piston is connected by three spring-loaded sliding rods to the slip ring mounted behind the propeller. Slip ring movement is transmitted by two carbon blocks through the propeller reversing lever and linkage to the primary governor pilot valve. As the propeller blades approach the low pitch angle selected with the propeller control and power levers, the movement of the propeller slip ring, through a yoke and the attendant linkage, lifts the override rod to center the pilot valve plunger and shut off the flow of oil to and from the propeller. This prevents the propeller from reversing by hydraulically locking the blades at the prescribed low pitch setting.

A secondary low pitch stop system prevents the propeller from reversing should the normal low pitch stop system malfunction. An arm assembly for a proximity sensor, mounted in a bracket on front of the engine. rides on the propeller slip ring. When the blade angle of the propeller is at approximately the 12° position, the slip ring moves the target area of the arm assembly through the magnetic lines of flux emitted by the sensor. This actuates a transistorized switch mounted on a bracket with the fire detector amplifiers and secondary stop relays under the center aisle floorboard aft of the main spar and temperature control panel. The switch closes to complete the circuit to ground through the secondary stop relay. The relay then actuates the

annunciator panel lights and the governor solenoid valve that shuts off oil pressure to the propeller to prevent the blades from reversing. The circuit is protected by a 5 ampere circuit breaker in the side panel beside the copilot that provides power to the other components of the secondary low pitch stop system through power switches in the pedestal that are actuated by the movement of the power levers. A test switch in the LH subpanel, adjacent to the control column, provides a means of bypassing the pedestal switches to provide power for an operational check of the secondary low pitch stop system.

When the cockpit controls are lifted to the reversing range, power levers actuate pedestal switches that break circuit to the secondary low pitch stop system. This enables servo piston to reverse pitch of propeller blades with the aid of engine oil pressure from the governor pump. The cockpit controls are connected to engine control cam, which is also linked to fuel control unit. A push-pull cable extends from the control cam forward to reversing lever to which are attached two brushes that ride in the propeller slip ring. Movement of cockpit controls is transmitted through the cam and interconnecting linkage to the fuel control unit and governors to regulate propeller speed and pitch. The reversing lever is connected to the power turbine governor to limit propeller rpm in the reverse position. The power turbine governor is progressively reset to below 100% N_2 speed as the cockpit power lever is retarded below the 90% position. The power turbine governor also senses the power turbine (or propeller) speed in the normal pitch range and is set to govern at 106% N_2 speed (approximately 2332 rpm). It indirectly controls propeller speed at this setting by limiting the flow of fuel through the fuel control unit. The power turbine governor is mounted on top of the sandwich-type propeller tachometer-generator, which is located on the right side of the reduction gear housing.

As a precaution against overtorquing engines on developing asymmetrical thrust, a DO NOT REVERSE light in the annunciator panel warns against reversing while the blades of either propeller are in the high pitch (low rpm) position. Power to the warning light switches is supplied through the landing gear control switch when in the DOWN position. The warning light switches are mounted in the pedestal adjacent to the propeller lever bellcranks. When either propeller lever is moved from the high rpm (low pitch) position, its respective warning switch closes to illuminate the DO NOT REVERSE light in the annunciator panel. When both propeller levers are in the high rpm (low pitch) position, the switches are open and the warning light is out.

2. REVERSING PROPELLER CONTROLS (PT6A-27, PT6A-28 AND PT6A-36 ENGINES)

An increase in the flow of oil regulated by the primary governor passes through the oil transfer housing and the hollow center of the propeller shaft to move the propeller blades toward the low pitch (high rpm) hydraulic stop and reverse positions. The primary governor, which is mounted on top of the gear reduction housing, is a constant speed unit that regulates power turbine (N_2) speed by varying propeller blade pitch. The governor consists of a gear type oil pump with a pressure relief valve, a pair of pivoted flyweights mounted on a rotating head and a spring-loaded pilot valve that regulates the flow of oil to and from the propeller servo piston. The position of the pilot valve is controlled by the rotating flyweights, in conjunction with the spring load imposed by the externally mounted speed control lever through the control cable from the propeller control lever in the pilots compartment. When engine rpm drops to an underspeed condition below the control setting, speeder spring force overcomes flyweight force to lower the pilot valve plunger and open the port in the governor drive gear shaft, through which oil flows to the propeller servo piston and decreases blade angle. The decrease in pitch reduces the load on the engine. The resultant increase in engine rpm also increases the centrifugal force of the rotating flyweights, which then lift the pilot valve plunger to cover the port in the governor drive gear shaft and shut off the flow of oil to the propeller. The forces exerted on the pilot valve plunger by the flyweights and speeder spring then balance to initiate the on-speed cycle of the governor. An overspeed condition occurs with a decrease in propeller load or with movement of the propeller control to decrease rpm. Flyweight force then overcomes speeder spring force and raises the pilot valve plunger to open the port through which oil drains from the propeller through the governor to the sump. The load on the engine increases and rpm drops as the counterweights and feathering spring increase propeller pitch. The pilot valve then centers in the governor drivegear shaft to block the flow of oil to and from the propeller as governor flyweight and speeder spring force reach a state of equilibrium.

A propeller overspeed governor, mounted on the left side of the reduction gear housing, acts as a safeguard against propeller overspeed should the primary governor fail. The overspeed governor regulates the flow of oil to the propeller pitch-change mechanism by means of a flyweight and speeder spring arrangement similar to that of the primary governor. The overspeed unit governs at 104% N_2 speed (approximately 2288 rpm). Since it has no mechanical controls, the overspeed governor is equipped with a testing solenoid that resets the normal overspeed setting to approximately 2000 $\pm$ 50 rpm for ground testing.

Low pitch propeller position is determined by a mechanically monitored hydraulic stop. corresponding to the fixed low pitch mechanical stop used with non-reversing propellers. The propeller servo piston is connected by three spring-loaded sliding rods to the slip ring mounted behind the propeller. A carbon block riding on the slip ring transfers the movement of the latter through the propeller reversing lever to the beta valve of the governor. The initial forward motion of the beta valve blocks off the flow of oil to the propeller. Further motion forward dumps the oil from the propeller into the reduction gearbox sump. A mechanical stop limits the forward motion of the beta valve. Rearward movement of the beta valve does not affect normal propeller control. When the propeller is rotating at a speed lower than that selected on the governor, the governor pump provides oil pressure to the servo piston and decreases the pitch of the propeller blades until the feedback of motion from the slip ring pulls the beta valve into a position blocking the supply of oil to the propeller, thus preventing further pitch changes.

3. SECONDARY LOW PITCH STOP SYSTEM

A. (Airplane Serials Prior to U-165, except U-50, Unless Modified by Installation of Kit 99-3026- 1S)

NOTE: The secondary low pitch stop systems have been removed from airplane equipped with PT6A-27 or PT6A-28 engines after modification by Kit 99-3026-1s in compliance with S.I. 0844-247.

A secondary low pitch system installed prior to U-165 prevents the propeller from reversing should the normal low pitch stop system malfunction.

B. Airplane Serials Prior to U-151, except U-50 (Unless Modified By Kit 99-3026-1S)

An arm assembly for a proximity sensor mounted on a bracket on the front of the engine, rides on the propeller slip ring. When the blade angle of the propeller is at approximately the 12° position, the slip ring moves the target area of the arm assembly through the magnetic lines of flux emitted by the sensor. This actuates a transistorized switch mounted on a bracket with the fire detector amplifiers and secondary stop relays under the center aisle floorboard aft of the main spar and temperature control panel. The switch closes to complete the circuit to ground through the secondary stop relay. The relay then actuates the annunciator panel lights and the governor solenoid valve that shuts off oil pressure to the propeller to prevent the blades from reversing. The circuit is protected by a 5 ampere circuit breaker in the side panel beside the copilot that provides power to the other components of the secondary low pitch stop system through power switches in the pedestal that are actuated by the movement of the power levers. A test switch in the LH subpanel. adjacent to the control column. provides a means of bypassing the pedestal switches to provide power for an operational check of the secondary low pitch stop system.

C. Airplane Serials Prior to U-152 thru U-164 (Unless Modified By Kit 99-3026-1S)

On U-152 through U-164, which have not been modified by Kit 99-30261S, an electro-optic switch is used in the secondary low pitch stop system. An arm assembly, mounted on a bracket on the front of the engine, rides on the propeller slip ring. When the blade angle of the propeller is in approximately the 12° position, the slip ring moves the outer area of the arm assembly into the slot of the horseshoe shaped electro-optic switch, blocking the light rays, emitted by the light emitting diode, from the light sensor. This transistorized switch actuates the annunciator panel lights and the governor solenoid valve that shuts off oil pressure to the propeller to prevent the blades from reversing. The circuit is protected by a 5 ampere circuit breaker in the side panel beside the copilot that provides power to the other components of the secondary low pitch stop system through power switches in the pedestal that are actuated by the movement of the power levers. A test switch in the LH subpanel, adjacent to the control column, provides a means of bypassing the pedestal switches to provide power for an operational check of the secondary low pitch stop system.

When the controls are lifted for placement in the reversing range, the power levers actuate the pedestal switches that break the circuit to the secondary low pitch stop system. This enables the propeller servo piston to reverse the pitch of the propeller blades with the engine oil pressure from the governor pump. The cockpit controls are connected to the engine control cam, which is also linked to the fuel control unit. A push-pull cable extends from the control cam forward to the reversing lever to which are attached the two brushes that ride in the propeller slip ring. Movement of the cockpit controls is transmitted through the cam and interconnecting linkage to the fuel control unit and governors to regulate propeller speed and pitch.

4. PRIMARY PROPELLER GOVERNOR

The primary propeller governor contains a fuel section designed to protect the engine against overspeeding in the event of a malfunction, such as sticking propeller blades. During reverse thrust operation, the fuel section is set below the speed selected by the governor to permit control of governor speed by the FCU (fuel control unit) servo system. As speed is increased, a pneumatic orifice is opened by a yoked bellcrank operated off the flyweight head to reduce the fuel metered to the engine. The speed of governor fuel section operation is determined by the speed selected by the governor and the position of the reset lever. which is normally set so that the fuel section fully regulates power turbine (N_2) speed at approximately 6% higher than the maximum speed setting of the governor and at 4% lower than the minimum speed setting. In its minimum speed position, the reset lever causes the yoke to contact the pilot valve. The resultant spring load, in addition to the speeder spring force, must be overcome by the flyweights in order for them to control propeller speed. This effect increases governed speed by about one percent over that nominally selected.

5. DO NOT REVERSE ANNUNCIATOR LIGHT

As a precaution against overtorquing the engines on developing asymmetrical thrust, a DO NOT REVERSE light in the annunciator panel warns against reversing while the blades of either propeller are in the high pitch (low rpm) position. Power to the warning light switches is supplied through the landing gear control switch when in the DOWN position. The warning light switches are mounted in the pedestal adjacent to the propeller lever bellcranks. When either propeller lever is moved from the high rpm (low pitch) position, its respective warning switch closes to illuminate the DO NOT REVERSE light in the annunciator panel. When both propeller levers are in the high rpm (low pitch) position, the switches are open and the warning light is out.

PROPELLERS
REVERSING PROPELLER CONTROLS
MAINTENANCE PRACTICES

1. PRIMARY GOVERNOR (PT6A-20)

A. Removal

(1) Remove the upper forward engine cowling.

(2) Remove the cotter pin, castellated nut, washer and bolt securing the governor control lever to the propeller speed control linkage.

(3) Remove the cotter pins, castellated nuts, washers and bolts securing the fork end of the two governor arms to the governor actuating lever.

(4) Remove the safety wire and disconnect the electrical plug from the solenoid on the governor.

(5) Remove the four nuts and washers anchoring the governor to the mounting pad on the reduction gearbox case.

(6) Remove governor and governor mounting pad gasket.

B. Installation

(1) Install a new gasket over the four studs on the governor mounting pad.

CAUTION: *Make sure the gasket is placed on the mounting pad with the raised side of the gasket screen up so that it will fit into the recess in the base of the governor.*

(2) Lightly coat the splined shaft of the governor with Plastilube No. 3.

(3) Position the governor on the mounting pad and secure it in place with the four attaching washers and nuts. Torque the attaching nuts 170 to 190 inch-pounds.

(4) Secure the fork end of each of the two governor arms to the actuating lever with the attaching bolt, washer, castellated nut and cotter pin.

CAUTION: *Do not exceed a torque of 25 inch-pounds when installing these attaching nuts and bolts.*

(5) Secure the governor control lever to the propeller speed control linkage with the attaching bolt, washer, castellated nut and cotter pin.

(6) Connect the electrical plug to the solenoid and safety wire.

(7) Check under the governor spool with the special feeler gage (P/N 50-590158-5) for a "Z" dimension of 0.084 to 0.002 inch as shown in Figure 208, Chapter 76-00-00. If necessary, make the adjustments described in Step (21) under PROPELLER REVERSING' MECHANISM RIGGING (PT6A-20) until the proper "Z" dimension is obtained.

(8) Install the upper forward engine cowling and perform the necessary engine run-up checks.

2. PRIMARY GOVERNOR (PT6A-27 AND PT6A-28)

A. Removal

NOTE: The power turbine governor is an integral part of the primary governor.

(1) Remove the upper forward cowling.

(2) Remove the cotter pin, castellated nut, washers and bolt securing the governor control lever to the propeller speed control.

(3) Remove the cotter pin, castellated nut, washer and bolt securing the fuel governor interconnecting rod to the fuel governor control lever.

(4) Remove the cotter pin, castellated nut, washer, bolt and spacer securing the front clevis end to the propeller reversing lever.

(5) Remove the cotter pin, washer, clevis pin and bushing securing the propeller reversing lever to the beta valve.

(6) Remove the safety wire and disconnect the electrical plug from the solenoid on the governor.

(7) Remove the safety wire and disconnect the sensor line from the governor union. Cap or plug open lines and ports.

(8) Remove the four nuts and washers anchoring the governor to the mounting pad on the reduction gear box case.

(9) Remove governor and governor mounting pad gasket.

B. Installation

(1) Install a new gasket over the four studs on the governor mounting pad.

CAUTION: *Make sure the gasket is placed on the mounting pad with the raised side of the screen up so that it will fit into the recess in the base of the governor.*

(2) Lightly coat the splined shaft of the governor with Plastilube No. 3.

(3) Position the governor on the mounting pad and secure it in place with the four attaching washers and nuts. Torque the nuts 170 to 190 inch-pounds.

(4) Secure the propeller reversing lever to the beta valve with the attaching bushing, clevis pin, washer and cotter pin.

(5) Secure the front clevis end to the propeller reversing lever with the attaching spacer, bolt, washers, castellated nut and cotter pin.

(6) Secure the fuel governor interconnecting rod to the fuel governor control lever with the attaching bolt, washer, castellated nut and cotter pin.

(7) Secure the governor control lever to the propeller speed control with the attaching bolt, washers, castellated nut and cotter pin.

(8) Connect the sensor line to the governor union and safety wire.

(9) Connect the electrical plug to the solenoid and safety wire.

(10) Install the upper forward engine cowling.

3. PRIMARY GOVERNOR (PT6A-36)

A. Removal

(1) Unplug the electrical lead from the synchrophaser.

(2) Disconnect the propeller reversing lever from the Beta control valve.

(3) Disconnect the propeller governor interconnect rod from the propeller governor airbleed link (reset arm).

(4) Disconnect the coupling nut of the pneumatic (Py) front tube from the straight nipple on the propeller governor.

(5) Remove the linkage from the speed select lever on the propeller governor.

(6) Using the governor retaining nut wrench (CPWA30114-091-16), remove the four self-locking nuts and plain washers securing the propeller governor to the reduction gearbox. Withdraw the propeller governor and gasket from the mounting pad.

(7) If the propeller governor is to be replaced by a new or overhauled unit, remove the straight nipple from the Py port on the governor. Remove and discard the preformed packing: retain the nipple for use on the replacement unit.

B. Installation

(1) If a new or overhauled propeller governor is to be installed, install the straight nipple as follows:

(a) Lightly lubricate a new preformed packing and install it on the nipple.

(b) Install the nipple in the Py port in the propeller governor. Torque the nipple 65 to 75 inch-pounds.

(2) Install a new gasket over the studs on the propeller governor mounting pad: ensure that the raised side of the gasket faces upward.

CAUTION: *Ensure that the drive splines are correctly engaged by checking that the flange of the governor rests squarely on the gasket with no gap. If necessary, rotate the propeller to assist engagement.*

(3) Install the propeller governor on the mounting pad with the four washers and self-locking nuts. Torque the nuts 170 to 190 inch-pounds with the retaining nut wrench (P/N CPWA30114-091-16).

(4) Connect the fork end of the Beta control valve to the propeller reversing lever with the clevis pin and secure with the washers and cotterpin.

(5) Connect the propeller governor interconnect rod to the propeller governor airbleed link.

(6) Connect the airframe linkage to the speed select lever on the propeller governor.

(7) Connect the coupling nut of the pneumatic (Py) front tube to the propeller governor. Torque the nuts 90 to 100 inch-pounds and lockwire.

(8) Connect the electrical leads to the lock pitch solenoid valve (if installed) on the propeller governor.

(9) Pressure test the pneumatic system as follows:

(a) Disconnect the compressor discharge air pressure line at the inlet of the P_3 air filter.

(b) Connect a supply of clean, dry compressed air, (regulated to 75 to 80 psi) to the inlet of the P_3 air filter.

(c) Apply leak checking fluid (107, Table 6, 91-00-00) to the entire surface of all connecting points of the pneumatic system. The only leaks permitted are from the two bleed holes in the fuel control unit and one bleed hole in the propeller governor.

(d) If leaks are detected, the affected parts must be inspected for defects before retorquing. Replace any defective parts. If no defects are found, retorque the connections as indicated in the engine maintenance manual and repeat the leak test.

(e) When no leaks are found, wipe the leak check fluid from the pneumatic system.

(f) Disconnect the air supply.

(g) Connect the compressor discharge air line to the inlet of the P_3 air filter. Torque the coupling nut 90 to 100 inch-pounds. Safety wire the coupling nut.

After installation of a new or overhauled propeller governor, carry out the adjustment and tests as detailed in Chapter 71-03-00 of the Pratt & Whitney Aircraft of Canada LTD Maintenance Manual (PIN 3021242).

4. LOW RPM

A. Setting

CAUTION: *To avoid overtorquing the engine during the following procedure, see that torque does not exceed 1000 foot-pounds and that propeller rpm does not drop below 1625.*

To adjust the low rpm settings, first position the propeller control in the full high rpm position and advance the power lever to obtain 2075 rpm. Pull the propeller lever back against the detent, then adjust the detent as required to obtain the specified setting (1625 to 1675 rpm). If the proper setting cannot be obtained by this means, the governor linkage may be lengthened or shortened as a further adjustment.

CAUTION: *Check that propeller rpm linkage is adjusted to permit propeller feathering with the synchronizer (if installed) turned on and run out to the extremity of its travel.*

5. HIGH RPM

A. Setting

If a high rpm adjustment is required, turn the high rpm stop screw on the governor head in or out as required to obtain 2200 rpm. Make sure the linkage to the control lever will allow sufficient travel for the control arm to contact the stop. If the linkage will not allow sufficient travel, adjust the control linkage at either rod end to obtain the necessary travel. This will in turn require resetting the low rpm and the feathering adjustment.

6. FEATHERING

A. Adjustment

Position the propeller lever in the full high rpm position and advance the power lever to obtain 1900 rpm. Pull the propeller lever back against the detent, then adjust the detent (Ref. Figure 201, Chapter 61-21-00) as required to obtain 1625 to 1675 rpm (at the detent). Pull the propeller lever past the detent (FEATHERING POSITION) and observe that the propeller is in the fully feathered position.

7. POWER TURBINE GOVERNOR (PT6A-20)

A. Removal

(1) Disconnect the sensor line from the governor union.

(2) Remove the bolt and nut securing the governor control arm to the bracket or linkage.

(3) Remove the four nuts and washers securing the governor to the mounting pad on the tachometer generator.

B. Installation

(1) Apply Plastilube No. 3 sparingly to the shaft of the governor.

(2) Position the governor on the mounting pad and secure it in place with the four attaching nuts and washers. Apply a torque of 170 to 190 inch-pounds to the mounting nuts.

(3) Connect the sensor line to the governor union and lockwire the retaining nut.

(4) Secure governor control arm to the bracket or linkage with the attaching bolt, washer, castellated nut and cotter pin.

8. PROPELLER OVERSPEED GOVERNOR

A. Removal

(1) Remove the upper forward engine cowling.

(2) Remove the safety wire and disconnect the electrical plugs from the governor solenoids.

(3) Tag and disconnect the two propeller synchronizer sensor wires.

(4) Remove the four self-locking nuts and plain washers securing the governor and remove the governor from the left side of the reduction gear housing.

B. Installation

(1) Install a new gasket on the mounting pad.

(2) Apply Plastilube No. 3 sparingly to the governor splined drive.

(3) Position the governor on the mounting pad and install the four plain washers and self-locking nuts. Apply a torque of 170 to 190 inch-pounds to the mounting nuts.

(4) Connect the electrical plugs to the governor. The solenoid with the receptacle on top is the speed reset solenoid used to test governor operation and the remaining solenoid is used for autofeathering. Safety wire plugs.

(5) Connect the two propeller synchronizer wires.

(6) Install the upper forward engine cowling.

C. Check

The overspeed unit, mounted on the left side of the reduction gear housing, is preset at the factory to govern at 104% N_2 speed (approximately 2288 rpm) and should not normally require readjusting. This setting can be reduced to 2000 $\pm$ 50 rpm for testing purposes by means of a testing solenoid. A second solenoid attached to the governor provides a means of immediately dumping oil from the unit for autofeathering. Perform the overspeed governor check as follows:

(1) Set the power levers to below 1900 rpm with the propeller controls in the full HIGH rpm position. Hold the PROP TEST switch in the OVERSPEED GOV position and advance the power levers to INCREASE. The tachometer will show from 1900 to 2100 rpm when the overspeed governor is operating properly. Overspeed units that fail to govern at the prescribed settings should be replaced and returned to a qualified service station for repair.

CAUTION: *Do not exceed engine torque limits during this check.*

PROPELLERS
AUTOFEATHERING SYSTEM
DESCRIPTION AND OPERATION

1. PROPELLER AUTOFEATHERING

The automatic feathering system provides a means of immediately dumping oil from the propeller governor to enable the feathering springs to start feathering the propeller blades as soon as engine torquemeter oil pressure drops below 6.5 psi (approximately 200 foot-pounds of torque) at power setting of 85 to 90% N_1 or greater. The system is actuated by placing the autofeather arm switch in the LEFT SUBPANEL to the ARM position. This closes the circuit breaker panel beside the copilot to the power switches mounted on a bracket in the control pedestal; however, the autofeathering system will remain inoperative as long as the power levers are retarded below the 85 to 90% N_1 position. When the power levers are advanced to the 85 to 90% N_1 position, a mechanical actuator connected to each power lever will close its respective switch and complete the circuit to the high and low pressure switches mounted adjacent to the torque pressure transmitter just forward and above the left hand exhaust outlet on each engine. These pressure switches monitor torquemeter oil pressure. The high pressure switch is a single pole, double-throw switch that actuates under pressure exceeding 12 psi (approximately 400 foot-pounds of torque). When actuated, the LH high pressure switch provides power from its "C" contact through the RH arming relay to the RH autofeather armed light in the LH instrument panel directly above the altimeter. The RH high pressure switch is similarly hooked to its armed light in the instrument panel. The arming relay for each system is mounted on the inboard aft side of the first bulkhead forward of the firewall in each nacelle. When the torquemeter oil pressure of an engine drops below 12 psi, the high pressure, switch on that engine opens the circuit to the arming light relay on the opposite engine to disable the autofeather system of the latter. The high pressure switch closes the circuit through its "B" contact to the coil of its respective arming relay, which will remain de-energized as long as torquemeter oil pressure exceeds 6.5 psi because of the open circuit at the "C" contact of the lower pressure switch. When torquemeter oil pressure drops below 6.5 psi, a ground return for the arming light relay and the dump valve solenoid mounted on the overspeed governor is provided through the "B" contact of the low pressure switch on the affected engine. Now energized, the arming light relay breaks the circuit to the autofeather armed light of the affected engine. Current through the relay then energizes the solenoid dump valve that immediately reduces governor oil pressure to zero. This allows the feathering springs to move the propeller blades to the feathered position on the engine in which torquemeter oil pressure dropped below 6.5 psi. At the same time, the autofeathering system of the other engine is disarmed through the hookup between the high pressure switches and arming light relays of the right and left hand systems. For testing purposes, the autofeather arm switch is equipped with a TEST position by which the power switches in the pedestal can be bypassed to checkout the system with the power levers retarded to below the 85 to90% N_1 position.

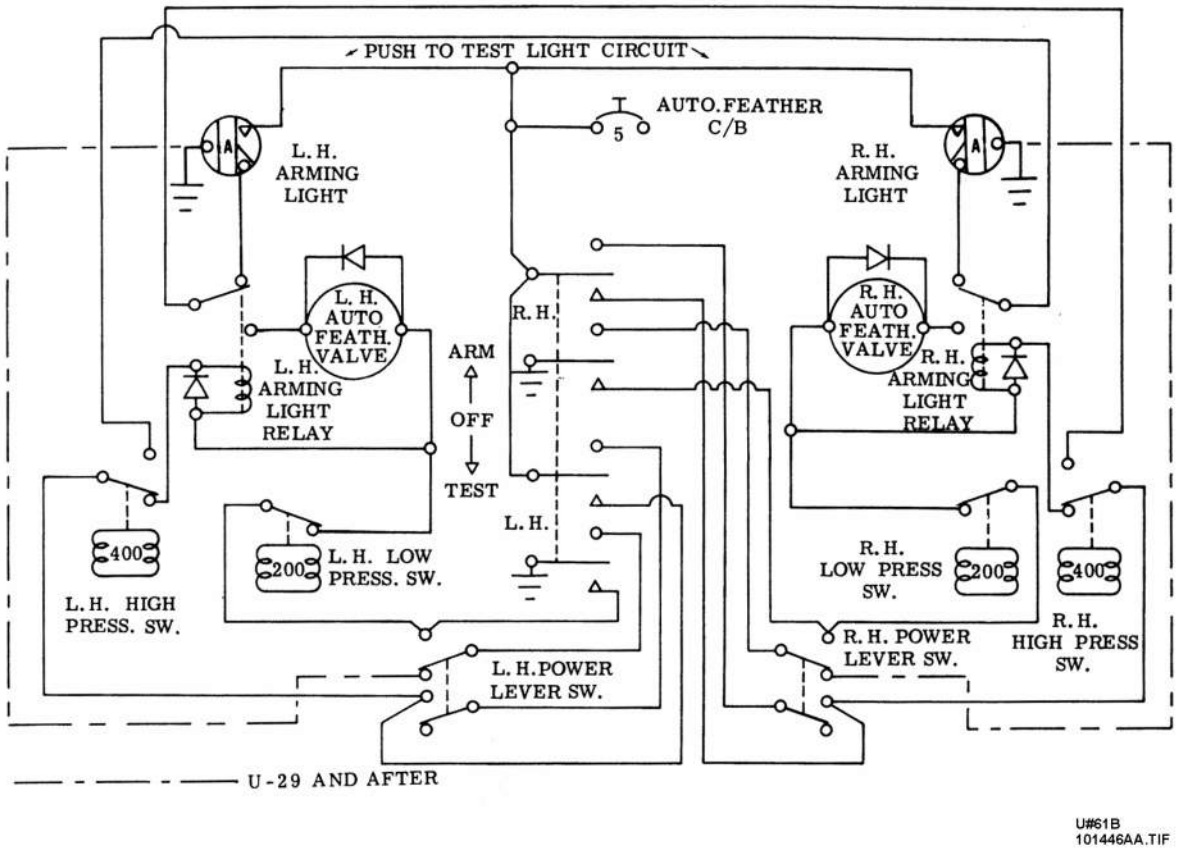


Figure 1
Auto Feather Electrical Schematic

PROPELLERS
AUTOFEATHERING SYSTEM
MAINTENANCE PRACTICES

1. AUTOFEATHERING ARMING RELAYS (U-1 THRU U-164, EXCEPT U-50)

A. Removal

An arming relay is mounted on the inboard aft side of the first bulkhead of the firewall in each nacelle.

(1) With the battery and generator switches OFF and the external power source disconnected, open the upper cowling.

(2) Remove the attaching hardware securing the relay to the bulkhead.

(3) Disconnect and tag the electrical wires and arc suppression diode from the relay. Clean the existing shrink tubing from the wire ends and diode.

B. Installation

(1) Slide new shrink tubing over the wire ends and arc suppression diode and connect wires to relay.

(2) Attach relay to bulkhead.

(3) Close upper cowling.

2. AUTOFEATHER ARMING RELAY PCB MODULE (U-50, U-165 AND AFTER)

A. Replacement

Replace the LH or RH autofeather arming relay PCB (printed circuit board) module located under the center aisle floorboard as follows:

(1) Disconnect all electrical power to the airplane.

(2) Remove the first center aisle floorboard panel forward of the main spar to expose the deice electrical equipment panel.

(3) Referring to Chapter 39-20-00, locate the PCB relay module assembly mounted on the deice panel.

(4) Refer to the Model C99 Airliner Wiring Diagram Manual and replace the appropriate autofeather arming relay PCB module.

(5) Install the center aisle floorboard panel removed in Step (2).

(6) Restore the electrical power to the airplane.

3. AUTOFEATHER DUMP VALVE

A. Removal

(1) With the battery and generator switches OFF and the external power source disconnected, remove the lower forward cowling.

NOTE: The dump valve is the lower of the two solenoid units mounted on the overspeed governor.

(2) Unsafety and disconnect the electrical connector of the dump valve.

(3) Unsafety and remove the screws securing the dump valve in place.

B. Installation

(1) Attach the dump valve to the over-speed governor.

(2) Connect the electrical connector to the dump valve and safety.

(3) Replace the lower forward cowling.

4. AUTOFEATHER POWER SWITCH

A. Adjustment

Access to the autofeather power switches is through the cover on the left side of the pedestal. The switches may be adjusted as follows (Ref. Figure 201):

With the autofeather control switch in the ARM position, adjust each power switch in its mounting slots until its respective AUTOFEATHER ARMED light on the LH instrument panel illuminates when the power levers are set to obtain an N_1 speed of 85 to 90%.

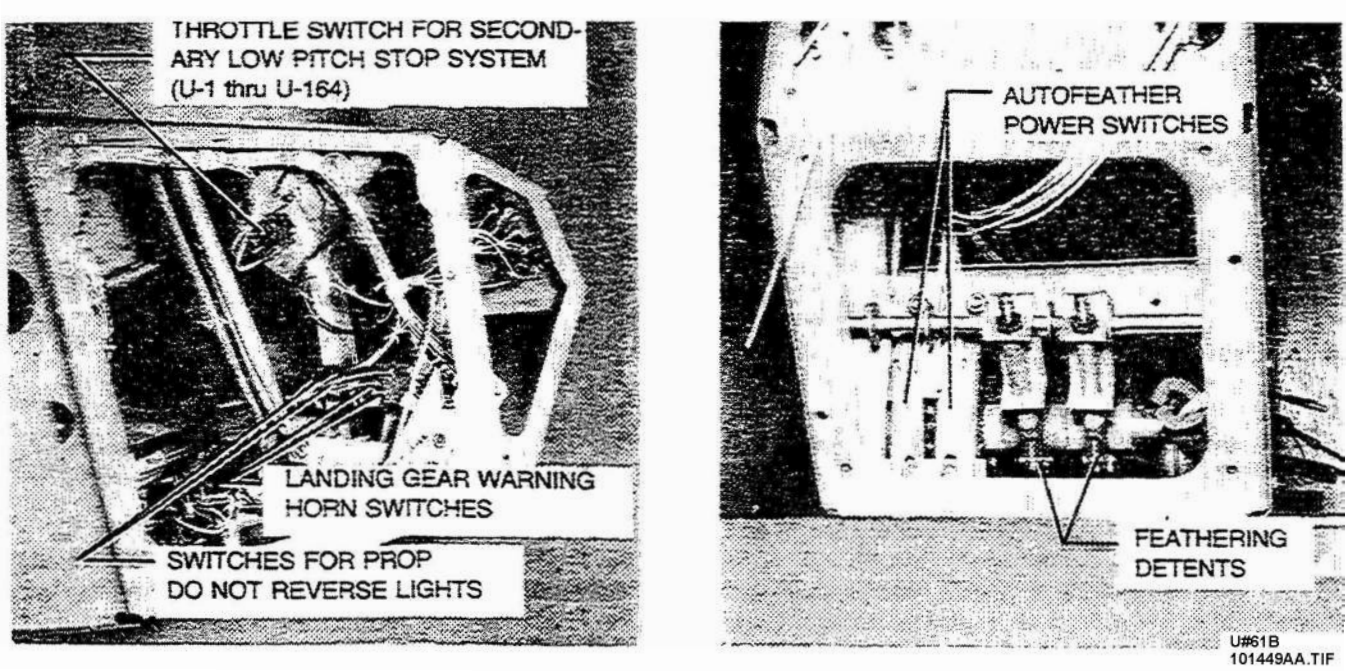


Figure 201
Pedestal Switches and Feathering Detent

5. AUTOFEATHER PRESSURE SWITCHES

A. Removal

NOTE: Removal procedures for the high pressure and low pressure switches are identical. The high pressure switch is mounted adjacent to the torque pressure transmitter just forward and above the left hand exhaust outlet on each engine. The low pressure switch is mounted just below and to the left of the high pressure switch.

(1) With the battery and generator switches OFF and the external power source disconnected, remove the upper forward cowling.

(2) Unsafety and remove the electrical connector from the switch.

(3) Unsafety the mounting nut at the base of the switch and remove the switch.

(4) Install a protective plug in the switch mounting hole to prevent contamination of the governor oil supply.

B. Installation

NOTE: Installation procedures for the high pressure and low pressure switches are identical. The high pressure switch is mounted adjacent to the torque pressure transmitter just forward and above the left hand exhaust outlet on each engine. The low pressure switch is mounted just below and to the left of the high pressure switch.

(1) Remove the protective plug from the switch mounting hole of the governor.

(2) Mount the switch to the governor.

(3) Safety the mounting nut.

(4) Install the electrical connector to the switch and safety.

(5) Install the upper forward cowling.

6. AUTOFEATHERING

A. Operational Check

(1) Move the autofeather arm switch to the TEST position with power levers set at idle. Check that propellers remain unfeathered and that AUTOFEATHER ARM lights in the LH instrument panel remain out.

(2) With the switch still in the TEST position and the engine controls set to obtain 500 foot-pounds of torque, both AUTOFEATHER ARM lights should illuminate.

(3) Slowly retard the LH engine power lever and check that the RH AUTOFEATHER ARM light extinguishes at 330 to 410 foot-pounds of torque. Continue retarding the LH engine power lever and check that both the LH and RH AUTOFEATHER ARM lights are extinguished and that the LH engine propeller starts to feather at 160 to 240 foot-pounds of torque.

(4) Repeat the preceding check with the RH engine.

(5) Move the autofeather arm switch to the ARM position and advance the LH engine power lever above the 85 to 90% N_1 position. Check that the propeller remains unfeathered and that the AUTOFEATHER ARM lights remain out on U-29 and After. On U-1 through U-28, the AUTOFEATHER ARM lights will illuminate.

(6) Retard the LH engine power lever, then repeat the proceeding check with the RH engine power lever advanced above the 85 to 90% N_1 position.

(7) Advance both power levers above the 85 to 90% N_1 position. Check that both the LH and RH AUTOFEATHER ARM lights are illuminated.

PROPELLERS
SYNCHRONIZER SYSTEM
DESCRIPTION AND OPERATION

1. PROPELLER SYNCHRONIZER

The propeller synchronizer automatically matches the right "slave" propeller rpm to that of the left "master" propeller. To prevent the right propeller from losing excessive rpm if the left propeller is feathered while the synchronizer is on, operation is limited to approximately ± 30 rpm from the manual governor setting. Normal governor operation is unchanged but the synchronizer will continuously monitor propeller rpm and reset the governor as required. A magnetic pickup mounted in each propeller overspeed governor transmits electric pulses to a transistorized control box installed behind the pedestal. The control box converts any pulse rate differences into correction commands, which are transmitted to a stepping type actuator motor mounted on the right engine cowl forward support ring. The motor then trims the right propeller governor through a flexible shaft and trimmer assembly to exactly match the left propeller rpm. The trimmer, installed between the governor control arm and the control cable, screws in or out to adjust the governor while leaving the control lever setting constant. A toggle switch installed on the left center floating panel below the synchroscope turns the system on. With the switch off, the actuator automatically runs to the center of its range of travel before stopping to assure that when next turned on, the control will function normally. To operate the system, synchronize the propellers in the normal manner and turn the synchronizer on. The right propeller rpm will automatically be adjusted to correspond with the left. To change rpm, adjust both propeller control at the same time. This will keep the right governor setting within the limiting range of the left propeller. If the synchronizer is on but is unable to adjust the right propeller rpm to match the left, the actuator has reached the end of its travel. Turn the synchronizer switch off (allowing the actuator to run to the center of its range and the right propeller to be governed by the propeller lever), synchronize the propellers manually and turn the synchronizer switch on (Ref. Figure 1).

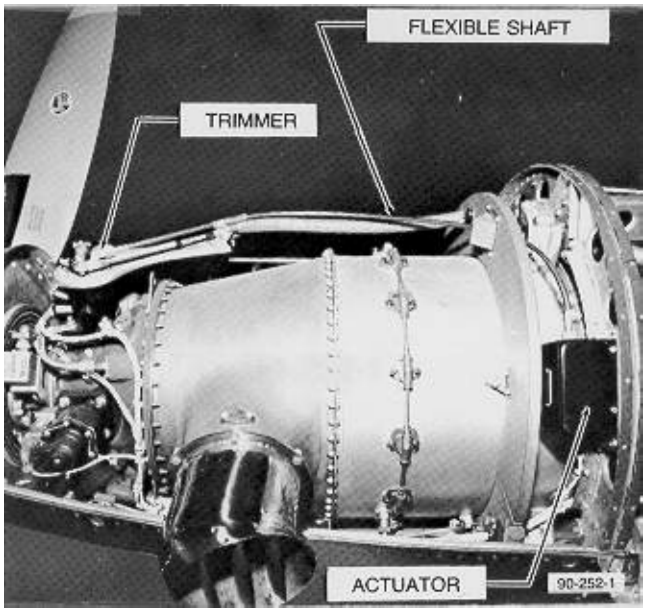


Figure 1
Propeller Synchronizer Actuator

PROPELLERS
PROPELLER SYNCHRONIZER
TROUBLESHOOTING

1. PROPELLER SYNCHRONIZER

Troubleshooting - Propeller Synchronizer

PROBLEM	POSSIBLE CAUSE	REMARKS
1. Synchronizer inoperative		
a. Circuit breaker trips	a. Short in aircraft wiring	a. Isolate and repair faulty component
b. Intermittent readings on pins 6 & 8 or 7 & 8 (circuit breaker may also trip)	b. Intermittent short or open magnetic pickup in governor	b. Repair or replace magnetic pickup
c. Slave (right) governor pickup gives open or short circuit reading on pins 6 & 8	c. Broken or grounded wire in magnetic pickup of slaved governor	c. Repair or replace magnetic pickup
d. Master (left) governor pickup gives open or short circuit reading on pins 7 & 8	d. Broken or grounded wire in magnetic pickup of master governor	d. Repair or replace magnetic pickup
2. Poor Synchronization		
a. Pickup voltage exceeds 3 volts at cruise rpm	a. Insufficient pickup to flyweight head clearance	a. Reset pickup to give specified voltage output
b. Pickup voltage is less than 0.5 volt at cruise rpm	b. Clearance between pickup and flyweight head too great	b. Reset pickup to give specified voltage output
c. Synchronizer pulses out of synchronization when turned on but returns to center when turned off	c. Leads No. 3 and No. 4 reversed or master and synchronizer leads reversed at Jones plug	c. Rewire correctly
d. Synchronizer pulses out of synchronization when turned on but returns to center when turned off and pins 6 & 8 or 7 & 8 have fluctuating readings	d. Intermittent open or short in pickup or wiring	d. Replace faulty pickup or wiring
e. Synchronizer action is sluggish	e. Excessive friction in trimmer or flexible rotary shaft	e. Clean, lubricate and check for misalignment of shaft in guide tube
f. Synchronizer action too limited in range	f. Excessive friction at one end of trimmer or actuator and trimmer were not centered when flexible rotary shaft was connected	f. Clean, lubricate. and check for misalignment, recenter actuator and rod end and reengage shaft
3. Actuator inoperative but magnetic pickups and control box function properly	a. Shorted or open actuator motor winding	a. Replace

Troubleshooting - Propeller Synchronizer

PROBLEM	POSSIBLE CAUSE	REMARKS
4. Actuator not recentering		
a. Actuator will not return to center after shutting synchronizer control off in flight	a. Defective centering mechanism in actuator or incorrect clearance between flyweight head and pickup	a. Replace actuator or reset magnetic pickup output as necessary
b. Actuator has improper travel (should make 3 revolutions)	b. Defective centering mechanism in actuator	b. Replace
c. Actuator dead band determined by ohmmeter (pins 5 and 1 open circuit) is 7 to 26 steps wide (actuator will probably not recenter when switch is turned off)	c. Actuator switches stuck in depressed position	c. Replace
5. Oil leaks from magnetic pickup connection in governor body	a. Defective packing between pickup and lock nut	a. Replace
	b. Defective gasket under lock nut	b. Replace
	c. Nut loose	c. Tighten to 25 inch-pounds
	d. Defective pickup	d. Replace pickup ***CAUTION*** *Never turn pickup into flyweight head. Check voltage output.*

PROPELLERS
PROPELLER SYNCHRONIZER SYSTEM
MAINTENANCE PRACTICES

1. PROPELLER SYNCHRONIZER

A. Functional Check

Proper operation of the propeller synchronizer can be determined by the following method. Turn the synchronizer on in the normal manner, then decrease the left propeller rpm in small increments. Both propellers should decrease together until the actuator reaches the end of its travel, then the right propeller should stabilize its rpm while the left continues to decrease. Increase the left propeller rpm until both propellers commence to increase rpm together. Turn the system off and an unsynchronized condition will develop as the actuator runs to its midrange position. When the synchronizer is turned on, the right propeller should again synchronize with the left.

2. SYNCHRONIZER

A. Rigging

(1) Disconnect the flexible trimmer shaft from the actuator.

(2) Rotate the shaft in one direction until it reaches the internal stop in the trimmer. Check the propeller control to assure that it has full travel in both directions.

(3) Rotate the shaft in the other direction until it reaches the stop, counting the turns. Again check the propeller control for full travel in both directions. This ensures that the aircraft is rigged to permit the governor adjusting lever to travel from its minimum to its maximum rpm stops at any possible trimmer setting.

(4) Rotate the shaft to the center of its range.

(5) Count total turns available in the actuator motor and turn it to the center of its range. The motor can be turned by inserting a square shaft end into the drive.

(6) With both the trimmer and the actuator motor centered, connect the flexible shaft.

(7) Rig the propeller controls in the normal manner.

3. CHECKS

These checks will help locate the source of trouble should the synchronizer system malfunction. If no malfunctions are found among the units being tested, the transistorized control box is probably the source of trouble. An ohmmeter and voltmeter are required to conduct the following ground and flight checks.

A. Ground Check

(1) To eliminate the most obvious causes for malfunction, make sure that the aircraft master switch is ON, that the system circuit breaker is not tripped, and that the Jones plug receptacle is properly mated with the plug in the aircraft electrical system.

(2) Unplug the control box, turn the aircraft master switch OFF, and pull the synchronizer circuit breaker before proceeding further with these checks.

CAUTION: *Do not plug in the control box until this test has been satisfactorily completed to prevent damage to the unit.*

(3) Connect the actuator assembly and the magnetic pickups to the cables. Either pickup wire may be connected to pin receptacle number "8" of the Jones cinch socket. Connect the other wire to pin receptacle number "6" for the right (slave) engine and to pin receptacle number "7" for the left (master) engine. Before tying down the pickup wires, complete the following checks:

CAUTION: *Zero the ohmmeter and read on the X1 or X10 scale during the following checks. Do not use a probe greater than 0.045 inch in thickness. Insert and remove the probe carefully to avoid damaging the pin connectors.*

(a) If the synchronizer pickup assembly has an entirely black sheathing lead, no part number stamped on the body and a ceramic core, check as follows:

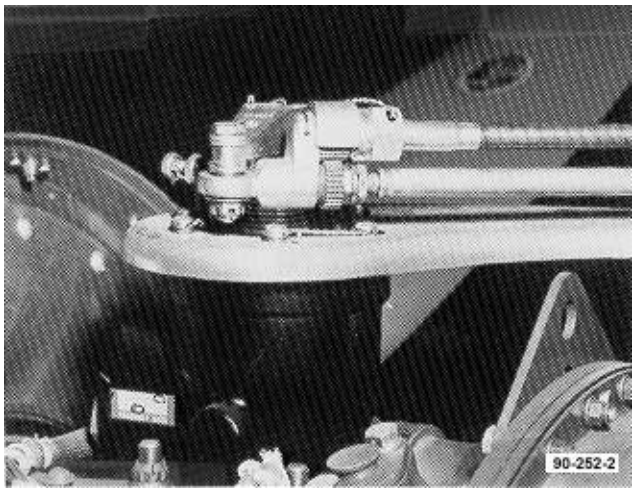


Figure 201
Propeller Synchronizer Trimmer

Table 201
Test For Defective Synchronizer Actuator (With Jones Cinch Socket)

OBTAIN

	With Actuator Centered	*With Actuator Uncentered 180°*	
TEST BETWEEN RECEPTACLES OF JONES CINCH SOCKET		*Turn Clockwise (Facing Drive End) To Uncenter*	*Counterclockwise (Facing Drive End) To Uncenter*
5 and 1	open circuit (high resistance)	6.5 to 8.5 ohms	6.5 to 8.5 ohms
5 and 3	open circuit (high resistance)	closed circuit 0 to 1ohm	13 to 17 ohms
5 and 4	open circuit (high resistance)	13 to 17 ohms	closed circuit 0 to 1.0 ohms
4 and 1	6.5 to 8.5 ohms	6.5 to 8.5 ohms	6.5 to 8.5 ohms
4 and 3	13 to 17 ohms	13 to 17 ohms	13 to 17 ohms
3 and 1	6.5 to 8.5 ohms	6.5 to 8.5 ohms	6.5 to 8.5 ohms

Table 202
Bench Test of Synchronizer Actuator (Without Jones Cinch Socket)

OBTAIN

	With Actuator Centered	*With Actuator Uncentered 180°*	
TEST BETWEEN RECEPTACLES OF JONES CINCH SOCKET		*Turn Clockwise (Facing Drive End) To Uncenter*	*Counterclockwise (Facing Drive End) To Uncenter*
D & C	Open Circuit (high resistance)	6.5 to 8.5 ohms	6.5 to 8.5 ohms
D & B	Open Circuit (high resistance)	closed circuit (0 ohms)	13 to 17 ohms
D & A	Open Circuit (high resistance)	13 to 17 ohms	Closed Circuit (0 ohms)
A & C	6.5 to 8.5 ohms	6.5 to 8.5 ohms	6.5 to 8.5 ohms
A & B	13 to 17 ohms	13 to 17 ohms	13 to 17 ohms
B & C	6.5 to 8.5 ohms	6.5 to 8.5 ohms	6.5 to 8.5 ohms

Table 203
AIRCRAFT WIRING CHECKS

TO CHECK AIRCRAFT WIRING WITH ACTUATOR REMOVED

Jones Pin No. 1 to aircraft ground	0 Ohms
Jones Pin No. 3 to aircraft ground	Open Circuit (Very high resistance)
Jones Pin No. 4 to aircraft ground	Open Circuit (Very high resistance)
Jones Pin No. 5 to aircraft ground	Open Circuit (Very high resistance)

1 Using an ohmmeter, test that the resistance between pin receptacle "7" and "8" of the Jones cinch socket is 90 to 110 ohms with pin receptacle "6" disconnected at the pickup.

2 Check that the resistance between pin receptacle "6" and "8" is 90 to 110 ohms with pin receptacle "7" disconnected at the pickup.

(b) If the synchronizer pickup assembly has a section of white sheathing, no part number stamped on the body and a solid steel closed pickup, check as follows:

1 Using an ohmmeter, test that the resistance between pin receptacle "7" and "8" of the Jones cinch socket is 52 to 68 ohms with pin receptacle "6" disconnected at the pickup.

2 (b) Check that the resistance between pin receptacle "6" and "8" is 52 to 68 ohms with pin receptacle "7" disconnected at the pickup.

(4) Make the following checks with an ohmmeter connected to the pin receptacles of the Jones cinch socket (P/N 233032).

(5) Check that an open circuit (very high-resistance) exists between pin receptacle "8" and aircraft ground and between pin receptacle "2" and ground.

(6) Check that a closed circuit (zero ohms) exists between pin receptacle "1" and ground.

(7) Turn the aircraft master switch ON and reset the synchronizer circuit breaker, but leave the control box unplugged.

(8) Using a DC voltmeter, check that the voltage between pin receptacles "1" and "2" is the same as the supply voltage and that the polarity of pin number "1" is negative while that of pin number "2" is positive.

(9) Using a 5000 ohm/volt AC voltmeter and with the engines running near cruise rpm. probe pins "6" and “8" for pickup voltage on the right (slave) engines and pins "7" and "8" for pickup voltage on the left (master) engine. These values should be between 1/2 volt minimum and 3 volts maximum. If these values cannot be obtained, replace the overspeed governor or on PT6A-27 and PT6A- 28 engines, the magnetic pickup can be replaced or adjusted by an FAA approved overhaul station.

NOTE: To check magnetic pickup resistance at the governor, disconnect both connectors and measure the resistance with an ohmmeter set at the X1 or X10 scale. Should a resistance other than 90 to 110 ohms be indicated, magnetic pickups with a wholly black sheathing must be replaced. Magnetic pickups with a section of white sheathing must be replaced if a resistance other than 52 to 68 ohms is indicated.

(10) When the system is in compliance with the preceding check values, plug the control box into the synchronizer system. If the synchronizer system still malfunctions, the source of trouble is probably either in the control box itself or is the result of a mechanical failure of the flexible shaft or governor speed trimming device.

(11) Check the speed pickups mounted on the governor for oil leaks or looseness.

(12) Remove the flexible rotary shaft and rotate the actuator through its range. It should stop positively at each end of its range and should rotate freely except for the ratcheting effect of the detent wheel. Do not exceed a torque of 24 ounce-inches against the stops. Make sure the actuator is returned to the center of its range.

(13) Adjust the governor trimmer by turning the squared end of the flex shaft by hand (a turning fixture may be required to turn the shaft in the direction to decrease rpm). Under any circumstances, a turning fixture 1/4 inch in diameter should be adequate for rotating the trimmer freely throughout its range. After thus verifying that the friction between the rotating parts is at an acceptable level, recenter the trimmer and attach it to the actuator.

(14) With the actuator removed, zero the ohmmeter and use the X1 scale when running the preceding checks on the actuator.

(15) Replace actuators that do not meet the preceding values. Should the actuator meet these values, check the four leads of the actuator for continuity.

CAUTION: *You must replace both the control box and the actuator if the actuator is being replaced because it will not meet the ohm values specified. The replacement actuator could be damaged if these parts are not changed in pairs.*

If the response of the system to the preceding checks has been satisfactory, the aircraft is ready for flight.

B. Flight Checks

(1) Check the effect of rpm and/or power setting (particularly in the lower cruise range) on synchronizer action. If operation at lower rpm resulted in improved synchronization, inspect the drives to the governors.

(2) Reduce the electrical load and turn off the generator and all other electrical units, except the master switch, and synchronizer. If synchronizing improves, abnormal voltage spikes on the aircraft bus from some other electrical accessory may have been upsetting the synchronizer. Isolate the offending accessory and repair it. If the trouble lies in the control box, replace it.

PROPELLERS
SYNCHROPHASER
DESCRIPTION AND OPERATION

1. PROPELLER SYNCHROPHASER

The propeller synchrophaser automatically matches the rpm of both propellers, and maintains a pre-set phase angle relationship between the left and right engine. Input signal pulses occurring once per revolution of the propeller are obtained from magnetic pickups (located on brackets bolted to the front of the engine) when the target (mounted on the aft side of the spinner bulkhead) passes the magnetic pickup. The control box converts this signal pulse rate difference into correction commands, which are transmitted to the appropriate governor. Speed trim is accomplished by pulse width modulation of the electro-magnetic coil in each governor. The electro-magnetic coil can increase the speed set by the propeller control lever, but never decrease the speed set by the control lever. The rpm of one engine will follow the changes in rpm of the other engine over a predetermined range (approximately 20 rpm). This limited range prevents either engine from losing more than a limited rpm if the other engine is feathered with the synchrophaser on.

A toggle switch (placarded PROP SYNCH - ON - OFF) installed near the synchroscope turns the system on. To operate the system, synchronize the propellers in the normal manner and turn the synchrophaser on. To change rpm, adjust both propellers at the same time. This will keep the setting within the limited range of the system. If the synchrophaser is on, but unable to match the propeller rpm, the system has reached its range limit. Turn the synchrophaser off, synchronize the propellers manually, then turn the synchrophaser on.

PROPELLERS
PROPELLER SYNCHROPHASER
TROUBLESHOOTING

1. PROPELLER SYNCHROPHASER

A. Type II Propeller Synchrophaser (U-50, U-165 and After)

The first indication that a fault may exist in the synchrophaser system will most likely be squawks from the flight crew; however, performing a functional check of the operation of the synchrophaser system before troubleshooting may yield information essential to differentiating faults within the system. The functional check which follows should be performed as the first phase of troubleshooting.

The second phase of troubleshooting will include performing continuity checks of the synchrophaser system wiring and peripheral components. A breakout box (Ref. Figure 105) can be fabricated in the shop and will provide convenient access to the various circuits of the synchrophaser system. The breakout box can also be used to check the operation of the propeller governors. Should continuity checks fail to reveal the nature of the fault, it will be necessary to begin the third phase of troubleshooting which will be to evaluate the operation of the propeller governors. For Jones plug type control boxes only, Woodward test instrument P/N 213618 may be purchased in lieu of fabricating the test box shown in Figure 105.

The fourth phase of troubleshooting, if no faults became apparent during previous troubleshooting attempts, is to evaluate the character of the outputs from the magnetic pickups.

In the event none of the previous checks or procedures reveals the exact nature of the fault, the fault may be assumed to be in the control box, and a control box, known to be operational, should be installed.

Troubleshooting Table 104 proposes a logical sequence of checks and activities to be performed during the troubleshooting process and may aid the technician in eliminating confusion and maintaining logical order. All manners of faults have been investigated, but only the most probable causes of faults in the system are detailed in this Table.

B. Functional Check

The proper operation of the synchrophaser system may be ground checked as follows:

(1) Start the engines and increase power until the speed on both propellers is at 2000 rpm with the propeller levers in high rpm.

(2) Using the propeller levers, reduce propeller rpm on both engines to as nearly 1900 rpm as possible.

CAUTION: *Do not exceed torque red line.*

(3) Turn the synchrophaser switch to the ON position.

(4) Using the propeller lever, increase the rpm of the left propeller slightly and check that the synchrophaser increased the rpm on the right propeller to match.

(5) Should the previous check yield negative results, turn the synchrophaser off and repeat the above procedure.

(6) If the previous check indicated that the synchrophaser is operating, continue increasing the propeller rpm with the propeller lever until synchronization is lost and record this rpm.

(7) Using the propeller lever, slowly decrease the propeller rpm on the left engine below the rpm of the right engine until synchronization is again lost and record this rpm.

(8) The difference between the two propeller speeds at which synchronization was lost is the holding range and should be at least 31 to 37 rpm with the 8210-025 governor or 36 to 42 rpm with the 8210-030 governor. This check evaluates the ability of the right propeller governor to respond to correction commands from the control box, as well as the control box's ability to issue commands properly.

NOTE: Failure of this check will necessitate performing the synchrophaser CONTINUITY CHECK.

(9) Repeat the functional check procedure by varying the propeller rpm of the right engine while letting the left propeller governor respond.

(10) During this functional check, determine that the communications and navigation equipment is not adversely affected by the synchrophaser operation and that synchrophaser operation is not adversely affected by the operation of other electrical equipment.

C. Continuity Check

This check will help locate the source of the fault should the synchrophaser system fail the functional check. A sensitive volt/ohmmeter will be required to perform these continuity checks.

(1) Check that the battery master switch is off, and open the synchrophaser circuit breaker.

NOTE: If another accessory uses the same circuit breaker, it must be disconnected.

(2) Remove the connector from the control box.

(3) Check the synchrophaser system wiring for conformity to the schematic in Figure 101, 102 or 103 with respect to connector plug configuration.

(4) Perform the continuity checks as indicated in Table 101 for Jones plug control boxes or Tables 102 or 103 for round control boxes.

NOTE: Resistance readings noted in Tables 101, 102 or 103 may be as much as 20% higher during heat soak following engine shutdown.

(5) After performing the continuity checks, close the synchrophaser circuit breaker, turn the battery master switch on and check for battery voltage between pins "1" and "2" of the Jones plug or between pins "A" and "B" on airplanes with the round plug installed.

(6) Correct any wiring faults that may have been revealed during these continuity checks. Replace any component which failed to meet the proper resistance specified for that component. Restore the system to operational status and perform the functional check again.

CAUTION: *DO NOT connect the plug to the control box before turning the battery master switch off and opening the synchrophaser circuit breaker.*

(7) If no faults were found by the continuity checks, perform the propeller GOVERNOR OPERATIONAL CHECK.

2. GOVERNOR

A. Operational Check

The following checks assess the operation of the governors and should be performed before evaluating the outputs of the magnetic pickups:

(1) Connect the breakout box between the airplane harness and the airplane control box.

(2) Start the engines and increase the propeller rpm to 2000 rpm with the propeller levers in high rpm.

(3) Using the propeller levers, reduce the rpm to 1900 rpm to ensure that the propellers are being controlled by the governors.

CAUTION: *Do not exceed torque red line.*

(4) Install a jumper between jacks "4" and "1" of the breakout box on airplanes with Jones plug control boxes or between ins "R" and "A" (Ref. Figure 102) or "K" and 'A" (Ref. Figure 103) on airplanes with round plug control boxes. The speed of the right propeller should noticeably increase.

(5) Install the jumper between jacks "6" and "1" of the breakout box on airplanes with Jones plug control boxes or between pins "K" and "A" (Ref. Figure 102) or "J" and "A" (Ref. Figure 103) on airplanes with round plug control boxes. The speed of the left propeller should noticeably increase.

(6) A failure of either propeller governor to increase propeller speed indicates that there is a fault in that governor.

(7) If no faults were revealed by this operational check, evaluate the character of the outputs from the pickups before condemning the control box.

Table 101
SYNCHROPHASER CONTINUITY CHECKS

Jones Plug Control Box

Test Between Receptacle Pins	Ohmmeter Indication*	Component
1 and ground	Continuity	
2 and ground	Infinity	
3 and ground	52 to 68 ohms	Right Pickup
4 and ground	Infinity	
5 and ground	52 to 68 ohms	Left Pickup
6 and ground	Infinity	
8 and ground	Infinity	
4 and 8	115 to 205 ohms	Right Governor
6 and 8	115 to 205 ohms	Left Governor

* Resistances may be as much as 20% higher during heat soak after engine shutdown.

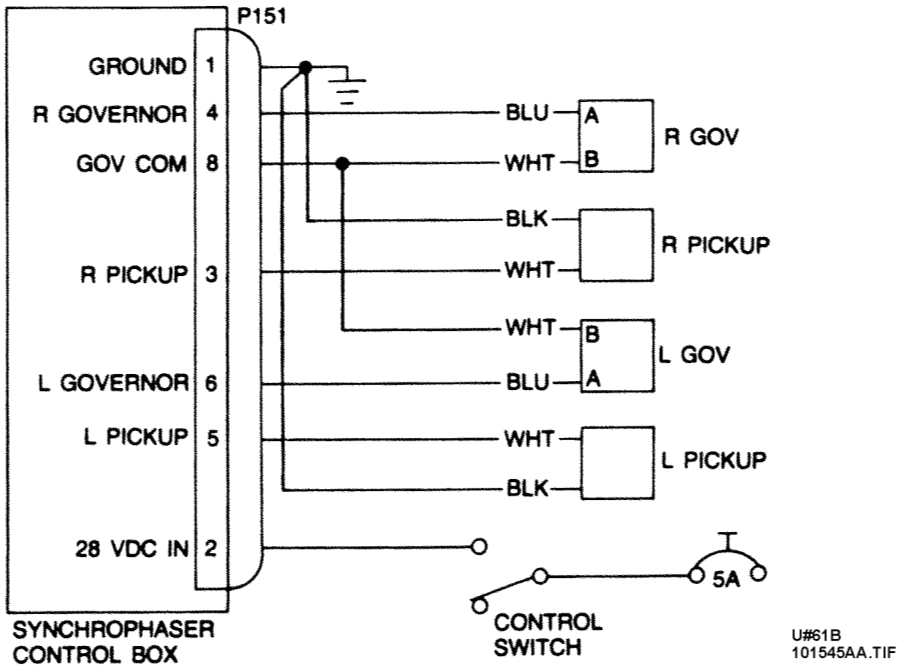


Figure 101
Jones Plug Control Box Wiring Schematic

Table 102
SYNCHROPHASER CONTINUITY CHECKS

Round Plug Control Box
(Airplanes prior to U-225 that have not accomplished Service Bulletin No. 2239)

Test Between Receptacle Pins	Ohmmeter Indication*	Component
A and ground	Continuity	
A and B	Infinity	
C and G	52 to 68 ohms	Right Pickup
D and E	52 to 68 ohms	Left Pickup
J and R	115 to 205 ohms	Right Governor
K and L	115 to 205 ohms	Left Governor

* Resistances may be as much as 20% higher during heat soak after engine shutdown.

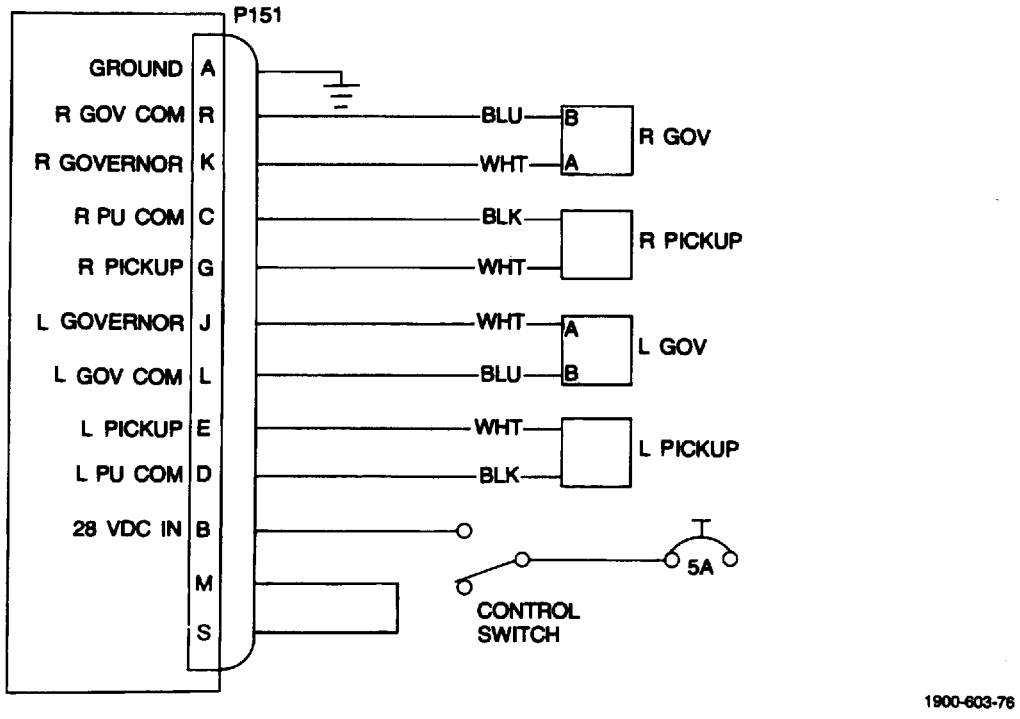


Figure 102
Round Plug Control Box Wiring Schematic
(Airplanes prior to U-225 that have not accomplished Service Bulletin No. 2239)

Table 103
SYNCHROPHASER CONTINUITY CHECKS

Round Plug Control Box
(U-225 and After and prior serials that have accomplished Service Bulletin No. 2239)

Test Between Receptacle Pins	Ohmmeter Indication*	Component
A and ground	Continuity	
A and B	Infinity	
C and G	52 to 68 ohms	Right Pickup
D and E	52 to 68 ohms	Left Pickup
J and R	115 to 205 ohms	Right Governor
K and L	115 to 205 ohms	Left Governor

* Resistances may be as much as 20% higher during heat soak after engine shutdown.

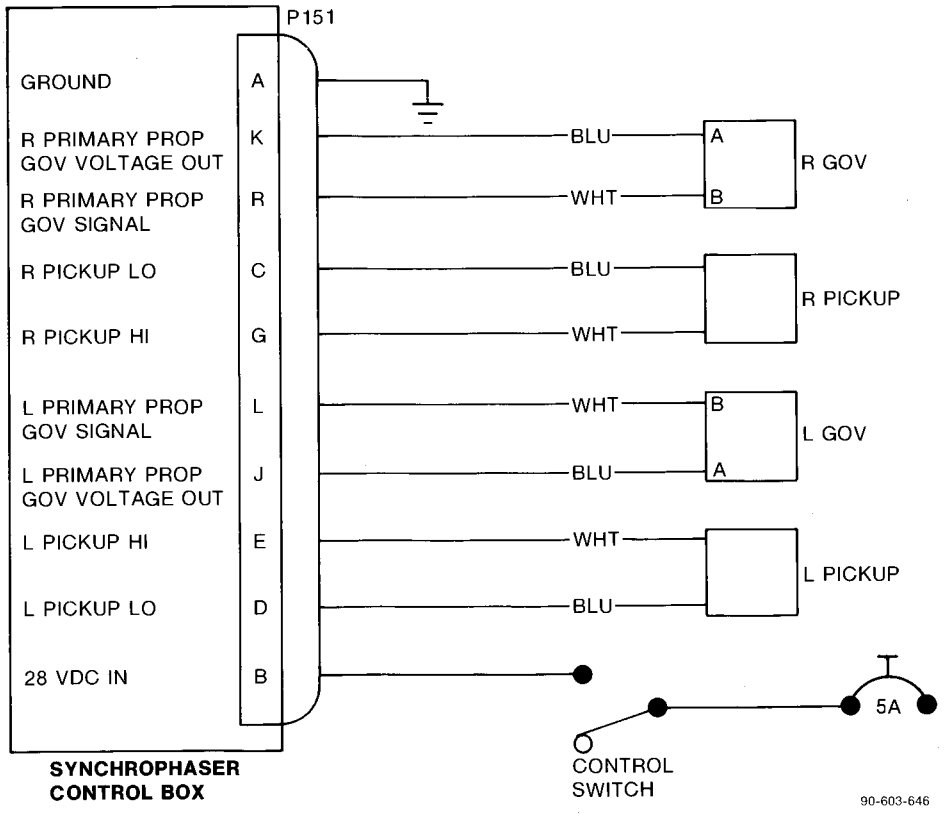


Figure 103
Round Plug Control Box Wiring Schematic
(U-225 and After and prior serials that have accomplished Service Bulletin No. 2239)

B. Pickup Output Check

This procedure makes it possible to evaluate the character of the pickup outputs and to make adjustments as required. An oscilloscope is required to display the output waveform and measure the peak-to- peak voltages.

(1) Be sure the battery master switch is turned to OFF and the synchrophaser circuit breaker is open.

(2) Disconnect the plug from the control box.

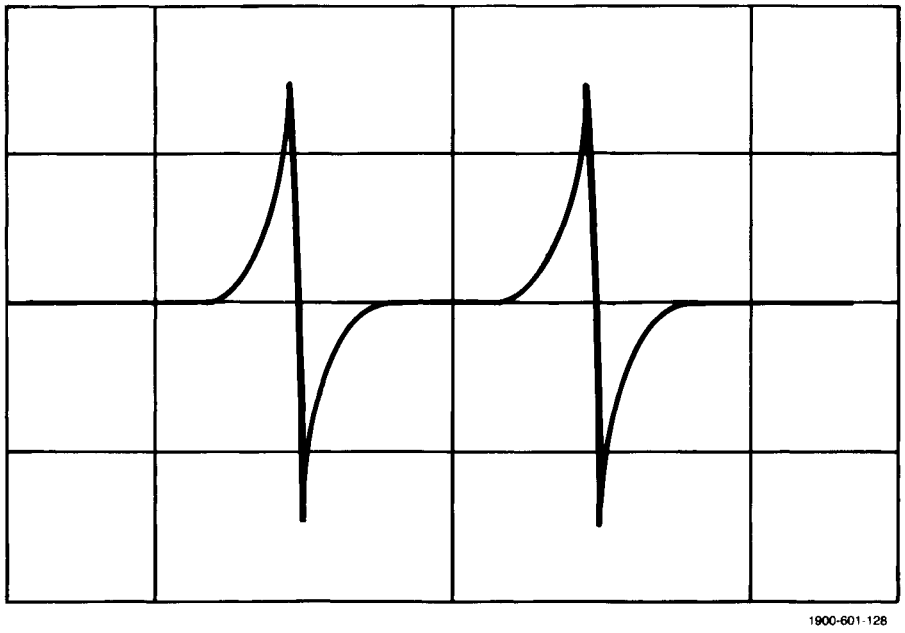
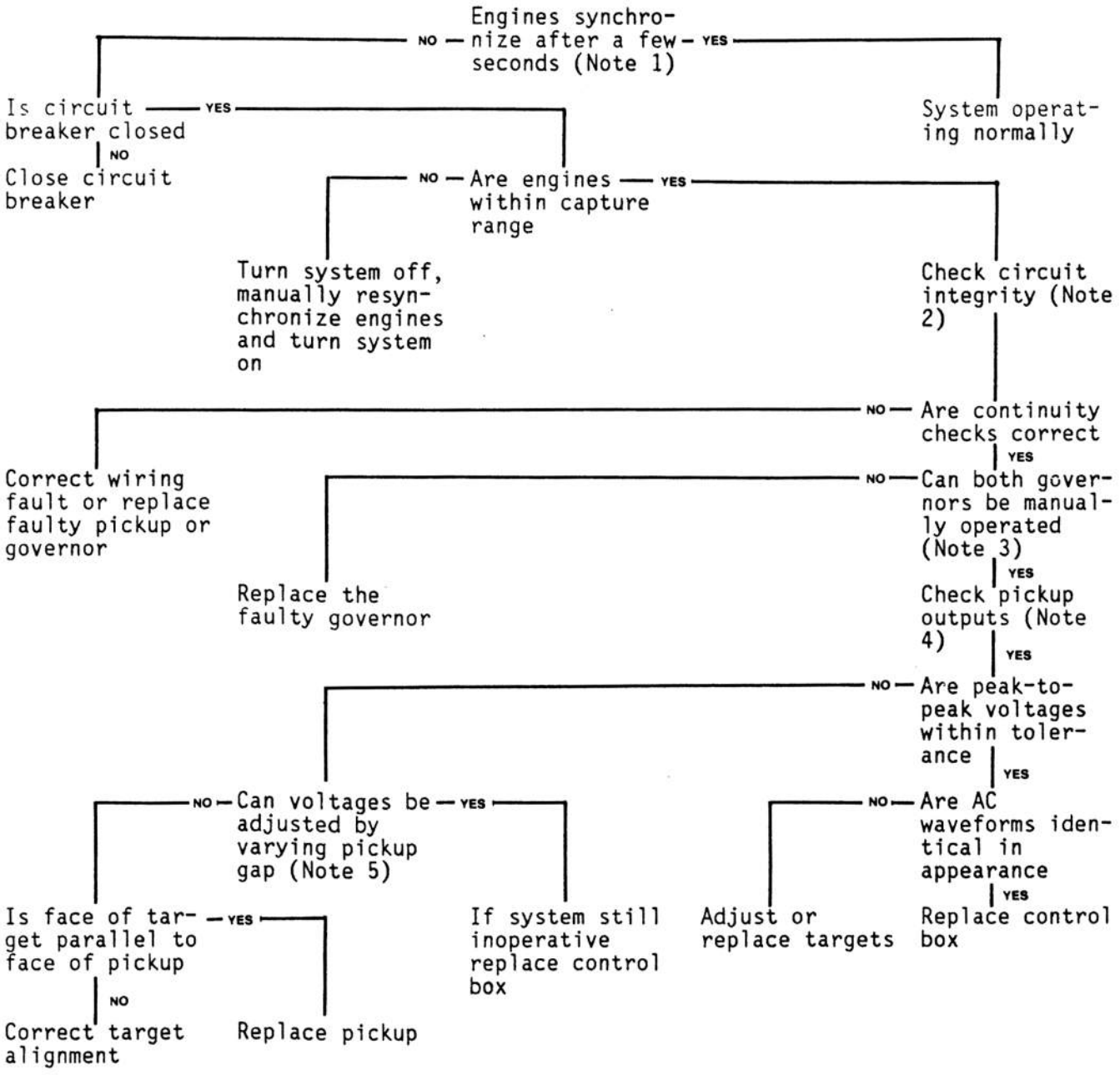


Figure 104
Typical Pickup Waveform



NOTES:
1. Perform synchrophaser FUNCTIONAL CHECK procedure.
2. Perform synchrophaser CONTINUITY CHECK procedure.
3. Perform propeller GOVERNOR OPERATIONAL CHECK procedure.
4. Perform synchrophaser PICKUP OUTPUT CHECK procedure.
5. Refer to synchrophaser PICKUP INSTALLATION AND ADJUSTMENT procedure.

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101546AA.TIF

Table 104
Troubleshooting - Propeller Synchrophaser

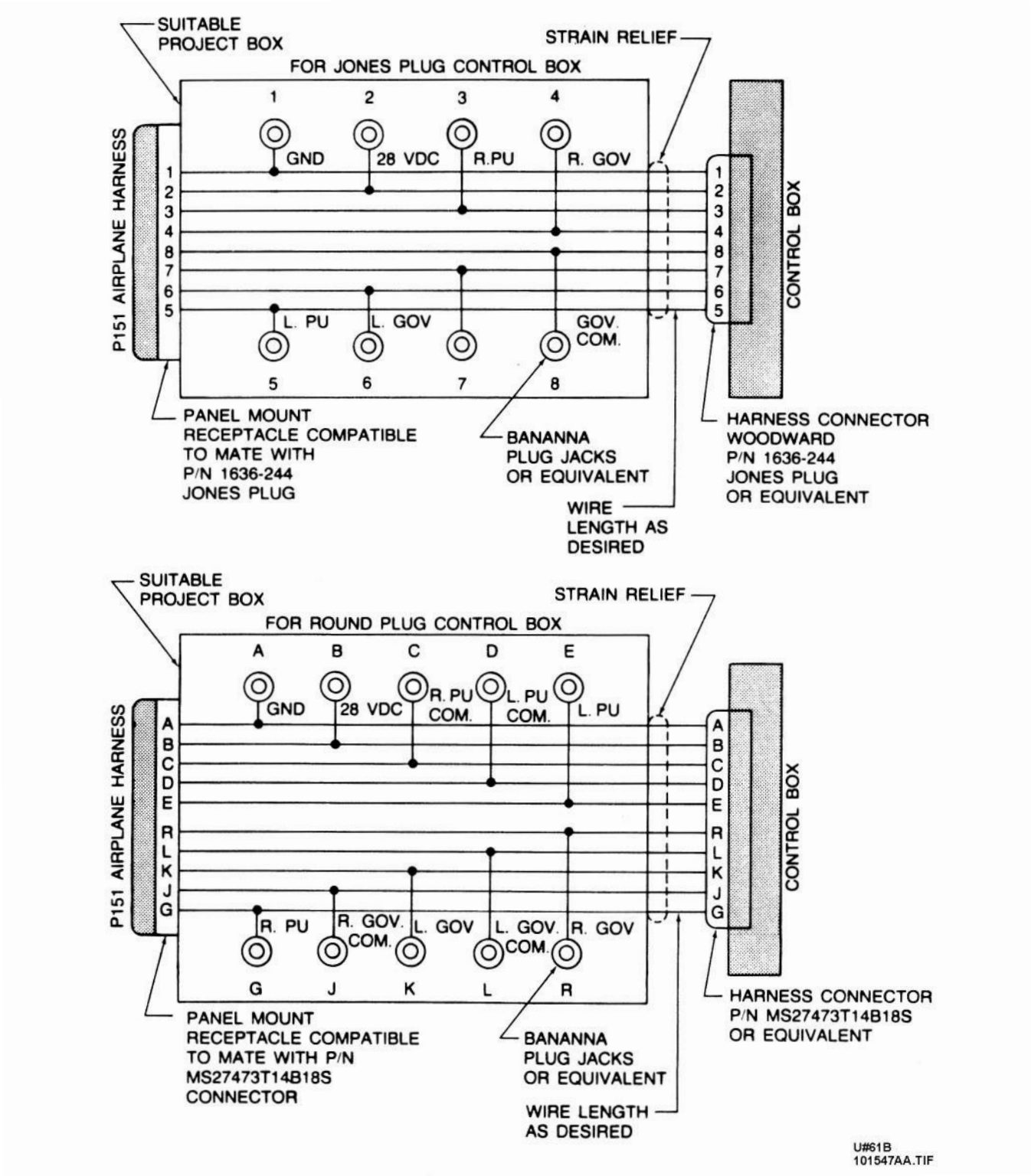

Figure 105
Synchrophaser Breakout Box Schematic

(3) Connect the oscilloscope between pin "3" and ground of the Jones plug for the right pickup, and between pin "5" and ground for left pickup. If the control box is the round plug type, connect the oscilloscope between pins "C" and "G" (common) for the right pickup and between pins "D" and "E" for the left pickup.

(4) Start the engines and adjust the propeller speeds to 1900 rpm as nearly as possible.

CAUTION: *Do not exceed torque red line.*

(5) The waveform generated by the pickups should depart smoothly from the baseline, rise smoothly to the positive peak, drop off to the negative peak and return to the baseline (Ref. Figure 104). There should be no secondary peaks at either the positive or negative peaks of the waveform. The ascending and decending sides of the peaks should be symmetrical:

(a) A ragged waveform may indicate an intermittent fault in the pickup.

(b) An asymmetrical waveform may indicate that the target is not parallel to the tip of the pickup.

(c) A secondary peak in the waveform may indicate that the target has been damaged.

(6) Record the peak-to-peak voltage of the waveforms from each pickup. Ideally, the peak-to-peak voltage should be 3 volts, within 0.5 volt; this will keep the variation of voltage from side-to-side within the necessary one volt.

(a) Should the peak-to-peak voltage of the pickups not be within tolerance, perform the MAGNETIC PICKUP INSTALLATION AND ADJUSTMENT procedure.

PROPELLERS
SYNCHROPHASER SYSTEM
MAINTENANCE PRACTICES

1. TYPE II PROPELLER SYNCHROPHASER (U-50, U-165 AND AFTER)

2. MAGNETIC PICKUP

A. Removal

(1) Remove the lower forward engine cowl to gain access to the pickup.

(2) Unplug the electrical leads of the pickup from the fireseal receptacle.

(3) Unsafety and remove the pickup from its mounting bracket.

B. Installation and Adjustment

The synchrophaser pickups are located in a bracket adjacent to each propeller deicer brush block. The pickup should be installed and adjusted as follows:

(1) Remove the lower forward engine cowl to provide access to the pickup.

(2) Screw the pickup into the mounting bracket.

(3) Adjust the pickup to provide the following:

(a) Clearance of 0.060 inch between the pickup and the target.

(4) Tighten the pickup locknut to a maximum torque of 25 inch-pounds.

(5) Connect the two electrical leads.

(6) Measure the output of the pickup according to the synchrophaser PICKUP OUTPUT CHECK procedure.

(7) If necessary, adjust the gap between the pickup and target Table 201may be used as a guide. The minimum gap should be 0.030 inch.

(8) Torque the locknut on the pickup to a maximum of 25 inch-pounds and safetywire the locknut.

(9) If the connector plug was previously disconnected, connect the plug to the control box.

(10) Perform the synchrophaser FUNCTIONALCHECK procedure.

(11) Install the lower forward engine cowl.

Table 201
Pickup Gap Adjustment

Peak-To- Peak Voltage At 0.05 Inch Gap	Increase Gap To (Inches)
3.0	0.050
3.5	0.057
4.0	0.064
4.5	0.070
5.0	0.075
5.5	0.079
6.0	0.082
6.5	0.085
7.0	0.088
7.5	0.090
8.0	0.093
8.5	0.095
9.0	0.097
9.5	0.098
10.0	0.100

PROPELLERS
INDICATING
MAINTENANCE PRACTICES

1. SECONDARY LOW PITCH STOP (U-1 THRU U-164, EXCEPT U-50, WITHOUT KIT 99-3026-1S)

A. Check

NOTE: The secondary low pitch stop systems have been removed from airplanes equipped with PT6A-27 or PT6A-28 engines after modification by Kit 99-3026- 1S, in compliance with S.I. 0844-247.

(1) Run the engine with the power lever at the idle detent and set the condition lever to 70% gas generator (N_1) speed, then read propeller rpm.

(2) Hold the momentary test switch on the left center subpanel in the SEC LOW PITCH STOP position. Lift the power lever and pull it aft until it is over the top horizontal line marking on the pedestal. After the initial deflection of its pointer in the increase rpm direction, the propeller should indicate a maximum fluctuation of 50 rpm. The midpoint of the fluctuation range should be 170 to 250 rpm higher than the reading noted in Step (1) and the PROP LOW PITCH light (white) in the annunciator panel should be illuminated.

NOTE: If propeller rpm does not increase within the prescribed limits, readjust the position of the sensor on its mounting bracket (Ref. Figure 202). Move the sensor forward to decrease rpm or aft to increase rpm. U-152 thru U-164, readjustment of the electro-optic switch is the same except as noted in Figure 203.

(3) Release the momentary test switch and rpm should increase. Return the power lever to its normal idle position.

2. ELECTRO-OPTIC SWITCH (U-152 THRU U-164 WHICH HAVE NOT INCORPORATED KIT NO. 99-3026- 75)

A. Functional Test

Functional testing of the electro-optic switch (U-152 thru U-164) may be accomplished by placing +28 volts DC between wires K72D18 and K74A18N. Connect a voltmeter between wires W82C18 and K72A18N. (A GE327 lamp or equivalent may be used in lieu of voltmeter). Insert an Opaque object in the slot of the horseshoe shaped switch, assuring the light ray is blocked. Observe 28 volt indication on voltmeter or that the lamp glows.

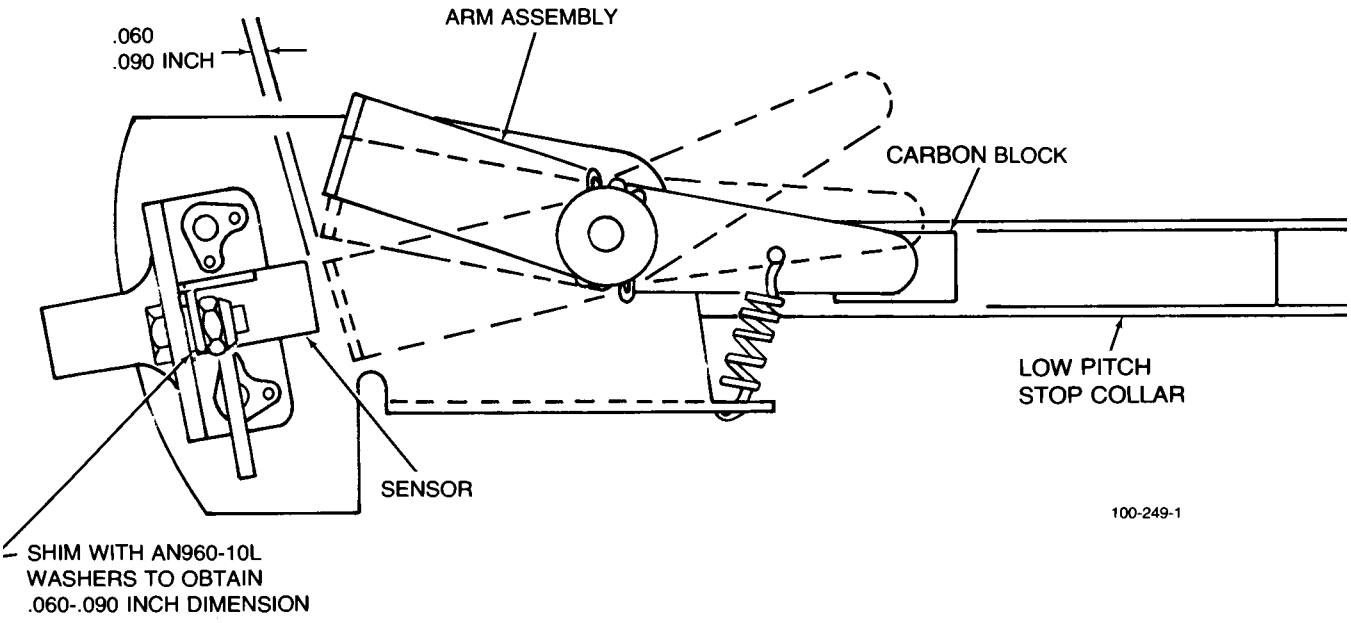

Figure 201
Target Clearance Check

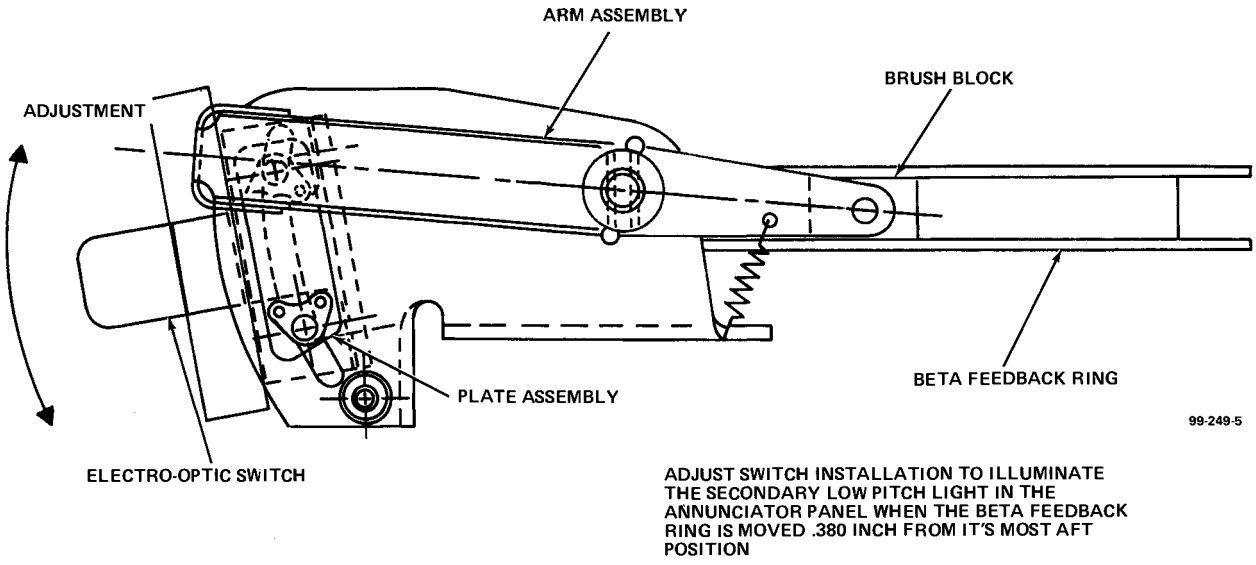

Figure 202
Switch Rigging (U-152 thru U-164 Which Have Not Incorporated Kit No. 99-3026-1S)

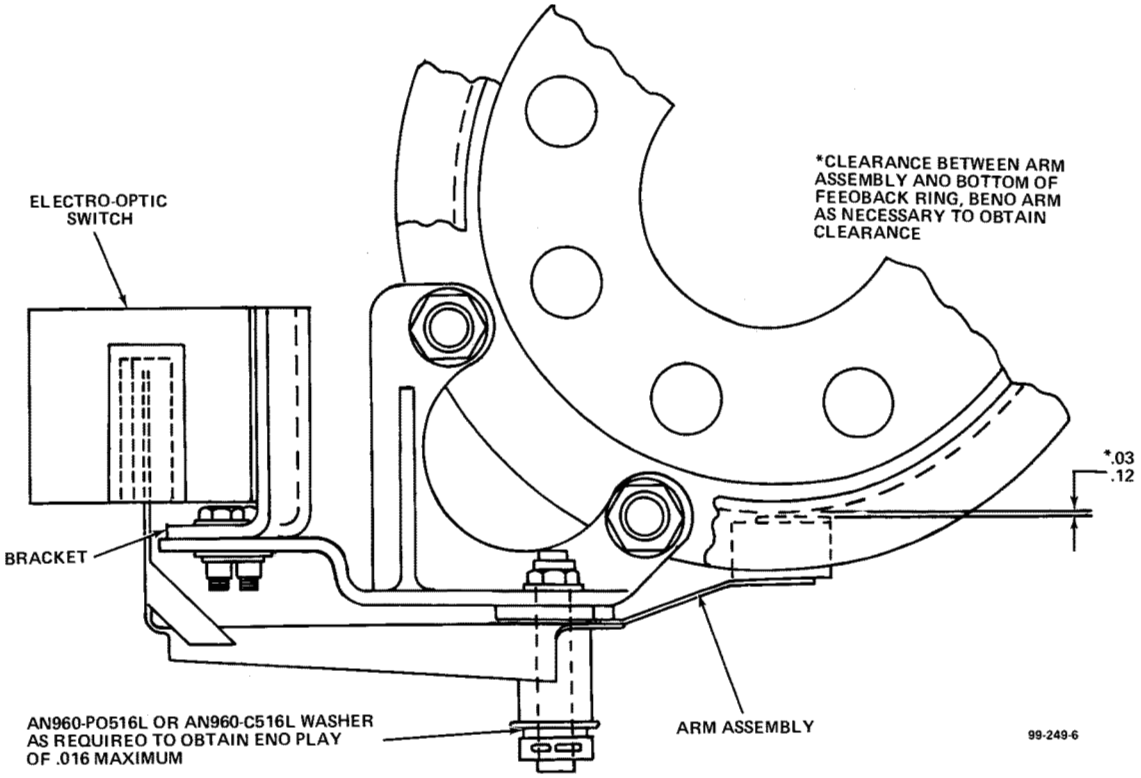

Figure 203
Electro-Optic Switch Installation (U-152 thru U-164 Which Have Not Incorporated Kit No. 993026-1S)

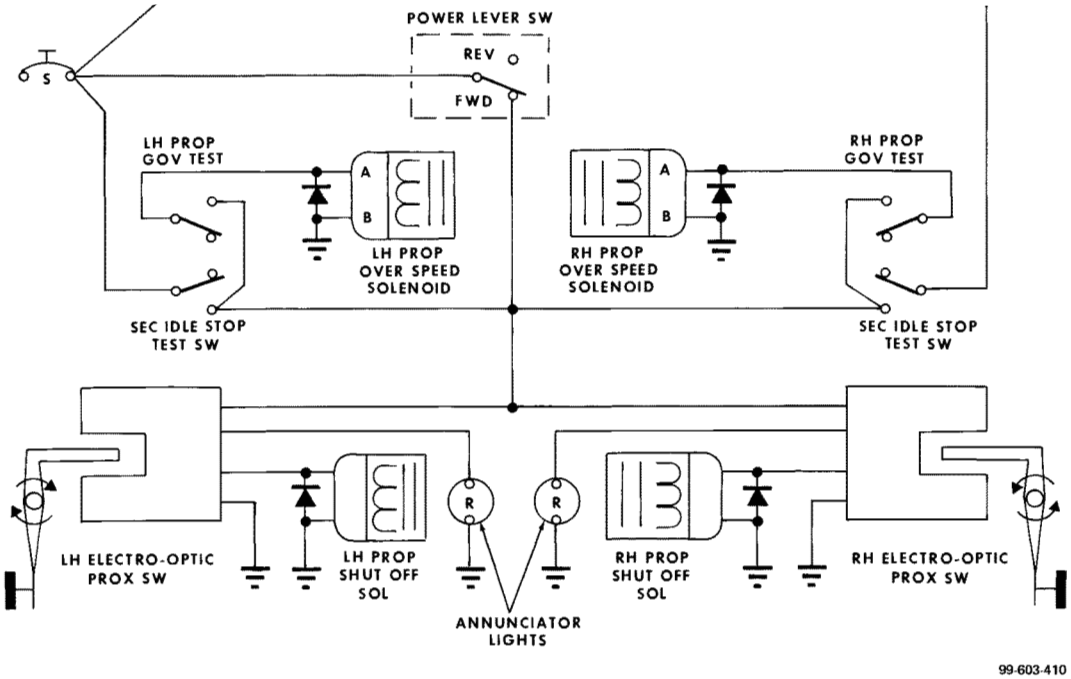

Figure 204
Propeller Governor and Secondary Low Pitch Schematic (U-152 thru U-164) (Prior to Incorporating Kit No. 99-3026-1S)

3. THROTTLE SWITCH (FIGURE 201, CHAPTER 61-21-00) (U-1 THRU U-164, EXCEPT U-50, WHICH HAVE NOT INCORPORATED KIT NO. 99-3026- 1S)

A. Adjustment

Power to the secondary low pitch stop system is provided through the throttle switch actuated by the power levers.

(1) Remove the panel from the left side of the pedestal.

(2) Adjust the throttle switch in its mounting slots until it is in the full overtravel position (approximately 0.06 inch beyond the point of actuation) with the power levers in the forward position.

(3) Lift either power lever and check that the switch clicks at a point approximately 0.20 inch before the end of travel by the lever attached to the actuating arm.

4. DO NOT REVERSE LIGHT SWITCHES (FIGURE 201, CHAPTER 61-21-00)

A. Adjustment

The switches for the DO NOT REVERSE light in the annunciator panel are mounted in the pedestal adjacent to the propeller lever bellcranks. Access to the switches is through the cover on the left side of the pedestal. Adjust the switches as follows:

With the landing gear control switch in the DOWN position and the propeller control levers in the full high rpm (low pitch) position, advance the power levers to obtain 2200 rpm. Adjust each switch in its mounting slot so that the DO NOT REVERSE light (white) illuminates at the point when the propeller lever for that switch is set to obtain 2075 rpm.

5. AUTOFEATHER POWER SWITCH (FIGURE 201, CHAPTER 61-21-00)

A. Adjustment

Access to the autofeather power switches is through the cover on the left side of the pedestal. The switches may be adjusted as follows:

With the autofeather control switch in the ARM position, adjust each power switch in its mounting slots until its respective AUTOFEATHER ARMED light illuminates when the power levers are set to obtain 90% N_1 speed. On airplanes prior to U-165 except U-50, the light is in the left instrument panel. At U-50. U-165 and After, the light is in the annunciator panel above the pedestal.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 71 - POWER PLANT

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POWER PLANT
GENERAL
DESCRIPTION AND OPERATION

1. GENERAL

The Model 99 Airliner Series Airplanes may be equipped with the Pratt and Whitney turbine engines, PT6A-20, -27, -28 or -36, as follows:

	-20	-27		-28	-36
99 Airliners	X				
99 Airliners (U- 1 thru U-55, Except U-50) With Kit 99-5014 installed		X	or	X	
99A, A99A and B99		X	or	X	
C99 Airliners					X

The Pratt and Whitney engines installed on the 99 Airliner Series Airplanes are of the "Free Turbine" type and are so named because the two turbines are free of one-another (not interconnected) and function independently. One turbine drives the compressor and accessory gear train while the other drives the propeller: through reduction gears.

PT6A-27 and -28 engines are designed in accordance with the QEC (Quick Engine Change) concept which provides for LH and RH engine packages, including cowlings and all equipment forward of the engine firewall, to be identical and interchangeable, with the exception of the exhaust stacks and the air conditioner compressor. The PT6A-36 engine is a modular concept design which allows for change out of certain engine parts without the necessity of removing the engine from the airplane. The PT6A-28 engine may be used as a spares replacement for the PT6A-27 engine by installing the U.A.C.L. scavenge oil tee (P/N 3014775) and hose (P/N 3014776). Both engine installations use the same engine monitoring instruments and may be used on the same airplane. The same engine controls are used on all installations: however, slight rigging differences are necessary for the proper operation of each installation.

NOTE: For detailed instructions on module replacement for PT6A-36 engines, refer to the appropriate Pratt and Whitney Engine Maintenance Manual.

Engine and propeller functions are controlled by conventional levers and push-pull cables: the power lever controls the engine power and the propeller reversing mechanism, the propeller control lever controls the pitch of the propeller, the fuel condition lever controls fuel flow to the fuel control unit or start control unit. Refer to the engine maintenance manual for complete description and maintenance information on the engine and accessories.

POWER PLANT
ENGINES
MAINTENANCE PRACTICES

1. ENGINE

A. Removal

NOTE: Airplanes idle for more than 7 days due to maintenance or other reasons should either have the engines operated or preserved in accordance with the engine manufacturers recommendations, as found in the ENGINE MAINTENANCE MANUAL, to avoid contamination of the internal mechanism.

Engine removal is accomplished as described in the following procedure:

(1) Preliminary Steps.

(a) Position a suitable stand under the tail of the airplane.

CAUTION: *With both engines removed, the airplane becomes tail heavy. Failure to install a tail stand may result in the airplane tail settling to the ramp. With only one engine removed, the airplane may be towed if done carefully. It is advisable to position a stand under the tail while the airplane is parked with one engine removed.*

(b) Drain the engine oil as described in Chapter 12.

(c) Make sure all electrical power to the airplane is disconnected.

(2) Remove the propeller as described in PROPELLER REMOVAL, Chapter 61.

(3) Remove the engine cowling as indicated in 71-10-00.

(4) Place a suitable cover over the engine inlet air screen to prevent foreign matter from entering the air inlet.

(5) Disconnect the following plumbing hoses from the engine:

NOTE: Tag or identify all hose to facilitate and ensure correct installation on the new engine. Cap all open hoses and engine ports to prevent contamination.

(a) Oil inlet hose - flare fitting.

(b) Oil overboard breather hose - clamp.

(c) Oil outlet hose - flare fitting.

(d) Fuel inlet hose - flare fitting.

(e) Fuel return hose - flare fitting.

(f) Forward combustion chamber drain hose - flare fitting.

(g) Aft combustion chamber drain hose - flare fitting.

(h) Manifold dump valve drain hose - flare fitting.

(i) Fuel pump drain hoses - clamps.

(j) Air bleed hose - flare fittings.

(k) EPA system lines.

(6) Except when the existing engine is to be replaced with another, all electrical leads but that for the thermocouple harness may be disconnected by removing the electrical plugs from the firewall receptacles. If the engine harness is to be used on a replacement engine, disconnect the electrical leads and plugs from the components listed below and remove the electrical harness clamps necessary to allow engine removal.

NOTE: Tag or identify all electrical connectors gnd note harness clamp locations to facilitate correct reinstallation. Cap all plugs and receptacles to prevent contamination.

(a) Starter-generator.

(b) Ignitor control box.

(c) lnterstage turbine temperature thermocouple.

(d) Gas generator tachometer generator.

(e) Oil pressure transmitter.

(f) Oil temperature transmitter.

(g) Propeller tachometer generator.

(h) Propeller overspeed governor.

(i) Propeller deicer brushes.

(j) Torque pressure transmitter.

(k) High pressure autofeather switch.

(l) Low pressure autofeather switch.

(m) Autofeather dump solenoid.

(n) PT6A-20 installations only, proximity sensors.

(7) Disconnect the engine controls as follows:

(a) Disconnect the propeller control cable from he governor control arm. If the airplane is equipped with a propeller synchronizer, disconnect the control cable from the trimmer immediately aft of the driven trimmer gear. Remove the firesleeve from the control cable.

(b) Disconnect the power control cable from the power control lever in the mounting bracket for the cam and attendant control linkage.

(c) Disconnect the condition lever control cable from the linkage at the fuel control unit, or start control unit.

NOTE: Tag and retain the throttle attaching bolt, washer, and nut for installation on the new engine.

(8) Remove the engine from the airplane as follows:

NOTE: Identify and retain all engine mounting bolts, washers, and nuts to facilitate and insure a correct new engine installation.

(a) Attach an engine sling to the engine hoisting lugs. Position a suitable hoist directly over the eye of the hoisting sling.

(b) Remove the screws on both sides of the forward and aft cowling bulkheads.

(c) Remove the bolts attaching the lower engine mount adapters to the engine case. The large bolt through the center of the engine mount need not be removed unless new mounts are to be installed.

(d) Remove the two taper pins securing the upper engine mount V-brace.

(e) The engine may now be hoisted to clear the lower cowl structure.

B. Installation

(1) Attach an engine sling to the engine hoisting lugs.

(2) Hoist the engine over the airplane engine mount and slowly lower it into the mount bed.

CAUTION: *Take care to prevent damage to the engine components, plumbing, and wiring while lowering the engine into position.*

(3) Attach the lower engine mounts. Refer to Step (1) of ENGINE BUILD UP for correct torque.

(4) Position the upper mount rods in the fittings at the firewall and install the taper pins and secure with the washers and nuts. Torque the nuts 30 to 40 inch-pounds and safety with cotter pins.

(5) Install the bolts and screws at the two cowl bulkheads.

(6) Connect the following hoses to the engine fittings:

NOTE: Apply a light coat of anti-seize compound to the male threads of all fittings as specified in Chapter 91, Table 4.

(a) Oil inlet hose - flare fitting.

(b) Oil overboard breather hose - clamp.

(c) Oil outlet hose - flare fitting.

(d) Fuel inlet hose - flare fitting.

(e) Fuel return hose - flare fitting.

(f) Forward combustion chamber drain hose - flare fitting.

(g) Aft combustion chamber drain hose - flare fitting.

(h) EPA system lines.

(i) Fuel control drain hoses - clamps.

(j) Air bleed hose - flare fitting.

(7) Install the engine harness plugs into the firewall receptacles. Route the electrical harness and secure it with the attaching clamps as it was installed on the replaced engine. Make all electrical connections finger tight to the engine components listed below and safety wire the electrical plug.

(a) Starter-Generator.

(b) Ignitor control box.

(c) Interstage turbine temperature thermocouple.

CAUTION: *Torque the thermocouple harness connections according to the ENGINE MAINTENANCE MANUAL.*

(d) Gas generator tachometer generator.

(e) Oil pressure transmitter.

(f) Oil temperature transmitter.

(g) Propeller tachometer generator.

(h) Propeller overspeed governor.

(i) Propeller deicer brushes.

(j) Torque pressure transmitter.

(k) High pressure autofeather switch.

(l) Low pressure autofeather switch.

(m) PT6A-20 installations only, proximity sensors.

(n) Autofeather dump solenoid.

(8) Control cables installation.

(a) Connect the power control cable to the power lever at the mounting bracket for the cam and attendant linkage.

(b) Connect the condition lever control cable to the linkage at the fuel control unit, or start control unit.

(c) Connect the propeller governor control cable. Rig engine and propeller reversing mechanism per instructions in Chapter 61 and Chapter 76.

(9) Install the propeller as described in Chapter 61.

(10) Remove the cover over the engine inlet air screen.

(11) Engine cowling installation:

(a) Install the forward bottom cowling and secure with the attaching screws. Secure the rod end of the control cable to the linkage for the movable vane with the attaching bolt, washer and bushing.

(b) Position the upper aft cowling on the cowling hinges and install the attaching screws.

(c) Position the upper forward cowling on the engine and secure the four latches.

(12) If necessary refer to the ENGINE MAINTENANCE MANUAL for depreservation of the engine.

2. QUICK ENGINE CHANGE (QEC) (U-50, U-80 AND AFTER)

A. Removal

The Quick Engine Change may be accomplished by removing the complete engine unit, including cowling, as follows:

NOTE: Airplanes which are expected to be idle for more than 7 days due to maintenance or other reasons should either have the engines operated or preserved in accordance with the engine manufacturers recommendations, as found in the ENGINE MAINTENANCE MANUAL, to avoid contamination of the internal mechanism.

(1) Position a suitable stand under the tail of the airplane.

CAUTION: *With both engines removed, the airplane becomes tail heavy. Failure to install a tail stand may result in the airplane settling to the ramp. With only one engine removed, the airplane may be towed if done carefully. It is advisable to position a stand under the tail while the airplane is parked with one engine removed.*

(2) On the LH side of the engine, disconnect the following items:

(a) Propeller control cable from the governor control arm. If the airplane is equipped with a propeller synchronizer, disconnect the control cable from the trimmer immediately aft of the driven trimmer gear. Remove the firesleeve from the control cable (Ref. Figure 201).

(b) Pneumatic line at the firewall.

(c) Two wiring cables at the firewall.

(d) Fire Detector.

(e) The air conditioning compressor and mount adapter (if installed).

NOTE: Step (e) applies only to those airplanes equipped with an air conditioner. The compressor is mounted on the RH engines.

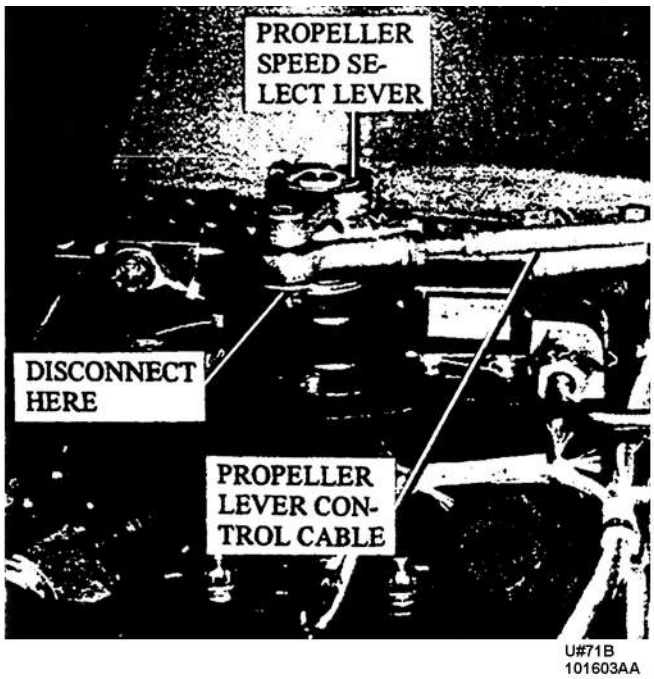

Figure 201
Propeller Control Cable

(3) On the RH side of the engine, disconnect the following Items:

(a) Power control cable from the power control lever in the mounting bracket for the cam and attendant control linkage (Ref. Figure 202).

(b) Condition lever cable from the linkage at the fuel control or start control unit (Ref. Figure 203).

(c) Control cable rod end from the linkage to the movable vane of the internal anti-icer.

(d) One wiring harness at the firewall.

(e) One ground cable at the firewall.

(f) Main fuel line at the firewall.

(g) EPA system lines.

(h) ITT lead wire at the probe and at the cable attach points.

(i) Two fire detectors.

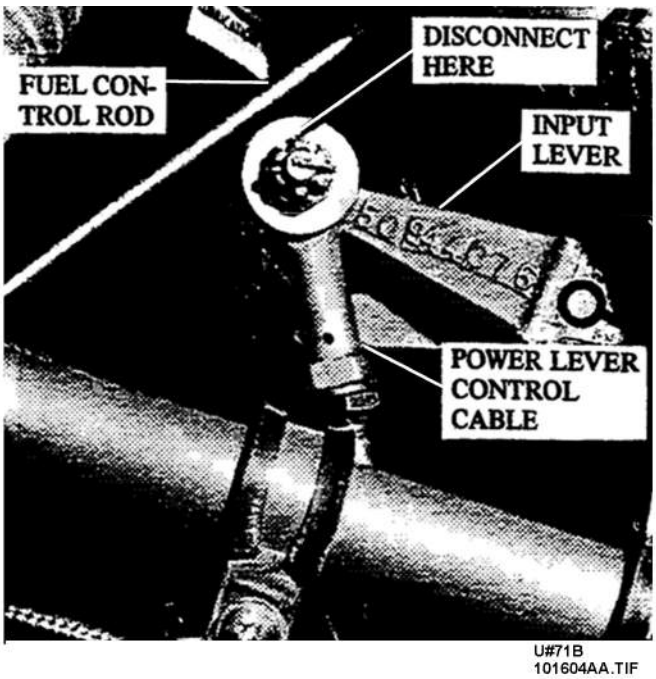

Figure 202
Power Control Cable

(4) Remove the englne from the airplane as follows:

(a) Attach an englne sling to the engine hoisting lugs. Postilion a suitable hoist directly over the eye of the hoisting sling.

(b) Remove the four engine attach bolts at the firewall.

(c) The englne may now be removed from the airplane.

B. Installation

NOTE: Airplane serials U-158 through U-164 are manufactured with either PT6A-27 or PT6A-28 engines. All airplanes prior to U-158, except U-50 and the basic Model 99, may use the PT6A-28 as a spare replacement for the PT6A-27 englne by replacing the existing scavenge oil tee and hose (U.A.C.L. P/N's 3014775 and 3014776) with oil tee and hose (U.A.C.L. P/N's 3011351 and 3011362) respectively. The PT6A-27 and PT6A-28 engines are interchangeable and it is permissible to use a PT6A-27 and PT6A-28 engine on the same airplane.

Apply a light coat of anti-seize compound to the male threads of all fittings as specified in Chapter 91, Table 4.

(1) Attach an englne sling to the engine hoisting lugs. Position a suitable hoist directly over the eye of the hoisting sling. Hoist the englne into the attach position.

(2) Install the four englne bolts attaching the V-brace mount to the firewall and torque 365 to 390 inch-pounds.

(3) Connect the electrical wiring harnesses to the firewall electrical plugs and safety the connectors.

(4) Connect the fire detectors.

(5) Connect the ITT lead were at the probe and at the cable attach points.

CAUTION: *Torque the thermocouple harness connections according to the ENGINE MAINTENANCE MANUAL.*

(6) Connect EPA system lines.

(7) Connect the main fuel line at the firewall.

(8) Connect the ground cable at the firewall.

(9) Connect the control cable rod end to the linkage on the movable vane of the inertial anti-icer.

(10) Connect the condition lever control cable to the linkage at the fuel control unit, or start control unit (Ref. Figure 203).

(11) Connect the power control cable to the power control lever in the mounting bracket for the cam and attendant control linkage (Ref. Figure 202).

(12) Replace the firesleeve on the propeller control cable and connect the cable to the governor control arm. If the airplane is equipped with a propeller synchronizer, connect the control cable to the trimmer immediately aft of the driven trimmer gear.

(13) Connect the pneumatic line at the firewall.

(14) Install the air conditioning compressor mount adapter and compressor.

NOTE: Step (14) applies only to those airplanes equipped with an air conditioner. The compressor is mounted on the RH engine. Refer to Chapter 21 for COMPRESSOR INSTALLATION.

If necessary, refer to the ENGINE MAINTENANCE MANUAL for depreservation of the engine.

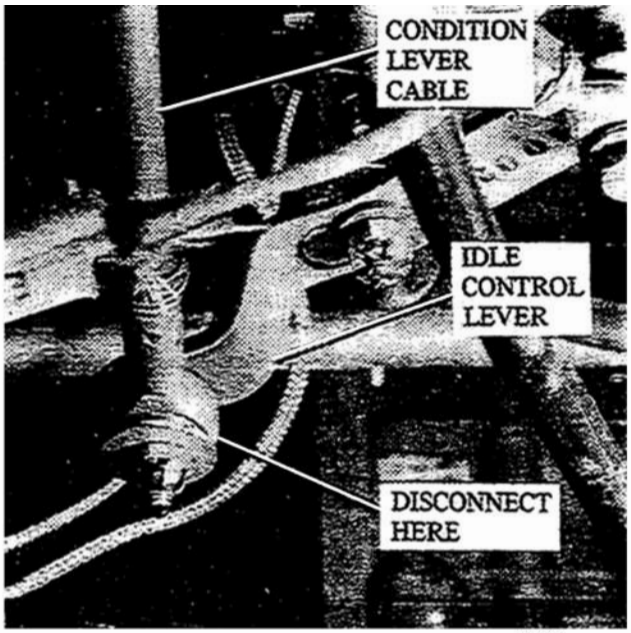

Figure 203
Condition Control Cable

3. ENGINE BUILDUP

CAUTION: *Consult the Engine Maintenance Manual before attempting to remove the new engine from the shipping container.*

Engine build-up consists of the installation of accessories and equipment on the new engine. Refer to the ENGINE MAINTENANCE MANUAL for removal and replacement procedures. After all accessories have been installed, complete the engine build-up as follows:

NOTE: Tag or identify all hose, bolts, nuts, washers, electrical connectors, and note harness clamp locations for installation on the new engine. Cap all open hoses and engine ports to prevent contamination.

(1) Remove the engine mount V-brace, engine mount (vibration isolators), and upper halves of the nacelle bulkheads as a unit and install on the new engine.

NOTE: All engine mounts (vibration isolators) on an engine must be of the same manufacturer and type. The engine mount (vibration isolator) may be rebuilt by the replacement of the rubber cushion.

Torque the engine mount (vibration isolator) attaching bolts to engine case 180 to 225 inch-pounds (275 to 300 inch-pounds on PT6A-36 installation only) If new engine mounts (vibration isolators) are installed, torque the large bolt through the center of the mount 400 to 550 inch-pounds (480 to 600 inch-pounds on PT6A-36 installation only).

(2) Remove the bleed air adapter from the old engine and install on the new engine using the same bolt and washer combination, with a new gasket. Torque bolts 50 to 70 inch-pounds.

(3) Remove engine control brackets and supports from the old engine and install on the new engine. Torque nuts 20 to 30 inch-pounds.

(4) Remove the exhaust stacks and install on corresponding (left or right) exhaust ports. Torque the bolts 50 to 70 inch-pounds.

4. FUEL CONTROL UNIT

A. Purge and Soak Procedure

CAUTION: *Observe the starter operating limits throughout the following procedure.*

(1) To purge the fuel control unit of preservation oil, install the unit on the engine and connect it to the airplane fuel supply. Install a suitable line between the fuel control outlet and a clean container.

(2) With the ignition OFF, and the fuel control lever in the IDLE position, motor the engine until clean fuel flows from the fuel outlet. During the motoring run, move the fuel lever from the IDLE to CUTOFF position to ensure fuel shutoff.

(3) Remove the line used to purge the Fuel Control Unit and connect the engine plumbing to the fuel control outlet. Check the installation for leakage.

NOTE: An eight hour soaking period is recommended following the purging procedure. This will enable the bypass valve diaphragm to regain the pliability it possessed during unit calibration and will permit the synthetic packing in the unit to swell for a satisfactory seal.

CAUTION: *Before attempting a start, motor the engine with the fuel shutoff to ensure that all fuel has been purged from the engine.*

5. FUEL NOZZLES

A. Handling

Extreme care should be exercised when handling fuel nozzles; even fingerprints on the orifice may produce a poor spray pattern. If fuel nozzles are not to be reinstalled in the engine immediately, they should be placed in a covered container to prevent exposure to dirt. If nozzles are to be shipped or stored, they must be separately and securely wrapped. Packing nozzles together in a single container which allows relative movement will damage orifice faces.

6. ENGINE MOUNTS

A. Reconditioning

The firewall engine mount is a welded assembly fabricated from 4130 chrome-moly steel, so normally the reconditioning of this type of mount would be as follows:

(1) Strip paint from the mount with Turco 5351 or equivalent and magnaflux inspect for cracks.

(2) Remove the taper pins and check the condition of pin holes.

(3) On the reinstallation of the taper pin, torque 30 to 40 inch pounds.

(4) Some cracks may be repaired by approved heliarc welding. The Hawker Beechcraft Service Department should be consulted on an individual mount basis to determine if welding is possible or feasible. If cracks are to be welded, pre-heat the section to be repaired 400 to 800°F and post-heat after repair 700 to 800°F for stress relief.

(5) Check the straightness of tubes.

(6) Clean mount for painting and cover all bushings to prevent paint from adhering to the inside of the bushing ends.

(7) Prime and paint with aluminum epoxy paint.

POWER PLANT
COWLING
MAINTENANCE PRACTICES

1. ENGINE COWLING

A. Removal

(1) Remove the upper aft cowling by removing the two bolts from each upper hinge attach point.

(2) Remove the upper forward cowling by disengaging the four cowl latches.

(3) Remove the lower forward cowling as follows:

CAUTION: *To prevent damage to both the spinner edge and the edge of the cowling it is imperative that the cowling be kept level throughout the removal procedure.*

(a) Mask off the forward end of the cowling to protect it against scratches from the spinner.

(b) Disconnect the deicer boot wiring.

(c) Disconnect the rod end of the control cable from the linkage to the movable vane of the inertial anti-icer.

(d) Rotate the propeller until one blade points straight up.

(e) Remove all retaining screws from both sides except for two at the top aft edge of the cowling.

(f) With a man positioned on each side of the cowling, hold the cowling level and remove the remaining two screws.

(g) Move the cowling forward and down until the vane control arm is clear and the cowling clears the spinner, then lower the aft end of the cowling.

B. Installation

(1) With a man positioned on each side of the cowling, raise the cowling into position on the nacelle. Take care to avoid contacting either the propeller spinner or the control arm of the anti-icer vane.

(2) Holding the cowling level, install the two screws at the top aft edge of the cowling.

(3) Install the retaining screws along both sides of the cowling.

(4) Connect the rod end of the control cable to the linkage for the movable vane of the inertial anti-icer.

(5) Connect the deicer boot wiring.

(6) Install the upper forward cowling on the lower aft cowling and engage the cowl latches.

(7) Install the upper aft cowling and secure it in place with the two hinge bolts at each upper attach point.

POWER PLANT
MOUNTS
DESCRIPTION AND OPERATION

1. GENERAL

The engine is supported by a welded tubular frame bolted to the nacelle at four attachment points. Three vibration isolators are utilized to secure the engine to the tubular support and to dampen engine vibration. The vibration isolators are each secured to the engine combustion casing by four bolts and by a single bolt to the welded tubular support. A vibration isolator can be replaced without removing the engine provided that the weight of the engine is adequately supported. The vibration isolators may be renewed by replacing the rubber cushions. All engine mounts (vibration isolators) on any engine MUST be of the same manufacturer and type. For detailed removal and installation of the engine mounts and support structure, refer to ENGINE BUILD UP and ENGINE INSTALLATION.

POWER PLANT
FIRESEALS
DESCRIPTION AND OPERATION

1. GENERAL

The fireseals are fitted to the engine, one forward and the other aft of the engine compressor intake. Each fireseal is constructed of 2 semi-circular sections, which are bolted to the engine fireseal flange and to each other to form a complete fireseal between the engine and cowlings. The fireseals also provide a mounting location and support for all lines, controls, and ducts that pass from one engine fire zone to another.

POWER PLANT
ELECTRICAL HARNESS
DESCRIPTION AND OPERATION

1. GENERAL

The power plant electrical harness distributes power to or from the starter-generator, fire detectors and all engine electrical accessories. The harness is connected at the engine firewall by 3 connectors. The harness is routed from the engine firewall, through the rear and forward fireseals to the reduction gear case and propeller electrical systems.

POWER PLANT
ENGINE DRAINS
DESCRIPTION AND OPERATION

1. EPA SYSTEM

Fuel from the drain on the fuel control or start control unit of the engine is routed to the fuel drain collector tank mounted on the center of the fire seal directly under the accessory section of the engine. This tank contains an inertial float for a normally open switch that closes the circuit to a pump mounted on the fireseal just to the right of the tank. When fuel is dumped, the tank fills and closes the switch. At the first activation of the fuel line heater from which the power for the pump is derived, fuel is pumped aft to the purge line leading to the nacelle tank. A check valve in the line prevents the back flow of fuel during purging of the fuel system at engine start. The drain is teed into the fuel return line aft of the fuel purge valve to prevent interruption of pressure in the fuel system. A vent line is plumbed from the top of the collector tank downward to the fuel drain manifold on the underside of the nacelle. The vent line is also routed through a flame arrestor.

POWER PLANT
ENGINE DRAINS
TROUBLESHOOTING

1. GENERAL

A. Troubleshooting - Engine Fuel and Drain Collector System

	TROUBLE	PROBABLE CAUSE	REMARKS
(1)	Pump inoperative.	a. Improper voltage and polarity.	a. Check.
		b. Improper ground.	b. Check.
		c. Improper or loose electrical connections.	c. Check.
		d. Defective float or float switch.	d. Check float or replace switch.
(2)	Pump operates but fails to deliver fuel.	a. Air leak in lines or fittings.	a. Check lines and fittings.
		b. Kinked or clogged lines restrict fuel flow.	b. Check and correct as necessary.

B. Smoke Detection System (U-1 thru U-164, except U-50)

Table 101
Troubleshooting Smoke Detection System (U-1 thru U-164, except U-50)

PROBLEM	PROBABLE CAUSE	REMARKS
(1) Failure to test.	a. Warning lamp inoperative.	a. Check circuit continuity and warning bulb.
	b. Defective test lamp or beacon lamp in detector.	b. Replace required item.
	c. Faulty wiring.	c. Repair wiring.
	d. Defective detector.	d. Replace detector.
	e. Defective control amplifier.	e. Replace control amplifier.
(2) Failure to recycle after test.	a. 28 volt leakage to signal lead.	a. Check detector pin C for more than 10 volts.

POWER PLANT
ENGINE DRAINS
MAINTENANCE PRACTICES

1. FUEL DRAIN COLLECTOR FILTER

A. Servicing

Clean the filter of the fuel pump as follows at 100-hour intervals:

(1) Remove the fuel pump from the airplane as indicated in FUEL DRAIN COLLECTOR PUMP REMOVAL.

CAUTION: *Do not tamper with the seal in the center of the mounting bracket attached to the bottom of the pump. This seal retains the dry gas that surrounds the electrical components in the upper part of the pump.*

(2) Release the bottom cover from the bayonet fittings of the pump with a 5/8-inch wrench, then twist the cover off the pump by hand.

(3) Carefully remove the filter from the open end of the pump.

(4) Wash the filter in cleaning solvent (38, Table 6, 91-00-00) and blow out dirt and cleaning solvent with air pressure.

(5) Replace the cover gasket if deteriorated. Clean the cover and blow the magnet in the center of the cover clean.

(6) Place the cover gasket and magnet in the bottom of the cover, then place the filter screen around the magnet in the bottom of the cover. The screen must fit snugly at both ends.

(7) Taking care to avoid pinching or distorting the filter screen, insert the filter into the pump and twist the cover by hand to hold it in position on the pump housing. Tighten the cover securely in place with a 5/8-inch wrench.

(8) Install the fuel pump in the nacelle as indicated in FUEL DRAIN COLLECTOR PUMP INSTALLATION.

2. FUEL DRAIN COLLECTOR PUMP

A. Removal

(1) Disconnect the electrical connector at the center of the fuel pump.

(2) Disconnect the inlet tube at the elbow on the bottom of the fuel pump.

(3) Disconnect the outlet hose at the nipple on top of the fuel pump.

(4) Remove the mounting bolts, washers, and nuts securing the base of the fuel pump to the fireseal immediately forward of the engine accessory section.

B. Installation

(1) Attach the base of the fuel pump to the fireseal with the retaining bolts, washers and nuts.

(2) Connect the outlet hose to the nipple on top of the fuel pump.

(3) Connect the inlet tube to the elbow on the bottom of the fuel pump.

(4) Connect the airplane wiring to the electrical connector at the center of the fuel pump.

3. FUEL DRAIN COLLECTOR TANK

A. Removal

(1) Make certain that all electrical power is disconnected from the airplane and that all switches are off.

(2) Disconnect the electrical connector on the right side of the tank.

(3) Loosen the clamp and remove the vent tube from the top of the tank.

(4) Disconnect the inlet hose at the fitting on the aft side of the tank.

(5) Disconnect the outlet tube at the fitting on the aft side of the tank.

(6) Remove the three mounting screws on the forward side of the tank (forward side of the fireseal) and remove the tank from the airplane.

B. Cleaning

Remove the drain collector tank and clean every 12 months as follows:

(1) Refer to FUEL DRAIN COLLECTOR TANK REMOVAL and remove tank from airplane.

(2) Remove the cover plate from the tank and clean with solvent (38, Table 6, 91-00-00).

(3) Refer to FUEL DRAIN COLLECTOR TANK INSTALLATION and install tank in airplane.

C. Installation

NOTE: Prior to installation, check the tank for leaks. Plug all openings and apply a pressure of three psi to the tank. Use a mild soap solution for the leakage check. Leaks in the tank can be repaired providing the weld does not interfere with the structural integrity of the tank or with the function of the float inside the tank. The tank is manufactured of 0.040 annealed sheet steel. The welding heat and rods used for repair must he compatible with the material.

(1) Mount the tank on the aft side of the fireseal and secure it in place with the mounting screws and washers.

(2) Secure the inlet hose to the fittings on the aft side of the tank with the retaining nut and washer.

(3) Secure the outlet tube to the fitting on the aft side of the tank with the retaining nut and washer.

(4) Connect the vent line on top of the tank and tighten the clamp in place.

(5) Connect the electrical connector on the right side of the tank and tighten the retaining nut.

4. FUEL DRAIN COLLECTOR TANK FLOAT SWITCH

A. Replacement

(1) Remove the tank per instructions in FUEL DRA!N COLLECTOR TANK REMOVAL.

(2) Remove the cover plate and gasket from the top of the tank.

(3) Remove the nut and washer that secures the switch end of the float switch assembly.

(4) Install a new float switch assembly (with "TOP" placard facing up) making certain that a washer is installed on the inside of the tank and a washer outside of the tank before securing the switch with the locking nut.

(5) lnstall the cover plate using a new gasket.

(6) After replacing the switch assembly and securing the cover plate, plug off the tank openings and test for leaks at three psi.

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72-00-00	1	Nov 1/10

CHAPTER 72 - ENGINE

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ENGINE

72-00-00

Description and Operation 1
General 1

ENGINE
ENGINE
DESCRIPTION AND OPERATION

1. GENERAL

NOTE: Refer to the following Pratt and Whitney Engine Maintenance Manuals for specific engine information:

Publication 3015442 for m6A-20 Engines

Publication 3027342 for PT6A-21 Engines

Publication 3013242 for PT6A-27 and PT6A-28 Engines

Publication 3021242 for PT6A-36 Engines

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74-CONTENTS	1	Nov 1/10
74-00-00	201 and 202	Nov 1/10

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CHAPTER 74 - IGNITION

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Auto-Ignition Pressure Switch 201
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Auto-Ignition 202
Check 202

IGNITION
AUTO-IGNITION
MAINTENANCE PRACTICES

1. AUTO-IGNITION SYSTEM

The auto-ignition system is designed to ensure instantaneous ignition under adverse conditions of icing weather and during all night flights above 14,000 feet. The auto-ignition control switch on the left subpanel is wired to a pressure switch mounted adjacent to the torque pressure transmitter just forward and above the exhaust outlet on each engine. This switch monitors torquemeter oil pressure. When the control switch is in the ARM position and engine torque exceeds 425 foot-pounds, power is applied from a one ampere circuit breaker on the right subpanel through the control switch and normally open contacts of the pressure switch to illuminate the green ARM lights adjacent to the control switch on the left subpanel. Should engine torque drop below 350 to 400 foot-pounds, the pressure switch closes to provide power to the coil of the ignition power relay. The relay closes to provide power from a 20 ampere circuit breaker to the glow plugs or high tension ignitors and the IGNITION ON light in the annunciator panel.

2. AUTO-IGNITION PRESSURE SWITCH

A. Removal

This switch is mounted adjacent to the torque pressure transmitter just forward and above the left hand exhaust outlet on each engine.

NOTE: On airplanes equipped with both auto-ignition and autofeathering, a single double-pole, double-throw switch functions both as the auto-ignition pressure switch and the high pressure switch for the auto-feathering system. In the latter instance, the low pressure switch for the autofeathering system is mounted just below and to the left of the auto-ignition pressure switch.

(1) With the battery and generator switches OFF and the external power source disconnected, remove the upper forward cowling.

(2) Unsafety and remove the electrical connector from the switch.

(3) Unsafety the mounting nut at the base of the switch and remove the switch.

(4) Install a protective plug in the switch mounting hole to prevent contamination.

B. Installation

(1) Remove the plug from the switch mounting hole and screw the switch into the hole. Tighten the switch mounting nut to seat the packing seal and safety wire the nut.

(2) Connect the electrical plug. The plug is to be tightened finger tight and then safety wired.

(3) Install the upper forward cowling.

3. AUTO-IGNITION

A. Check

(1) Set the power levers at IDLE and place the auto-ignition switches in the ARM position. The IGNITION ON light in the annunciator panel should illuminate.

(2) Advance the power lever to obtain a torque exceeding 425 foot-pounds. The auto-ignition ARM lights in the subpanel should illuminate and the annunciator lights should go out. Turn the auto-ignition switches off and shut down the engine.

CAUTION: *Make sure the auto-ignition system is not armed except for a check of the system and just prior to take-off. If the system is left armed and the battery switch is turned on, the glow plugs or ignitors may be damaged by overheating.*

NOTE: For detailed maintenance procedures and checks of the engine ignition system, refer to the appropriate Pratt-Whitney Engine Maintenance Manual.

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CHAPTER 76 - ENGINE CONTROLS

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ENGINE CONTROLS
GENERAL
DESCRIPTION AND OPERATION

1. GENERAL

WARNING: Do not lubricate engine control cables under any circumstances. These cables are prelubricated by the manufacturer in accordance with their specifications. The manufacturer recommends that NO further lubrication be applied.

Engine and propeller functions are controlled by conventional levers and push-pull cables. The PT6A-20, PT6A-27, PT6A-28 and PT6A-36 engines are controlled by three levers: The power lever which controls engine power and propeller reversing mechanism, the propeller control lever and the condition (idle and fuel cutoff) lever. Refer to the engine maintenance manual for complete description and maintenance information on the engine and accessories supplied by Pratt & Whitney Aircraft of Canada Ltd.

CAUTION: *Any time an engine torque indicator or an engine torque transmitter is replaced, calibrate the new instruments as indicated in Chapter 77.*

ENGINE CONTROLS
POWER CONTROLS
MAINTENANCE PRACTICES

1. ENGINE CONTROLS

CAUTION: *All figures and dimensions noted in these rigging procedures are approximations and may vary with the individual engine. One engine should be rigged as though it were the master engine and the other engine rigged to match.*

When rigging the engine, each end of the engine is rigged separately then coupled with the interconnect cable.

2. CONTROL CABLE

A. Purging

The engine control cables are prelubricated by the manufacturer with a special lubricant compounded for effective lubrication in the low temperatures encountered during flight at high altitudes. For this reason, any oil, grease, solvent, etc., used to lubricate the sealed cables will only hinder the cable operation. The cables may be periodically purged with DRY NITROGEN to remove moisture and prevent subsequent freezing by the following Steps.

(1) Remove the rubber seal at each end of the control cable.

(2) Connect a DRY NITROGEN source to the end of the control cable that attaches to the engine.

(3) Purge the control cable using 18 psi DRY NITROGEN, for 15 minutes. Observe the opposite end of the control cable to ensure that all moisture has been purged.

NOTE: If the control cable has been contaminated with a lubricant that congeals at a low temperature, the nitrogen purge is ineffective and the affected cable must be replaced.

3. IDLE ADJUSTMENTS (PT6A-20)

A. Ground Idle Adjustment

Ground idle speed adjustment of the N_1 governor can be performed satisfactorily only at pressure altitude (from sea level to approximately 1500 ft.) at which the fuel flow required to produce 50/53% N_1 is greater than the minimum fuel flow setting of the fuel control unit. At higher altitudes, the minimum fuel flow will produce N_1 speeds above the N_1 governor setting. The N_1 governor cannot reduce fuel flow below the minimum flow setting and N_1 speed will increase in proportion to altitude when operating above approximately 1500 ft.

CAUTION: *Do not perform checks at barometric pressures below 28.0 in Hg. or altitudes in excess of 1500 feet. Checks must be performed with zero power extraction.*

(1) Move power lever to ground idle position. Allow gas generator speed to stabilize. If idle speed is too fast, move idle lever through the ground idle detent toward fuel cutoff. This will assure that the idle speed is being controlled by the idle adjusting screw and not by the position of the power or idle lever.

(2) Remove safety wire from idle adjusting screw and loosen lock nut.

(3) Turn idle adjusting screw clockwise to increase ground idle speed or counterclockwise to decrease ground idle speed.

CAUTION: *If interturbine temperature or turbine inlet temperature rises abnormally at any time during idle speed check, shut down the engine immediately.*

NOTE: The idle speed adjustment is very sensitive and should not be moved more than 1/8 of a turn between idle speed checks. Adjust ground idle speed to 50/53% N_1.

(4) Tighten the locknut and safety the idle adjusting screw after the adjustment has been completed.

(5) Move the power lever through the idle gate and then back against the pedestal ground idle stop. If idle speed increases from previous setting, adjust the pedestal ground idle stop or the power lever rigging to obtain the idle speed obtained in Step (3).

B. High Idle Adjustment

Place the pedestal lever in HIGH-IDLE position. The lever on the engine must contact the stop bolt. Adjust the bolt as necessary to obtain 70% N_1 (Ref. Figure 201).

C. Fuel Cutoff Lever Adjustment

(1) Back off the overtravel nut several rounds (Ref. Figure 201).

(2) Move pedestal lever to fuel cutoff position. The fuel cutoff linkage must contact the fuel cutoff stop bolt with some springback in the pedestal lever. If rigging is out of adjustment, adjust the power control cable rigging. This adjustment is made with the fuel cutoff stop bolt.

(3) With the pedestal lever in the fuel cutoff position, adjust the cutoff valve by placing the power lever against its minimum stop and check the clearance between the nut and spacer as indicated in Figure 201. Reposition the nut as necessary to obtain a clearance of 0.005 to 0.010 inch.

(4) If a new engine or fuel control unit has been installed, check for positive cutoff as follows before operating the engine:

(a) Disconnect the fuel line at the rear of the fuel control unit. A suitable length of hose or tubing may be routed overboard from the fuel control unit to prevent fuel from being pumped into the cowling.

(b) Motor the engine with the starter.

(c) Advance the power lever until fuel flows from the fuel control unit. Allow the fuel to flow until the system is purged of vapor.

(d) Move the power lever to the fuel cutoff position. Fuel should immediately cease flowing from the fuel control unit.

(e) Reconnect the fuel line.

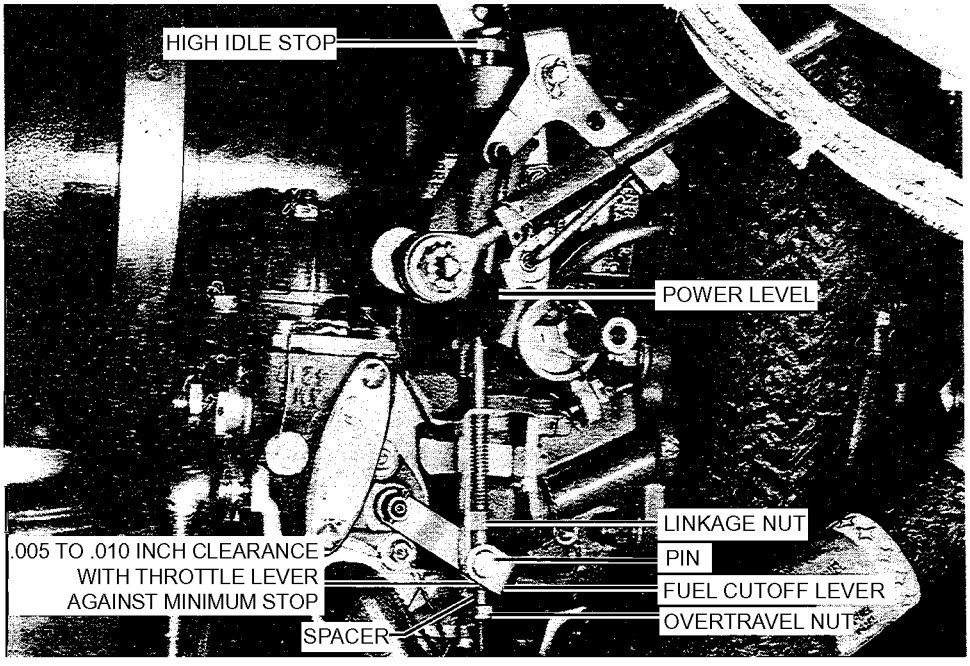


Figure 201
Idle Adjustment (PT6A-20)

4. ENGINE AND PROPELLER REVERSING MECHANISM (PT6A-20 ENGINES)

A. Rigging

NOTE: If a new interconnect cable is to be installed secure the cable to the front lifting attachment and to the center and rear mount rings with the cable couplings and nuts. Apply a torque of 80 to 100 inch-pounds to the attaching nuts at the front lifting attachment and rear mount ring. Torque the attaching nut at the center mount ring 60 to 80 inch-pounds, BUT DO NOT CUT THE INTERCONNECT CABLE TO LENGTH AT THIS TIME.

(1) Set propeller control in HI RPM position, then verify that the speed control arm is against the maximum speed adjusting screw on top of the governor.

NOTE: If the interconnect cable is not centered in this housing, loosen the clamp bolts on the push-pull control terminal (4) and push-pull control cable (27), center the interconnect cable and retighten the clamp bolts before proceeding to the next Step (Ref. Figure 202).

(2) Disconnect the front fork end (2) and adjustable fork end (16) of the interconnect cable and adjust the low pitch stop adjuster (3) jam nut until the travel at the forward end of the cable is from 1.1 to 1.2 inches.

(3) Connect the adjustable fork end (16) at the top hole in the propeller control cam (29). Adjust the front fork end (2) then retighten the front fork end (2) jam nuts and disconnect the adjustable fork end (16) from the propeller control cam (29).

NOTE: There should be at least four turns of thread engagement between each fork end and the interconnect cable.

(4) Wire the lever (18) against the maximum N_1 stop (20) and remove the control rod (17).

(5) Position the 50-590158-25 template and insert a 3/16 x 3/4 inch bolt through hole number 1 and the hole in the end of the shaft for the FCU control arm (Ref. Figures 203 and 204). The slot in the template slips behind the power control lever and the unmarked hole in the tool fits over the pin.

(6) Position the cam pin in the bottom of the cam slot (Ref. Figure 205).

(7) Align the power control lever with hole number 2 and insert a 1/4 inch bolt through the power control lever and hole number 2 in the template.

NOTE: If the power control lever is splined in such a manner that it cannot be aligned with hole number 2, install the power control lever one spline clockwise beyond hole number 2.

Securely tighten the clamp bolt in the power control lever and check that the cam pin is still bottomed out in the cam slot.

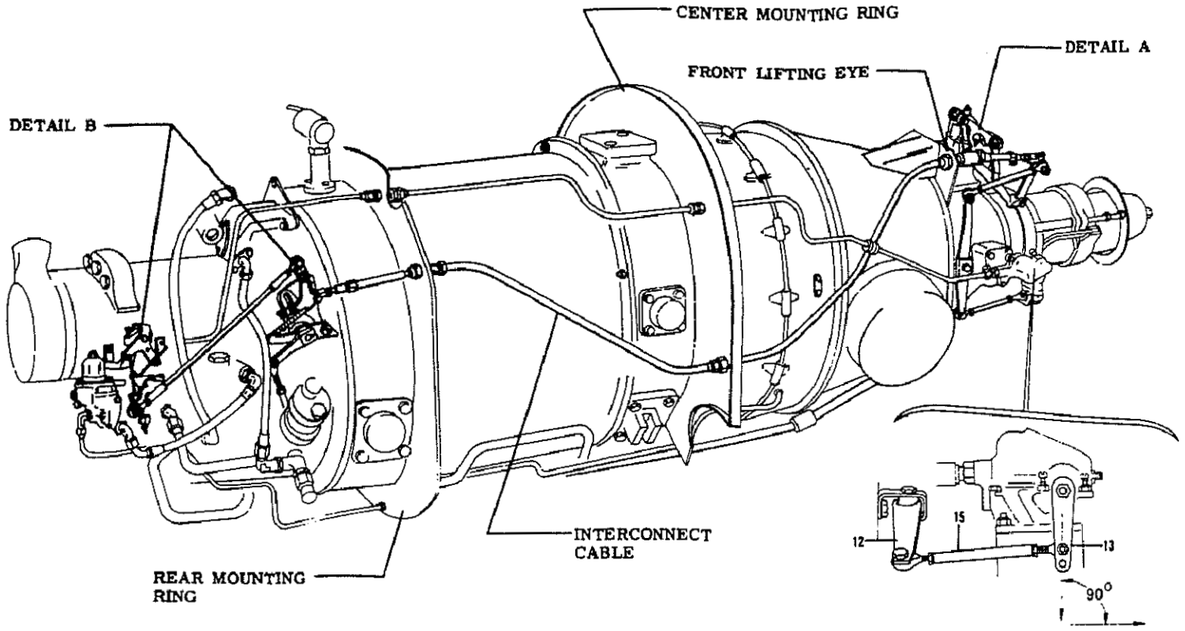
NOTE: When the rigging template was designed, hole number 3 was incorporated for a crosscheck of power control lever and cam pin position that was later found unnecessary.

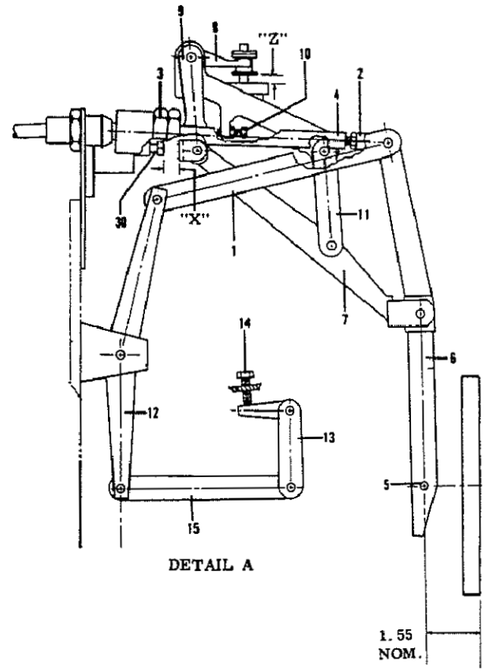
(8) Align the power control lever with hole number 6 and insert a 1/4 inch bolt through the power control lever and hole (Ref. Figure 207). The cam pin should be centered on the cam track point (Ref. Figure 205). If necessary, reposition the cam pin as required to center it on the track point, but be sure to maintain the angle of the power control lever at hole number 6 as set by the rigging template.

(9) Align the FCU control arm (21) with template hole number 4 by adjusting the vernier nut (Ref. Figure 206), then tighten the vernier nut to lock the FCU control arm in place.

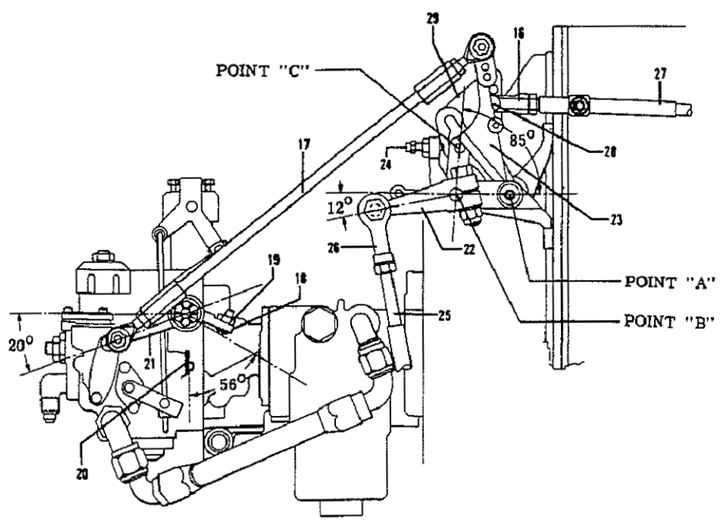
NOTE: Estimate the number of degrees between the end of the FCU control arm (21) and hole number 4 in the template. Rotate the serrated spacer and FCU control arm (21) toward the hole for the number of serrations (each serration changes the position of the FCU control arm (21) by 1/2°) required to make up the estimated degrees, then rotate the FCU control arm (21) alone in the opposite direction for the same number of serrations as both the spacer and FCU control arm (21) were originally moved. If necessary, make further minor adjustments in the same manner until the FCU control arm (21) is aligned with hole number 4 in the template.

(10) Remove the wire securing the lever (18) against the maximum N_1 stop (20). Align the FCU control arm (21) with template hole number 5 and install a 3/16 inch bolt to secure the arm in place.





DETAIL A



DETAIL B

99-247-13

1. Propeller Reversing Interconnecting Rod
2. Front Fork End
3. Low Pitch Stop Adjuster
4. Push-Pull Control Terminal
5. Carbon Blocks
6. Propeller Reversing Lever
7. Propeller Governor Actuating Lever
8. Governor Lever
9. Governor Arm
10. Stop
11. Actuating Lever Support Arm
12. Power Turbine Governor Lever
13. Power Turbine Governor Arm
14. Power Turbine Governor Maximum Stop
15. Interconnecting Rod
16. Adjustable Fork End
17. Control Rod
18. Lever
19. N_1 Speed Adjustment Control
20. Maximum N_1 Stop
21. FCU Control Arm
22. Power Control Lever
23. FCU Actuating Lever
24. Reverse Adjust Screw
25. Power Control Cable
26. Adjustable Rod End
27. Push Pull Control Cable
28. Propeller Control Cam Pin
29. Propeller Control Cam
30. Adjustable Screw

Figure 202
Rigging Propeller Reversing Mechanism (PT6A-20)

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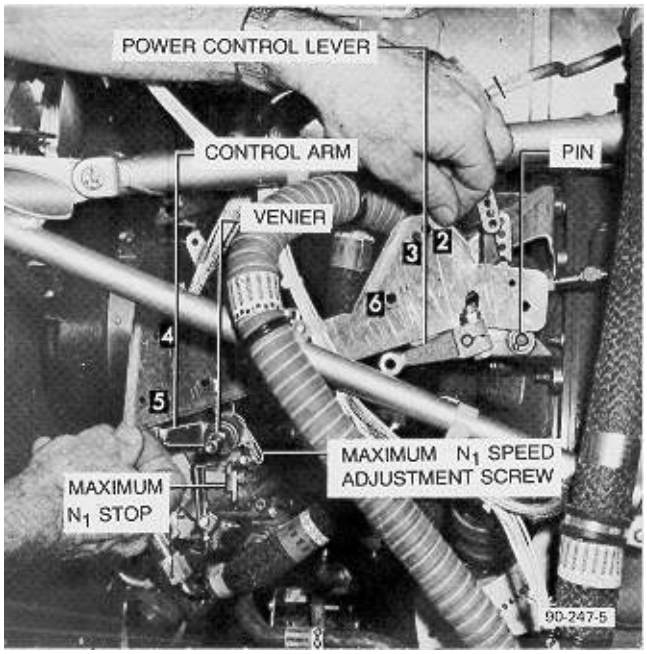


Figure 203
Positioning the Rigging Template

Figure 204
Installing the Rigging Template

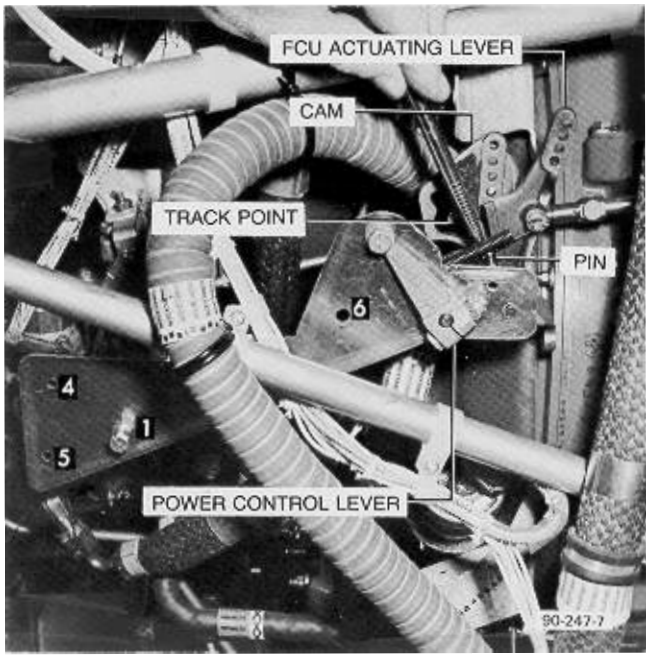


Figure 205
Positioning Power Control Lever and Cam Pin

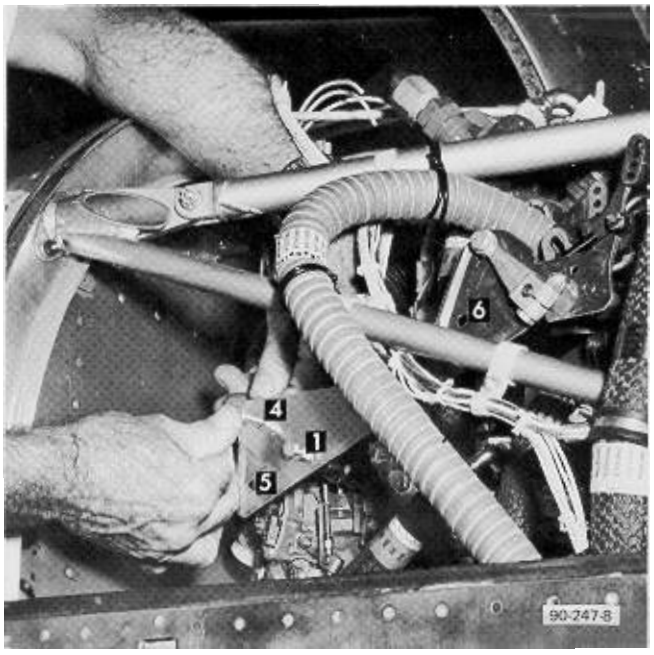


Figure 206
Aligning FCU Control Arm

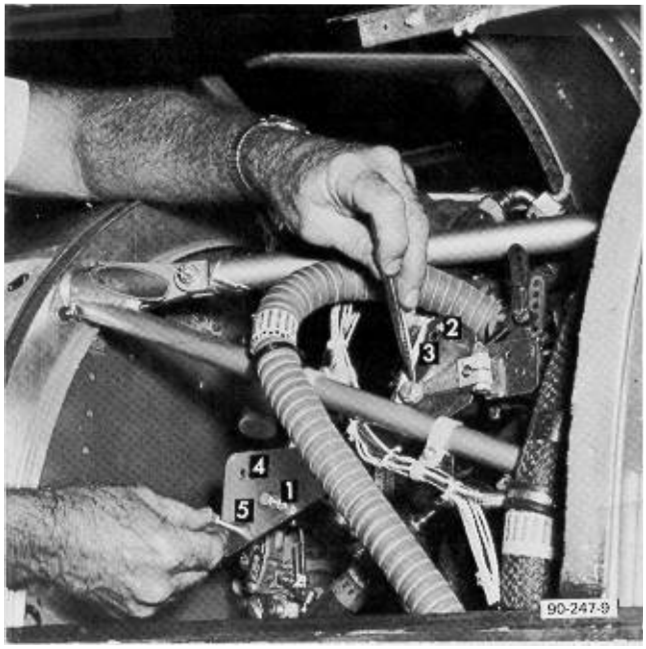


Figure 207
Rigging FCU Arm and Power Lever

(11) Adjust the control rod (17) until the distance between the center of the hole on each end of the control rod (17) is 11 3/16 inches. Connect the control rod (17) to the top of the FCU actuating lever (23). Place the free end of the control rod (17) between the template at hole number 5 and the FCU control arm (21), then insert a 3/16 inch bolt through the template, control rod (17), and inboard hole of the FCU control arm (21). Check that the distance between the control rod (17) and stop is approximately the same on each engine.

(12) Align the power control lever (22) with hole number 6 in the template. With the power control lever (22) held in this position, connect it to the adjustable rod end (26) of the power control cable (25).

NOTE: To rig all lost motion of the power control cable (25) in one direction, move both power levers in the pedestal full forward, then to the idle position, before connecting the power control cable (25) to the power control lever (22).

(13) Remove the template and install the attaching bolts at the respective connection points for the engine controls as positioned by the rigging template.

(14) Place the power levers in the pedestal in the forward range and cut the interconnect cable to length.

(15) Turn the adjustable fork end (16) until it is 1/2 turn longer than perfect alignment with the top hole in the propeller control cam (29), then install the attaching propeller control cam pin (28) from the outboard side through the adjustable fork end (16) and the top hole in the propeller control cam (29).

(16) Attach the carbon block (5) to the propeller reversing lever (6), and check that the clearance between the carbon block (5) and the low pitch stop collar (brass ring) is 0.001 to 0.002 inch. If necessary, the new carbon block (5) may be lightly sanded to obtain this clearance.

CAUTION: *With the carbon block (5) held against the low pitch stop collar, check the clearance between the low pitch stop collar and the metal retaining clip of the carbon block (5). If at any point this clearance is 0.005 inch or less, replace the carbon block (5) to prevent damage to the low pitch stop collar by the carbon block retaining clip.*

(17) Secure the propeller reversing lever (6) to the propeller governor actuating lever (7) and to the front fork end (2).

(18) With the governor lever (8) held in position against the upper flange of the governor spool rod, hold the governor arm (9) vertically, or slightly forward of vertical, while attaching it to the governor.

(19) Adjust the stop (10) to obtain a Z dimension of 0.200 to 0.210 inch on the governor shaft when the governor arm (9) is in contact with the stop (10). The proper dimension may be ascertained with the governor spool feeler gage (P/N 50-590158-5).

CAUTION: *Damage to the governor will result if the governor arm (9) is forced into a position where dimension Z is greater than 0.230 inch.*

(20) Connect the governor arm (9) to the governor actuating lever (7) and install the actuating lever support arm (11). Do not exceed 25 inch-pounds of torque when installing the attaching nuts.

(21) Adjust the front fork end (2), until dimension Z is 0.084 $\pm$ 0.002 inch with the push-pull control terminal (4) pulled forward against the low pitch stop adjuster (3) and the carbon block (5) against the aft flange of the low pitch stop collar (brass ring). Check for the proper Z dimension with the governor spool feeler gage (P/N 50-590158-5) (Ref. Figure 208). If the proper Z dimension cannot be obtained with the fork end adjustment, adjust the low pitch stop adjuster (3) to obtain a Z dimension of 0.084 $\pm$ 0.002 inch. The preload on the adjustable fork end (16) must then be rechecked.

NOTE: Travel of the push-pull control terminal (4) must not exceed one inch when the 0.084 dimension is set. If the proper Z dimension cannot be obtained by adjusting the front fork end (2) and low pitch stop adjuster (3), reposition the governor arm (9) on the spline of the shaft in a vertical or slightly forward of vertical position. If necessary, a finer adjustment can be made by removing the nut from the end of the governor shaft (Ref. Figure 209), loosen the screw in the top of the governor lever, and rotate the shaft (a half turn of the shaft results in one serration of change).

(22) Connect the propeller reversing interconnect rod (1) to the front fork end (2) and propeller reversing lever (6), taking care not to lose the spacer on the propeller reversing lever (6). Adjust the propeller reversing interconnecting rod (1) until the lower portion of the power turbine governor lever (12) is parallel to the front flange of the engine and the power turbine governor arm (13) is resting lightly on the power turbine governor maximum stop (14).

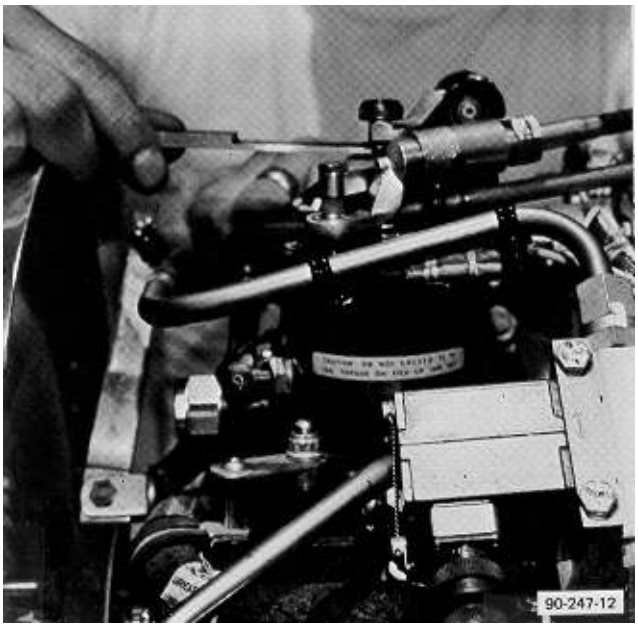


Figure 208
Checking Z Dimension

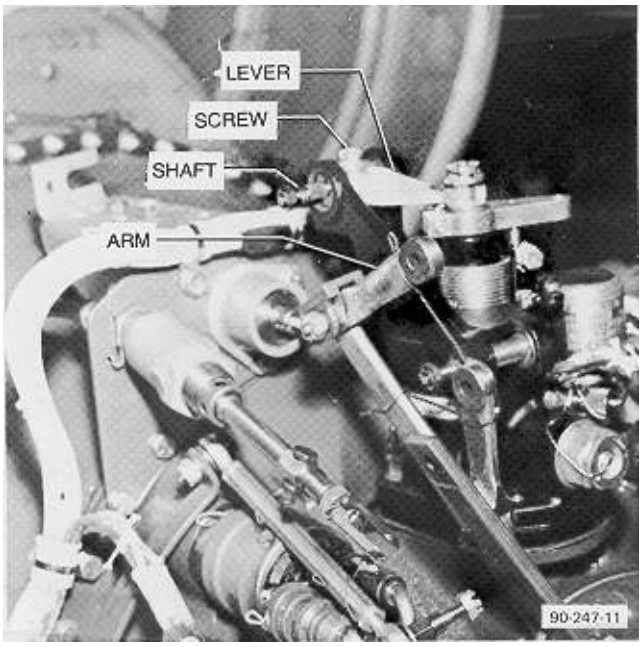


Figure 209
Checking Z Dimension

NOTE: If the power turbine governor arm (13) does not touch the power turbine governor maximum stop (10), the power turbine governor may surge or cause a loss of power on takeoff. If the power turbine governor arm (13) exerts too much pressure against the power turbine governor maximum stop (10), the Z dimension may be changed and binding or rough power control movement may be experienced. The power turbine governor maximum stop (10) setting must be checked anytime the Z dimension is changed.

Check under the governor spool for a Z dimension of 0.084 ± 0.002 inch. If necessary, readjust the front fork end (2) to obtain this dimension.

(23) Apply a torque of 20 to 30 inch-pounds to all nuts except those securing the governor arm (9) and actuating lever support arm (11), then lock the nuts in place with cotter pins and safety wire.

CAUTION: *Do not apply a torque greater than 25 inch-pounds to the retaining nuts for the governor arm (9) and actuating lever support arm (11).*

(24) Move the pilot's lever through its forward range to check for smooth operation. An excessive load during forward movement indicates that the adjustable fork end (16) was preloaded too much. If the adjustable fork end (16) was not preloaded enough, the propeller blade angle will be too low when the engine is test run.

(25) With the power control lever (22) forward of the idle position, set dimension X at 0.010 with the adjustable screw (30), then back the screw off enough to align the holes for the cotter pin.

NOTE: To avoid increasing dimension X too much, do not back off the screw more than a half turn.

(26) Pull the low pitch stop collar (brass ring) forward 0.380 inch from its rearmost position with the feedback ring puller (P/N TK-1573). Monitor the clearance between the governor arm (9) and stop (10) while operating the feedback ring puller. To avoid damaging the linkage while setting the 0.380 inch gap on the low pitch stop rod, disconnect the front fork end (2) from the propeller reversing lever (6).

NOTE: Check the 0.380 dimension with a feeler gage at the base of the spring on the low pitch stop (Ref. Figure 210).

With the low pitch stop collar (brass ring) in this position, check that the PROP LOW PITCH light (white) in the annunciator panel illuminates. (Sensors must be connected to amplifier and hooked into the airplane electrical system before check is possible).

CAUTION: *Damage to the solid state switch will result if the two wires running between it and the sensor are shorted.*

If necessary, adjust each sensor on its mounting bracket until the PROP LOW PITCH light (white) in the annunciator panel illuminates.

NOTE: To protect the low pitch stop collar (brass ring) against scoring, the clearance between the bottom of the collar and the rivet head on the brush block should be from 0.030 to 0.120 inch. Bend the arm assembly as necessary to obtain this clearance. Check for a clearance of 0.060 to 0.090 inch between the sensor and the end of the arm assembly opposite that on which the brush block is installed. Add AN960-10L washers to obtain this clearance (Ref. Figure 211).

Check the system for proper operation by performing the ENGINE RUN-UP CHECKS.

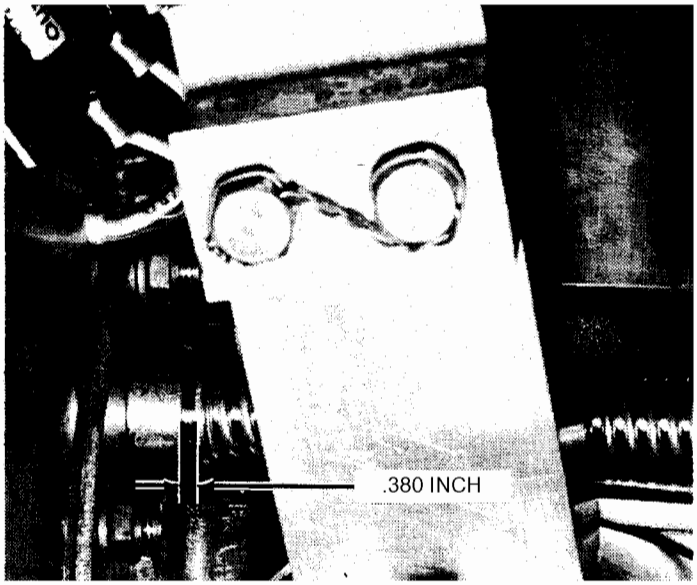


Figure 210
Checking Movement of Low Pitch Collar (Brass Ring)

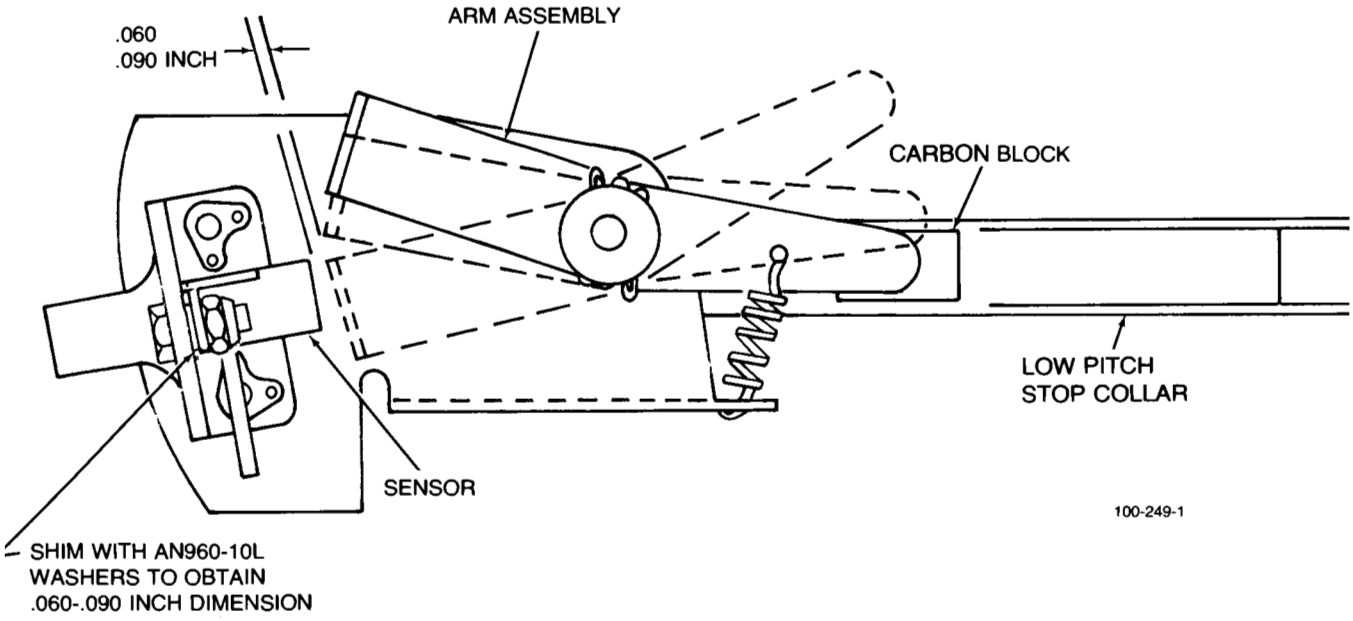


Figure 211
Checking Movement of Low Pitch Collar (Brass Ring)

5. ENGINE RUN-UP (PT6A-20)

A. Check

(1) With the propeller controls full forward, run up the RH engine and increase the power lever to obtain 2000 propeller rpm. The torque meter reading should be within 60 foot-pounds of the value designated by the Pressure Altitude and Temperature Conversion Chart (Ref. Figure 214) for the ambient conditions under which the engine is operating. Repeat this check on the LH engine. The difference between the torque meter readings on the RH and LH engines should not exceed 20 foot-pounds.

NOTE: If this check is not being performed under zero wind conditions, take an average of the upwind and downwind readings to obtain valid results.

(2) If the readings just taken fail to fall within the range prescribed in the preceding step, check that the Z dimension (Ref. Figure 202) on the governor speed control lever is 0.084 ± 0.002 inch.

If necessary, the Z dimension itself may be adjusted at the front fork end (2). Screw the front fork end (2) in to decrease the Z dimension and the resultant torque, or out to increase the Z dimension and resultant torque.

NOTE: The front fork end (2) should not be adjusted more than 1/2 turn at a time. Observe the Z dimension tolerances of 0.084 ± 0.002 inch.

If the torque prescribed by the chart cannot be obtained by adjusting the front fork end (2), a further adjustment may be made at the low pitch stop adjuster (3) of the interconnect cable.

CAUTION: *The Z dimension must not be changed from 0.084 inch by more than 0.002 inch. If the low pitch stop adjuster (3) is adjusted, the aft control linkage must be rigged and preload on the interconnect cable must be reset.*

(3) Position the power levers in the pedestal in the beta range and check for a simultaneous pickup of N_2 on each engine without the addition of power. If the pickup is not simultaneous, adjust the cam pin forward on the engine that picked up first until the engines pick up N_2 simultaneously.

NOTE: This may be accomplished by loosening the power control lever (22) on the shaft and sliding the cam pin forward, then retightening the power control lever (22) on the shaft. DO NOT CHANGE THE ANGLE OF THE POWER CONTROL LEVER (22) WHEN REPOSITIONING THE CAM PIN. Make sure that the cam pin on the engine that picked up last is back to the apex of the cam track before repositioning the cam pin on the engine that picked up first.

(4) Advance the power levers in the pedestal evenly (within 1/4 inch) while checking for an even pickup of power on both engines throughout the usable power range. If the power pickup of the engines is not even, adjust the vernier of the FCU control arm (21) forward on the low engine until its pickup is even with that of the other engine throughout the normal operating range.

NOTE: The adjustment just described will result in a slightly slower initial pickup from idle but a faster pickup in the normal operating range. It should also be noted that this adjustment will necessitate readjusting the control rod (17) to maintain the correct position of the FCU actuating lever (23).

(5) With the propellers fully reversed, set the reverse adjust screw (24) to obtain 85 ± 3% N_1. Adjust the screw in to increase or out to decrease gas generator (N_1) speed. The reverse adjust screw will normally change the (N_1) speed by 3% for each full turn of the screw. Make this check with the airplane in both the upwind and downwind positions to ascertain that even torque is obtained from each engine with the propellers fully reversed.

NOTE: The power levers in the pedestal will not necessarily be even at the point where the engines pick up reverse power.

6. FLIGHT IDLE TORQUE (PT6A-20)

A. Check

(1) Set the altimeter at 29.92 in. Hg. and read the pressure altitude. Take reading (in degrees Centigrade) from the aircraft OAT indicator.

NOTE: Make certain that the temperature reading has stabilized on airplanes recently removed from a hangar.

(2) At the top of the graph entitled Pressure Altitude and Temperature Conversion Chart, (Ref. Figure 214) locate the pressure altitude curve corresponding to the reading taken in the preceding Step. Trace this curve down to the point where it intersects the OAT reading (in degrees Centigrade) indicated on the LH side of the chart. A line dropped perpendicular from this point to the bottom of the graph will indicate the torque value in foot-pounds at which the engine should operate under the existing conditions.

(3) With the governor controls forward, run up the RH engine and increase the throttle to obtain 2000 propeller rpm. The torque meter reading should be within 60 foot-pounds of the value obtained in the preceding Step. Repeat this check on the LH engine. The difference between the torque meter readings on the RH and LH engines should not exceed 20 foot-pounds.

NOTE: If this check is not being performed under zero wind conditions, take an average of the upwind and downwind readings to obtain valid results.

(4) If the readings just taken fail to fall within the range prescribed in the preceding Step, check that the Z dimension (Ref. Figure 202) on the governor speed control lever is 0.084 ± 0.002 inch. If necessary the Z dimension itself may be adjusted at the front fork end (2). Screw the front fork end (2) in to decrease the Z dimension and the resultant torque or out to increase the Z dimension and resultant torque.

NOTE: The front fork end (2) should not be adjusted more than 1/2 turn at a tim. Observe the Z dimension tolerances of 0.084 ± 0.002 inch.

If the torque prescribed by the chart cannot be obtained by adjusting the front fork end (2), a further adjustment may be made at the low pitch stop adjuster (3) of the interconnect cable.

CAUTION: *The Z dimension must not be changed from 0.084 inch by more than 0.002 inch. If the low pitch stop adjuster (3) is adjusted, the aft control linkage must be rigged and preload on the interconnect cable must be reset.*

7. SECONDARY LOW PITCH STOP (PT6A-20)

A. Check

(1) Run the engine with the power lever at the idle detent and set the condition lever to 70% gas generator (N_1) speed, then read propeller rpm.

(2) Hold the momentary test switch on the left center sub-panel in the SEC LOW PITCH STOP position. Lift the power lever and pull it aft until it is over the top horizontal line marking on the pedestal. After the initial deflection of its pointer in the increase rpm direction, the propeller should indicate a maximum fluctuation of 50 rpm. The midpoint of the fluctuation range should be 170 to 250 rpm higher than the reading noted in Step (1) and the PROP LOW PITCH light (white) in the annunciator panel should be illuminated.

NOTE: If propeller rpm does not increase within the prescribed limits, readjust the position of the sensor on its mounting bracket (Ref. Figure 211). Move the sensor forward to decrease rpm or aft to increase rpm.

(3) Release the momentary test switch and rpm should increase. Return the power lever to its normal idle position.

8. IDLE ADJUSTMENTS (PT6A-27 AND PT6A-28)

A. Ground Idle Adjustment

Ground idle speed adjustment of the N_1 governor can be performed satisfactorily only at pressure altitudes (from sea level to approximately 3500 ft.) at which the fuel flow required to produce 50/53% N_1 is greater than the minimum fuel flow setting of the fuel control unit. At higher altitudes, the minimum fuel flow will produce N_1 speeds above the N_1 governor setting. The N_1 governor cannot reduce fuel flow below the minimum flow setting and N_1 speed will increase in proportion to altitude when operating above approximately 3500 ft.

CAUTION: *Do not perform checks at barometric pressure below 25.20 in. Hg. or altitudes in excess of 3500 feet. Checks must be performed with zero power extraction.*

(1) Move power lever to ground idle position. Allow gas generator speed to stabilize. If idle speed is too fast, move idle lever through the ground idle detent toward fuel cutoff. This will assure that the idle speed is being controlled by the idle adjusting screw and not by the position of the power or idle lever.

(2) Remove safety wire from idle adjusting screw and loosen lock nut.

(3) Turn idle adjusting screw clockwise to increase ground idle speed or counterclockwise to decrease ground idle speed.

NOTE: The idle speed adjustment is very sensitive and should not be moved more than 1/8 of a turn between idle speed checks. Adjusting ground idle speed to 50/53% N_1.

(4) Tighten the locknut and safety the idle adjusting screw after the adjustment has been completed.

(5) Move the power lever through the idle gate and then back against the pedestal ground idle stop. If idle speed increases from previous setting, adjust the pedestal ground idle stop or the power lever rigging to obtain the idle speed obtained in Step (3).

B. High Idle Adjustment

Set the condition lever at high idle and the power lever at idle, then check the tachometer for a reading of 70% gas generator (N_1) speed (Ref. Figure 212). If the gas generator speed is not at 70%, check for a clearance of 0.030 inch between the idle control rod and its adjustable end. Adjust the idle control rod at its lower end until the proper gas generator speed is obtained. An increase in clearance between the rod and its lower adjustable end will lower the gas generator speed while a decrease in the clearance will increase gas generator speed. If the proper gas generator speed cannot be obtained by the foregoing means, reset a clearance of 0.030 inch between the idle control rod and its adjustable end, then turn the high idle adjusting screw on the starting control unit until a gas generator speed of 70% is obtained. Turn the screw counterclockwise to increase or clockwise to decrease gas generator speed.

C. Fuel Cutoff Adjustment

(1) Disconnect the primary fuel delivery tube from the front side of the starting control unit (Ref. Figure 212). A suitable length of hose or tubing may be routed overboard from the starting control unit to prevent fuel from being pumped into the cowling.

(2) Place one man in the cockpit to motor the engine and another at the nose to observe fuel flow.

(3) With the electrical boost pumps operating and the ignition off, motor the engine while moving the cockpit condition lever from the CUTOFF position, then slowly back to CUTOFF.

CAUTION: *Do not exceed the starter time limits of 40 seconds ON, 60 seconds OFF, 40 seconds ON, 60 seconds OFF, 40 seconds ON, then 30 minutes OFF.*

Fuel should flow from the hose, then stop flowing when the condition lever reaches CUTOFF.

(4) If the fuel continued to flow after the condition lever was returned to CUTOFF, check the low and high idle adjustment.

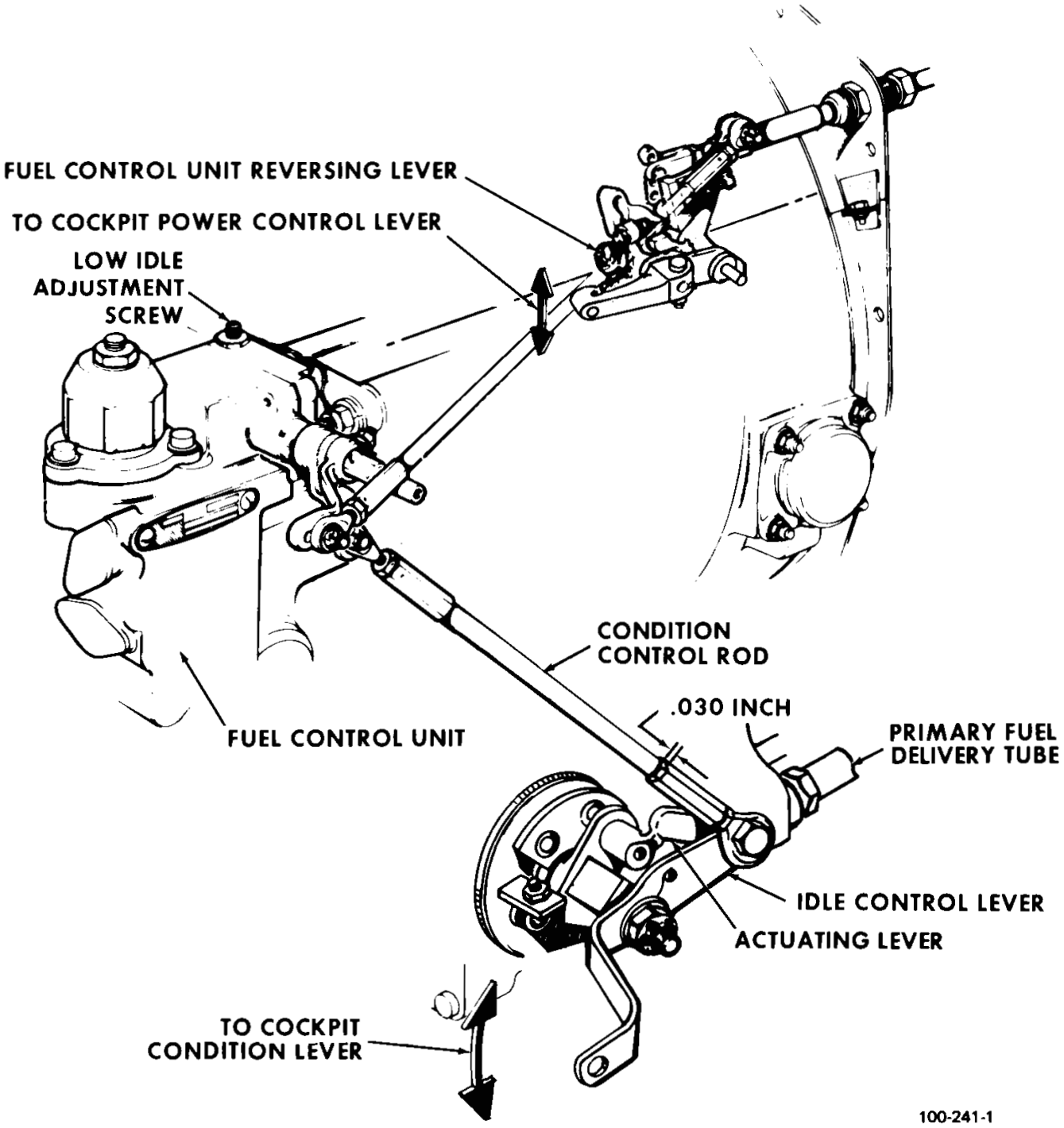


Figure 212
Idle Adjustments (PT6A-27 and PT6A-28)

9. ENGINE AND PROPELLER REVERSING MECHANISM (PT6A-27 AND PT6A-28 ENGINES)

A. Rigging

CAUTION: *If the cockpit controls are moved into the reversing position after the engine controls are hooked up, damage to the reversing linkage will result.*

(1) Check the propeller speed select lever (1) of the governor for full travel between the maximum and minimum stops without interference with the control rod end (Ref. Figure 213). If full travel is impossible, loosen the lever retaining bolt, lift the lever free, and reinstall it on the governor sleeve in a position 3 serrations counterclockwise from its original position.

(2) Attach the carbon block (2) to the propeller reversing lever (3) and check that the clearance between the carbon block (2) and the low pitch stop collar (4) is 0.001 to 0.002 inch at the narrowest portion of the low pitch stop collar (4).

CAUTION: *With the carbon block (2) held against the low pitch stop collar (4), check the clearance between the low pitch stop collar (4) and the metal retaining clip of the carbon block (2). If at any point this clearance is 0.005 inch or less, replace the carbon block (2) to prevent damage to the low pitch stop collar (4) by the carbon block retaining clip.*

(3) With the cockpit power control lever in the idle position (at the detent), install the engine power lever (5) on the grooved pin of the cam assembly so that the hole in the end of the engine power lever (5) is in a straight line with respect to the grooved pin and the pin immediately forward of it when the cam follower pin (6) is directly above (90° with respect to) the grooved pin. Tighten the clamp bolt in the cockpit power control lever to a torque of 90 to 100 inch-pounds.

CAUTION: *If the clamping portions of the cockpit power control lever contact one another, replace the cockpit power control lever with a new one to ensure a proper grip on the throttle rod.*

(4) Adjust the FCU (fuel control unit) control rod (7) until the distance between the center of the holes on each end of the FCU control rod (7) is 11 ± 0.05 inches. Attach one end of the FCU control rod (7) to the top hole of the FCU actuating lever (8) and the other end of the FCU control rod (7) to the inner hole of the fuel control unit arm (9).

(5) With the cockpit power control lever fully forward, check that the distance between the center line of the fuel control arm extension nut (10) and the center line of the FCU control rod (7) is 1.65 ± 0.05 inches when measured perpendicular to the fuel control rod (7). If the measurement is not within these limits, loosen the fuel control arm extension nut (10) and adjust the serrated spacer (11) between the levers until the gap is within limits. Retighten the fuel control arm extension nut (10) and secure it with lock wire.

(6) With the cockpit condition lever set at low idle, place the idle control lever (12) of the engine in the low idle position. This may be accomplished by rotating the idle control lever (12) fully counterclockwise (the high idle position), then rotating it clockwise to the first detent (the low idle position) on engines with the ball detent type of control. On engines with a slotted detent plate (26), align the rigging pin hole in the actuating lever (25) with the slot in the detent plate (26). Insert a pin or wire (a straightened coat hanger will suffice) through the rigging pin hole and the slot in the detent plate to lock the idle control lever (12) in low idle.

(7) Adjust the rod end of the control cable for the cockpit condition lever to align with the hole in the aft arm of the idle control lever (12). Insert the retaining bolt through the small washer, idle control lever (12), control cable rod end, and large washer. Install the retaining nut on the bolt and secure it in place with the cotterpin.

(8) With the condition lever set at low idle and the cockpit power control lever at idle, adjust the condition control rod (13) to fit between the idle control lever (12) and the fuel control unit arm (9) with a clearance of 0.030 inch between the lower end of the condition control rod (13) and its adjustable rod end. This clearance will give a gas generator (N_1) speed of approximately 70% at high idle. Secure each end of the condition control rod (13) in place with the attaching bolts, washers, castellated nuts, and cotter pins. Remove the rig pin that was inserted through the actuating lever (25) and detent plate (26) in Step (6).

(9) With both the fuel governor interconnecting rod (14) and the rear clevis (15) of the push-pull control cable disconnected, adjust the low pitch stop adjusters (16A, 16B) until the clevis slot end (17) of the beta control valve (18) is flush with the front face of the cap nut (19). Adjust the fuel governor interconnecting rod (14) until the retaining bolt will align with the ball rod end at the lower end of the interconnecting rod, then shorten the rod one-half turn to preload the override stop (20). Secure the upper end of the fuel governor interconnecting rod (14) to the clamp (21) of the push-pull control terminal (22) with the two washers, castellated nut, and cotter pin. Insert the retaining bolt through the lower arm on the fuel control portion of the governor, the lower end of the fuel governor interconnecting rod (14), and washers, then secure the bolt in place with the castellated nut and cotter pin.

(10) Adjust the rear clevis (15) of the push-pull control cable until the rear clevis (15) aligns with the second hole from the top of the beta control cam (23). Insert the retaining pin through the rear clevis (15), beta control cam (23), and washer, then lock the pin in place with a cotter pin.

10. RUN-UP (PT6A-27 AND PT6A-28 ENGINES)

A. Checks

(1) With the propeller control levers fully forward, advance the power levers to obtain 2000 rpm (Ref. Figure 213). Check that the power levers are even; if not, adjust the FCU control rod (7) on each engine until the power levers are even. Shorten the FCU control rod (7) to increase or lengthen it to decrease engine speed. The torque meter reading should be within 40 foot-pounds of the value designated by the Pressure Altitude and Temperature Conversion Chart (Ref. Figure 214) for the ambient condition under which the engines are operating. If adjustment is necessary, proceed as follows:

(a) Disconnect the fuel governor interconnecting rod (14) and disconnect the rear clevis (15) from the beta control cam (23). Adjust the low pitch stop adjuster (16) in or out until engine torque is within the prescribed limits and the torque of each engine is the same. One full turn in of the low pitch stop adjuster (16) will reduce torque by approximately 92 foot-pounds and one full turn out will increase torque by the same amount. Reconnect the fuel governor interconnecting rod (14) and rear clevis (15) as indicated in Step (9) of the procedure under ENGINE AND PROPELLER REVERSING MECHANISM RIGGING (PT6A-27 AND PT6A-28 ENGINES).

(2) With the cockpit condition lever set at high idle and the cockpit power control lever at idle, check the tachometer for a reading of 70% gas generator (N_1) speed. If necessary, adjust the condition control rod (13) at the lower end, where a clearance of 0.030 inch was set between the condition

control rod (13) and its adjustable end, until the proper gas generator speed is obtained. An increase in clearance at this point will result in lower gas generator speed and a decrease in clearance will result in higher gas generator speed.

(3) With the propellers fully reversed, check the tachometer for a gas generator (N_1) speed of 87%. If adjustment is necessary, turn the reverse adjust screw (24) in to increase or out to decrease gas generator (N_1) speed.

11. FLIGHT IDLE (PT6A-27 AND PT6A-28)

A. Check

(1) Set the altimeter at 29.92 in. Hg. and read the pressure altitude. Take reading (in degrees Celsius) from the airplane OAT indicator.

NOTE: Make certain that the temperature reading has stabilized on airplanes recently removed from a hangar.

(2) At the top of the graph entitled Pressure Altitude and Temperature Conversion Chart (Ref. Figure 214), locate the pressure altitude curve corresponding to the reading taken in the preceding Step. Trace this curve down to the point where it intersects the OAT reading (in degrees Celsius) indicated on the LH side of the chart. A line dropped perpendicular from this point to the bottom of the graph will indicate the torque value in foot-pounds at which the engine should operate under the existing conditions.

(3) With the governor controls forward, run up the RH engine and increase the throttle to obtain 2000 propeller rpm. The torque meter reading should be within 40 foot-pounds of the value obtained in the preceding Step. Repeat this check with the LH engine. There should be no difference between the torque meter readings on the RH and LH engines.

NOTE: If this check is not being performed under zero wind conditions, take an average of the upwind and downwind readings to obtain valid results.

(4) If the readings just taken fail to fall within the range prescribed in the preceding Step, disconnect the fuel governor interconnecting rod (14) and disconnect the rear clevis (15) from the beta control cam (23) (Ref. Figure 213). Adjust the low pitch stop adjuster (16) until engine torque is within the prescribed limits. One full turn in of the low pitch stop adjuster (16) will reduce torque by approximately 92 foot-pounds and one full turn out will increase torque by the same amount. Reconnect the fuel governor interconnecting rod (14) and rear clevis (15) to the linkage for the reversing mechanism.

12. SECONDARY LOW PITCH STOP (PT6A-27 AND PT6A-28 ENGINES) (AIRPLANE SERIALS PRIOR TO U-164, EXCEPT U-50, UNLESS MODIFIED BY INSTALLATION OF KIT NO. 99-3026-1S)

A. Check

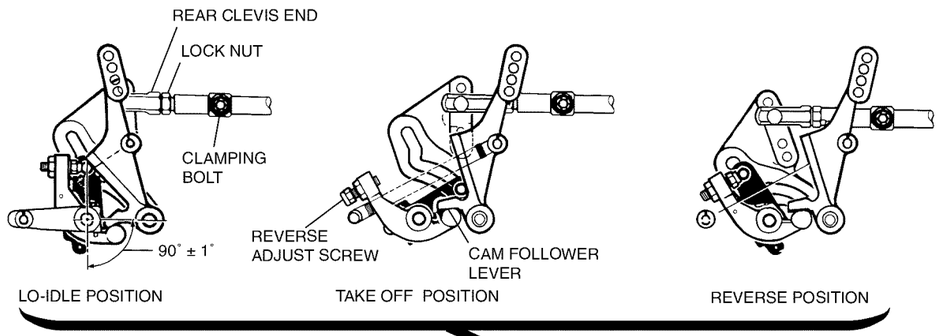
NOTE: The secondary low pitch stop systems have been removed from airplanes equipped with PT6A-27 or PT6A-28 engines after modification by Kit No. 99-3026-1S in compliance with S.I. 844-247.

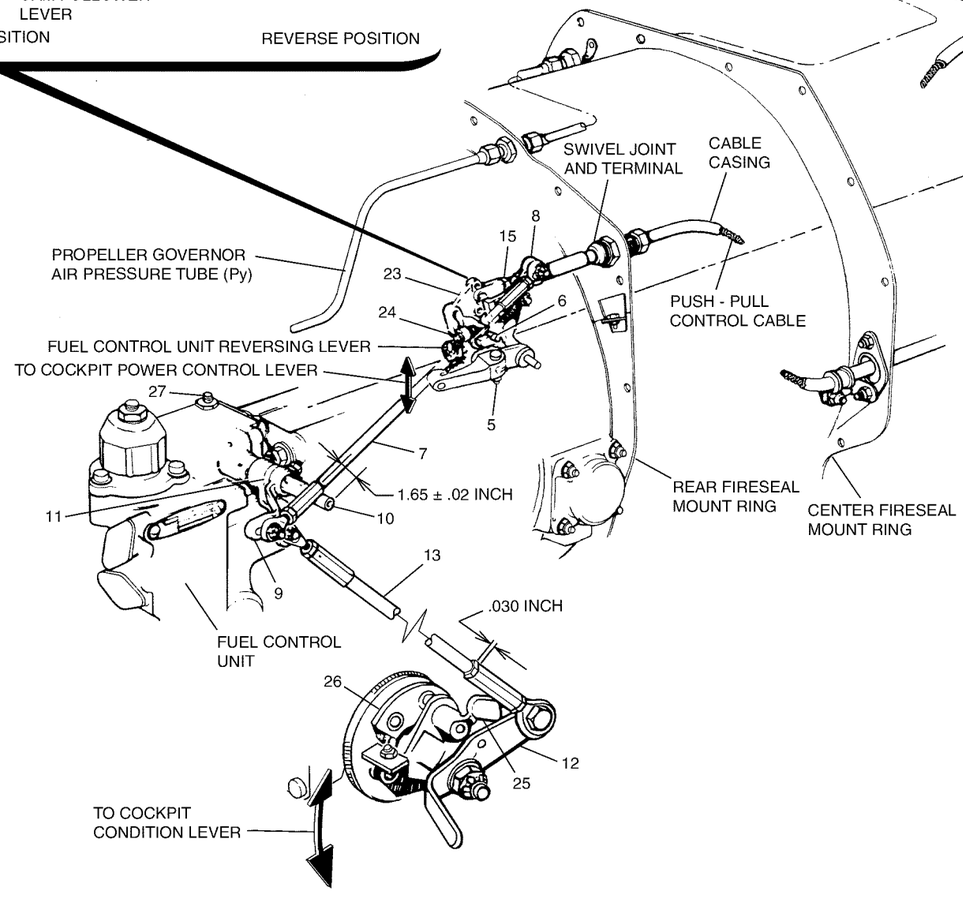
(1) Run the engine with the power lever at the idle detent and set the condition lever to 70% gas generator (N_1) speed, then read propeller rpm.

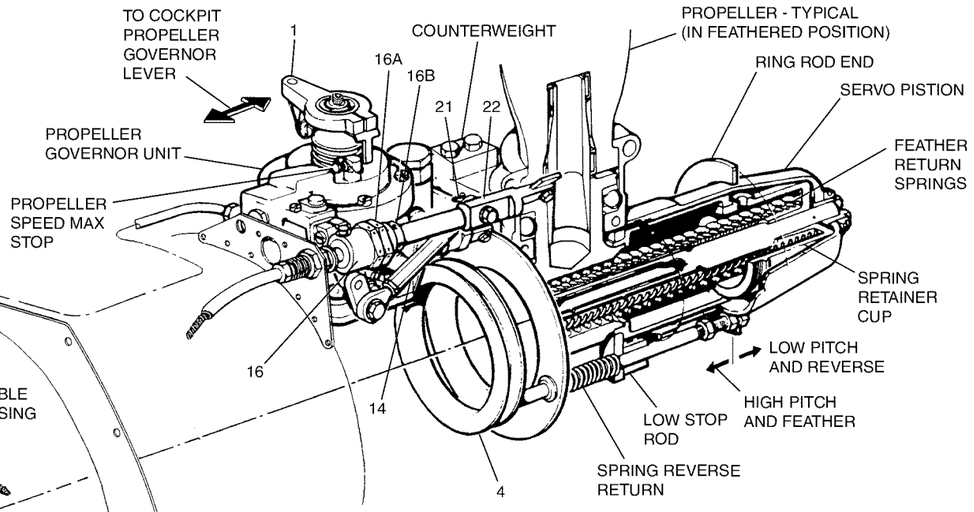
(2) Hold the momentary test switch on the left center subpanel in the SEC LOW PITCH STOP position. Lift the power lever and pull it aft until it is over the top horizontal line marking on the pedestal. After the initial deflection of its pointer in the increase rpm direction, the midpoint of the fluctuation range should be 170 to 250 rpm higher than the reading noted in Step (1) and the PROP LOW PITCH light (white) in the annunciator panel should be illuminated.

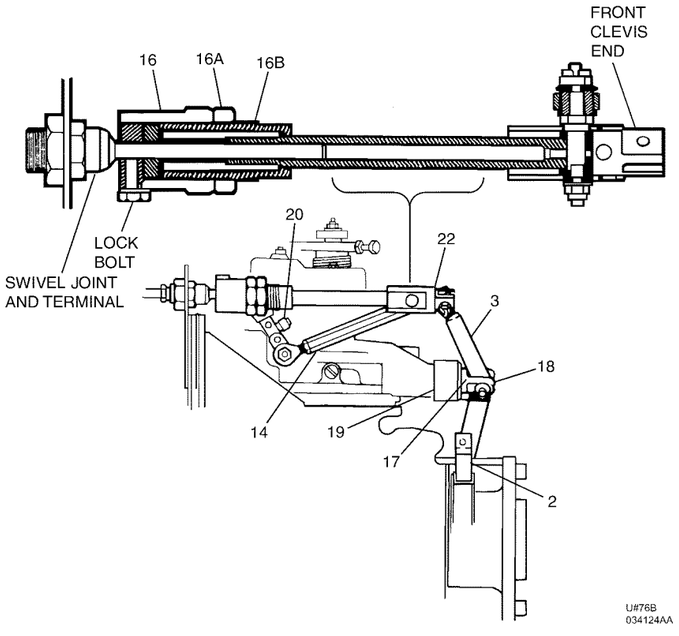
NOTE: If propeller rpm does not increase within the prescribed limits, readjust the position of the sensor on its mounting bracket (Ref. Figure 211). Move the sensor forward to decrease rpm or aft to increase rpm. On U-152 thru U-164, readjustment of the electro-optic switch is the same, except as indicated in Figure 217.

(3) Release the momentary test switch and rpm should increase. Return the power lever to its normal idle position.









1. PROPELLER SPEED SELECT LEVER
2. CARBON BRUSH BLOCK
3. PROPELLER REVERSING LEVER
4. LOW PITCH STOP COLLAR
5. ENGINE POWER LEVER
6. CAM FOLLOWER PIN
7. FCU CONTROL ROD
8. FCU ACTUATING LEVER
9. FUEL CONTROL UNIT ARM
10. FUEL CONTROL ARM EXTENSION NUT
11. SERRATED SPACER
12. IDLE CONTROL LEVER
13. CONDITION CONTROL ROD
14. FUEL GOVERNOR INTERCONNECTION ROD
15. REAR CLEVIS
16. LOW PITCH STOP ADJUSTER
17. CLEVIS SLOT END
18. BETA CONTROL VALVE
19. CAP NUT
20. OVERRIDE STOP
21. CLAMP
22. PUSH-PULL CONTROL TERMINAL
23. BETA CONTROL CAM
24. REVERSE ADJUST SCREW
25. ACTUATING LEVER
26. DETENT PLATE
27. LOW IDLE ADJUSTING SCREW

Figure 213
Rigging Propeller Reversing Mechanism

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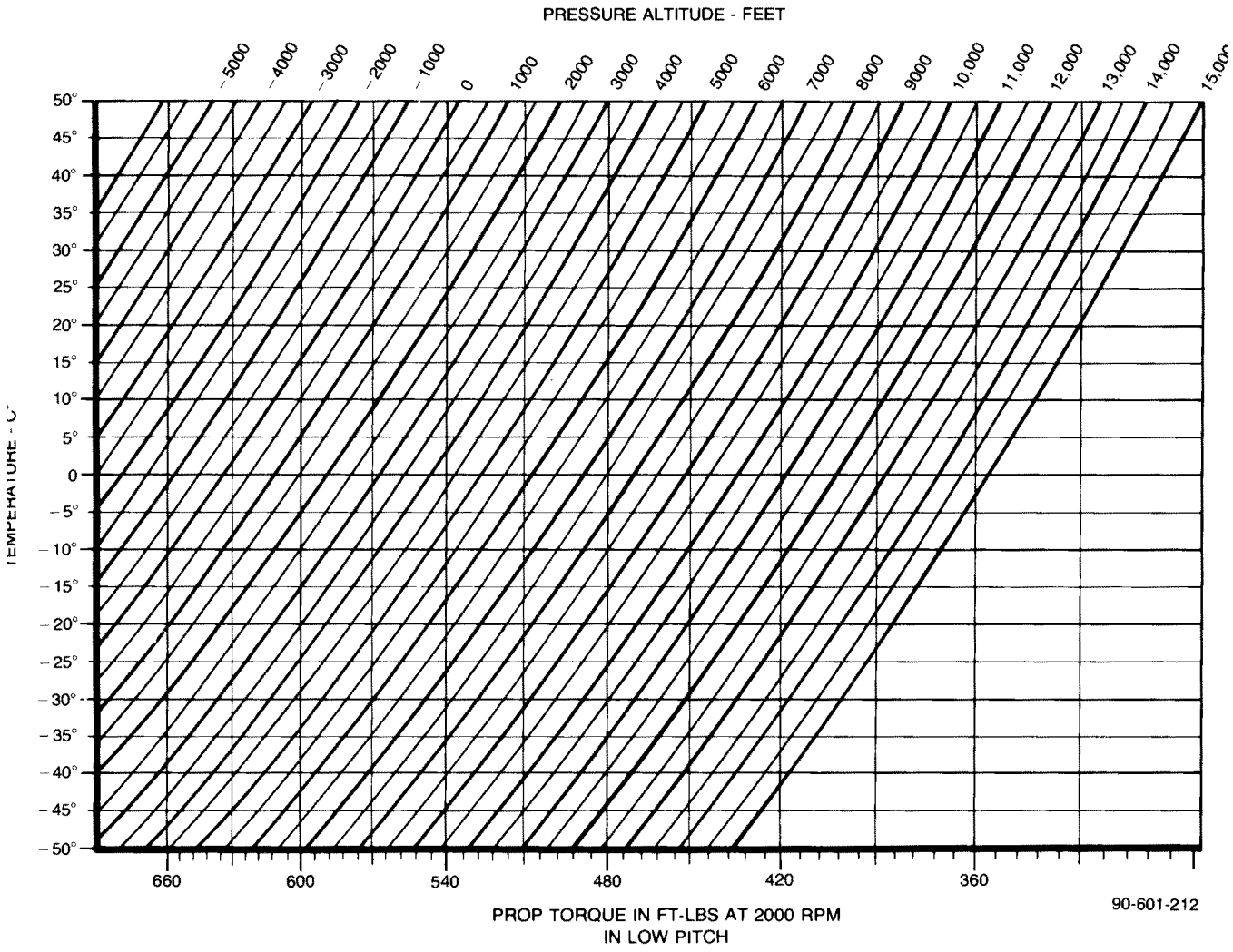

Figure 214
Pressure Altitude and Temperature Conversion Chart

13. ENGINE AND THE PROPELLER REVERSING MECHANISM (PT6A-36 ENGINE)

A. Rigging

CAUTION: *If the pedestal controls are moved into the reversing position after the engine controls are hooked up, damage to the reversing linkage will result.*

(1) Check the propeller speed adjusting lever (13) (Ref. Figure 218) of the governor for full travel between the maximum and minimum stops without interference with the control rod end. If full travel is impossible, loosen the lever retaining bolt, lift the lever free and reinstall it on the governor sleeve in a position 3 serrations from its original position.

(2) Attach the carbon block (45) to the propeller reversing lever (43) and check that the clearance between the carbon block (45) and the propeller feedback ring (23) is 0.001 to 0.002 inch at the narrowest portion of the propeller feedback ring (23).

CAUTION: *With the carbon block (45) held against the propeller feedback ring (23), check the clearance between the propeller feedback ring (23) and the metal retaining clip of the carbon block (45). If at any point this clearance is 0.005 inch or less, replace the carbon block (45) to prevent damage to the propeller feedback ring (23) by the carbon block retaining clip.*

(3) With the pedestal power control lever in the idle position (at the detent), install the input lever (26) on the grooved pin of the cam assembly so that the hole in the end of the input lever (26) is in a straight line with respect to the grooved pin and the pin immediately forward of it when the cam follower pin (36) is almost directly above (85° with respect to) the line formed by the grooved pin and the pin immediately forward of it. Tighten the clamp bolt in the input lever to a torque of 90 to 100 inch-pounds.

CAUTION: *If the clamping portions of the input lever contact one another, replace the input lever with a new one to ensure a proper grip on the grooved pin. New levers do not have grooves in the bore. They are formed when clamped to the grooved pin.*

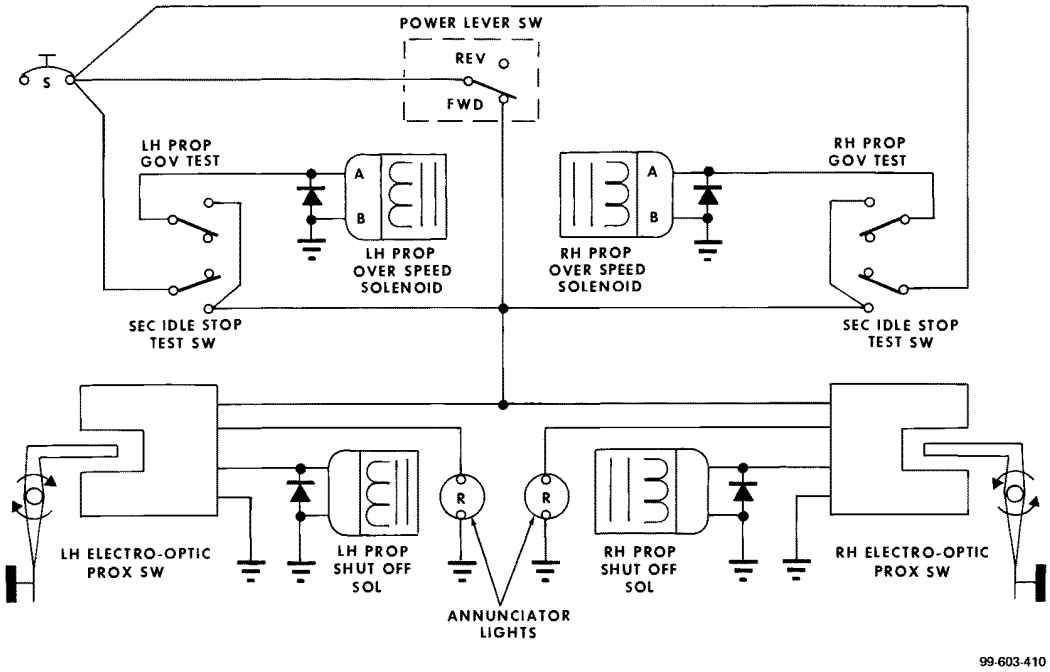

Figure 215
Propeller Governor and Secondary Low Pitch Schematic
(U-152 thru U-164, Prior to Incorporating Kit No. 99-3026-1S

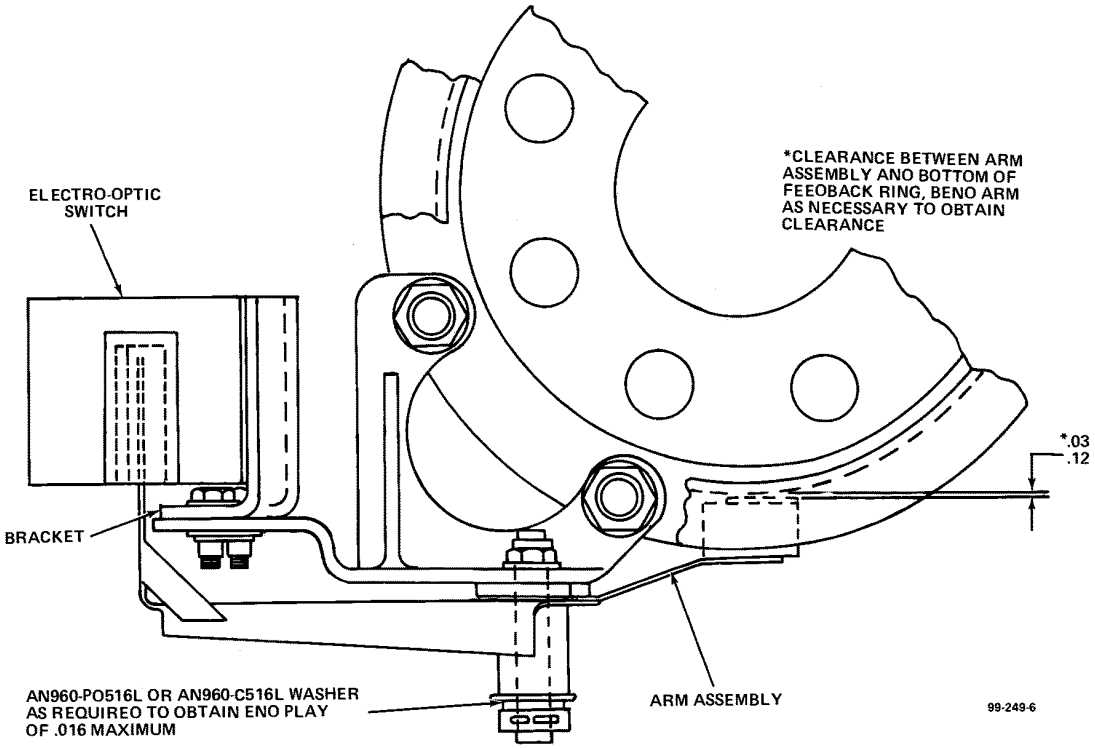

Figure 216
Electro-Optic Switch Installation
(U-152 thru U-164 Which Have Not Incorporated Kit No. 99-3026-1S)

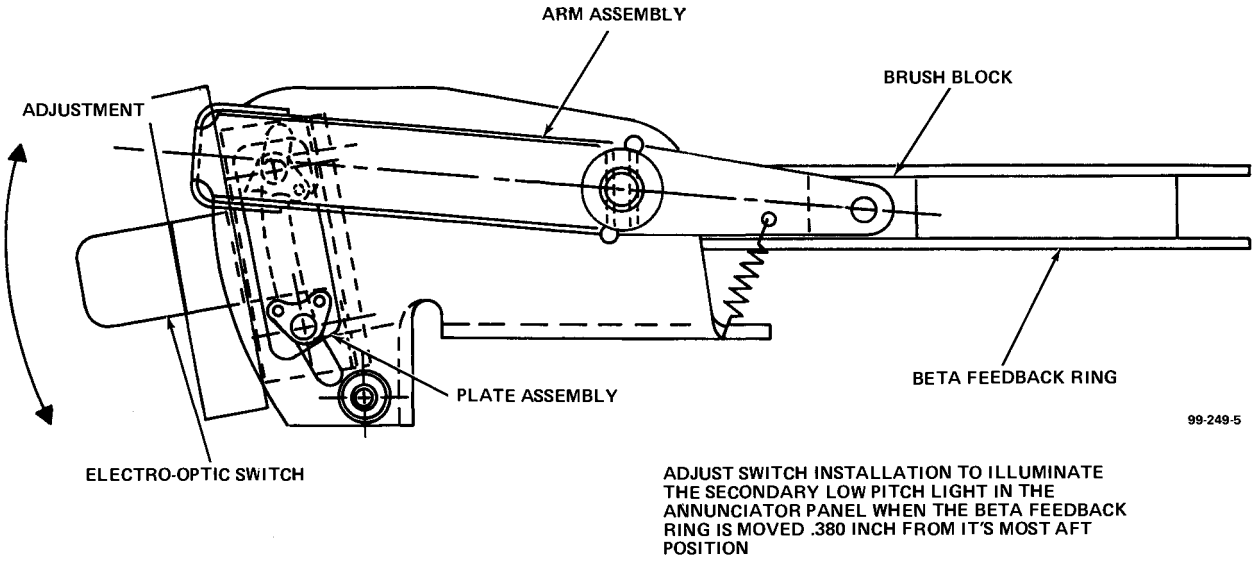

Figure 217
Electro-Optic Switch Rigging
(U-152 thru U-164 Which Have Not Incorporated Kit No. 99-3026-1S)

NOTE: Adjustment of the pedestal power levers to the idle position simultaneously established the ground fine setting.

(4) Adjust the FCU (Fuel Control Unit) control rod (29) until the distance between the center of the holes on each end of the FCU control rod (29) is 11 ± 0.05 inches. Attach one end of the FCU control rod (29) to the top hole of the FCU actuating lever (6) and the other end of the FCU control rod (29) to the inner hole of the FCU arm (31).

(5) With the pedestal power control lever fully forward (Ref. to TAKEOFF POSITION of Figure 218), the cam follower pin (36) should not contact the bottom of the slot in the reversing cam. If the pin does bottom, shorten the FCU control rod (29), as necessary to prevent bottoming.

(6) With both the propeller governor interconnecting rod (24) and the rear clevis (5) of the push-pull control cable disconnected, adjust the low pitch stop adjuster (25) until the clevis slot end (44) of the beta control valve is flush with the front face of the cap nut (46). Adjust the propeller governor interconnecting rod (24) until the retaining bolt will slide easily through the ball rod end with the ball rod end at the lower end of the interconnecting rod. Secure the upper end of the propeller governor interconnecting rod (24) to the clamp (48) of the push-pull control terminal (15) with the two washers, castellated nut and cotter pin. Insert the retaining bolt through the lower arm on the fuel control portion of the propeller governor, the lower end of the propeller governor interconnecting rod (24) and washers, then secure the bolt in place with the castellated nut and cotter pin.

(7) Adjust the rear clevis (5) of the push-pull control cable until the rear clevis (5) aligns with the second hole from the top of the propeller control cam (4). Insert the retaining pin through the rear clevis (5) propeller control cam (4) and washer, then lock the pin in place with a cotter pin.

14. RUN-UP CHECKS (PT6A-36 ENGINES)

A. Low Idle Adjustment (PT6A-36)

Low idle speed adjustment of the N_1 governor can be performed satisfactorily only at pressure altitude (from sea level to approximately 3500 ft.) at which the fuel flow required to produce 52 to 55% N_1 is greater than the minimum fuel flow setting of the fuel control unit. At higher altitudes, the minimum fuel flow will produce N_1 speeds above the N_1 governor setting. The N_1 governor cannot reduce fuel flow below the minimum flow setting and N_1 speed will increase in proportion to altitude when operating above approximately 3500 ft.

CAUTION: *Do not perform checks at barometric pressures below 25.2 in. Hg. or altitudes in excess of 3500 ft. Checks must be performed with zero power extraction except the generators must be loaded to 0.50 or greater.*

(1) Start the engines in accordance with the Pilot's Operating Handbook and FAA Approved Airplane Flight Manual.

(2) Move the power lever to low idle position. Allow the gas generator speed to stabilize. If the idle speed is too fast, move the power lever through the low idle detent toward reversing. This will assure that the idle speed is being controlled by the idle adjusting screw and not by the position of the power lever.

CAUTION: *If the interturbine temperature rises abnormally at any time during the idle speed adjustment, immediately shut down the engine.*

(3) Check the gas generator tachometer for a reading of 52 to 55% N_1 speed, with 54% being preferred while the generators are loaded to 0.50. If the proper N_1 speed cannot be obtained, the low idle adjustment screw (49) on the fuel control unit must be adjusted as indicated in the following Steps (Ref. Figure 218):

(a) Shut down the engine.

(b) Remove the safety wire from the idle adjusting screw (49) and loosen the locknut.

(c) Turn the idle adjusting screw (49) clockwise to increase low idle speed or counterclockwise to decrease the low idle speed.

NOTE: The idle speed adjustment is very sensitive and should not be moved more than 1/8 of a turn between idle checks.

(d) When low idle is 52 to 55%, ideally 54%, tighten the lock nut on the idle adjusting screw. Torque 20 to 29 inch-pounds.

(e) Check the low idle speed after the locknut is tight. If the idle speed is still 52 to 55%, shut down the engine and safety the low idle adjustment screw.

(f) Start the engine and move the power lever through the idle gate and then back against the idle stop. If idle speed increases from the previous setting, adjust the pedestal low idle stop or the power lever rigging to obtain the idle speed of 52 to 55% N_1.

B. High Idle Adjustment (PT6A-36)

(1) Start the engines in accordance with the Pilot's Operating Handbook.

(2) Place the condition lever at the high idle position.

(3) Place the power lever at the idle position.

(4) Check the tachometer reading. The N_1 reading should be 70 ± 1%. If the reading is not in this range, proceed with the following Steps to adjust the high idle:

(a) Shut down the engine.

(b) Loosen the locknut on the adjusting nut (34) (Ref. Figure 218).

(c) To increase the N_1 speed, turn the adjusting nut clockwise. To decrease the N_1 speed, turn the adjusting nut counterclockwise. One full turn of the nut will increase or decrease the N_1 about 1%.

(d) When the N_1 speed is in the recommended range of 70 ± 1% using the above procedures, tighten the locknut on the adjusting nut.

C. Setting Maximum Gas Generator Speed

NOTE: Fuel control units are calibrated during bench test to provide maximum power and temperature rating on an 18°C (65°F) day. If engine inlet temperature is other than 18°C (65°F), it is impossible to set the maximum gas generator speed stop without exceeding either temperature or torque limits. To overcome this, a movable spacer is placed between the gas generator maximum speed (N_1) stop and the power lever anvil. This spacer is called the part power trim stop and represents a constant N_1 speed decrement. Permanently attached to the gas generator maximum speed stop, the part power trim stop may be moved from its stowed position for engine trimming as follows:

(1) Loosen the screw securing part power trim stop to gas generator maximum speed stop.

(2) Position the part power trim stop between the power lever anvil and the maximum gas generator speed stop.

(3) Prior to adjusting, run the engine with power lever full forward for approximately two minutes to obtain normal operating temperatures.

With the power lever and propeller control lever full forward, adjust the maximum gas generator control to obtain 97.1 gas generator (N_1) speed. One complete turn of the gas generator speed control changes gas generator speed by approximately one percent.

15. MAXIMUM REVERSE N_1 (PT6A-36)

A. Adjustment

(1) Start the engines in accordance with the Pilot's Operating Handbook.

(2) Move the power levers to full reverse. Check for 85 +3/ -0% N_1 speed. If adjustment is required, proceed with the following Steps.

(3) Shut the engine down.

(4) Loosen the lock nut.

(5) To increase the N_1 speed, turn the reverse adjusting screw in. A full turn of the adjusting screw will increase the N_1 about 3%. To decrease the N_1 speed, turn the adjusting screw out.

NOTE: If necessary, the adjusting screw may be turned around to get the required N_1 speed. Figure 219 shows the adjusting screw in both directions.

16. POWER LEVER (PT6A-36)

A. Adjustment

(1) Check low idle speed as instructed in this chapter under the heading LOW IDLE ADJUSTMENT (PT6A-36).

(2) Check high idle speed as instructed in this chapter under the heading HIGH IDLE ADJUSTMENT (PT6A-36).

(3) Run both engines at low idle until the gas generator (N_1) speed of each engine stabilizes. Slowly advance the pedestal power lever for the LH engine until N_1 speed just starts to increase, then advance the pedestal power lever for the RH engine until the N_1 speed just starts to increase. The two pedestal power levers should be even. If they are not, advance each LH and RH pedestal power lever in equal increments to ascertain if the stagger in the power levers of each engine is constant or progressive.

(a) The stagger between the pedestal power levers is constant if the distance between them (for example, 1/8 inch) remains constant through the full range of power for both engines. If this condition exists, the FCU rod (29) requires adjustment (Ref. Figure 218). Lengthen the FCU rod on the engine that increases power first until engine power is symmetrical as the pedestal power levers are advanced evenly together.

(b) The stagger between the pedestal power levers is progressive if the distance between them increases as the power settings are advanced (for example, at 53 to 55% N_1, one pedestal power lever leads the other by 1/8 inch, at 60% N_1, the pedestal power lever leads the other by 3/16 inch and at 70% N_1, the pedestal power lever leads the other by 1/4 inch). If this condition exists, the FCU (fuel control unit) arm (31) must be adjusted. Adjust the FCU arm counterclockwise on the engine that increased power first (the one that continually followed the forward pedestal power lever) until the levers both advance at the same N_1 with both pedestal control levers even. This adjustment decreases the angle of the FCU arm to decrease the acceleration of the engine that increased N_1 speed faster. To perform this adjustment, remove the safety wire, loosen the FCU arm extension nut (30), reposition the FCU arm (31) using the serrations. The serrations are course and fine allowing the minimum adjustment of 0.6° of the FCU arm. After adjustment is made, torque the FCU arm extension nut (30) 25 to 35 inch-pounds. Safety wire the nut.

(4) With propeller levers fully forward, advance the pedestal power lever for the engine until the propeller reaches 2000 rpm. The torque meter reading for the engine should be within 40 foot-pounds of the value designated by the Pressure Altitude and Temperature Conversion Graph

(Ref. Figure 214) for the ambient conditions under which the engine is operating. (Repeat this check on the other engine. The difference between the torque meter readings on the RH and LH engine should not exceed 20 foot-pounds.

NOTE: If this check is NOT being performed under zero wind conditions, take an average of the upwind and downwind readings to obtain valid results.

If the torque meter readings do not fall within the limitations designated by the graph or if the difference between the LH and RH engines exceeds 20 foot-pounds, the following adjustments are required:

(a) Disconnect the propeller governor interconnecting rod (24) (Ref. Figure 218).

(b) Disconnect the rear clevis (5) from the propeller control cam (4).

(c) Loosen the low pitch stop adjuster locknut (51).

(d) Rotate the low pitch stop adjuster stop in to decrease, or out to increase the engine torque. One full rotation of the low pitch stop adjuster stop will increase or decrease the engine torque approximately 92 foot-pounds. Adjust as necessary to obtain the proper engine torque as determined in this Step.

(e) Torque the low pitch stop adjuster locknut (51) 150 to 250 inch-pounds and safety wire to the low pitch stop adjuster.

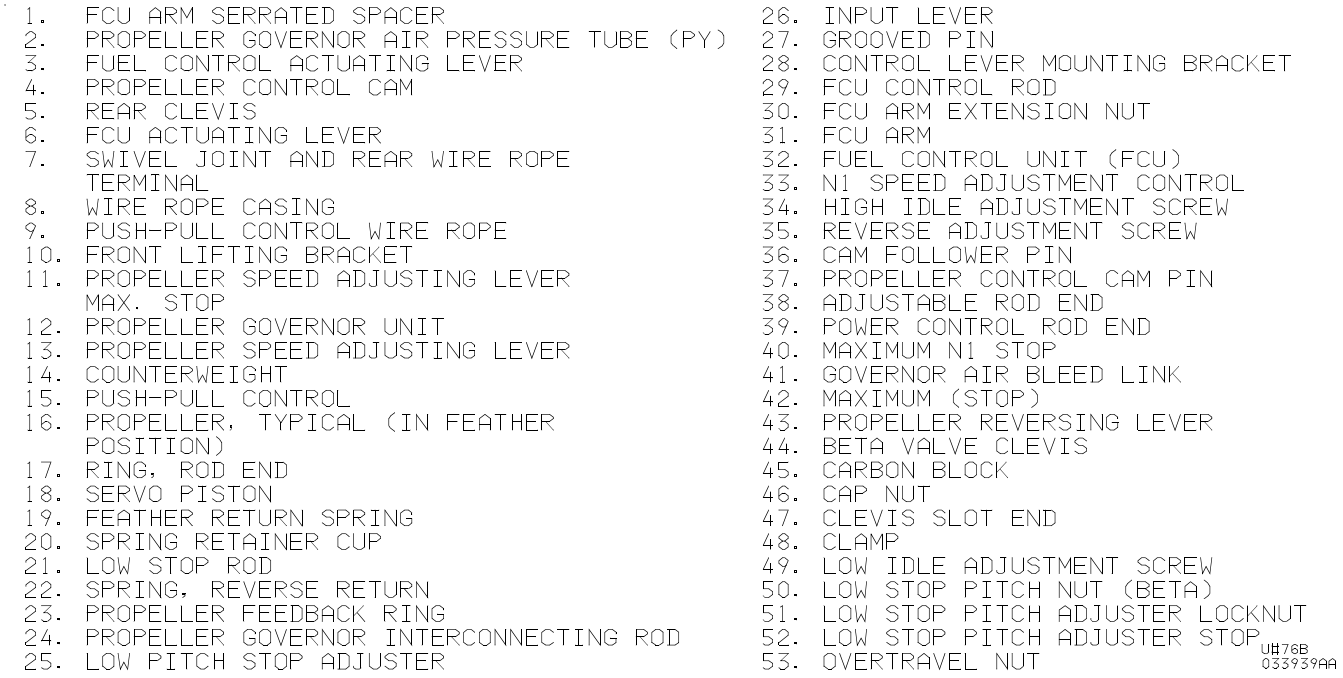

Figure 218 (Sheet 1 of 2)
Rigging Propeller Reversing Mechanism PT6A-36 Engines

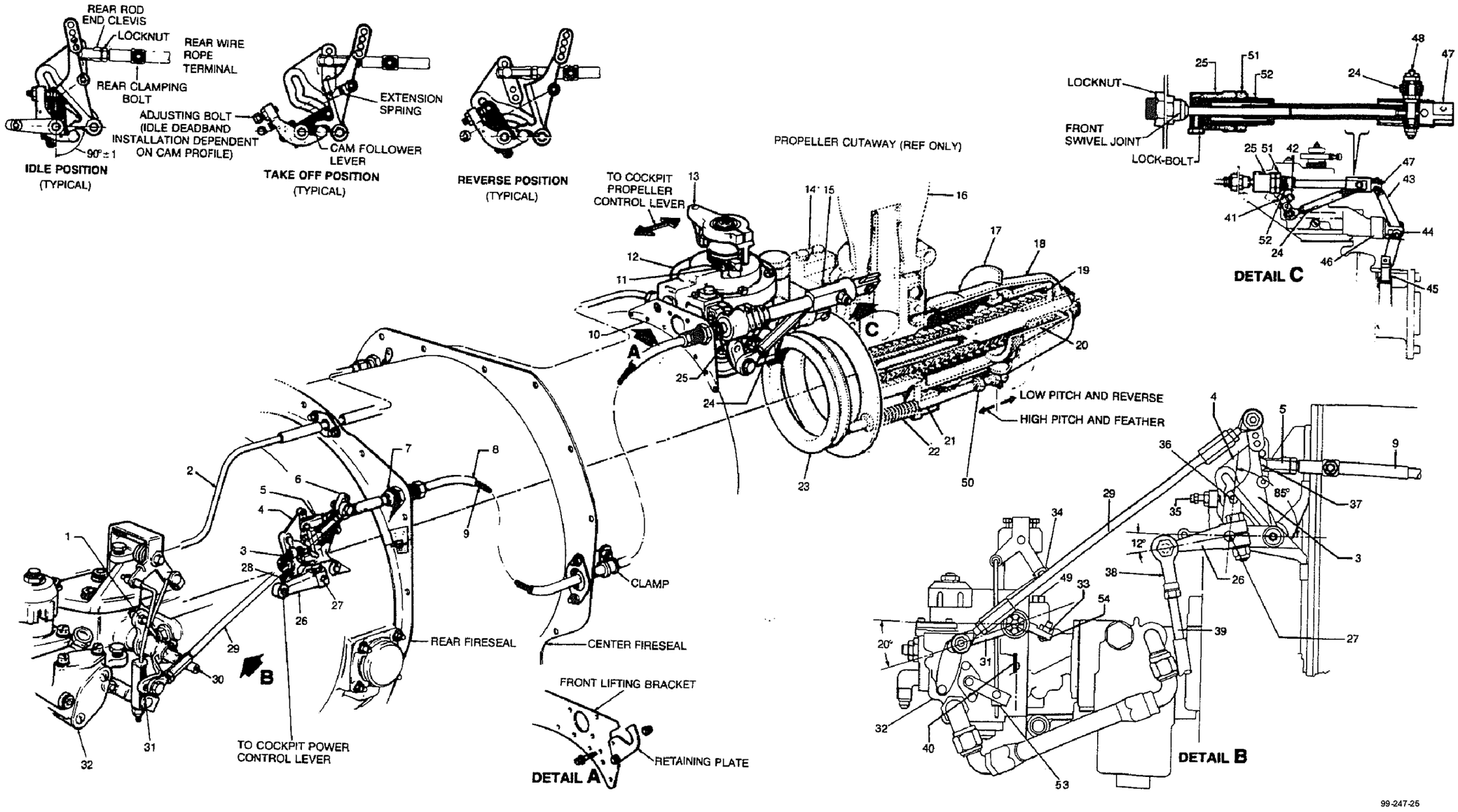

Figure 218 (Sheet 2 of 2)
Rigging Propeller Reversing Mechanism - PT6A-36 Engines

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(5) With the condition levers set at low idle and the propeller control levers full forward, pull both pedestal power levers over the flight idle detent to the ground fine detent, N_1 should remain at the flight idle setting. N_1 speed should not begin to increase on either engine until the power levers are moved over the ground fine detent and into reverse.

(6) Pull the pedestal power levers fully aft and check for a maximum N_1 speed of 85 to 88% in reverse for each engine. Adjust the reverse adjust screw (35) IN to increase or OUT to decrease N_1 speed until both engines are within the 85 to 88% N_1 and the engines are within 2% N_1 speed of each other. The reverse adjust screw (35), when turned one full turn, changes the N_1 speed approximately 3%.

(7) With engine shut down and the propeller in the feathered position, check that the beta valve is not bottomed out. There should be a minimum of 0.01 aft movement of the beta valve clevis possible when rigging is completed. If the valve is bottomed out, move the low stop nuts on the propeller out one turn. Check that nuts are evenly spaced from the piston. Repeat Steps (3) thru (7).

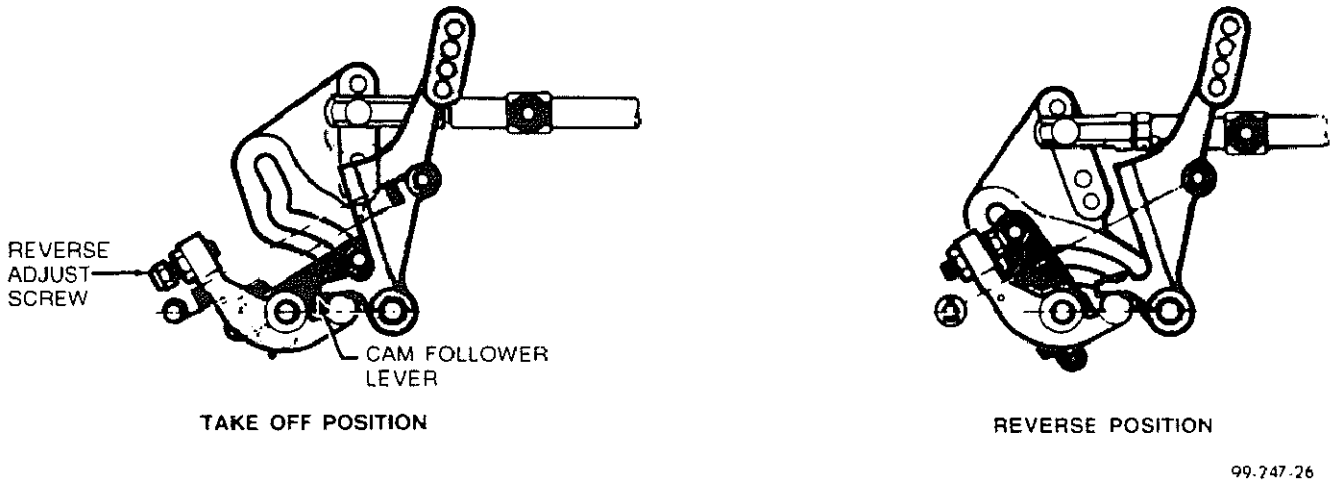

Figure 219
A Reverse Adjustment Screw

17. FUEL CUTOFF LEVER (PT6A-36)

A. Adjustment

(1) Back off the overtravel nut (53) several full turns (Ref. Figure 218).

(2) Move the pedestal condition lever to the fuel cutoff position. The fuel cutoff linkage must contact the fuel cutoff stop bolt with some spring back in the pedestal lever. If rigging is out of adjustment, adjust the power control cable rigging. This adjustment is made with the fuel cutoff stop bolt.

(3) With the pedestal condition lever in the fuel cutoff position, adjust the cutoff valve by placing the pedestal power lever against its minimum stop (idle detent), and check the clearance between the nut and spacer (Ref. Figure 201). Reposition the overtravel nut (53) as necessary to obtain the 0.005 to 0.010 inch clearance shown in Figure 201 (Ref. Figure 218).

(4) If a new engine or fuel control unit has been installed, check for positive cutoff as follows before operating the engine:

(a) Disconnect the fuel line at the rear of the fuel control unit. A suitable length of hose or tubing may be routed overboard from the fuel control unit to prevent fuel from being pumped into the cowling.

(b) Place one man in the flight compartment of the airplane to motor the engine and another at the nose to observe the fuel flow.

(c) With the electrical boost pumps operating and ignition OFF, motor the engine while moving the flight compartment condition lever from the CUTOFF position, then slowly return the condition lever to the CUTOFF position.

CAUTION: *Do not exceed the starter time limits of 40 seconds ON, 60 seconds OFF, 40 seconds ON, 60 seconds OFF, 40 seconds ON, then 30 minutes OFF.*

Fuel should flow from the hose, then stop flowing when the condition lever reaches the CUTOFF position.

(d) If the fuel continued to flow after the condition lever was returned to the CUTOFF position, check the low and high idle adjustment.

(e) Reconnect the fuel line to the rear of the fuel control unit.

(f) Pressure the fuel line and check for leaks.

18. FLIGHT IDLE TORQUE (PT6A-36)

A. Check

(1) Set the altimeter at 29.92 inches Hg. and read the pressure altitude. Take reading (in degrees Celsius) from the airplane OAT indicator.

NOTE: Make certain that the temperature reading has stabilized on airplanes recently removed from a hangar.

(2) At the top of the graph entitled Pressure Altitude and Temperature Conversion Chart (Ref. Figure 214), locate the pressure altitude curve corresponding to the reading taken in the preceding Step. Trace this curve down to the point where it intersects the OAT reading (in degrees Celsius) indicated on the LH side of the chart. A line dropped perpendicular from this point to the bottom of the graph will indicate the torque value in foot-pounds at which the engine should operate under the existing conditions.

(3) With propeller levers fully forward, advance the pedestal power lever for the engine until the propeller reaches 2000 rpm. The torque meter reading for the engine should be within 40 foot-pounds of the value designated by the Pressure Altitude and Temperature Conversion Graph for the ambient conditions under which the engine is operating.

(4) Repeat this check on the other engine. The difference between the torque meter readings on the RH and LH engine should not exceed 20 foot-pounds.

NOTE: If this check is NOT being performed under zero wind conditions, take an average of the upwind and downwind readings to obtain valid results.

(5) If the torque meter readings do not fall within the limitations designated by the graph or if the difference between the LH and RH engines exceeds 20 foot-pounds, the following adjustments are required:

(a) Disconnect the propeller governor interconnecting rod (24) (Ref. Figure 218).

(b) Disconnect the rear clevis (5) from the propeller control cam (4).

(c) Loosen the low pitch stop adjuster locknut (51).

(d) Rotate the low pitch stop adjuster stop in to decrease, or out to increase the engine torque. One full rotation of the low pitch stop adjuster stop will increase or decrease the engine torque approximately 92 foot-pounds.

Adjust as necessary to obtain the proper engine torque as determined in this Step.

(e) Torque the low pitch stop adjuster locknut (51) 150 to 250 inch-pounds and safety wire to the low pitch stop adjuster.

(6) With the condition levers set at low idle and the propeller control levers full forward, pull both pedestal power levers over the flight idle detent to the ground fine detent and check for an increase in N_1 speed. N_1 speed on neither engine should increase until the power levers are moved over the ground fine detent and into reverse.

(7) Pull the pedestal power levers fully aft and check for a maximum N_1 speed of 85 to 88% in reverse for each engine. Adjust the reverse adjust screw (35) IN to increase or OUT to decrease N_1 speed until both engines are within the 85 to 88% N_1 and the engines are within 2% N_1 speed of each other. The reverse adjust screw (35), when turned one full turn, changes the N_1 speed approximately 3%.

(8) With engine shut down and the propeller in the feathered position, check that the beta valve is not bottomed out. There should be a minimum of 0.01 inch aft movement of the beta valve clevis possible when rigging is completed. If the valve is bottomed out, move the low stop nuts on the propeller out one turn. Check that nuts are evenly spaced from piston. Repeat Steps (3) thru (8).

19. POWER LEVER DETENT PINS

A. Inspection

(1) Disconnect all electrical power from the airplane.

(2) Remove the pilot's and copilot's seat assemblies from the airplane.

(3) Remove the upholstery panels from the LH and RH sides of the pedestal.

(4) Remove the small instrument panel forward of the pedestal control levers.

(5) Using a mirror and flashlight, visually inspect the detent pin on each throttle lever for excessive wear where the pin contacts the slot in the idle/reverse stop assembly between the levers (Ref. Figure 220). The detent pins may wear a maximum of 0.03 inch. If this visual inspection reveals any grooving in the detent pins, remove and replace the affected power lever as follows:

(a) Remove the engine control lever knobs, the engine control friction brake knobs and the flap control knob.

(b) Remove the engine control lever edge lighted panel and the printed circuit board from the pedestal.

(c) Remove the screw and washer that secure the left end of the control lever pivot shaft to the left side of the pedestal.

(d) Remove the four screws that secure the right end of the control lever pivot shaft to the right side of the pedestal.

(e) Place a drop cloth under the pedestal to prevent items dropped from falling below the floor.

(f) AN960-816 or AN960-816L washers are installed between the pedestal levers to center the levers in the pedestal slots; when removing the power levers, note the locations of these washers so they can be installed in these same positions.

(g) Fabricate a dummy control lever pivot shaft 0.495 inch in diameter and at least nine inches long.

(h) Insert the dummy shaft into the pivot shaft opening in the left side of the pedestal, and push the control lever pivot shaft to the right until the dummy shaft is ready to enter the power lever which is to be removed.

NOTE: If both power levers are to be replaced, it is best to remove and replace them individually starting with the RH lever.

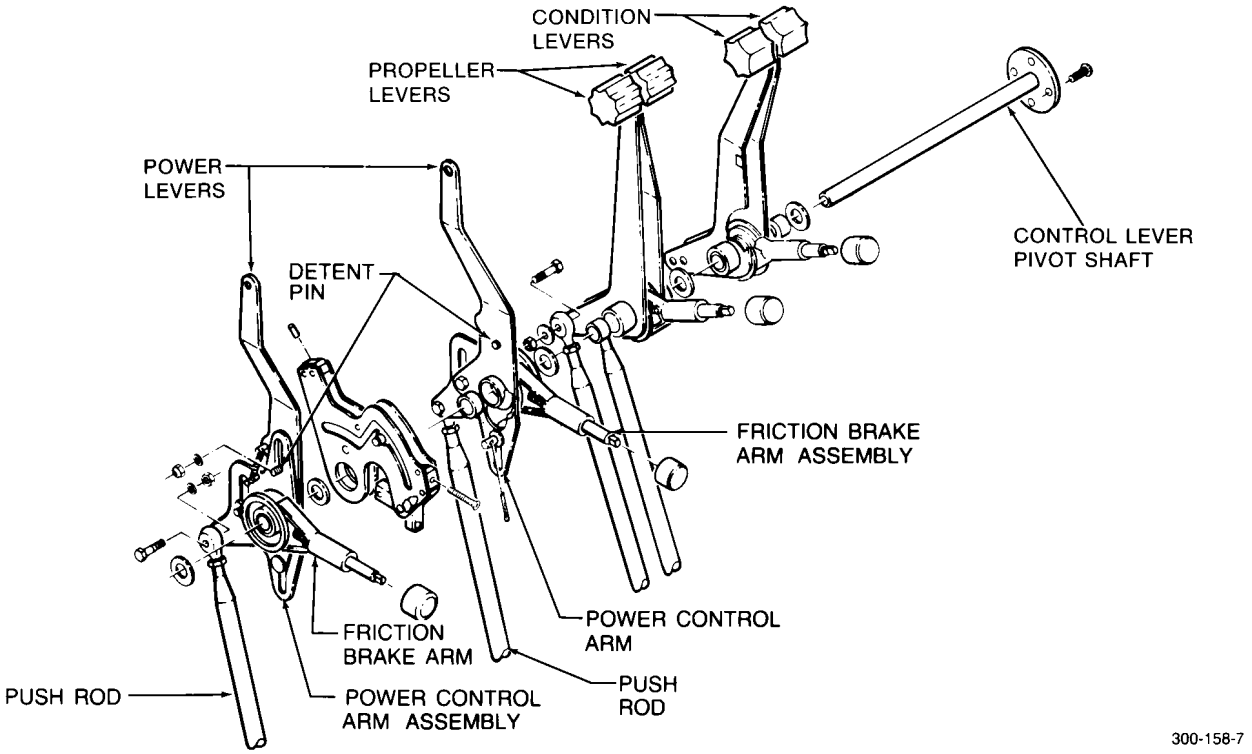

Figure 220
Engine Control Levers

(i) Disconnect the push rod from the power lever to be removed.

(j) Catching the washers that fall, pull the pivot shaft to the right until the power lever assembly is released, then remove the assembly from the pedestal.

(k) Remove the tension spring and arm assembly and friction brake from the power lever.

(l) Assemble the arm assembly and friction brake to the new power lever. Tighten the nuts to allow the minimum amount of clearance between the arm assembly and the power lever that will permit the pins to move freely in the full length of the slots. Install the tension spring.

(m) Position the power lever in the pedestal, and push the control lever pivot shaft through the throttle control lever assembly until contact is made with the dummy shaft. Install any washers that were removed while removing the power lever.

(n) Connect the push rod to the power lever.

(o) If both power levers are being replaced, repeat Steps (i) thru (n) to replace the LH power lever.

(p) Push the engine control lever pivot shaft to the left to remove the dummy shaft from the pedestal. Ensure that the washers stay in position while moving, the pivot shaft.

(q) Install the four screws which secure the RH end of the pivot shaft to the pedestal. Secure the LH end of the pivot shaft to the pedestal with the screw and washer.

(r) Install the engine control lever edgelighted panel and the printed circuit board on the pedestal.

(s) Install the engine control lever knobs, the friction brake knobs, and the flap control knob.

(6) Install the small instrument panel forward of the pedestal control levers.

(7) Install the LH and RH pedestal upholstery panels.

(8) Install the pilot's and copilot's seat assemblies.

(9) Check all switches and controls in the pedestal area for proper operation.

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77-10-00	201 thru 218	Nov 1/10
77-11-00	101 thru 103 201 thru 210	Nov 1/10 Nov 1/10
77-20-00	201 thru 209	Nov 1/10
77-21-00	201 thru 207	Nov 1/10

Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

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ENGINE INDICATING
GENERAL
DESCRIPTION AND OPERATION (U-1 THRU U-164, EXCEPT U-50)

The information and procedures contained in this chapter are supplemental to the instructions contained in the Engine Maintenance Manual and to the instructions supplied with the test equipment being used.

The engine performance check curves are presented as constant torque versus ambient temperature so that engine performance can be checked over a wide range of ambient temperature without overtorquing the reduction gearbox. Periodic engine performance checks must be carried out and changes in engine performance noted. All forms of engine deterioration will be accompanied by an increase in interstage turbine temperature and fuel flow at a given power. Compressor deterioration will, in most cases, be due to dirt deposits and will effect an increase in gas generator speed to obtain a given power setting. This form of deterioration can be remedied by compressor washing as described in the engine maintenance manual. Hot section deterioration will cause a decrease in the gas generator speed required for a given power setting. Perform the Engine Ground Test in accordance with the instructions of the Jetcal Unit.

ENGINE INDICATING
GENERAL
DESCRIPTION AND OPERATION (U-50, U-165 AND AFTER)

Checks performed at Hawker Beechcraft facilities on engine indicating systems include the inter turbine temperature indicating system (ITT), gas generator speed (N_1) indicating system, torquemeter system, fuel flow meter system, and the propeller speed (N_2) indicator system.

Various analyzers and equipment are available for the purpose of checking and calibrating the engine indicating systems; however, only the equipment used by Hawker Beechcraft facilities will be discussed in this chapter.

The Jetcal Analyzer, a product of Howell Instruments, Inc. of Fort Worth, Texas, is used by Hawker Beechcraft facilities to check and calibrate the ITT indicator and N_1 and N_2 indicator systems.

The torquemeter system is checked by a precision pressure regulator gage assembly available through Hawker Beechcraft parts and service operations under the part number 101-000000/934.

A mechanical flowrator is used in the check procedure for the fuel flowmeter system. Any precision flowrator with a scale up to at least 700 pph can be used for this check.

ENGINE INDICATING
POWER
MAINTENANCE PRACTICES (U-1 THRU U-164, EXCEPT U-50)

1. GROUND TESTS (WITHOUT JETCAL)

A. Test

The engine performance check curves are presented as constant torque versus ambient temperature so that engine performance can be checked over a wide range of ambient temperatures without overtorquing the reduction gearbox. These curves should not be used above 37.8°C (100°F), or overtemperature could occur.

Periodic engine performance checks must be carried out and changes in engine performance noted. All forms of engine deterioration will be accompanied by an increase in interstage turbine temperature and fuel flow at a given power. Compressor deterioration will, in most cases, be due to dirt deposits and will effect an increase in gas generator speed to obtain a given power setting. When no Jetcal unit is available, engine check parameters may be determined as follows:

NOTE: The thermocouples are wired in parallel to produce an average reading. Protect the thermocouples from direct sunlight while taking T_1 temperature readings.

(1) Install the inlet air temperature (T_1) harness (P/N 50-000065 TC935) in each engine air inlet opening behind the baffle in order to pick up the most accurate reading necessary for this check. The thermocouple connectors are located approximately one-half inch from the inlet screen on the fore and aft centerline at the two, four, eight and ten o'clock positions. Connect the thermocouple harness leads to the temperature indicator.

(2) Disconnect the engine bleed air line and cap the compressor outlet.

(3) Perform a prestart check and start the engines as described in the appropriate Operator's Manual.

(4) Warm up the engines at ground idle until the engine instruments are stabilized. Turn the generator switch off on the engine being checked. Turn off the air conditioning system (if installed).

(5) Determine the pressure altitude by setting the altimeter at 29.92 inches Hg.

NOTE: The basic 99 Airliner is equipped with two PT6A-20 engines. The 99 Airliners With Kit 99-5014 installed, serials U-1 through U-55 except U-50, are equipped with PT6A-27 or PT6A-28 engines.

(6) Obtain the required torque settings for each engine at your pressure altitude from the Pressure Altitude vs Torque curves (Ref. Figure 209 for the PT6A-20 or Figure 210 for the PT6A-27 and PT6A-28).

(7) Set the propeller at 2000 rpm and advance the power lever to the engine torque setting determined above.

(8) Allow the engine to stabilize for two minutes at the required torque, then record the observed fuel flow, gas generator speed, ITT and inlet air temperature.

(9) Retard the power lever to FLIGHT IDLE, perform the engine shutdown check, and stop the engines as described in the servicing section.

(10) Check that the fuel flow observed during engine operation is within ± 15 lbs/hr of the Chart value (Ref. Figure 207 for the PT6A-20 or Figure 208 for the PT6A-27 and PT6A-28), designated for the inlet air temperature and pressure altitude at which the engine was operated.

(11) On the PT6A-20 engine, check that the ITT observed during engine operation is within ± 30.5°C (± 55°F) on the value designated by the Chart in Figure 213. On the PT6A-27 and PT6A-28 engine, check that the ITT reading does not exceed the maximum value designated by the Chart in Figure 214 for the inlet air temperature at which the engine was operated. If the reading is more than 75°C below the temperature designated in Figure 214, check the instrumentation.

(12) Determine the N_1 ratio of each engine by entering the values (data plate speed and gas generator speed) observed during engine operation in the DPS Conversion Chart in Figure 212. Check that this value is within ± 2% of the values designated in the Chart (Ref. Figure 213 for PT6A-20 PT6A-20 or Figure 214 for the PT6A-27 and PT6A-28) for the temperatures at which the engine was operated.

NOTE: If the observed values deviate from those noted in the preceding Steps, an instrument or engine malfunction should be suspected.

2. GROUND TESTS (WITH JETCAL)

A. Equipment Required

The following equipment is used at the factory in checking engine performance. Any equipment meeting the specified accuracy requirements may be used.

	Accuracy Requirements
N_1 RPM	± 0.1%
N_2 RPM	± 0.1%
T_5 TEMP	± 5.5°C

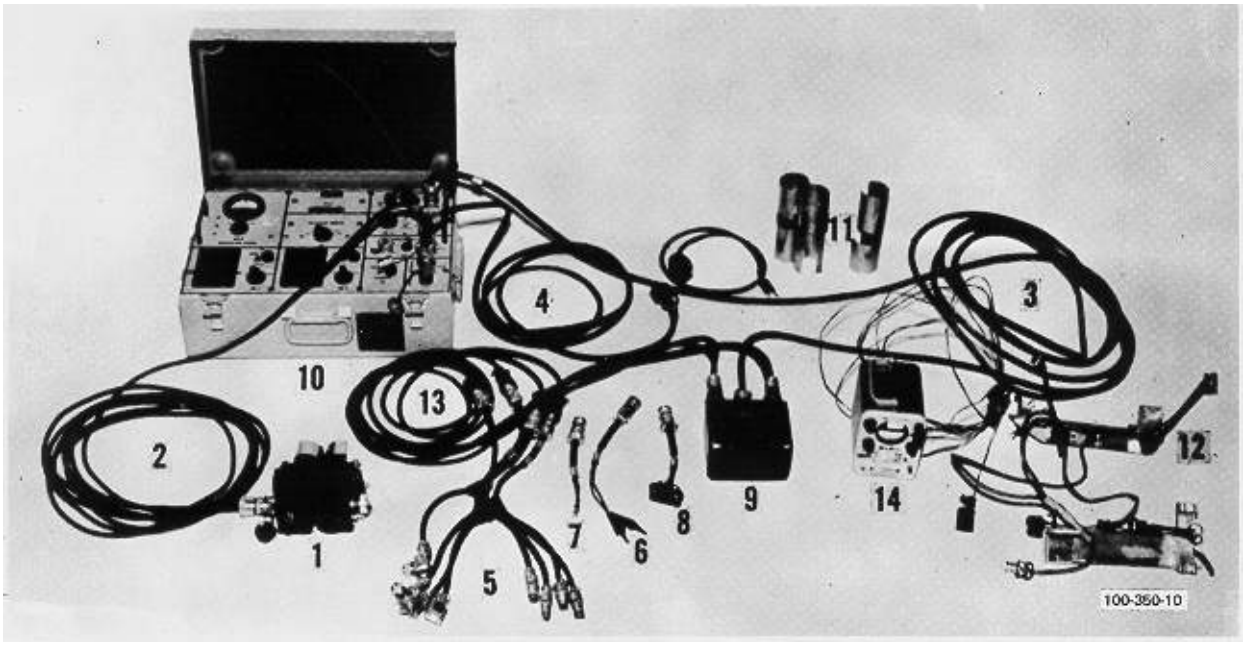


Figure 201
Ground Test Equipment

NOTE: An order for BH112JR Jetcal Unit called out in (10), will include the items listed in (1) thru (9).

	Item	Quantity
(1)	*BH134 Switch Box.	2 each
(2)	*BH450 Check Cable.	1 each
(3)	*BH485 Instrument Check Cable.	1 each
(4)	*BH499 Power Cable.	1 each
(5)	*BH820 Tachometer Check Adapter.	4 each
(6)	*BH821 Insulation Check Adapter.	1 each
(7)	*BH822 EGT Indicator Adapter.	1 each
(8)	*BH223 Resistance Check Adapter.	1 each
(9)	*BH5165 RPM Check Box.	1 each
(10)	*BH112JR Jetcal Unit.	1 each
(11)	Tachometer Support Brackets (Ref. Figure 202).	4 each
(12)	Inlet Air Temperature Thermocouple Harness (P/N 50-000065 TC935) or equivalent.	1 each

	Item	Quantity
(13)	Adapter Cable (4 Conductor Cable No. 18 gage wire used with BH820 RPM Check Cables). Consists of: 6 feet in length (4 required). MS3106A14S2S Plug (4 required). MS3106A14S2P Plug (4 required). AN3057-6 Plug (8 required).	4 each
(14)	Potentiometer Pyrometer, Model No. 80200 with a range of 0° to 600°F (product of Thermo Electric Co., Inc., Saddle Brooke, New Jersey) or equivalent.	1 each
(15)	“Y” Cable (for Toque Indicator and Transmitter Check). Consists of: MS3106A10SL3P Plug (2 required). MS3102A10SL3P Plug (1 required).	
(16)	Pressure Gage with range of 0 to 50 psi and accuracy of 1/4% throughout range (Model 0-60 GP-65.8 by Ashcraft is satisfactory) Regulated nitrogen or dry air supply for calibration of the torque system is also required.	1 each
(17)	Flowmeter (Capable of 1% accuracy in 100 to 400 pph range). Flowrator, Model 10A 2735CA (product of Fisher and Porter, Warminister, Pennsylvania) with tube No. FP 1/2 27G 10/80 6509 A1 671 A1 is satisfactory.	1 each
(18)	Power Cable (Ref. Figure 203).	1 each

* Products of Howell Instruments Inc. 3479 West Vikery, Fort Worth. Texas

3. INSTRUMENT CALIBRATION

Perform the following tests before the power check is initiated.

A. Calibration

(1) Perform the ITT CALIBRATION procedure (Ref. 77-20-00).

(2) Fuel Flow System Check

(a) Install the test flowmeter in the main fuel line at the inlet to the fuel heater (Ref. Figure 204).

(b) Turn on the ship fuel boost pump and open the firewall shutoff valve.

(c) Open the needle valve in the test system and let the fuel recirculate at the full flow rate until no bubbles are present in the flowmeter glass before starting the system check.

(d) Check each flowmeter at the rates indicated in Table 201.

(e) Remove the test equipment and connect the fuel lines at the fuel heaters.

(3) Torquemeter System Calibration

(a) Remove the torque transmitter from its mounts on the engine and connect it to a regulated nitrogen or dry an supply that has a pressure gage with a range of 0 to 50 psi plumbed into the system (Ref. Figure 205).

(b) With the transmitter still connected to the system electrical harness, turn on the 400 cycle inverter and carefully adjust the regulator to apply pressure to the transmitter while observing the torquemeter. Check the torquemeter at the readings indicated in Table 202.

(c) Remove the test system and install the torquemeter transmitters on the engine.

(d) After attaching the tachometer adapter cables to the tachometer, connect the Y adapters to the BH5165 RPM switch box with a BH820 cable, then connect the BH450 cable between the RPM switch box and the Jetcal unit (Ref. Figure 206).

(e) Install the inlet air temperature (T_1) harness (P/N 50-000065 TC935) in each engine air inlet opening, behind the baffle in order to pick up the most accurate reading necessary for this check. The thermocouple connectors are located approximately one-half inch from the air inlet screen on the fore and aft centerline at the two, four, eight, and ten o'clock positions.

(f) Route the thermocouple harness out through the cowling, along the wing leading edge, and through the storm window. Clamp or tape the thermocouples and harness to adjacent structure and along the wing as required to provide support during engine operation.

NOTE: The thermocouples are wired in parallel to produce an average reading. Protect the thermocouples from direct sunlight while taking T_1 temperature readings.

(g) Connect the thermocouple harness leads to the temperature indicator. After completing the check on one engine, repeat it on the other.

4. POWER CHECK

A. General Information

(1) The bleed air line must be capped off at the engine during the following ground tests.

(2) The generator switches on the engine being tested must be in the OFF position during the ground test.

(3) If air conditioning is installed on the aircraft, turn the air conditioning mode switch to the OFF position.

(4) Look directly into the gage being read to eliminate parallax.

(5) Use extreme care in obtaining T_1 temperatures and averages. A small change in T_1 results in a large ITT error on the Chart data.

(6) Let all power runs stabilize at least two minutes before reading the instruments.

(7) Check panel instruments constantly during engine power points. Hold all power settings as close as is possible on the gages being used.

(8) The following Charts and information are required to perform the engine power check:

(a) Temperature Conversion Chart °F to °C (Ref. Figure 2 11).

(b) N_1/DPS Ratio Conversion Chart (Ref. Figure 212).

(c) Engine Performance Check Curves (Ref. Figures 207, 209 and 213 for the PT6A-20 or Figures 208, 210 and 214 for the PT6A-27 and PT6A-28).

(d) Data plate speed from each engine.

(e) Power Check Data Sheet, Table 203.

B. Test Run

(1) Turn on the aircraft power and the instrument inverters.

NOTE: Check the galvanometer balance on the Jetcal unit after each reading. Should the attitude of the Jetcal unit be changed in any way during the test, the galvanometer will immediately become unbalanced.

(2) Turn on the Jetcal unit to the T/C position as directed by the operating instructions on the lid. Balance the temperature and rpm section manometers in accordance with the operating instructions noted in the preceding sentence.

(3) Start the left engine and repeat the check in the preceding Step.

(4) Set the altimeters to 29.92 and determine the elevation of the field. Average the reading of the pilot's and copilot's altimeters when they differ.

(5) Using the Pressure altitude reading obtained in the preceding Step, determine the torque values to be used for both the right and left engines from the engine performance check curve in Figure 213 for the PT6A-20 or Figure 214 for the PT6A-27 and PT6A-28.

NOTE: The 90.8% reading set on the Jetcal unit provides a common setting for each engine in case the engine instruments are not in agreement.

(6) With the Jetcal unit set at 90.8%, set the (N_2) propeller speed on one engine at 2,000 rpm and the torque value obtained in the preceding Step for that particular engine. If the needle on the Jetcal unit does not center, adjust the propeller rpm until the needle does center, then record that rpm on the Power Check Data Sheet (Ref. Table 203).

(7) After operating the engine at this power for at least one minute, read all T_1 (Air Inlet) Temperatures on the operating engine. When approximately two minutes of running time has elapsed, record the fuel flow and torque indicated on the pilot's panel.

NOTE: Operate the engine at the power setting until the ITT readings have stabilized before recording the result.

(8) Record the N_1 (gas generator speed), N_2 (propeller speed), and ITT readings from the Jetcal unit on the Power Check Data Sheet (Ref. Table 203).

(9) After all the required readings have been taken, reduce engine power to idle and repeat Steps (6) through (8).

(10) Using the average T_1 temperature, select the proper fuel flow rate, ITT and N_1/DPS (Data Plate Speed) ratio for the test conditions of each engine from the Power Charts (Ref. Figures 207 and 213 for the PT6A-20 or Figures 208 and 214 for the PT6A-27 and PT6A-28). Record these values on the Power Check Data Sheet. Check that the fuel flow rate obtained is within ± 15 lbs/hr of the Chart value.

(11) On the PT6A-20 engine, check that the ITT observed during engine operation is within ± 30.5°C (± 55°F) on the value designated by the Chart in Figure 213. On the PT6A-27 and PT6A-28 engine, check that the ITT reading does not exceed the maximum value designated by the Chart in Figure 214 for the inlet air temperature at which the engine was operated. If the reading is more than 75°C below the temperature designated in Figure 213 or 214, check the instrumentation.

(12) Check that the ratio obtained on the power check is within ± 2% of the value taken from the N_1/DPS Conversion Chart (Ref. Figure 212).

(13) If the results of the preceding data recorded on the Power Check Data Sheet do not fall within the limits on the power Chart, notify Parts and Service Operation, Hawker Beechcraft Corporation, for disposition.

C. Setting Maximum Gas Generator Speed

NOTE: Fuel control units are calibrated during bench test to provide maximum power and temperature rating on an 18°C (65°F) day. If engine inlet temperature is other than 18°C (65°F), it is impossible to set the maximum gas generator speed stop without exceeding either temperature or torque limits. To overcome this, a movable spacer is placed between the gas generator maximum speed (N_1) stop and the power lever anvil. This spacer is called the part power trim stop and represents a constant N_1 speed decrement. Permanently attached to the gas generator maximum speed stop, the part power trim stop may be moved from its stowed position for engine trimming as follows:

(1) Loosen the screw securing part power trim stop to gas generator maximum speed stop.

(2) Position the part power trim stop between the power lever anvil and the maximum gas generator speed stop.

(3) Prior to adjusting, run the engine with power lever full forward for approximately two minutes to obtain normal operating temperatures.

(4) With the power lever and propeller control lever full forward, adjust the maximum gas generator control to obtain 97.1 gas generator (N_1) speed. One complete turn of the gas generator speed control changes gas generator speed by approximately one percent.

Table 201
Fuel Flowmeter System Check

Indicator lbs/hr	Left Instrument S/N_____	Right Instrument S/N_____
100 ± 10	_____	_____
150 ± 10	_____	_____
200 ± 10	_____	_____
250 ± 10	_____	_____
300 ± 10	_____	_____
350 ± 10	_____	_____
400 ± 10	_____	_____

Table 202
Torquemeter System Check

Test PSI	Torquemeter ft-lbs	Tolerance	Left Instrument S/N_____	Right Instrument S/N_____
10	307	+43 to 60	_____	_____
20	614	+26 to 41	_____	_____
30	922	+18 to 33	_____	_____
35	1075	+15 to 30	_____	_____
40	1229	+12 to 28	_____	_____
45	1382	+16 to 31	_____	_____
50	1536	+19 to 34	_____	_____
55	1690	+25 to 34	_____	_____
60	1843	+29 to 34	_____	_____
65	1997	+31 to 35	_____	_____
70	2150	+32 to 37	_____	_____
80	2458	+35 to 40	_____	_____

Table 203
Power Check Data Sheet

Left Engine

Engine Serial No._____

Data Plate Speed_____%

Pressure Altitude_____ft

OAT_____°C

Inlet Air Temp (T_1)_____°C

Airplane Serial No._____

Date_____

Run by_____

Inspector_____

Chart Data		Run Data	Inst. Corr.	Corr. Reading	Deviation	Tolerance	Remarks
Torque _____lb-ft		Torque _____lb-ft					
ITT_____°C		Panel_____°C					
		JC_____°C					
Fuel Flow _____lbs/hr		Fuel Flow _____lbs/hr					
N_1/DPS	Ratio _____	Panel_____%					
		JC_____%					
N_2	2,000 rpm	Panel _____rpm					
	(90.8%)	JC_____%					

Right Engine

Engine Serial No._____

Data Plate Speed_____%

Pressure Altitude_____ft

OAT_____°C

Inlet Air Temp (T_1)_____°C

Chart Data		Run Data	Inst. Corr.	Corr. Reading	Deviation	Tolerance	Remarks
Torque _____lb-ft		Torque _____lb-ft					
ITT_____°C		Panel_____°C					
		JC_____°C					
Fuel Flow _____lbs/hr		Fuel Flow _____lbs/hr					
N_1/DPS	Ratio _____	Panel_____%					
		JC_____%					
N_2	2,000 rpm	Panel _____rpm					
	(90.8%)	JC_____%					

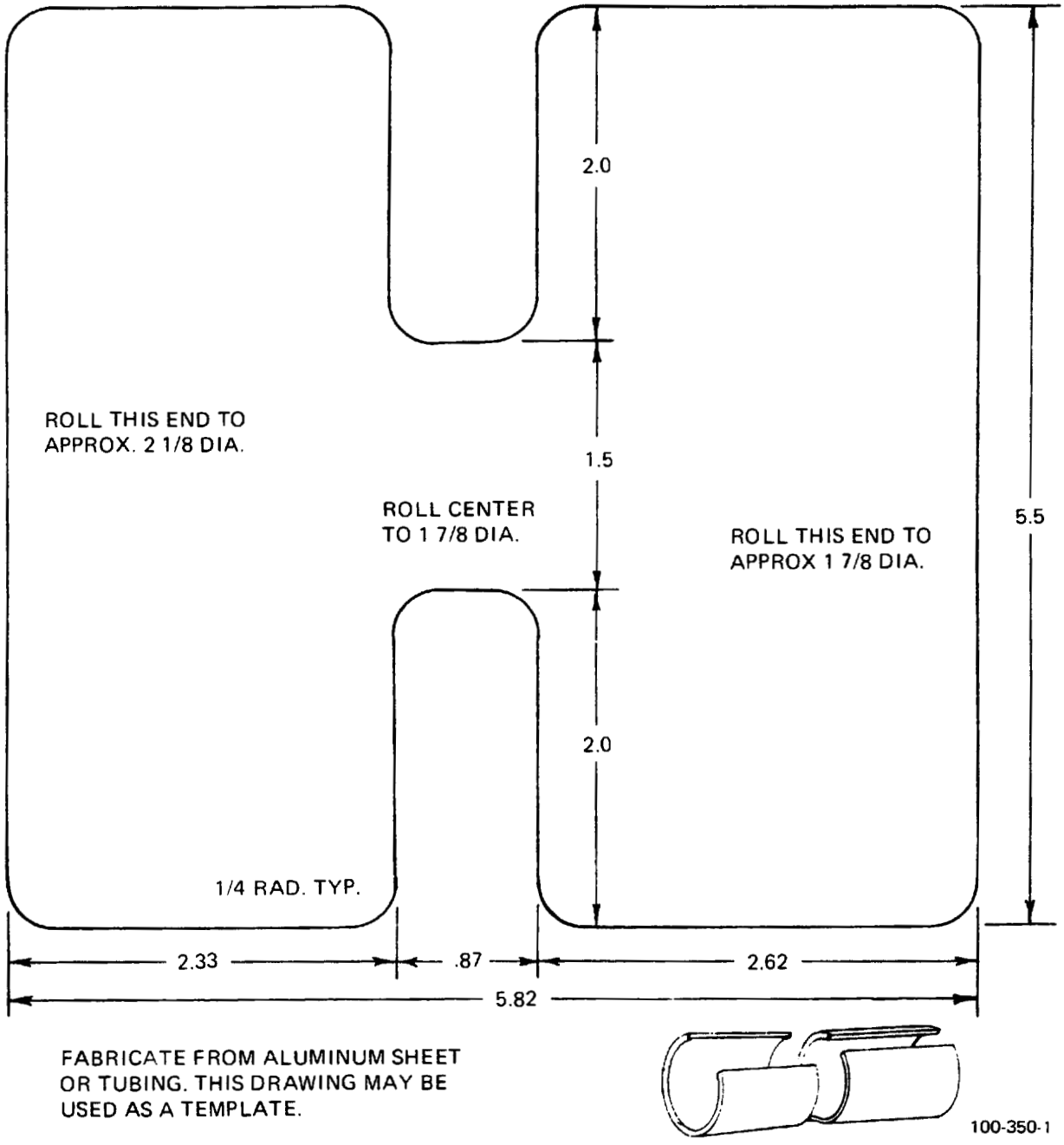


Figure 202
Tachometer Support Bracket

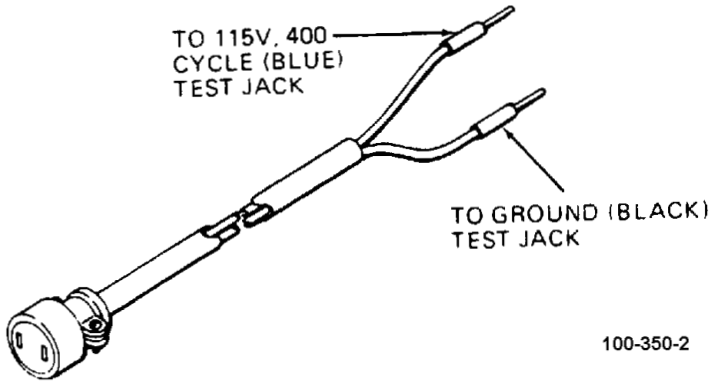


Figure 203
Jetcal Power Cables

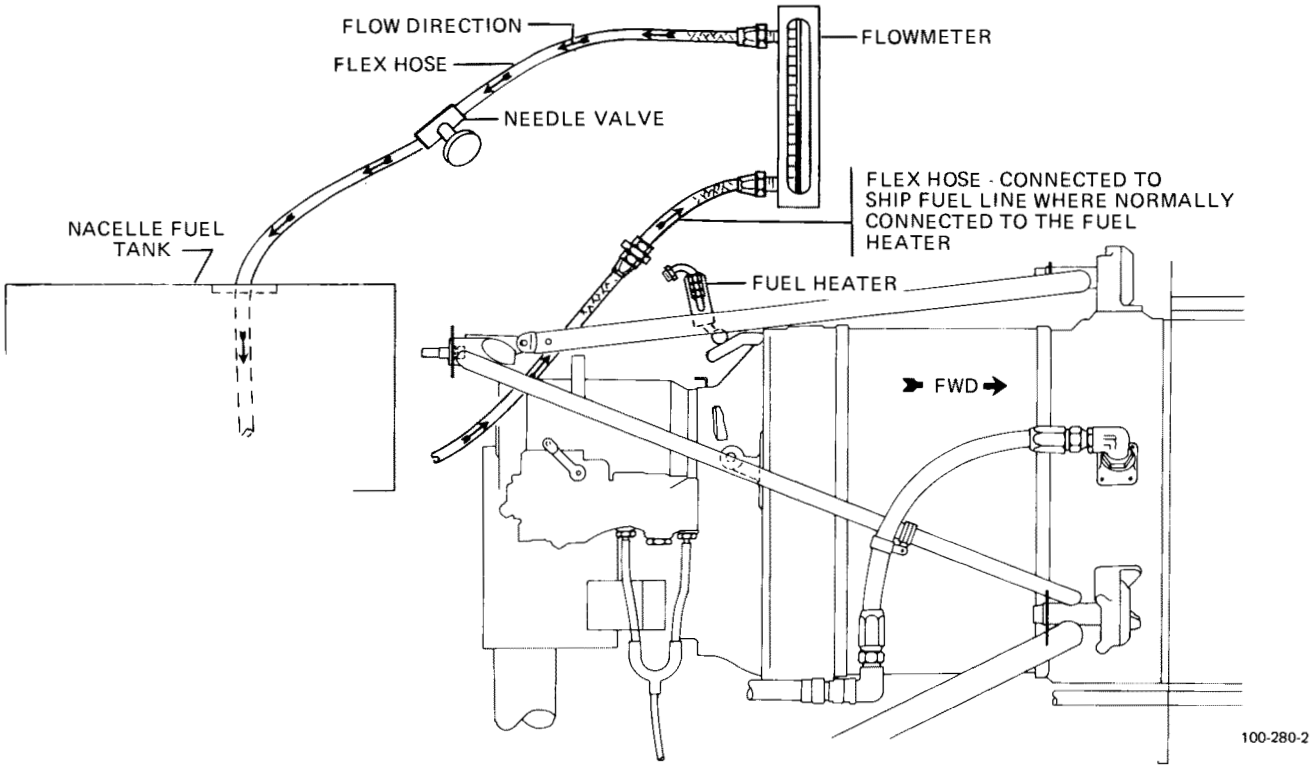


Figure 204
Fuel Flowmeter Test Equipment

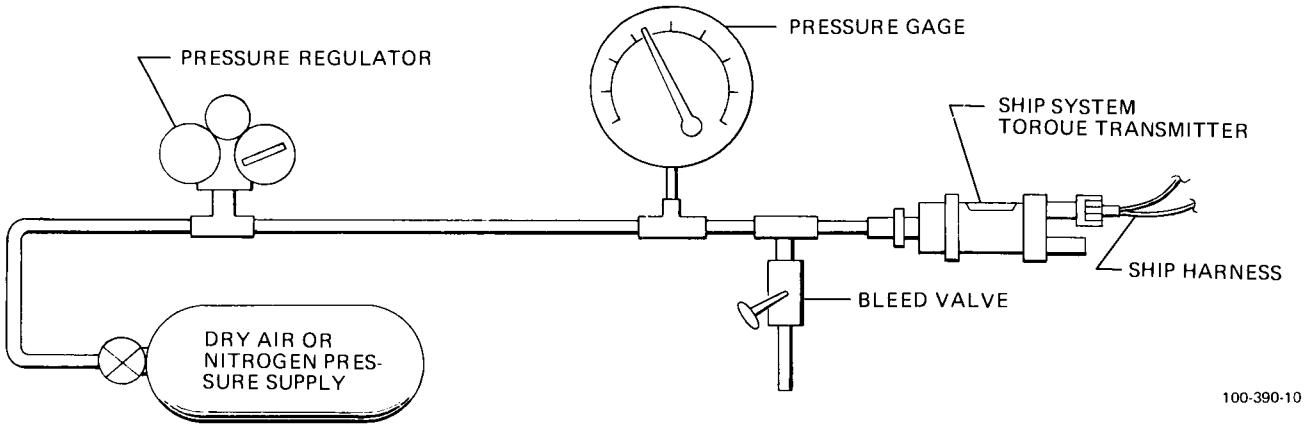


Figure 205
Torquemeter System Test Equipment

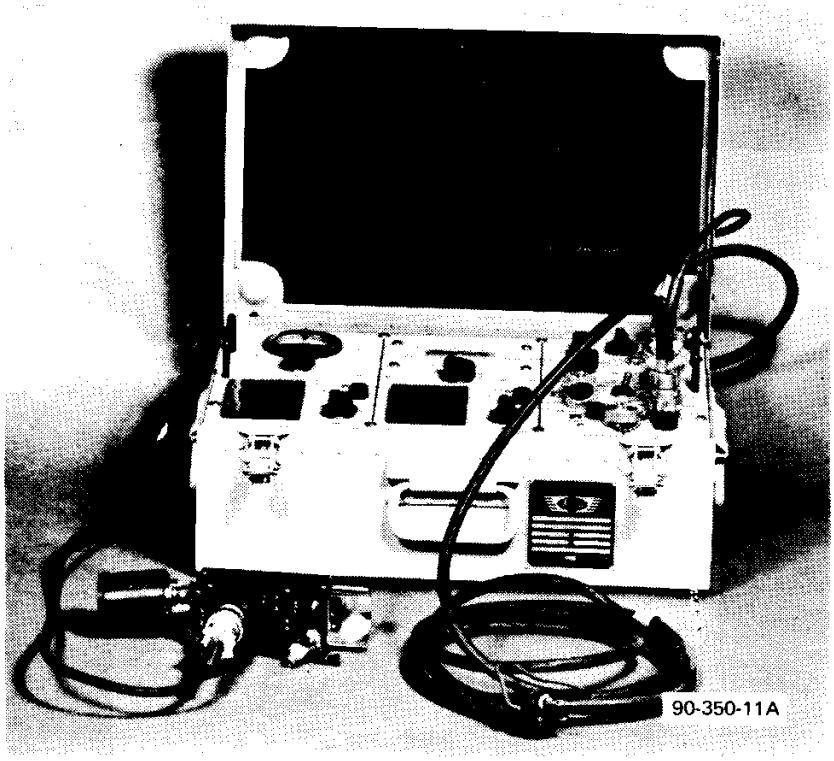


Figure 206
Jetcal Equipment for N_1 and N_2 Check

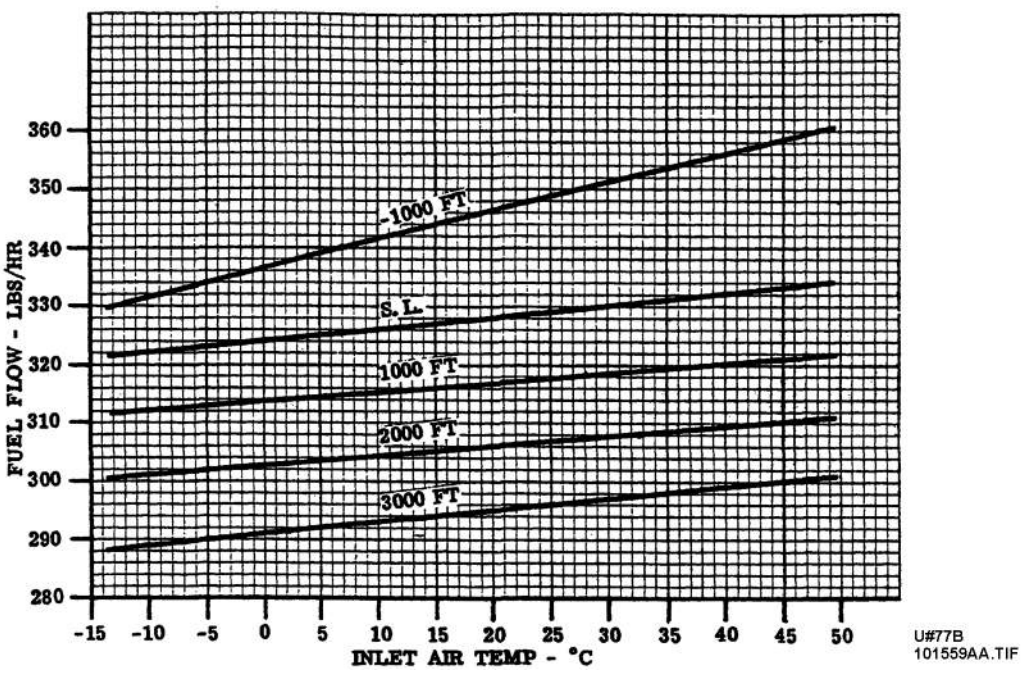


Figure 207
Fuel Flow Chart (PT6A-20 Engines)

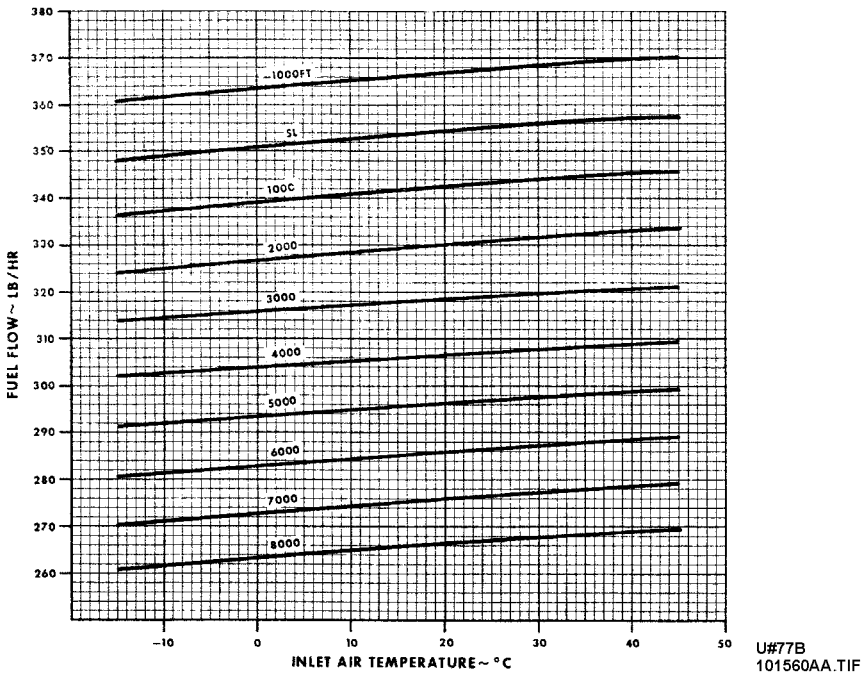


Figure 208
Fuel Flow Chart (PT6A-27 and PT6A-28 Engines)

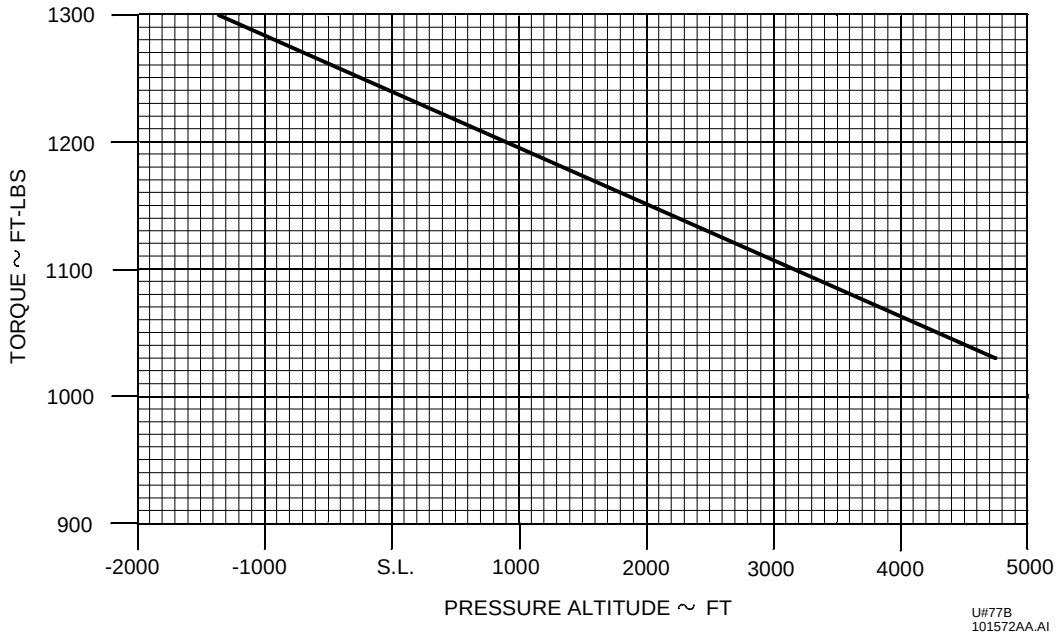


Figure 209
Pressure Altitude Vs Torque (PT6A-20 Engines)

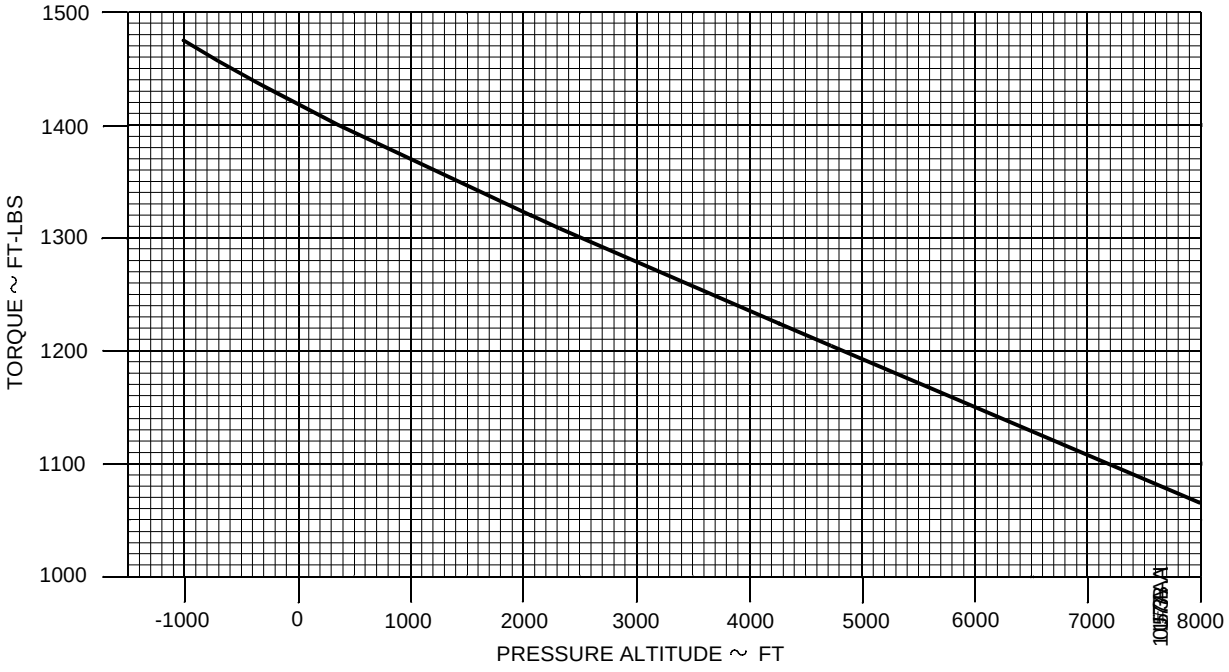


Figure 210
Pressure Altitude Vs Torque (PT6A-27 and PT6A-28 Engines)

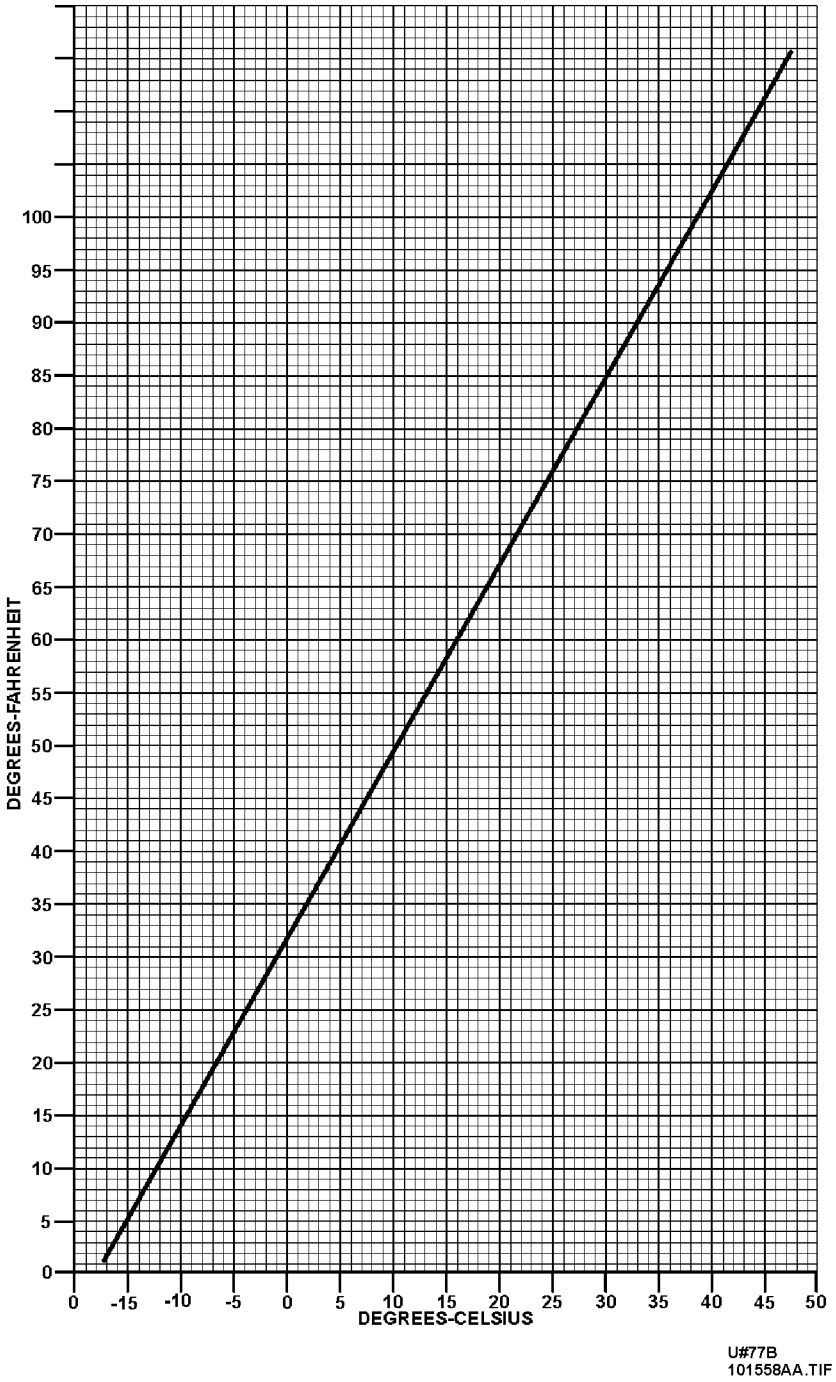


Figure 211
Temperature Conversion Chart

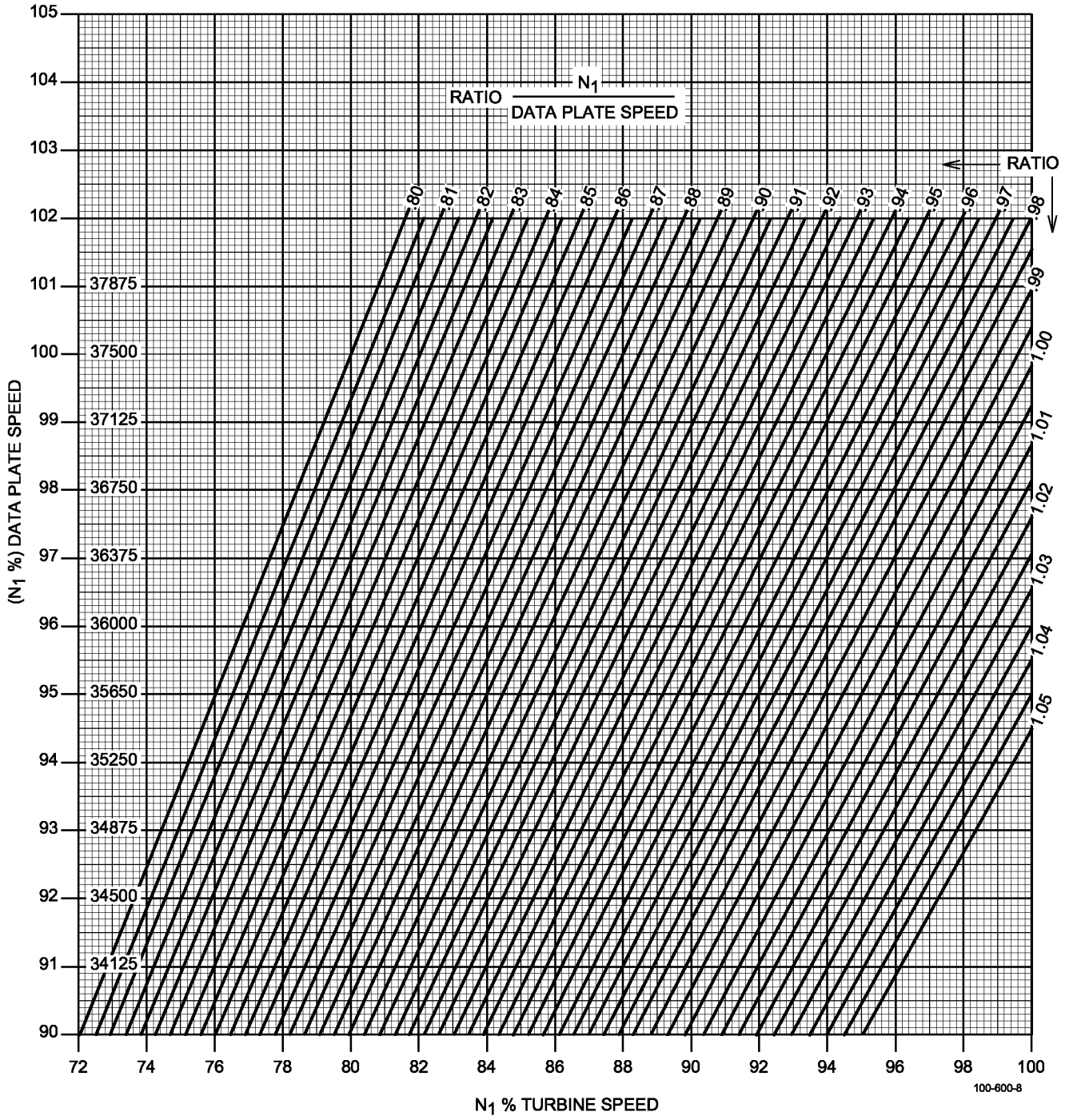


Figure 212
DPS Conversion Chart

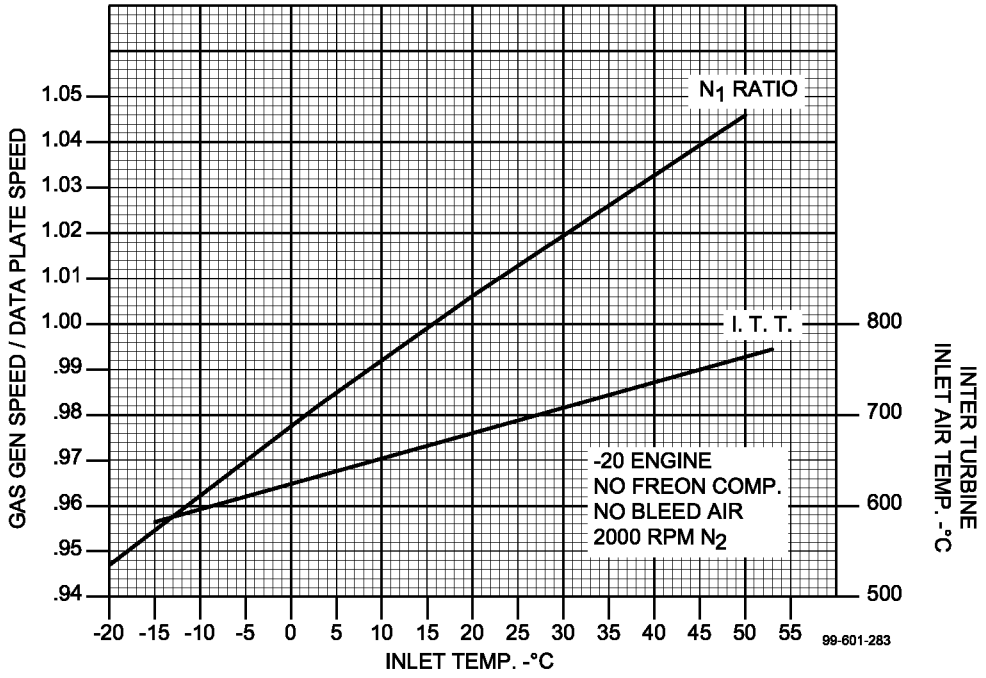
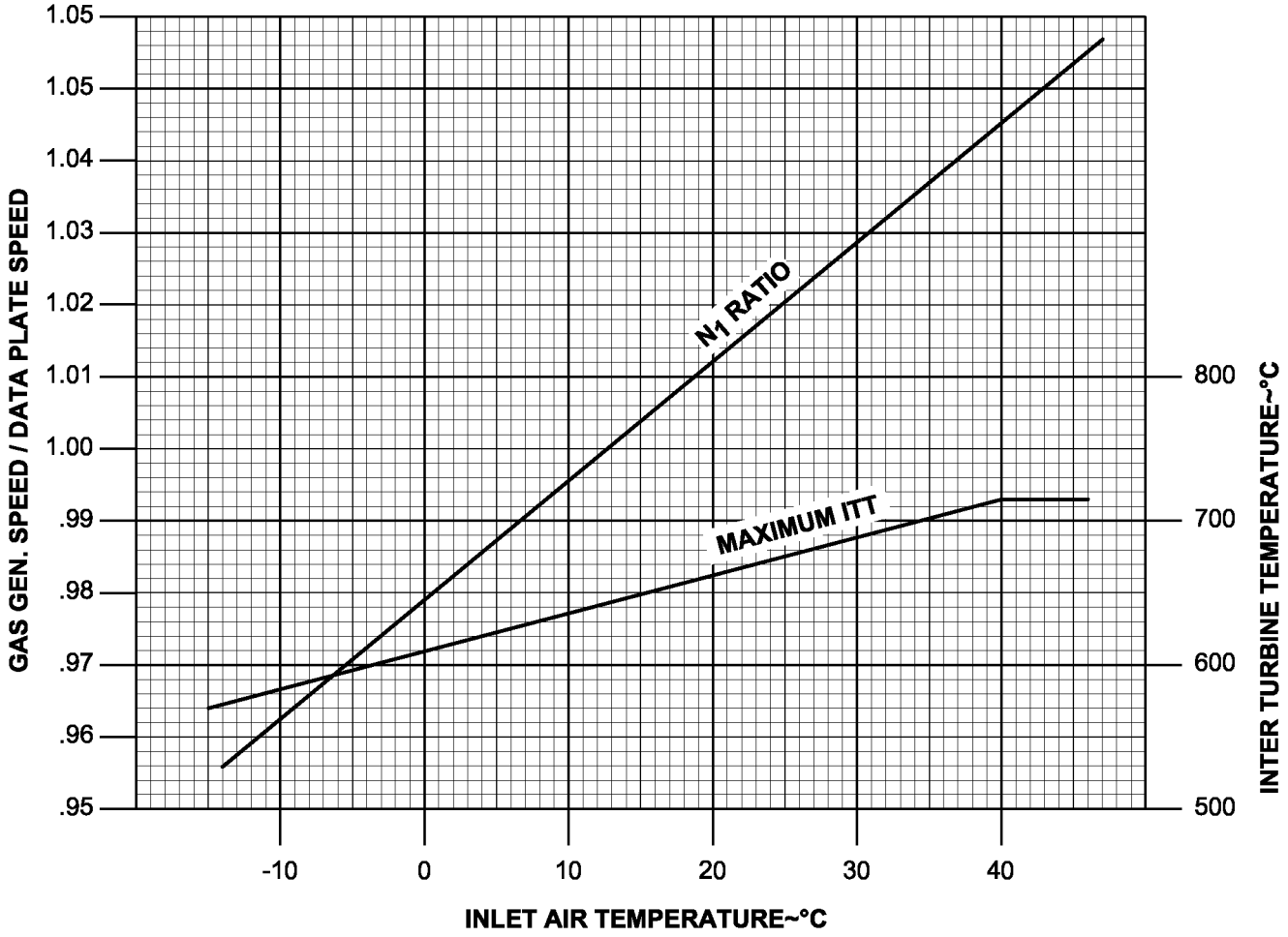


Figure 213
Engine Performance Check Curve (PT6A-20 Engines)

CAUTION

ON SOME DAYS OUTSIDE AIR TEMPERATURES MAY MAKE IT IMPOSSIBLE TO PERFORM THIS ADJUSTMENT WITHOUT EXCEEDING MAXIMUM TORQUE OR TEMPERATURE LIMITS. DO NOT EXCEED THE MAXIMUM TORQUE (1315 FOOT-POUNDS) OR TEMPERATURE LIMITS (750° C FOR FIVE MINUTES) WHEN PREFORMING THIS ADJUSTMENT.



CAUTION

THE I.T.T. VALUES DESIGNATED BY THE CHART ABOVE ARE THE MAXIMUM VALUES THAT MUST NOT BE EXCEEDED. IF THE READING IS MORE THAN 75°C BELOW THE TEMPERATURE DESIGNATED BY THE CHART, CHECK THE INSTRUMENTATION.

99-601-41

Figure 214
Engine Performance Check Curve (PT6A-27 and PT6A-28 Engines)

ENGINE INDICATING
POWER
TROUBLESHOOTING (U-50, U-165 AND AFTER)

The first indication that checks on the engine indicating systems may need to be performed would be that the airplane engine failed to meet the performance parameters of the ENGINE POWER ASSURANCE CHECK in this section or has had a shift in engine performance parameters.

Usually, when only one engine performance parameter is suspect, it can be assumed that a problem in the indicating system for that parameter is present and an immediate check of that indicating system must be performed. Normally, when the fault is in the engine, at least two of the performance parameters will be outside the expected limits. In either case, the engine should never be automatically condemned without a thorough investigation of the appropriate engine indicating systems.

Engine rigging should never be ruled out as a causative factor when an engine failed to meet the minimum performance parameters.

1. ENGINE POWER ASSURANCE CHECK

The engine power assurance check graphs (Ref. Figure 101) are presented as constant torque versus ambient temperature so that engine performance can be checked over a wide range of ambient temperatures without overtorquing the reduction gearbox or exceeding the normal operating limitations of the engine. Periodic engine performance checks must be carried out and changes in engine performance noted. All forms of engine deterioration will be accompanied by an increase in interstage turbine temperature and fuel flow at a given power. Compressor deterioration will, in most cases, be due to dirt deposits and will affect an increase in gas generator speed to obtain a given power setting. This form of deterioration can be remedied by compressor washing as described i n the engine maintenance manual. Hot-section deterioration will cause a decrease in the gas generator speed required for a given power setting.

A. Engine Power Assurance Worksheet

The Engine Power Assurance Worksheet (Ref. Table 101) should be used to record engine performance data. The worksheet may be retained following engine performance checks for the purpose of documentation of acceptable engine performance and trend monitoring.

(1) Set 29.92 in the barometric pressure window of the airplane's altimeter and record the field pressure altitude.

(2) Record the outside air temperature in degrees Celsius.

(3) Obtain the target torque reading from Graph 1 of Figure 101 by plotting temperature against field barometric pressure.

(4) Start the engines, bring the generators on line and allow the oil temperature to increase to normal.

(5) Check that the condition levers are in high idle and propeller levers are in high rpm.

(6) Turn the generator off on the engine to be checked and set up a torque indication equal to the torque value from Graph 1 of Figure 101.

(7) Check that engine propeller rpm is at 2200 rpm.

(8) Record the N_1 reading, ITT indication and the fuel flow from the engine being checked.

(9) Shut down the engines after checks are complete.

(10) Determine the target fuel flow from Graph 2 of Figure 101 by plotting the field pressure altitude against the outside air temperature. The fuel flow during the ground run check should be at, or below, the graph value.

(11) Observed N_1 should be at, or below, the target value taken from Graph 3 of Figure 101.

(12) Observed ITT should be below the ITT taken from Graph 4 of Figure 101.

Table 101
Engine Power Assurance Worksheet

Airplane S/N: _____ Engine S/N: LH _____ RH _____

Technician: _____ Date: _____ Field Elevation:_____

OAT: _____ Pressure Altitude: _____

Engine Start Comments: LH: _____

RH: _____

Engine Oil Temp: LH _____ RH _____

Engine Power Assurance Check (Ref. Figure 101)

Field Barometric Pressure: _____ OAT: _____

Torque from Graph 1: _____ Indicated: LH _____ RH _____

Propeller Speed
(Target 2,200 rpm) Indicated: LH _____ RH _____

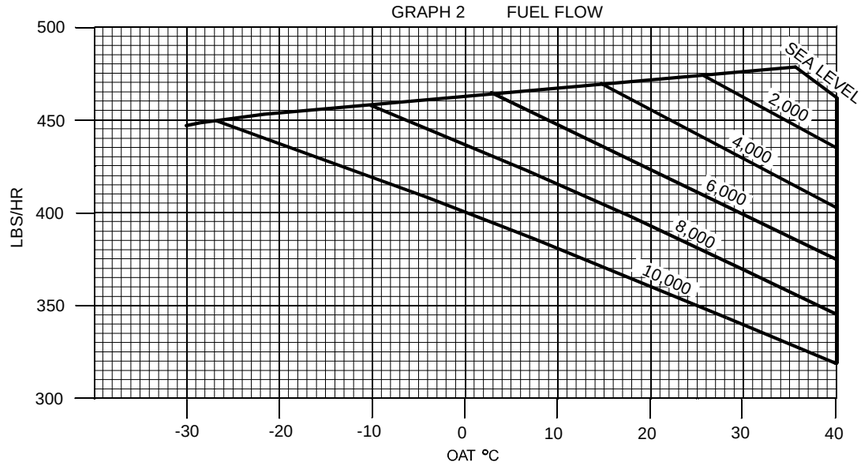
Fuel Flow from Graph 2: _____ Indicated: LH _____ RH _____

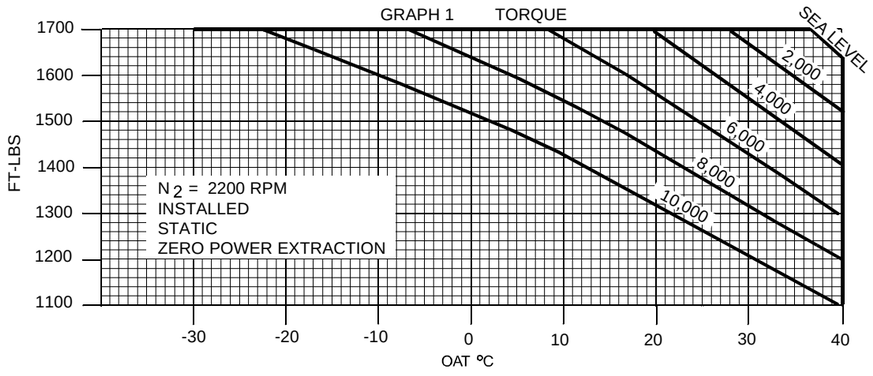
N_1 from Graph 3: _____ Indicated: LH _____ RH _____

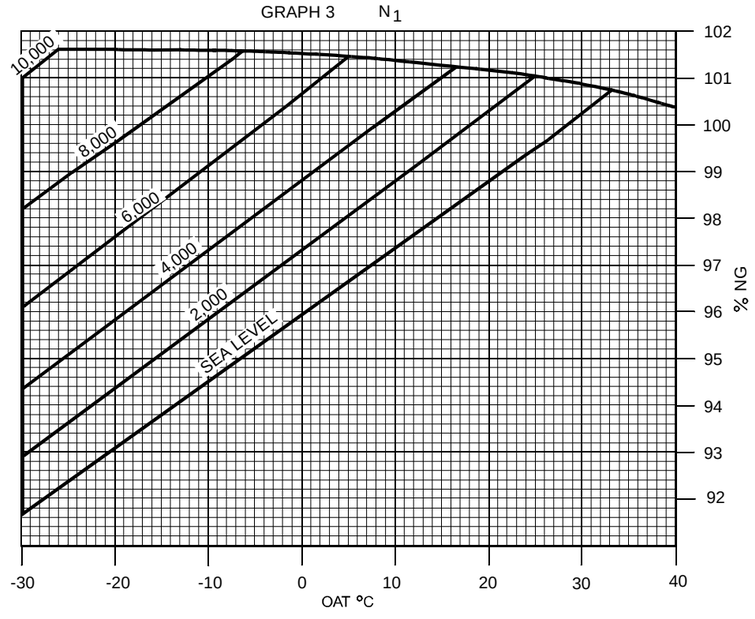
ITT from Graph 4: _____ Indicated: LH _____ RH _____

Engine Shutdown Comments: LH: _____

RH: _____







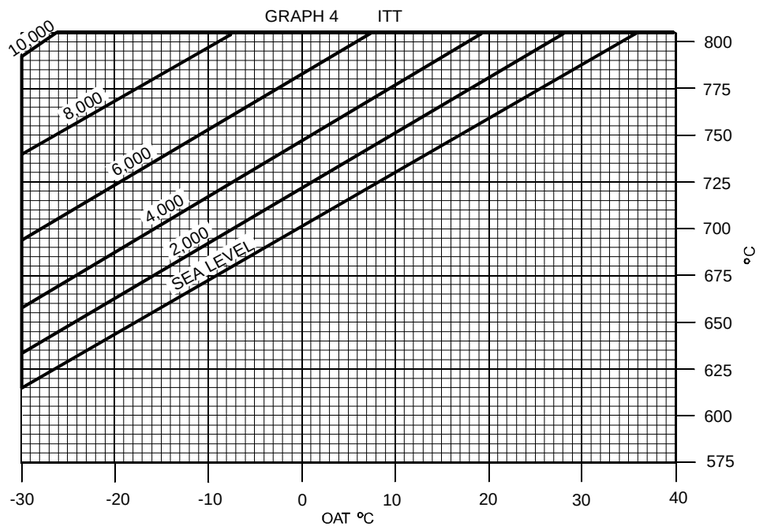


Figure 101
Engine Power Assurance Graphs

ENGINE INDICATING
POWER
MAINTENANCE PRACTICES (U-50, U-165 AND AFTER)

The following checks on the various engine indicating systems are compatible with analyzers and equipment being used in Hawker Beechcraft facilities. Some modifications in these procedures may be necessary if different equipment is used. All pressures, resistances and voltages are standard values and will remain unaffected, regardless of the type of equipment being used.

Use of these analyzers and equipment by Hawker Beechcraft facilities does not necessarily constitute an endorsement by Hawker Beechcraft, as other factors affecting the accuracy of the equipment, such as calibration, frequency of certification or misuse, must be taken into consideration before any equipment may be used to check or calibrate the engine indicating systems

1. RPM INDICATING SYSTEM (N_1 AND N_2)

Gas generator speed (N_1) is indicated as a percentage of the maximum allowable gas generator speed in rpm. The jetcal readout is expressed in percentage as well.

Propeller speed (N_2) is indicated in rpm and will have to be converted to percentage in order to correlate to jetcal readings.

Thee jetcal unit is capable of measuring rotating speeds within ± 0.1%; therefore, the jetcal unit lends itself well to being used as a calibrating standard.

The N_1 and N_2 indicator systems checks must be performed while the engine is running. The airplane inverter system is used to power the jetcal unit; therefore, a power adapter (Ref. Figure 202) will have to be fabricated in the shop to allow using the AC test jack, located beneath the RH subpanel immediately adjacent to the pedestal.

The N_1 or N_2 indicator has no adjustment and must be replaced in the event of a disagreement between the indicator and the jetcal reading.

A. N_1 and N_2 Indicator System Check

(1) Remove the interconnect cable from the jetcal unit and lift the unit out of the carrier.

(2) Place the jetcal unit in a convenient location in the airplane, and connect BH499A power cable to the POWER INTERCONNECT receptacle.

(3) Connect the shop fabricated power adapter to the power cable, and connect the unit to the black and blue AC test jacks under the RH subpanel.

(4) Place all switches on the jetcal unit in the OFF position.

(5) Connect the BH485A instrument cable to the jetcal unit.

(6) Connect the rpm check adapter BH820A to the instrument cable.

(7) Remove the N_1 or N_2 indicator from the instrument panel, and connect the rpm check adapter in series between the indicator and the indicator harness.

(8) Place the RPM switch in the N_1 or N_2 position.

(9) Start the engine according to the PILOTS OPERATING HANDBOOK and bring the generator on line.

(10) Turn on the airplane inverter and the power switch on the jetcal unit.

(11) Compare the N_1 or N_2 reading on the indicator to the N_1 or N_2 reading from the jetcal, and record these on the worksheet in Table 201. When making N_2 checks, refer to the PROPELLER RPM CONVERSION TABLE in Table 201 for converting propeller rpm to percentage.

(12) Place a l l switches i n the OFF position before disconnecting any cables from the airplane or the jetcal unit.

2. TORQUE INDICATING SYSTEM

The torque transmitter, attached to the torque manifold on the LH side of the reduction gearbox (Ref. Figure 203), senses pressure inside the reduction gearbox and transmits a signal to the torque indicator. The torque indicator displays torque applied to the propeller shaft in foot-pound units.

The torque transmitter and indicator is functionally tested by simulating gearbox pressure. Shop air pressure is applied to the torque manifold through a torque calibration unit.

The torque calibration unit is a precision pressure regulator and valve assembly. The face of the instrument is calibrated in pounds per square inch. While the calibration unit referenced in this chapter is accuracy certified, any precision pressure regulator accurate to within $\pm$ 0.25 psi in the range of 0 psi to 100 psi may be used in calibrating the torque indicator system.

The torque indicating system response to gearbox pressure is a linear one and translates to approximately 35.7 ft-lbs of torque to 1 psi of gearbox pressure.

The worksheet in Table 201 should be used to record test data and may be retained in the airplanes files for the purpose of documentation.

Adjustments to the torque indicating system may be accomplished by adjusting the adjustment screw which is under a plug in the case of the transmitter.

A. Torque Indicator Calibration Check

(1) Disconnect the vent line from the torque transmitter (Ref. Figure 203).

(2) Disconnect the elbow fitting from the torque manifold and connect the torque calibration unit to the manifold.

(3) Turn both valves on the torque calibration unit off and connect shop air to the unit.

(4) Connect an APU to the airplane with an output of 28 vdc.

(5) Turn the battery switch on and bring one of the inverters on line.

(6) Turn on the pressure source valve on the torque calibration unit.

(7) Adjust the pressure adjust valve to each of the values on the worksheet (Ref. Table 201) and record the torque readings.

(8) Calibration adjustments to the torque transmitter can be made by adjusting the screw under the adjustment plug.

3. FUEL FLOW INDICATING SYSTEM

The fuel flow indicating system makes use of a fuel flow transmitter and a precision indicator mounted in the instrument panel. The transmitter measures the volume of fuel actually flowing to the injectors. The volumetric measurement is converted to mass flow (pounds per hour, pph) at the rate of 6.74 pounds per gallon.

The fuel flow transmitter has two adjusting potentiometers, one for adjusting the slope of the scale and one for adjusting the offset of the scale. The slope adjusting potentiometer is rectangular in shape and the offset adjusting potentiometer is round. The adjusting pots are mounted on a printed circuit board which is concealed under a cover plate on the body of the fuel flow transmitter.

Adjusting the slope potentiometer will cause a greater change in the flowmeter reading at the top end of the scale than it will at the bottom end of the scale; the amount of change resulting from adjusting the slope potentiometer is directly proportional to the fuel flow indications. Adjustment of the offset potentiometer will result in the same amount of change in pph throughout the entire scale of the flowmeter. Usually a combination of adjustments of both potentiometers will be required to affect the best calibration.

A. Fuel Flow Indicator Calibration Check

(1) Disconnect the fuel line running between the fuel flow transmitter and the fuel heater from the fuel heater.

(2) Connect a flowrater, equipped with a needle valve for controlling the flow, to the fuel line.

(3) Run a line from the flowrater outlet back into the fuel tanks so that any fuel used during calibration will return to the tanks.

(4) Connect an APU with an output of 28 vdc to the airplane and turn the external power switch and the battery switch on.

(5) Turn the appropriate fuel standby pump on.

(6) Open the valve on the flowrater and purge all the air from the flowrater and associated lines.

(7) Adjust the flow rate on the flowrater to 150 pph and record the flowmeter indication.

(8) Adjust the flow rate on the flowrater to 450 pph and record the flowmeter indication.

(9) Any disagreement between the flowrater and the airplane flowmeter must not exceed $\pm$ 10 pph. If the calibration is out of tolerance, remove the lockwire from the four screws securing the cover of the flow transmitter and remove the cover.

(10) Locate the two adjusting potentiometers (Ref. Figure 204).

(11) Carefully remove the Glyptol sealant from the adjusting screws on the potentiometers.

(12) If the amount of error at the 150 pph flowrater setting is equal to the amount of error at the 450 pph flowrater setting, adjust the offset potentiometer. If the amount of error at the 150 pph setting was approximately one-third of the amount of error at the 450 pph setting, adjust the slope potentiometer.

CAUTION: *Before disconnecting the electrical connector from the fuel flow transmitter or before removing the flowrater from the system. Turn the Battery switch, external power switch and the standby fuel pump switch off.*

(13) Repeat the above checks and adjustments until the flowmeter indications agree with the flowrater settings within 10 pph. If the flow transmitter cannot be adjusted properly, swap out the indicator and repeat the calibration procedure. If proper calibration still cannot be affected, replace the fuel flow transmitter.

(14) After adjusting the transmitter, reseal the potentiometers with Glyptol or an equivalent sealant.

(15) Install the cover on the transmitter, using RVP sealant for water proofing, and safety wire the retaining screws.

(16) Remove the flowrater and connect the fuel line to the fuel heater.

4. JETCAL CHECKS

Perform the Inter Turbine Temperature (ITT) Indicating System Checks before starting this procedure (Ref. 77-21-00).

A. Running Jetcal Indicator Checks

Occasionally, vibration, flight loads or varying flight attitudes may induce faults of an intermittent nature which may not show up in routine indicator system checks. It may be desirable, at times, to perform jetcal checks while the airplane engines are running, or perhaps while the airplane is in flight.

Performance graphs in the appropriate Pilot's Operating Handbook may be used to set up engine power as a reference for in-flight checks. Power assurance graphs in this Chapter (Ref. Figure 101), should be used for reference when performing static ground checks.

The following procedure provides instructions for connecting the jetcal unit to the airplane indicating systems for either static ground checks or the i n-flight checks.

(1) Place all switches on the jetcal unit in the OFF position.

(2) Remove the power interconnect cable, lift the jetcal unit out of the carrier and place the unit in a convenient position in the airplane.

(3) Connect the BH499A power cable to the POWER INTERCONNECT receptacle.

(4) Connect the shop fabricated power adapter (Ref. Figure 202) to the power cable, and connect the unit to the black and blue AC test jacks under the RH subpanel.

(5) Connect the BH485A instrument cable to the RPM INPUT INSTRUMENT CABLE receptacle on the jetcal unit.

(6) Connect two rpm check adapters BH820A to the rpm cable.

(7) Connect the two rpm check adapters in series with the N_1 and N_2 indicators. A support bracket (Ref. Figure 205) may be shop fabricated to support the indicators in the instrument panel.

(8) Connect the BH450 check cable to the CHECK CABLE receptacle on the jetcal unit.

(9) Connect the check cable to the top receptacle of the BH134 switch box.

(10) Remove the ITT indicator from the instrument panel and connect the indicator to the switch box.

(11) Connect the airplanes ITT harness to the rear receptacle of the switch box and slip the switch box into the instrument panel ITT opening.

(12) With the engines running and the battery and inverter on, turn on the power switch on the jetcal unit.

(13) Place the TEMPERATURE switch on the jetcal unit in the OPERATE position.

(14) Place the RPM switch in either the N_1 or N_2 position.

(15) Refer to the ENGINE POWER ASSURANCE CHECK procedure for instructions in setting up engine power.

(16) A switch on the ITT switch box is used to select readings from either the indicator or the jetcal unit.

(17) Rpm readings occur simultaneously on the indicator and the JETCAL unit; however, it is necessary to select either N_1 or N_2 readings with the RPM switch on the jetcal unit.

Table 201
PT6A-36 ENGINE INDICATING CHECK WORKSHEET

DATE: _____ AIRPLANE S/N _____ ENGINE S/N LH _____ RH _____

TECHNICIAN _____ COMMENTS: _____

ITT INDICATING SYSTEM

LOOP RESISTANCE: LH _____ RH _____ 8 ohms ± 0.05

HARNESS INSULATION: LH _____ RH _____ Greater than 5,000 ohms

ITT INDICATOR

	LEFT S/N_____	RIGHT S/N_____
JETCAL°C TOL	LH ITT	RH ITT
500 ± 20	_____	_____
600 ± 10	_____	_____
700 ± 5	_____	_____
800 ± 10	_____	_____
900 ± 20	_____	_____

RPM INDICATING SYSTEM

N_1 INDICATOR

JETCAL % TOL	LH N_1	RH N_1
55 ± 1	_____	_____
70 ± 1	_____	_____
85 ± 1	_____	_____
97 ± 1	_____	_____

N_2 INDICATOR

JETCAL % TOL	LH N_2 RPM	LH N_2 %	RH N_2 RPM	RH N_2 %
_____ ± 1	_____	_____	_____	_____
_____ ± 1	_____	_____	_____	_____
_____ ± 1	_____	_____	_____	_____

Table 201
PT6A-36 ENGINE INDICATING CHECK WORKSHEET (Continued)

TORQUE INDICATING SYSTEM

TEST PRESSURE	TORQUE	TOL	LH IND TORQUE	RH IND TORQUE
10	307	260 to 360	_____	_____
20	614	580 to 650	_____	_____
30	922	890 to 950	_____	_____
35	1075	1050 to 1100	_____	_____
42.5 (EZ)	1305.6	1300 to 1310	_____	_____
55	1690	1660 to 1720	_____	_____
60	1843	1800 to 1880	_____	_____

FUEL FLOW INDICATING SYSTEM

FLOWRATOR PPH	TOL	LH IND	RH IND
300	± 10	_____	_____
400	± 10	_____	_____
500	± 10	_____	_____
600	± 10	_____	_____

PROPELLER RPM CONVERSION TABLE

PROP RPM	JETCAL %	PROP RPM	JETCAL %
2250	102.3	1750	79.5
2200	100.0	1700	77.3
2150	97.7	1650	75.0
2100	95.5	1600	72.7
2050	93.2	1550	70.5
2000	90.9	1500	68.2
1950	88.6	1450	65.9
1900	86.4	1400	63.5
1850	84.1	1350	61.4
1800	81.8	1300	59.1

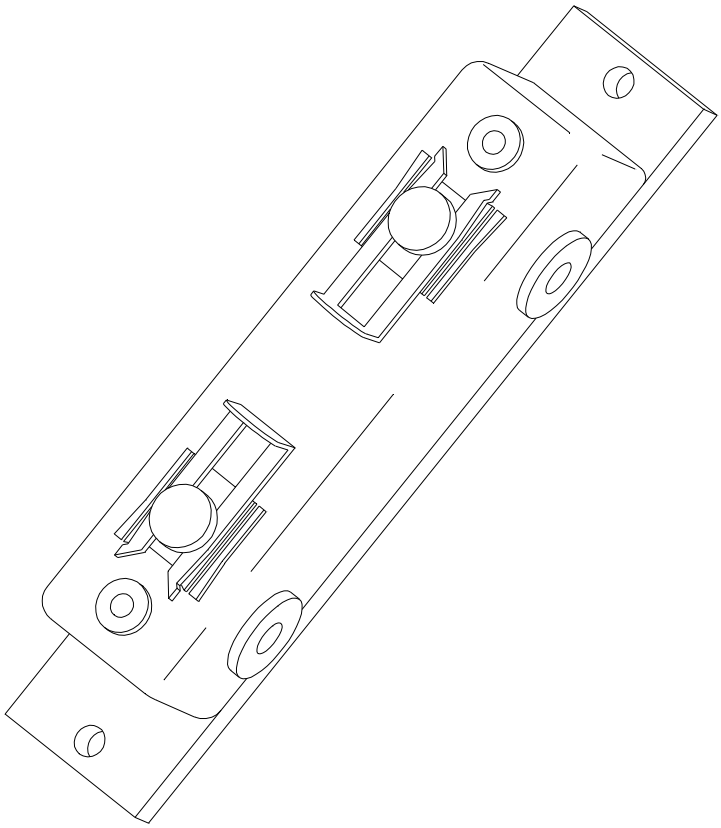


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Figure 201
Compensator Spool Assembly

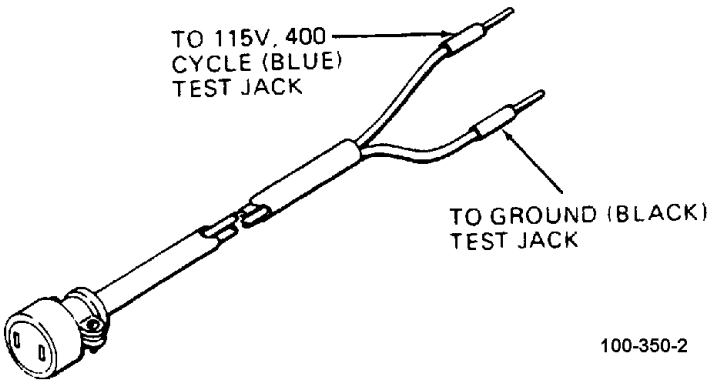

Figure 202
Power Adapter

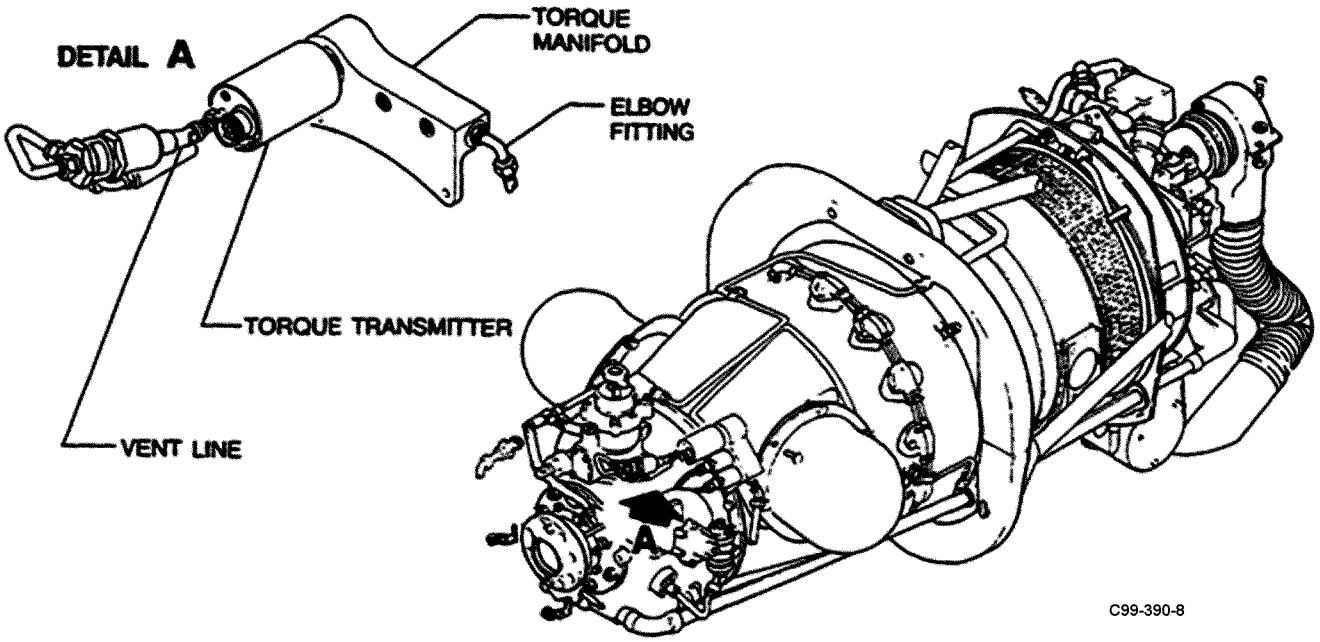

Figure 203
Torque Manifold-Transmitter

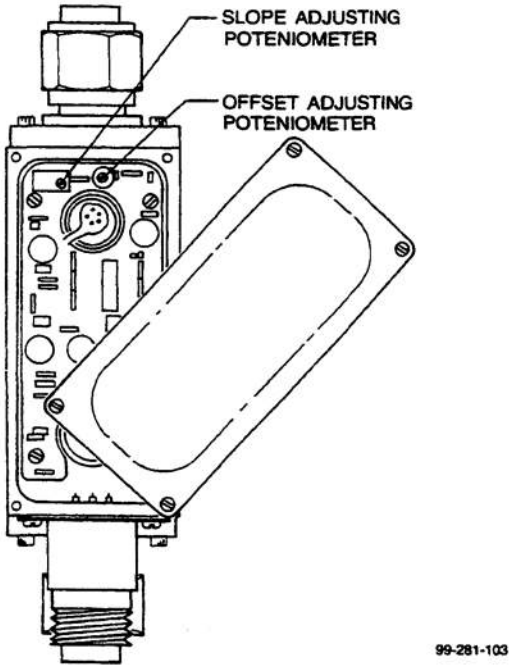

Figure 204
Fuel Flow Transmitter Adjustments

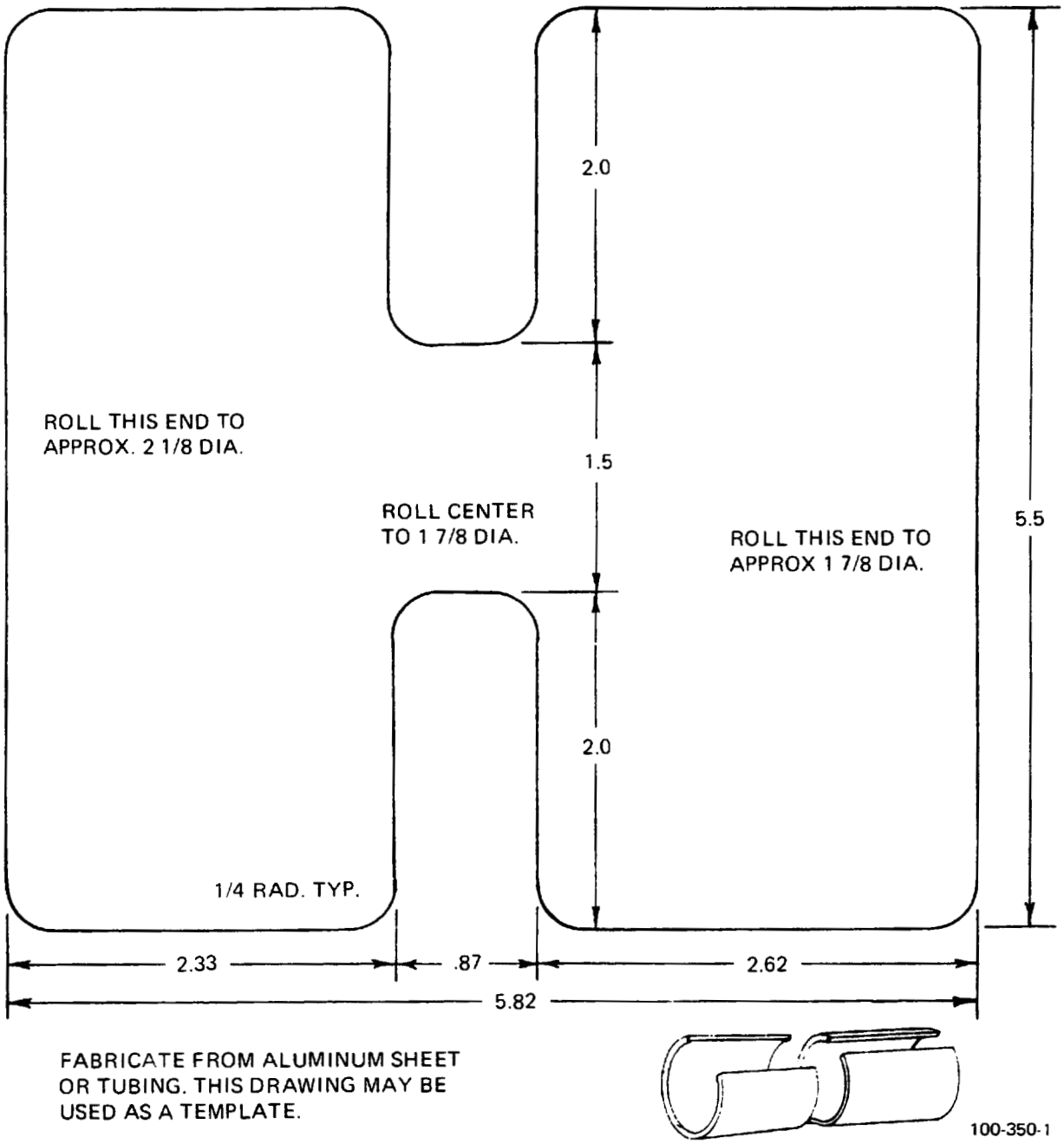

Figure 205
Indicator Support Bracket

ENGINE INDICATING
TEMPERATURE
MAINTENANCE PRACTICES (U-1 THRU U-164, EXCEPT U-50)

1. GROUND TESTS (WITH JETCAL)

A. Equipment Required

The following equipment is used at the factory in checking engine performance. Any equipment meeting the specified accuracy requirements may be used.

Accuracy Requirements	
N_1 RPM	± 0.1%
N_2 RPM	± 0.1%
T_5 TEMP	± 5.5°C

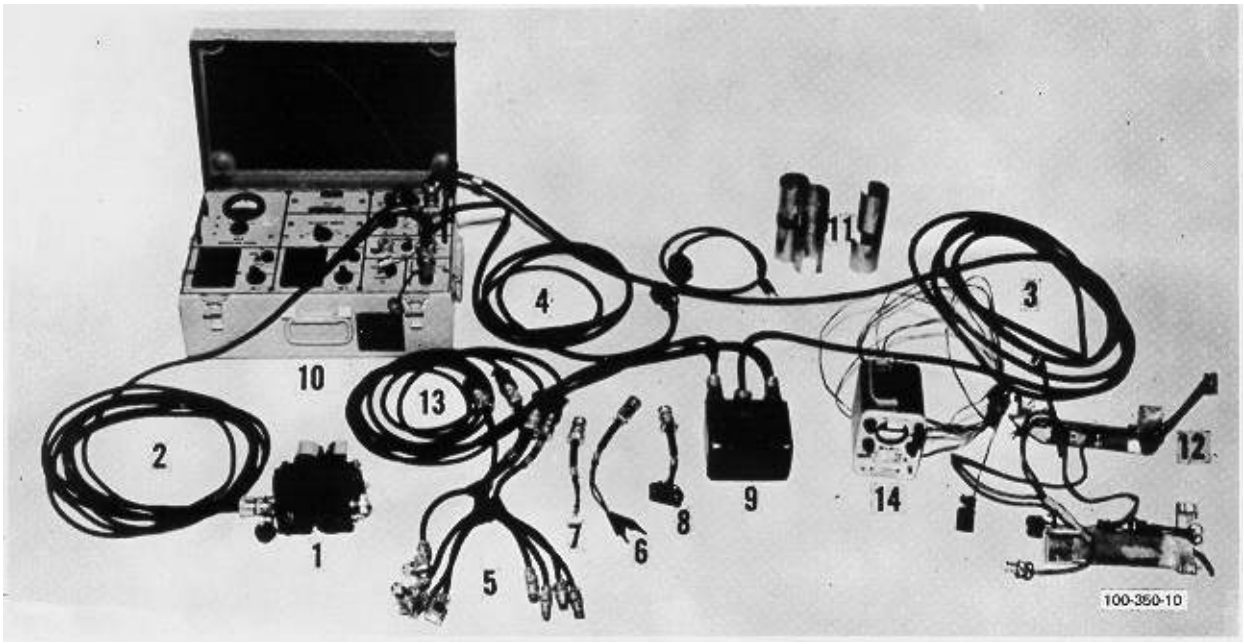

Figure 201
Ground Test Equipment

NOTE: An order for BH112JR Jetcal Unit called out in (10), will include the items listed in (1) thru (9).

Item	Quantity
(1) *BH134 Switch Box	2 each
(2) *BH450 Check Cable	1 each
(3) *BH485 Instrument Check Cable	1 each
(4) *BH499 Power Cable	1 each

	Item	Quantity
(5)	*BH820 Tachometer Check Adapter	4 each
(6)	*BH821 Insulation Check Adapter	1 each
(7)	*BH822 EGT Indicator Adapter	1 each
(8)	*BH223 Resistance Check Adapter	1 each
(9)	*BH5165 RPM Check Box	1 each
(10)	*BH112JR Jetcal Unit	1 each
(11)	Tachometer Holding Brackets	4 each
(12)	Inlet Air Temperature Thermocouple Harness (P/N 50-000065 TC935) or equivalent	1 each
(13)	Adapter Cable (4 Conductor Cable No. 18 gage wire used with BH820 RPM Check Cables) Consists of: 6 feet in length (4 required) MS3106A14S2S Plug (4 required) MS3106A14S2P Plug (4 required) AN3057-6 Plug (8 required)	4 each
(14)	Potentiometer Pyrometer, Model No. 80200 with a range of 0° to 600°F (product of Thermo Electric Co., Inc., Saddle Brooke, New Jersey) or equivalent	1 each
(15)	“Y” Cable (for Toque Indicator and Transmitter Check) Consists of: MS3106A10SL3P Plug (2 required) MS3102A10SL3P Plug (1 required)	
(16)	Pressure Gage with range of 0 to 50 psi and accuracy of 1/4% throughout range (Model 0-60 GP-65.8 by Ashcraft is satisfactory) Regulated nitrogen or dry air supply for calibration of the torque system is also required	1 each
(17)	Flowmeter (Capable of 1% accuracy in 100 to 400 pph range) Flowrator, Model 10A 2735CA (product of Fisher and Porter, Warminister, Pennsylvania) with tube No. FP 1/2 27G 10/80 6509 A1 671 A1 is satisfactory	1 each
(18)	Power Cable (Ref. Figure 202)	1 each

* Products of Howell Instruments Inc. 3479 West Vikery, Fort Worth. Texas

2. INSTRUMENT CALIBRATION

Perform the following tests before the power check is initiated.

A. ITT Calibration

NOTE: The following resistance, insulation and indicator checks may be accomplished as described in the operating instructions located in the lid of the Jetcal unit. Do not perform these checks unless the components being checked and the Jetcal unit are at the same temperature. If possible, store them together for 12 hours before initiating the following checks:

(1) Resistance Check

(a) Remove the ITT indicators from the instrument panel.

NOTE: If thermocouple resistance is not within acceptable limits, perform an INSULATION CHECK and/or the ITT THERMOCOUPLE HARNESS FUNCTIONAL CHECK that follow this procedure to eliminate other possible causes for the resistance discrepancy prior to initiating the adjustment in the following Step.

(b) Connect the Jetcal unit to the thermocouple lead wire for the left engine through cable BH485 and adapter BH823 as indicated in Figure 203. Record the resistance reading for the thermocouple harness on Table 201. Repeat this check on the thermocouple harness for the right engine. The resistance registered for each thermocouple harness should be 8 ± 0.05 ohms.

NOTE: Do not cut off the excess wire. The best procedure is to insulate the excess wire with tape and stow it after the proper resistance has been obtained. This will facilitate later resistance adjustments without necessitating replacement of the balance resistor.

(c) Remove the center section leading edge to gain access to the balance resistor installed at the inboard end of the center section. Adjust the balance resistor (PM AN5534-2) until a resistance of 8 ± 0.05 ohms is obtained for the thermocouple harness. To adjust harness resistance, disconnect the wire of the thermocouple lead spool on the balance resistor at the terminal and reconnect it so that there is less (to reduce resistance) or more (to increase resistance) spool wire in the circuit with the harness. Remove the insulated coating at the point where the new solder connection is made. The balance resistor incorporates a second thermocouple lead spool for additional resistance if required.

(2) Insulation Check

(a) Using the Jetcal unit and cable BH485, check the insulation resistance to ground by connecting one lead of adapter BH821 to the thermocouple harness and the other lead to the airplane structure as indicated in Figure 204. The insulation meter should indicate a reading of 5,000 ohms or more. Perform this check on both the left and right engine harnesses.

(b) Check for shorts between the leads by disconnecting both leads of adapter BH821 from the harness. The resistance indicated should be 5,000 ohms or more. After this check has been completed, connect the thermocouple harness to the engine harness.

(3) ITT Indicator Check

(a) Connect the Jetcal unit to the ITT indicator through cable BH485 and adapter BH822 (Ref. Figures 206 and 207).

NOTE: Do not attempt to reset the indicator until the indicator and Jetcal unit have stabilized at the same room temperature If possible, store the two together overnight before resetting.

(b) Check each indicator at the temperature points indicated in Table 201.

(c) If the ITT indicator requires adjustment, perform the same check noted in the preceding Step with the signal strength from the Jetcal unit set at 700°C and the indicator adjusted to match that setting.

(4) Functional Check of the ITT Thermocouple Harness

(a) Remove the external leads from the gas generator case. Check the insulation resistance of the harness at the terminal block with the recommended test set (Barfield 2312G-8 or equivalent). The minimum acceptable resistance from the terminals to ground is 0.5000 ohms.

If several probes are broken or damaged internally, the loop resistance would not necessarily fall outside the required tolerance; however, erroneous temperature indications could occur because of the resultant unbalance in the harness circuit.

(b) Check the continuity and loop resistance across the terminals of the terminal block. This resistance must be 0.58 to 0.74 ohm.

(c) To check for the presence of defective probe(s), it is necessary to remove the power section as indicated in the Engine Maintenance Manual and check each probe as follows:

1 Cut the lockwire and remove one of two screws securing the leads to the terminals of the probe to be tested.

2 Check for a resistance of 0.24 to 0.46 ohm across the probe terminals.

3 Check for a minimum resistance of 5,000 ohms between either terminal to the vane housing.

4 Replace defective probe(s) in accordance with the instructions given in the Engine Maintenance Manual.

Table 201
ITT System Check

Airplane S/N_____

Date _____ **Checked By**_____ **Inspector**

Harness Resistance

Left _____ 8 ± 0.05 ohms

Right _____ 8 ± 0.05 ohms

Insulation Check

Left _____

Right _____

Must be over 20,000 ohms

Indicator Check

Left S/N _____ Right S/N _____

PT6A-20		PT6A-27/PT6A-28			PT6A-20		PT6A-27/PT6A-28		
ITT	TOL	ITT	TOL	Jetcal Reading	ITT	TOL	ITT	TOL	Jetcal Reading
500°C	± 20	200°C	± 30	_____	500°C	± 20	200°C	± 30	_____
600°C	± 10	400°C	± 20	_____	600°C	± 10	400°C	± 20	_____
660°C	± 7.5	640°C	± 10	_____	660°C	± 7.5	640°C	± 10	_____
700°C	± 5	700°C	± 5	_____	700°C	± 5	700°C	± 5	_____
760°C	± 7.5	760°C	± 10	_____	760°C	± 7.5	760°C	± 10	_____
800°C	± 10	1000°C	± 20	_____	800°C	± 10	1000°C	± 20	_____
900°C	± 20	1200°C	± 30	_____	900°C	± 20	1200°C	± 30	_____

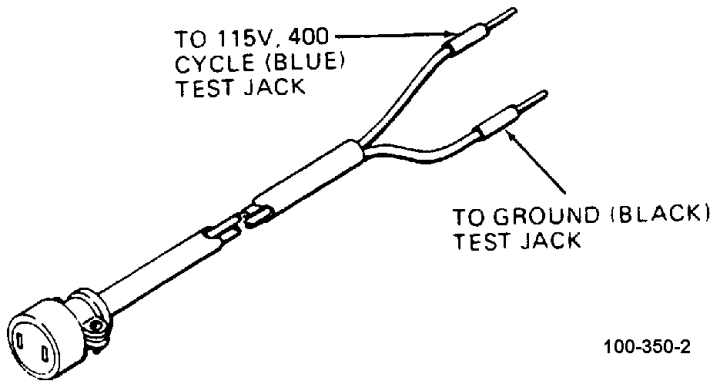

Figure 202
Jetcal Power Cables

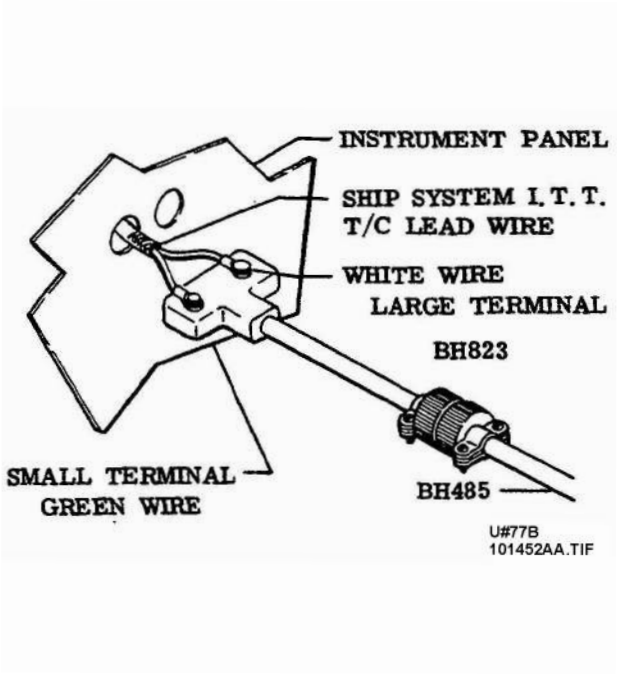

Figure 203
Resister Check Cable

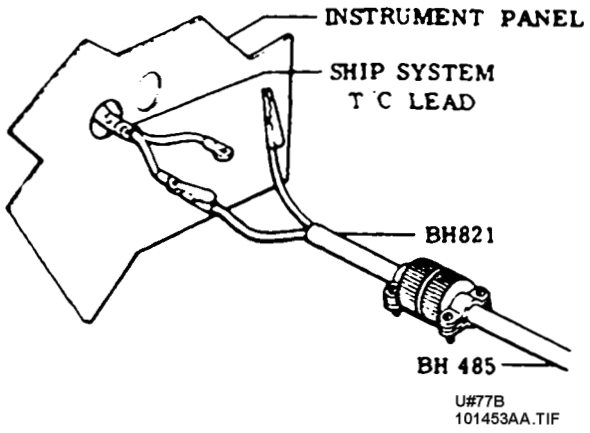

Figure 204
Insulation Check Cable

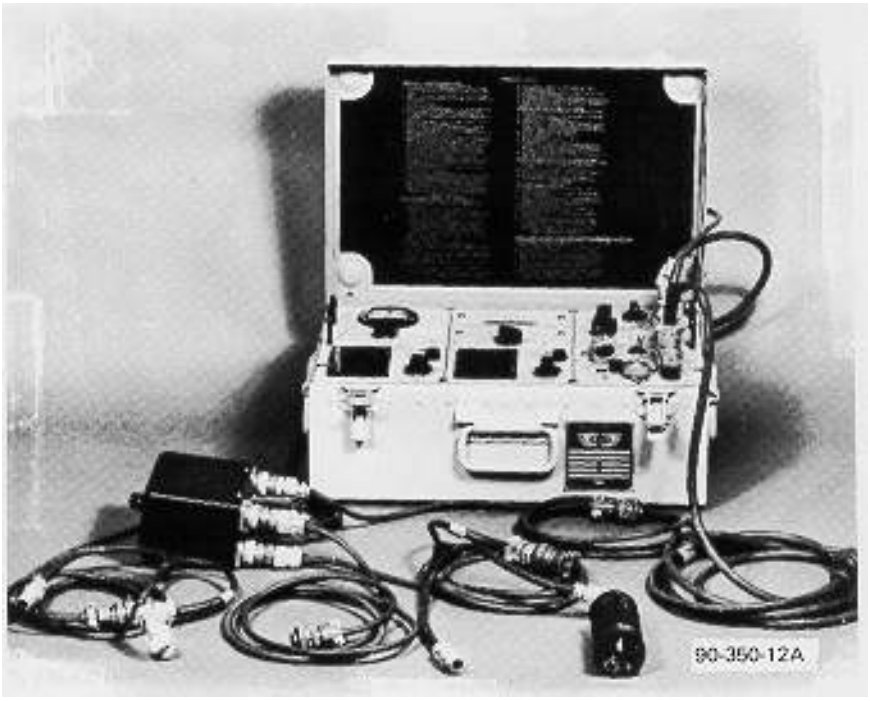

Figure 205
Jetcal Equipment for ITT Check

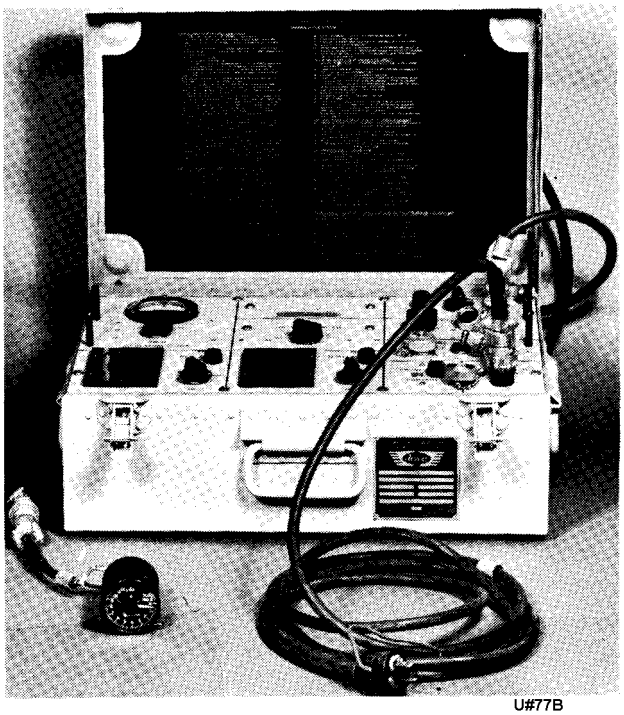

Figure 206
ITT Test Equipment

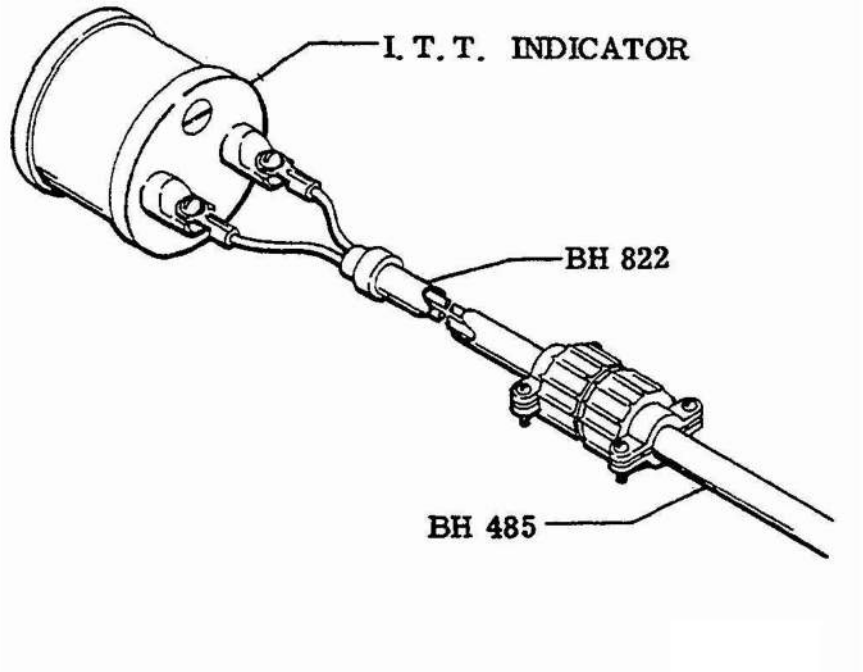

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Figure 207
ITT Indicator Cables

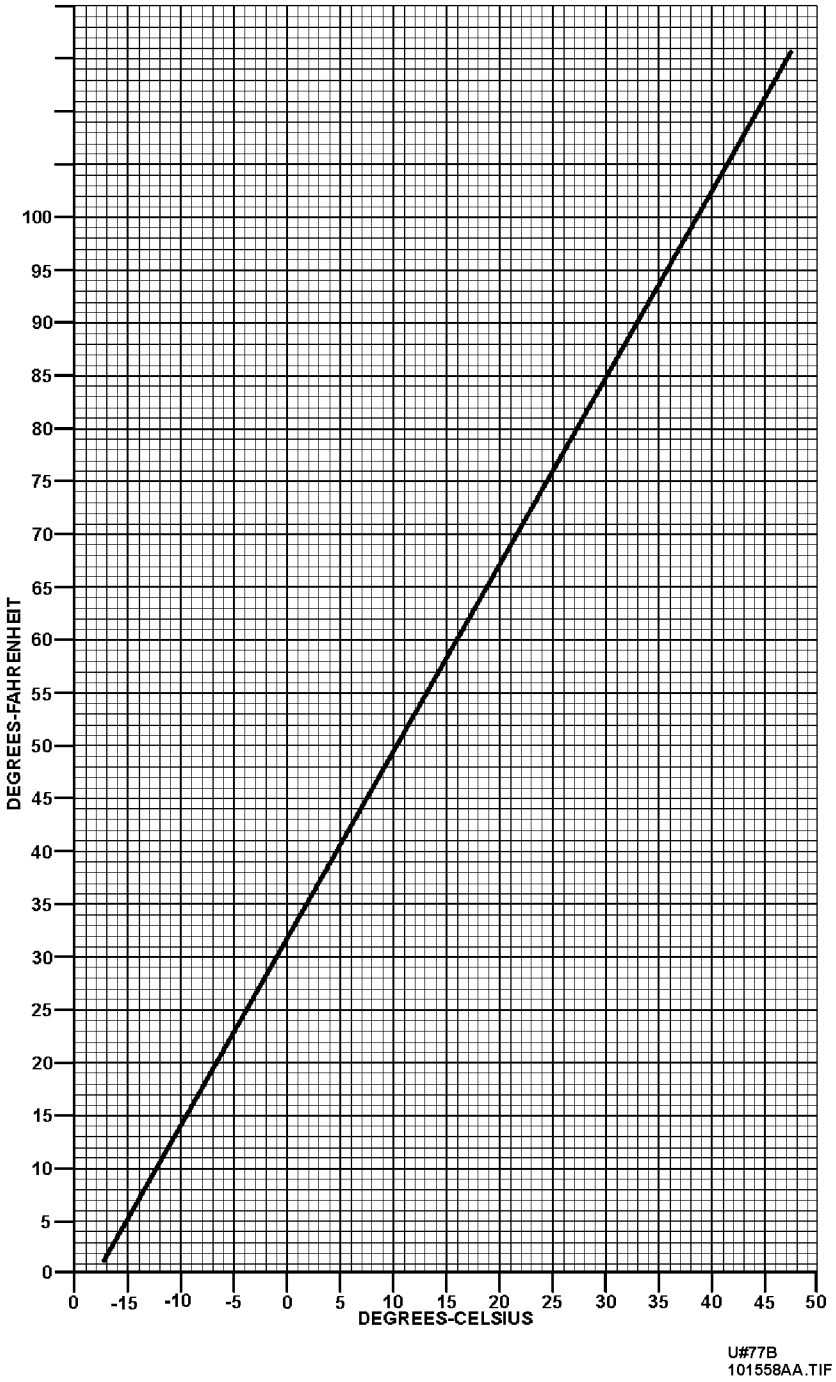

Figure 208
Temperature Conversion Chart

ENGINE INDICATING
TEMPERATURE
MAINTENANCE PRACTICES (U-50, U-165 AND AFTER)

The following checks on the various engine indicating systems are compatible with analyzers and equipment being used in Hawker Beechcraft facilities. Some modifications in these procedures may be necessary if different equipment is used. All pressures, resistances and voltages are standard values and will remain unaffected, regardless of the type of equipment being used.

Use of these analyzers and equipment by Hawker Beechcraft facilities does not necessarily constitute an endorsement by Hawker Beechcraft, as other factors affecting the accuracy of the equipment, such as calibration, frequency of certification or misuse, must be taken into consideration before any equipment may be used to check or calibrate the engine indicating systems

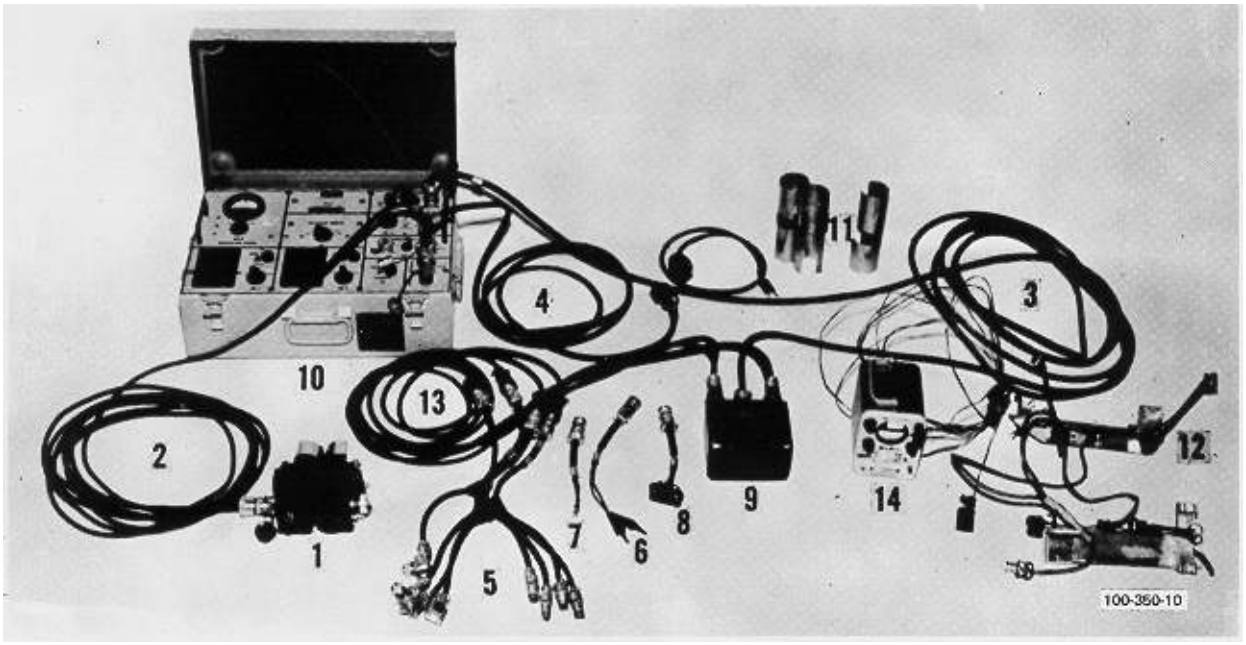

Figure 201
Ground Test Equipment

NOTE: An order for BH112JR Jetcal Unit called out in (10), will include the items listed in (1) thru (9).

Item	Quantity
(1) *BH134 Switch Box	2 each
(2) *BH450 Check Cable	1 each
(3) *BH485 Instrument Check Cable	1 each
(4) *BH499 Power Cable	1 each
(5) *BH820 Tachometer Check Adapter	4 each
(6) *BH821 Insulation Check Adapter	1 each
(7) *BH822 EGT Indicator Adapter	1 each

	Item	Quantity
(8)	*BH223 Resistance Check Adapter	1 each
(9)	*BH5165 RPM Check Box	1 each
(10)	*BH112JR Jetcal Unit	1 each
(11)	Tachometer Holding Brackets	4 each
(12)	Inlet Air Temperature Thermocouple Harness (P/N 50-000065 TC935) or equivalent	1 each
(13)	Adapter Cable (4 Conductor Cable No. 18 gage wire used with BH820 RPM Check Cables)	4 each
	Consists of:	
	6 feet in length (4 required)	
	MS3106A14S2S Plug (4 required)	
	MS3106A14S2P Plug (4 required)	
	AN3057-6 Plug (8 required)	
(14)	Potentiometer Pyrometer, Model No. 80200 with a range of 0° to 600°F (product of Thermo Electric Co., Inc., Saddle Brooke, New Jersey) or equivalent	1 each
(15)	“Y” Cable (for Toque Indicator and Transmitter Check)	
	Consists of:	
	MS3106A10SL3P Plug (2 required)	
	MS3102A10SL3P Plug (1 required)	
(16)	Pressure Gage with range of 0 to 50 psi and accuracy of 1/4% throughout range (Model 0-60 GP-65.8 by Ashcraft is satisfactory) Regulated nitrogen or dry air supply for calibration of the torque system is also required	1 each
(17)	Flowmeter (Capable of 1% accuracy in 100 to 400 pph range)	1 each
	Flowrator, Model 10A 2735CA (product of Fisher and Porter, Warminister, Pennsylvania) with tube No. FP 1/2 27G 10/80 6509 A1 671 A1 is satisfactory	
(18)	Power Cable (Ref. Figure 203)	1 each

* Products of Howell Instruments Inc. 3479 West Vikery, Fort Worth. Texas

1. INTER TURBINE TEMPERATURE (ITT) INDICATING SYSTEM CHECKS

A series of thermal resistive temperature sensing probes, located in the inter turbine gas flow between the first stage and second stage turbines, are interconnected by a harness to form a continuous loop. The combined resistance of the thermocouples, connected in parallel, and the harness is less than 8 ohms.

The resistance of the ITT loop is compensated to 8 ohms ± 0.05 ohms by the resistor thermocouple compensator spool assembly (Ref. Figure 202). The compensator spool is connected electrically in series with the thermocouple loop.

An adjustable ITT indicator, located in the instrument panel, completes the indicating system.

ITT loop resistance can be adjusted to a lower resistance by removing wire from the compensator spool, or increased, if excess wire has been left within the case and not separated from the spool, by winding wire back onto the spool. The compensator spool assembly contains two spools. Only one of the spools is used at a time. If wire has previously been removed from the spool and cut off, the alternate spool can be wired into the system.

An adjusting screw on the back of the ITT indicator is used to calibrate the indicator to the same temperature as indicated on the jetcal unit.

The following resistance checks should be performed according to the instructions which are a part of the particular jetcal unit being used.

These checks should not be performed until at least 10 hours have elapsed since the engine was last run so that the internal engine components and thermocouples will have cooled to ambient temperature.

The PT6A-36 ENGINE INDICATING CHECK WORKSHEET in 77-11-00, Table 201 should be used to record the results of these checks and may be retained in the airplanes records for the purpose of documentation.

A. ITT Thermocouple Loop Resistance Check

(1) Place all switches on the jetcal unit in the OFF position.

(2) Connect the power cable BH499A (Ref. Figure 203) to the power input receptacle and to the power source.

(3) Connect the power interconnect cable between the two interconnect receptacles on the unit.

(4) Connect the BH485A instrument cable to the instrument cable receptacle and to the resistance adapter BH823 (Ref. Figure 204).

(5) Remove the ITT indicator from the instrument panel.

(6) Connect the airplane thermocouple harness leads to the terminal block on the resistance adapter, observing proper polarity (Ref. Figure 204).

(7) Place the resistance and A/C indicator check switch in the 8-ohm position.

(8) Place the function select switch in the RES position and the master power switch in the ON position.

NOTE: Perform the ITT THERMOCOUPLE HARNESS INSULATION CHECK prior to making any adjustments to the thermocouple compensator spool.

(9) Thermocouple loop resistance must be 8 ohms ± 0.05 ohms; if not, adjust the thermocouple spool as necessary to bring the loop resistance within tolerance.

B. ITT Thermocouple Harness Insulation Resistance Check

If the thermocouple loop resistance is within acceptable limits, perform the following checks on the harness insulation resistance:

(1) Place all switches on the jetcal unit in the OFF position.

(2) Remove the resistance adapter from the instrument cable and connect the insulation resistance adapter, BH821 to the instrument cable (Ref. Figure 205).

(3) Place the function select switch in the INSUL position.

(4) Place the insulation check switch in the R X 1000 position.

(5) Place the master power switch in the ON position.

(6) Connect one lead of the insulation resistance adapter to one of thermocouple harness leads and connect the other lead of the insulation resistance adapter to an airplane ground (Ref. Figure 205).

(7) The insulation resistance meter should indicate 5,000 ohms or more.

(8) Connect the insulation resistance adapter leads to the other thermocouple harness lead and to an airplane ground.

(9) Resistance should read 5,000 ohms or more. If either resistance reading is not within tolerance, locate the short circuit and repair.

C. ITT Indicator Calibration Check

The following check on the ITT indicator should be performed any time the indicator is suspected to be in error or during jetcal checks of other indicating instruments.

Should thermocouple harness resistances be found to be within acceptable limits, the fault will probably be in the indicator. The indicator may be non-functioning or simply needing a calibration adjustment due to seasoning of internal components or minor shock damage.

An adjustment screw on the back of the indicator is used to align the temperature indications on the indicator with the temperature indications on the jetcal unit. In the event the indicator has lost its linear response to changes in thermocouple resistance, no adjustment can be made to correct this condition and the indicator must be replaced.

(1) Place all switches on the jetcal unit in the OFF position.

(2) Remove the insulation resistance adapter from the instrument cable.

(3) Connect the BH822 ITT indicator adapter to the instrument cable (Ref. Figure 206).

(4) Connect the ITT indicator adapter to the proper posts on the ITT indicator (Ref. Figure 204).

(5) Place the resistance and A/C indicator check switch in the 8-ohm position.

(6) Place the function select switch in the A/C IND CHK position.

(7) Place the temperature switch in the OPERATE position and turn the master power switch on.

(8) Set the test temperature on the jetcal unit with the A/C IND ADJ control.

(9) The ITT indicator readings should match the temperature settings on the jetcal unit within the tolerances listed in the PT6A-36 ENGINE INDICATING CHECK WORKSHEET in 77-11-00, Table 201.

(10) Adjust the ITT indicator as necessary.

(11) Glyptol seal the adjusting screw after calibration.

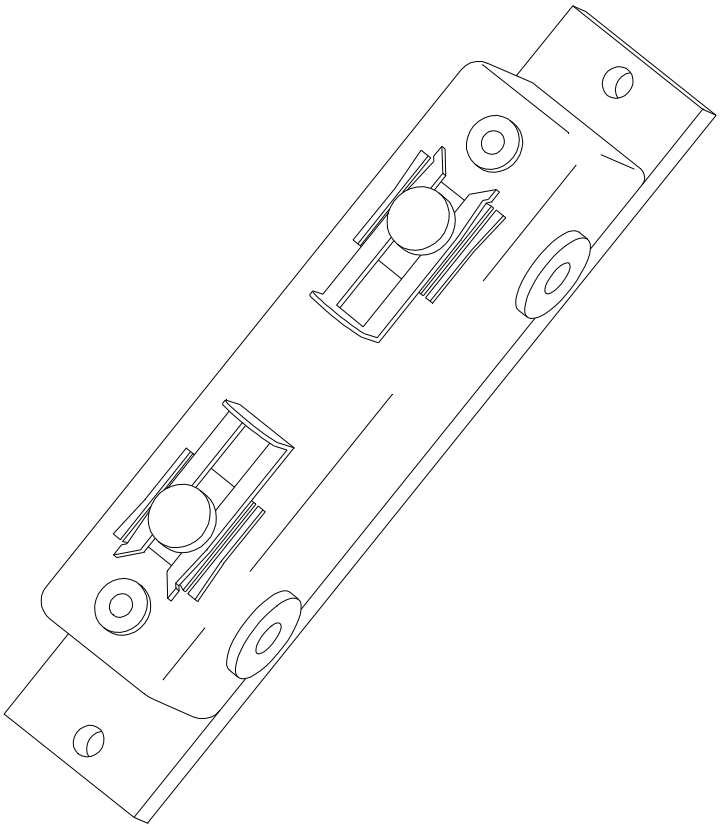

Figure 202
Compensator Spool Assembly

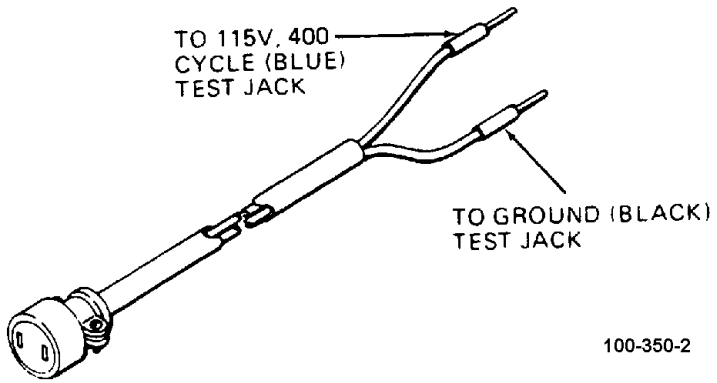

Figure 203
Power Adapter

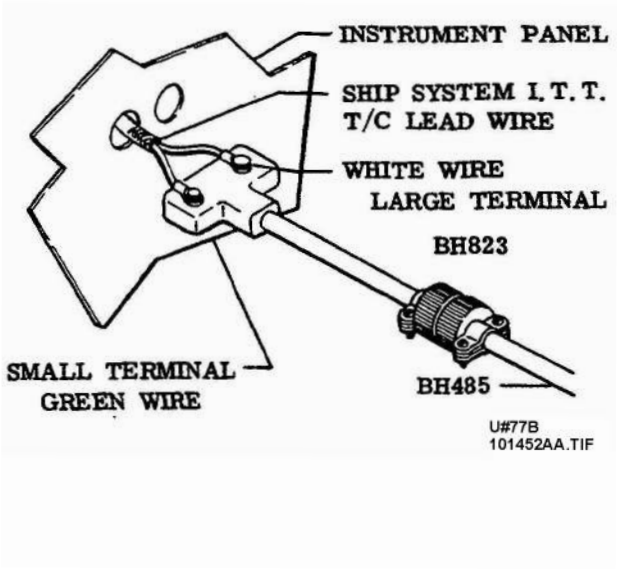

Figure 204
Resister Check Cable

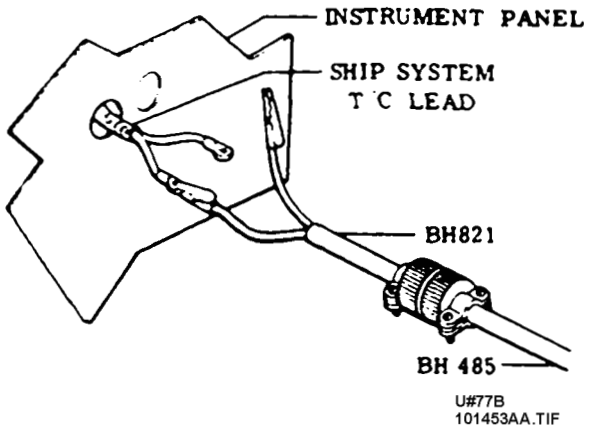

Figure 205
Insulation Check Cable

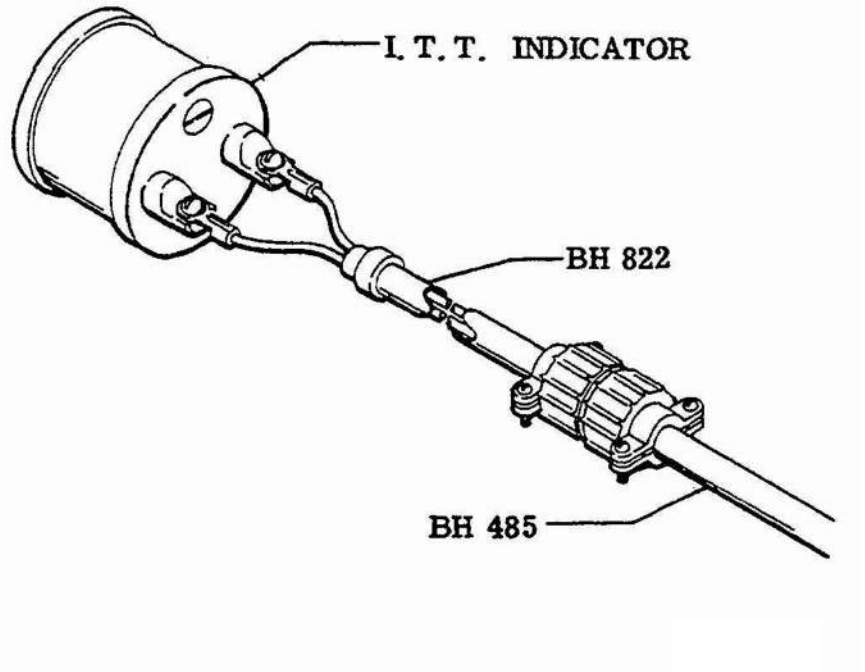

Figure 206
ITT Indicator Cables

List of Effective Pages

CH-SE-SU	PAGE	DATE
79-LOEP	1	Nov 1/10
79-CONTENTS	1	Nov 1/10
79-00-00	201 thru 205	Nov 1/10

CHAPTER 79 - OIL

TABLE OF CONTENTS

SUBJECT *PAGE*

OIL SYSTEM

79-00-00

Maintenance Practices 201
- Oil Filter 201
  - Changing 201
  - Cleaning 202
- Oil 202
  - List of Approved Engine Oils 202
  - Changing the Engine Oil 202
- Magnetic Drain Plug 203
  - Inspection 203

OIL
OIL SYSTEM
MAINTENANCE PRACTICES

Servicing the engine oil system primarily involves maintaining the engine oil at the proper level, inspecting the oil filter at 100 hour intervals, and cleaning or changing the filter as indicated in the applicable Pratt & Whitney Aircraft of Canada, Ltd., Maintenance Manual. The engine oil should be changed every 400 hours or 9 months (whichever occurs first) on airplanes operated 50 hours a month or less. On high utility type of operation, change the engine oil every 800 hours. The time for engine oil change may be extended to 1200 hours if 5-centistoke oils are used (Ref. Table 6, Chapter 91-00-00). It is recommended that 5-centistoke oils be used in airplanes normally operated in climates with ambient temperatures at or below 0°F (-18°C).

NOTE: The engine oil should be checked daily. Refer to the Pratt & Whitney Maintenance Manual (P/N 3013242) Chapter 72-00-00.

CAUTION: *Do not mix different brands, viscosities or types of oil since their different chemical structures may make them incompatible. If different types of oil become mixed, drain and flush system and fill with new oil.*

The oil tank is provided with an oil filler neck and quantity dipstick cap which protrude through the accessory gear-case at the eleven o'clock position. The dipstick is marked in U.S. quarts and indicates the amount of oil required to fill the tank. Access to the dipstick cap is gained by opening the aft engine cowl. Service the oil system with oil (Ref. Table 6, Chapter 91-00-00). Do not mix the oil brands or viscosities. The total oil tank capacity is 2.3 U.S. Gallons (9.2 quarts) of which only 1.5 gallons (6 quarts) is usable. An additional amount of approximately 5 quarts of oil is required to fill the lines and cooler, giving a total system capacity of 14 quarts; however, because of the residual oil trapped in the system, no more than 12 quarts should be added during an oil change.

CAUTION: *Remove spilled oil immediately to prevent the surface thus contaminated from causing tire damage.*

1. OIL FILTER

The engine oil filter is located under the square cover plate at the three o'clock position of the compressor inlet case and just behind the aft fire seal.

A. Changing

The filter element may be changed as follows:

(1) Remove the four self locking nuts and plain washers securing the filter cover to the compressor inlet case and remove the cover.

(2) Withdraw the element from the filter housing. Inspect the filter element for the presence of foreign particles.

(3) Insert a new filter or cleaned filter element (perforated flange end first) into the filter housing (Ref. Figure 201).

(4) Coat a new seal with engine oil and install the seal and cover on the engine.

(5) Secure the filter cover with four plain washers and self-locking nuts. Torque the nuts 32 to 36 inch-pounds above the torque necessary to turn the nuts before seating.

B. Cleaning

CAUTION: *Anytime the oil system is contaminated by metal particles, the oil cooler must be replaced and the oil system flushed to prevent engine damage. In addition, all airplane components and associated plumbing utilizing the engine oil system, such as propeller governor, must be flushed until free of contamination.*

Some filter elements may be cleaned and reused. Some of these filters may be reused as long as the filter is found to be serviceable upon inspection. Refer to the applicable Pratt & Whitney Aircraft of Canada Maintenance Manual or Service Bulletin for the cleaning procedures and limitations of these filters.

2. OIL

A. List of Approved Engine Oils

CAUTION: *Do not mix oils of different brands or oils of different types produced by the same manufacturer.*

Refer to Table 6, Chapter 91-00-00 for the list of approved engine oils.

B. Changing the Engine Oil

CAUTION: *When changing to a different brand of oil, completely drain the airplane oil system as indicated in the procedure below. Remove the oil filter and immerse it in the brand of oil to be used. Install the oil filter and drain plugs. Fill the system to the proper level, and ground run the engines for 20 minutes to thoroughly circulate the new brand of oil throughout the system. Completely drain the airplane oil system and again remove the oil filter and immerse it in the new brand of oil. Refill the airplane oil system as indicated below. This will thoroughly purge the system of the oil to prevent chemical interaction between it and the new brand.*

NOTE: Discard all old packings and replace with new packings.

(1) To gain access to the oil drain plug, remove the fiberglass duct from around the oil cooler and remove the metal bypass duct immediately aft of the oil cooler.

(2) Cut safety wire and remove the drain plug from the oil cooler and drain the oil into a container.

(3) Remove the safety wire from the oil drain cap.

(4) Position the oil drain funnel under the oil cap.

(5) Remove the oil drain cap. Allow all oil to drain from the engine.

(6) Remove the forward engine cowling (Ref. Chapter 71-10-00, ENGINE COWLING REMOVAL).

(7) Cut and remove safety wire then remove the chip detector from the nose case.

(8) Remove the drain plug from the 6 o'clock position on the rear face of the accessory gearbox housing.

CAUTION: *Limit motoring to the time required to accomplish the above because of limited lubrication available to the engine during this operation and starter motor operating time limits. To prevent damage to the fuel control unit, leave the condition lever in IDLE CUT-OFF while motoring the engine.*

(9) With all the drain plugs removed, motor the engine over with the starter only (no ignition) to permit the scavenge pumps to clear the engine.

(10) Install the cleaned or a new oil filter element as described in CHANGING THE OIL FILTER.

(11) Install and safety wire the engine oil drain cap.

(12) Install and safety the chip detector (torque 45 to 55 inch-pounds). Install the forward cowlings.

CAUTION: *Damage to the threads will result if the drain plug is tightened to a torque exceeding 15 to 20 inch-pounds.*

(13) Install and safety the oil cooler drain plug.

(14) Install and safety the accessory gearbox drain plug (torque 215 to 240 inch-pounds).

(15) Fill the engine with the correct amount and type of oil.

CAUTION: *Do not exceed the starter motor operating time limits of 40 seconds ON, 60 seconds OFF, 40 seconds ON, 60 seconds OFF, 40 seconds ON, then 30 minutes OFF.*

(16) Motor the engine over with the starter only, long enough to get an oil pressure reading.

(17) Check the engine for oil leaks.

(18) Fill the engine to the proper level.

(19) Install the metal bypass duct immediately aft of the oil cooler with the retaining screws and install the fiberglass duct around the oil cooler on the lower cowl.

3. MAGNETIC DRAIN PLUG

A. Inspection

A magnetic drain plug (chip detector) is installed in each engine. The plug attracts ferrous particles which may be present in the sump area.

Inspect the magnetic drain plug every 100 hours as follows:

(1) Remove the electrical connection.

(2) Remove the magnetic drain plug.

(3) Visually check the plug for metal particles.

(4) Check the plug for cracks or damaged threads.

(5) Check the electrical connector for bent, broken or burned pins.

(6) Check the resistance from pin to ground. The magnetic plug shall not indicate continuity to ground.

(7) Install new packing on magnetic plug.

(8) Install the magnetic plug and torque 45 to 55 inch-pounds.

(9) Install the electrical connection finger tight.

(10) Remove the scavenge oil filter screen from the lower case of the reduction gearbox. Some oil will be lost from the gearbox.

(11) Fabricate a magnetic chip detector check tool (Ref. Figure 202).

(12) Apply electrical power to the airplane.

(13) Insert the curved end of the fabricated tool through the scavenge oil filter screen hole and position the tip in the gap of the chip detector.

(14) Check to see if the chip detector light in the cockpit is illuminated. If not, repeat the above procedures.

(15) Remove the fabricated tool and check to ensure the light does not stay illuminated.

(16) Install the scavenge oil screen as instructed in the engine maintenance manual.

(17) Safety wire the chip detector.

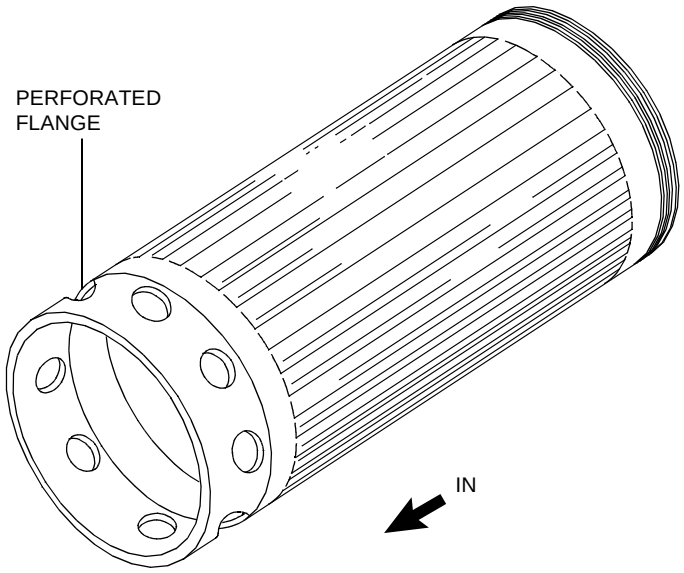

Figure 201
Engine Oil Filter

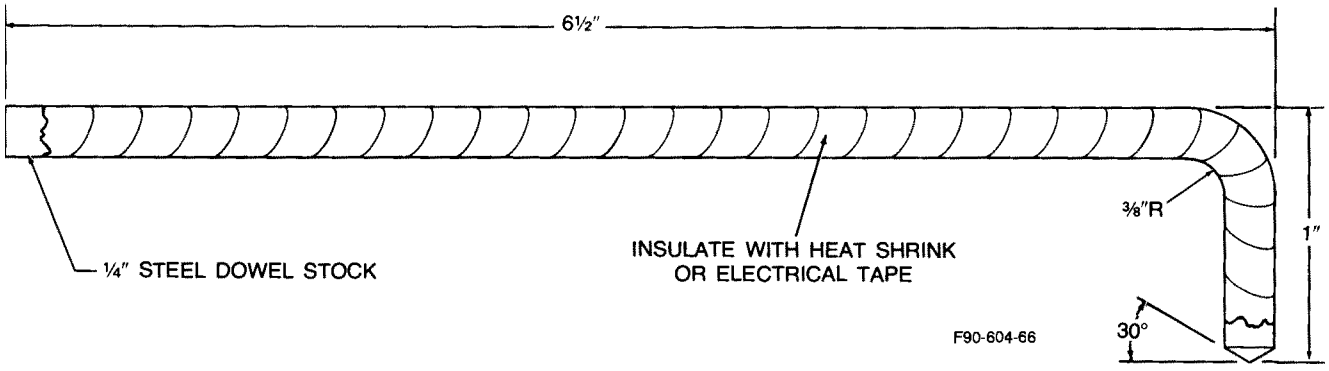

Figure 202
Magnetic Chip Detector Check Tool

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 82 - WATER INJECTION

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WATER INJECTION
GENERAL
DESCRIPTION AND OPERATION

1. DESCRIPTION AND OPERATION

In order to allow improved operation at high density altitudes with a minimum of required off-loading, an engine water/methanol injection system is available for C99 airplanes. By injecting water/methanol solution into the engine inlet, the inlet temperature is depressed, providing approximately 10% more horsepower than the dry condition, while maintaining engine parameters within the existing engine operating limits. The water/methanol injection system provides a temporary means of minimizing engine power loss during high altitude, hot ambient temperature conditions. The system is intended for use during takeoff so that payload and/or fuel weight need not be limited in order to meet FAA single engine rate of climb requirements; however, the system should not be used when the temperature is below 10°C.

The system consists of two water/methanol reservoirs (approximately three gallons each) located in the left and right center section leading edges; pressure regulators, switches and check valves to contain pressure in the system; compressor spray rings, low level sensors and connecting plumbing and wiring. The reservoirs are pressurized from engine bleed air to approximately 35 psi, controlled by factory preset regulators. No further readjustment of these regulators is required.

Prior to takeoff, the system is armed for operation by a 3-position control switch (placarded W/M INJECTION, ARM, OFF, TEST) located in the left outboard subpanel. The TEST position is to be used for preflight check to ensure the electrical components of the system are functional. Arming the water/methanol system opens a bleed air solenoid shutoff valve, allowing engine bleed air, tapped off of the instrument air line, to pressurize the water/methanol reservoir through a pressure regulator (Ref. Figure 1). Arming the system disengages the air conditioner compressor to minimize accessory power loss. Arming the system also precludes the use of the environmental bleed air to prevent the introduction of water/methanol vapors into the cabin. This is accomplished through a 60-second time delay circuit that renders the environmental bleed air inactive until the water/methanol system has been deactivated. A switch located between the pressure regulator and the solenoid shutoff valve senses the pressure and signals illumination of green L W/M ARM and R W/M ARM lights in the annunciator panel. Activation of the system occurs when power is advanced to 90% N_1, activating micro switches in the pedestal which energize and open the solenoid valves and allow the water/methanol solution to be sprayed into the engine inlets. A pressure switch located immediately upstream of the spray ring senses positive water/methanol solution pressure and illuminates green L W/M INJECT and R W/M INJECT lights in the annunciator panel. These pressure switches are set to actuate at a pressure of 28 +2 psi. The water/methanol solution pressure, transmitted via pressure signal lines to the fuel control units on each engine, causes the fuel control units to reset N_1 (gas generator) rpm up approximately 1%. Resetting the N_1 compensates for decreased rpm resulting from introduction of the water/methanol solution.

The system is deactivated by positioning the W/M INJECTION switch to OFF. This will shut off the engine bleed air and solenoid valves to the water/methanol system. The drop in pressure through the pressure signal lines to the fuel control units, in turn, resets the N_1 rpm down. ARM and INJECT signal lights will extinguish as the pressure is relieved. Loss of pressure for any reason, including depletion of the water/methanol solution, will also result in extinguishing the green ARM and INJECT annunciator lights and resetting the N_1 rpm.

The water/methanol solution is stored in separate reservoirs for each engine. The reservoirs have been sized to provide approximately five minutes of maximum added power. A liquid level sensor in each reservoir will illuminate yellow L W/M LOW and R W/M LOW annunciator lights when a minimum amount of solution remains. A minimum solution has been established for two minutes of operation. This two minutes has been determined as a worst condition; i.e. single engine, hot day 5000 ft. altitude, 50 ft. high obstacle, one minute operation required plus pad.

During ground operations the engine should be operated approximately 5 minutes after operation of the water/methanol system to ensure evaporation.

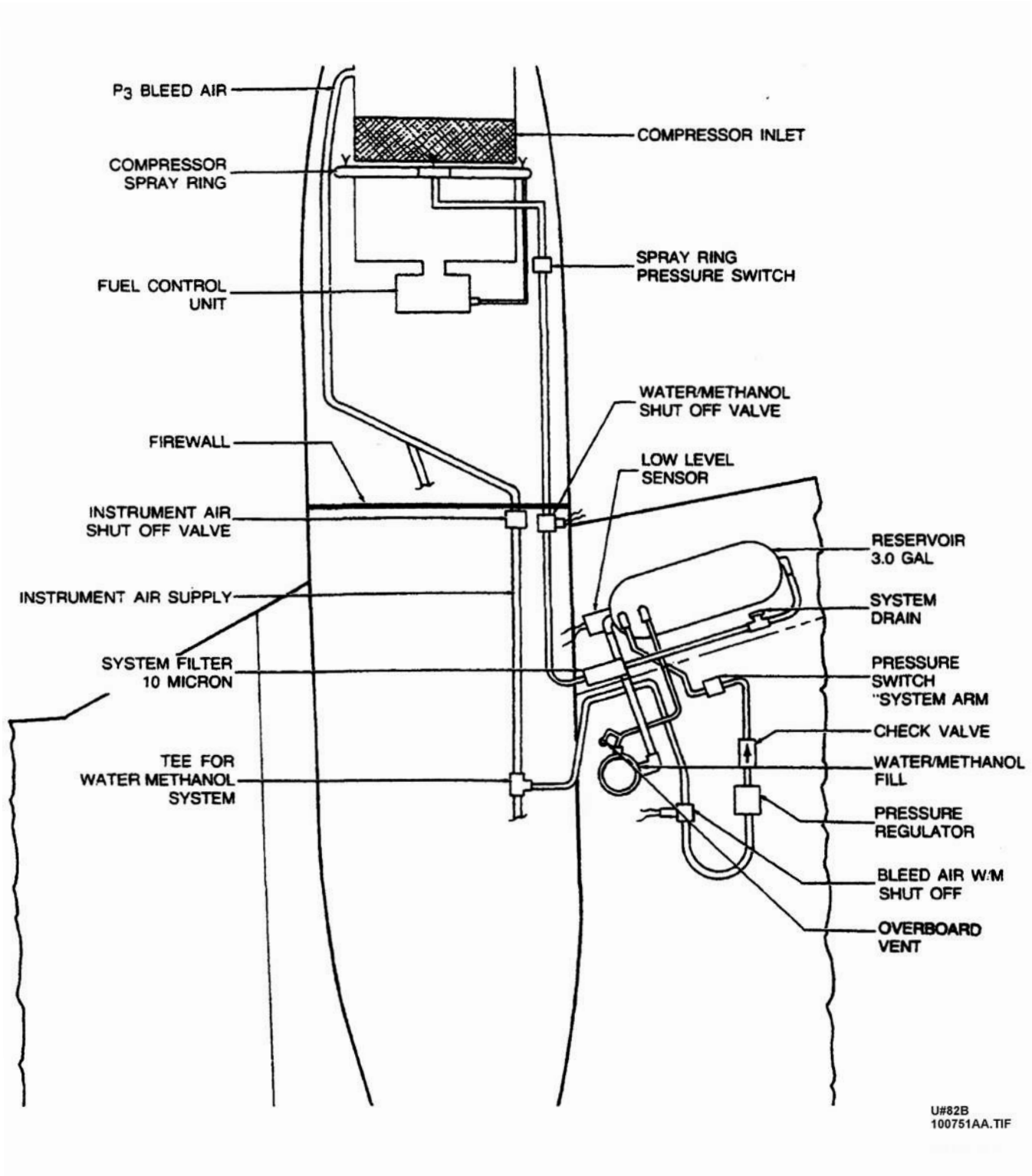

Figure 1
Water/Methanol Injection Schematic

WATER INJECTION
WATER/METHANOL INJECTION SYSTEM
TROUBLESHOOTING

1. WATER/METHANOL INJECTION SYSTEM

	Trouble	Probable Cause	Remarks
(1)	System totally inoperative (No annunciator light indication; No response to system ARM and TEST switch).	a. Faulty circuit breaker. b. Possible damage or loose wiring.	a. Replace circuit breaker. b. Check continuity of wire Q141A22. Repair or replace.
(2)	System totally inoperative (Switch in ARM position, Power Control lever above 80%. No system ARM light on either LH or RH).	a. Possible damaged or loose wire.	b. Check continuity of wire Q142A22. Repair or replace.
(3)	No System ARM light or system not pressurized on LH side. (Switch in ARM position. Power Control levers above 80%).	a. Lamp burned out. b. Damaged or loose wiring.	a. Replace lamp. b. Check continuity of wire Q142B22. Repair or replace.
(4)	No System ARM light or system not pressurized on RH side. (Switch in ARM position. Power Control lever above 80%).	a. Lamp burned out. b. Damaged or loose wiring.	a. Replace lamp. b. Check continuity of wire Q142C22. Repair or replace.
(5)	LH System inoperative (System armed. Power levers above 90%. No annunciator lights).	a. Lamp burned out. b. Damaged or loose wiring.	a. Replace lamp. b. Check continuity of wire Q142G22. Repair or replace.
(6)	RH System inoperative (System armed. Power levers above 90%. No annunciator lights).	a. Lamp burned out. b. Damaged or loose wiring.	a. Replace lamp. b. Check continuity of wire Q142F22. Repair or replace.

	Trouble	Probable Cause	Remarks
(7)	Strong methanol fumes in cabin during takeoff (All lights indicate normal operation).	a. P3 air being pumped into cabin saturated with methanol vapors. Environmental bleed air valves open after 60 second time delay.	a. Turn off water/methanol system. Fill cabin with fresh air. b. Check continuity of wires Q150A22 and Q151A22. Repair or replace.
(8)	Water/methanol system shuts down during takeoff (All lights out. System completely dead. No response from cycling circuit breaker. 60 second time delay inoperative).	a. Wiring loose or damaged.	a. Check continuity of wire Q141B22. Repair or replace.
(9)	Cabin air conditioner remains on during water/methanol injection cycle and takeoff and climb.	a. Loose or damaged wiring.	a. Check continuity of wires H300A20 and H151A20. Repair or replace.
(10)	No power increases on one engine. (ARM l i g h t s illuminate; Power levers past 90%; FCU responds on opposite engine but not on affected engine; Both W/M INJECT lights illuminated).	a. Blocked or plugged injection ring orifices on affected engine.	a. Repair or replace shutoff valve.
(11)	No power increase on one engine. (System armed; Power levers past 90%; No W/M INJECT lights on affected side)	a. W/M shutoff valve inoperative.	a. Repair or replace shutoff valve.
(12)	No indication of system being operational on one engine even though the affected engine responds with increased power with system armed, power levers past 90%.	a. Pressure switch, possibly inoperative.	a. Check pressure switch, repair or replace.
(13)	Low level indicator light not illuminated on one side after solution is depleted.	a. Low level sensor inoperative for affected side.	a. Remove affected low level sensor and inspect prism for clarity. If leached or damaged, replace.

	Trouble	Probable Cause	Remarks
(14)	Pressure remains in system and reservoir after deactivation.	a. Possible blockage in overboard vent line orifice.	a. Remove blockage and clean entire system.
(15)	Solution backs up into the fill port during filling and does not flow out through overboard vent line.	a. Blocked overboard vent line orifice.	a. Remove blockage and clean entire system.
(16)	Methanol fumes in cabin. System operation indicates normal. Some engine power loss.	a. Bleed air environmental shutoff valves not operating properly	a. Shut off system. Repair or replace environmental Meed air shutoff valves.
(17)	No ARM indicator light. 80% power. W/M switch in ARM position.	a. ARM pressure switch not operating properly. b. Regulator not operating properly.	a. Check and repair or replace ARM pressure switch. b. Check and repair or replace regulator.
(18)	Water/methanol mixture exhausting overboard through vent line with switches in ARM position (80% power).	a. Fill port standpipe seal leaking.	a. Inspect seal and replace as required.
(19)	No engine power increase with switch in ARM position, above 90% power. No W/M INJECT light.	a. Inline filter clogged.	a. Remove filter and clean or replace.

WATER INJECTION
WATER/METHANOL INJECTION SYSTEM
MAINTENANCE PRACTICES

1. SERVICING

See Chapter 12-10-00 for the servicing requirements for the water/methanol system.

2. WATER/METHANOL POWER LEVER SWITCHES

A. Adjustment

(1) Gain access to the microswitches by removing the panel from the side of the pedestal.

(2) Start and run the engines (according to the Pilots Operating Handbook) to 90% N_1; mark the position of the levers on the pedestal.

(3) Shut down the engines (according to the Pilots Operating Handbook).

(4) Return the power levers to the 90% N_1 position.

(5) Loosen the microswitch mounting screws and position the switches to just actuate.

(6) Remove the 90% N_1 marks and install the pedestal panel.

3. WATER/METHANOL INJECTION SYSTEM

A. Functional Test

(1) On the main DC power distribution panel located under the forward cabin center aisle subfloor, set the circuit breakers for the LH and RH No. 1 and No. 2 primary bus, the condenser blower, vent blower and the air conditioner compressor clutch.

(2) On the RH circuit breaker panel in the pilot's compartment, set the circuit breaker for the LH and RH No. 1 and No. 2 feeder bus, annunciator power and annunciator indicators.

(3) Turn the battery switch ON.

(4) Turn the Water/Methanol switch to TEST position.

(5) Observe that the following annunciator lights illuminate: L W/M LOW, R W/M LOW, L W/M ARM, R W/M ARM, L W/M INJECT, R W/M INJECT, L BL AIR OFF, R BL AIR OFF and observe that the L and R bleed air valves are not energized.

(6) Turn the Water/Methanol switch to ARM position.

(7) Observe that the L and R BL AIR OFF annunciator lights illuminate.

(8) Turn the Water/Methanol switch OFF.

(9) Connect a jumper wire between pins "A" and "C" on the L W/M ARM pressure switch connector.

(10) Connect a jumper wire between pins "A" and "C" of the R W/M ARM pressure switch connector.

(11) Set the L and R bleed air control switches OPEN; observe that the bleed air valves energize.

(12) Turn the Water/Methanol switch to ARM position.

(13) Observe that the L and R BL AIR OFF and L and R W/M ARM annunciator light illuminate and the L and R bleed air valves are not energized.

(14) Turn the Water/Methanol switch OFF.

(15) Connect a jumper wire between pins "A" and "C" of the L W/M INJECT switch connector.

(16) Connect a jumper wire between pins "A" and "C" of the R W/M INJECT switch connector.

(17) Turn the Water/Methanol switch to ARM position.

(18) Observe the power levers to 90% N_1 position; observe that the L and R W/M inject valves energize.

(19) Immediately turn the Water/Methanol switch OFF.

(20) Observe that the L and R W/M ARM, L and R W/M INJECT annunciator legends are not illuminated and the L and R BL AIR OFF legends remain on for approximately one minute.

(21) Apply ground to pin "17" of the connector on the air conditioner compressor clutch control printed wiring board module.

(22) Turn the Water/Methanol switch OFF.

(23) Turn the cabin temperature mode switch to MNL COOL.

(24) Turn the manual temperature control switch to DECR.

(25) Observe that the air conditioning compressor clutch is energized.

(26) Turn the Water/Methanol switch to ARM position.

(27) Observe that the air conditioning compressor clutch is not energized.

(28) Turn the Water/Methanol switch OFF.

4. WATER/METHANOL RESERVOIR

A. Removal

(1) Remove all screws which attach the wing center section leading edge skin assembly and remove the skin assembly.

(2) Disconnect all plumbing lines from the water/methanol reservoir and disconnect the wiring from the low liquid level sensor in the outboard end of the reservoir. Carefully remove the sensor from the reservoir.

(3) Remove the four attaching bolts, washers and nuts which secure the reservoir in the leading edge and remove the reservoir.

B. Installation

(1) Position the reservoir in the wing center section leading edge and install the four attaching bolts, washers and nuts. Torque the nuts 50 to 70 inch-pounds.

(2) Carefully install the low liquid level sensor in the outboard end of the reservoir, using a new P/N 2-016N674-70 packing. Tighten the sensor finger tight only.

(3) Connect the electrical wiring to the low level sensor and connect all plumbing lines to the reservoir. Torque all 1/4-inch diameter line fittings 135 to 150 inch-pounds and all 3/8-inch diameter line fittings 270 to 300 inch-pounds.

(4) Install the wing center section leading edge skin assembly and secure with the attaching screws.

5. LOW LIQUID LEVEL SENSOR

A. Inspection

The low liquid level sensor is located in the outboard end of the water/methanol reservoirs in the wing center section leading edges. Removal of the wing center leading edge skin assembly is necessary for access to the sensors.

Inspect the sensors at the beginning of each operational season and each 30 days thereafter until completion of the season. After two complete operational seasons, inspect the sensors at the beginning of the operational seasons and every 90 days thereafter until completion of the operational season.

At the time limits noted above, remove the low liquid level sensor from each reservoir and inspect the crystal prism for etching (a frosted or sandblasted appearance) and cleanliness. Check the liquid level sensor for proper operation by checking the fluid level against the indicator light. If etching or improper operation is indicated, the liquid level sensor must be replaced. Every time the sensor is removed, a new P/N 2-016N674-70 packing must be installed when installing the sensor. Tighten the sensor finger tight only.

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

CHAPTER 91 - CHARTS

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91-00-00

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CHARTS
GENERAL
DESCRIPTION AND OPERATION

1. TORQUING FINE THREAD SERIES BOLTS LOADED IN SHEAR

NOTE: The following torque values may be used as a guide when specific torques are not called out within this maintenance manual.

The following values apply to Class 3 threads, cadmium plated and nonlubricated. When tightening of the bolt-nut assemblies is accomplished by rotating the bolt, the high limits of the torque values given in columns one and two are recommended. When tightening castellated nuts on bolts to the recommended torque values listed in column one and two, it is possible that the cotter pin holes will not line up with the slots in the nuts. In such cases, the nut may be overtightened just enough to line up the nearest slot with the cotter pin hole, but the maximum torque shall not exceed the value listed in columns three and four.

Table 1
Torquing Fine Thread Series Bolts Loaded in Shear

Size	Torque Limits Recommended (Inch-Pounds)		Maximum Allowable Torque (Inch-Pounds)	
	Tension Type MS20365 and AN310 Nuts Column 1	Shear Type MS20364 and AN320 Nuts Column 2	Tension Type MS20365 and AN310 Nuts Column 3	Shear Type MS20364 and AN320 Nuts Column 4
8-36	12 to 15	7 to 9	20	12
10-32	20 to 25	12 to 15	40	25
1/4-28	50 to 70	30 to 40	100	60
5/16-24	100 to 140	60 to 85	225	140
3/8-24	160 to 190	95 to 110	390	240
7/16-20	450 to 500	270 to 300	840	500
1/2-20	480 to 690	290 to 410	1100	660
9/16-18	800 to 1000	480 to 600	1600	960
5/8-18	1100 to 1300	660 to 780	2400	1400
3/4-16	2300 to 2500	1300 to 1500	5000	3000
7/8-14	2500 to 3000	1500 to 1800	7000	4200
1-14	3700 to 5500	2200 to 3300	10000	6000
1 1/8-12	5000 to 7000	3000 to 4200	15000	9000
1 1/4-12	9000 to 11000	5400 to 6600	25000	15000

2. TORQUING COARSE THREAD SERIES BOLTS LOADED IN SHEAR

NOTE: The following values apply to Class 3 threads, cadmium plated and nonlubricated. When tightening of the bolt-nut assemblies is accomplished by rotating the bolt, the high limits of the torque values given in columns one and two are recommended. When tightening castellated nuts on bolts to the recommended torque values listed in column one and two, it is possible that the cotter pin holes will not line up with the slots in the nuts. In such cases, the nut may be overtightened just enough to line up the nearest slot with the cotter pin hole, but the maximum torque shall not exceed the value listed in columns three and four.

Table 2
Torquing Coarse Thread Series Bolts Loaded in Shear

SIZE	Torque Limits Recommended (Inch-Pounds)		Maximum Allowable Torque (Inch-Pounds)	
	Tension Type MS20365 and AN310 Nuts Column 1	Shear Type MS20364 and AN320 Nuts Column 2	Tension Type MS20365 and AN310 Nuts Column 3	Shear Type MS20364 and AN320 Nuts Column 4
8 - 32	12 to 15	7 to 9	20	12
10 - 24	20 to 25	12 to 15	35	21
1/4 - 20	40 to 50	25 to 30	75	45
5/16 - 18	80 to 90	48 to 55	160	100
3/8 - 16	160 to 185	95 to 110	275	170
7/16 - 14	234 to 255	140 to 155	475	280
1/2 - 13	400 to 480	240 to 290	880	520
9/16 - 12	500 to 700	300 to 420	1100	650
5/8 - 11	700 to 900	420 to 540	1500	900
3/4 - 10	1150 to 1600	700 to 950	2500	1500
7/8 - 9	2200 to 3000	1300 to 1800	4600	2700
1 - 8	3700 to 5000	2200 to 3000	7600	4500
1 1/8 - 8	5500 to 6500	3300 to 4000	12000	7200
1 1/4 - 8	6500 to 8000	4000 to 5000	16000	10000

3. FLARE FITTING TORQUE

NOTE: Installation of flared fittings:

When installing flare fittings, make sure the male threads are properly lubricated (Ref. Table 4). Torque the fittings in accordance with the following Table. Do not over-torque.

Table 3
FLARE FITTING TORQUE

Torque in Inch-Pound								
Tubing OD Inches	**Aluminum Alloy Tubing Flare AND1006 or AND10078**		**Steel Tubing Flare AND10061**		**Hose End Fitting and Hose Assemblies**		**Fuel System H Fittings Wheel Wells**	
	Minimum	**Maximum**	**Minimum**	**Maximum**	**Minimum**	**Maximum**	**Minimum**	**Maximum**
1/8	---	---	---	---	---	---	---	---
3/16	---	---	90	100	70	100	---	---
1/4	40	65	135	150	70	120	---	---
5/16	60	80	180	200	85	180	---	---
3/8	75	125	270	300	100	250	---	---
1/2	150	250	450	500	210	420	---	---
5/8	200	350	650	700	300	480	60	72
3/4	300	500	900	1000	500	850	---	---
1	500	700	1200	1100	700	1150	---	---
1-1/4	600	900	---	---	---	---	---	---
1-1/2	600	900	---	---	---	---	---	---
1-3/4	---	---	---	---	---	---	---	---
2	---	---	---	---	---	---	---	---

4. THREAD LUBRICANTS

NOTE: Lubricate engine fittings only with the fluid contained in the particular lines.

Table 4
Thread Lubricants

TYPE OF LINE	TYPE OF LUBRICANT
Hydraulic	MIL-H-5606 Hydraulic Fluid or Loctite Hydraulic Sealant Cat. No. 92 or Cat. No. 69
Fuel	VV-P-236, Anti-Seize, Graphite Petrolatum
Bleed Air System	M-77 (product of Dow Corning)
Oil	MIL-G-6032, Lubricating Grease (Gasoline and Oil Resistant)
Oxygen	Taper Threads: MIL-T-27730, Size 1 Teflon Tape. Straight Threads: None or MIL-T-27730, Size 1 Teflon Tape
Pitot and Static	Taper Threads: 92 Pipe Sealant with Teflon (product of Loctite Corp.)
Vacuum	TT-A-580, Anti-Seize Compound (White Lead Base)
Refrigerant	Oil of the System
Fire Extinguisher	92 Pipe Sealant with Teflon (product of Loctite Corp.)
Deicer (pneumatic)	92 Pipe Sealant with Teflon (product of Loctite Corp.)

5. FUEL BRANDS AND TYPE DESIGNATIONS

NOTE: Jet A - aviation kerosene-type fuel with -40°F (-40°C) freeze point.

Jet A-1 - aviation kerosene-type fuel with -58°F (-50°C) freeze point.

Jet B - a low grade, kerosene-type fuel with a freeze point of -60°F (-51°C), similar to MIL-T-5624 grade JP-4, which has a freeze point of -76°F (-60°C).

Table 5
Fuel Brands and Type Designations

PRODUCT NAME	DESIG	PRODUCT NAME	DESIG
AMERICAN OIL		PURE OIL COMPANY	
American Jet Fuel Type A	Jet A	Purejet Turbine Fuel Type A	Jet A
American Jet Fuel Type A-1	Jet A-1	Purejet Turbine Fuel Type A-1	Jet A-1
ATLANTIC REFINING COMPANY		RICHFIELD PETROLEUM COMPANY	
Arcojet-A	Jet A	Richfield Turbine Fuel A	Jet A
Arcojet-A-1	Jet A-1	Richfield Turbine Fuel A-1	Jet A-1
Arcojet-B	Jet B		
BP TRADING COMPANY		SHELL OIL COMPANY	
BP A.T.K.	Jet A-1	Aeroshell Turbine Fuel 640	Jet A
BP A.T.G.	Jet B	Aeroshell Turbine Fuel 650	Jet A-1
		Aeroshell Turbine Fuel JP-4	Jet B
CALIFORNIA TEXAS COMPANY		SINCLAIR OIL COMPANY	
Caltex Jet A-1	Jet A-1	Sinclair Superjet Fuel	Jet A
Caltex Jet B	Jet B	Sinclair Superjet Fuel	Jet A-1
CITIES SERVICE COMPANY		STANDARD OIL OF CALIFORNIA	
Turbine Type A	Jet A	Chevron TF-1	Jet A-1
		Chevron JP-4	Jet B
CONTINENTAL OIL COMPANY		STANDARD OIL OF KENTUCKY	
Conoco Jet-40	Jet A	Standard JF A	Jet A
Conoco Jet-50	Jet A	Standard JF A-1	Jet A-1
Conoco Jet-60	Jet A-1	Standard JF B	Jet B
Conoco Jet JP-4	Jet B		
GULF OIL COMPANY		STANDARD OIL OF OHIO	
Gulf Jet A	Jet A	Jet A Kerosene	Jet A
Gulf Jet A-1	Jet A-1	Jet A-1 Kerosene	Jet A-1
Gulf Jet B	Jet B		
EXXON OIL COMPANY		TEXACO	
Exxon Turbo Fuel A	Jet A	Texaco Avjet K-40	Jet A
Exxon Turbo Fuel 1-A	Jet A-1	Texaco Avjet K-58	Jet A-1
Exxon Turbo Fuel 4	Jet B	Texaco Avjet JP-4	Jet B
MOBIL OIL COMPANY		UNION OIL COMPANY	
Mobil Jet A	Jet A	76 Turbine Fuel	Jet A-1
Mobil Jet A-1	Jet A-1	Union JP-4	Jet B
Mobil Jet B	Jet B		

Table 5
Fuel Brands and Type Designations (Continued)

PRODUCT NAME	DESIG	PRODUCT NAME	DESIG
PHILLIPS PETROLEUM COMPANY Philjet A-50 Philjet JP-4	 Jet A Jet B		

6. CONSUMABLE MATERIALS

Only the basic number of each Military Specification is included in the Consumable Materials Table. No attempt has been made to update the basic number with the letter suffix that designates the current issues of the various specifications. It is the responsibility of the operator/user to determine the current revision of the applicable Military Specification prior to usage of that item. This determination may be made by contacting the supplier of a specific item. Suppliers listed as meeting Federal and Military Specifications are provided as reference only and are not specifically recommended by Hawker Beechcraft Corporation; consequently, any product conforming to the specification listed may be used. The products listed below have been tested and approved for aviation usage by Hawker Beechcraft Corporation, by the supplier, or by compliance with the applicable specifications. Other products that are locally procurable which conform to the requirements of the applicable Military Specification may be used even though not specifically included herein.

NOTE: For the various engine oils approved by Pratt and Whitney, refer to the Pratt and Whitney Service Bulletin No. 1001.

Take precautions when using MIL-G-23827, MIL-G-7711 and engine oil as these lubricants contain chemicals harmful to painted surfaces.

Table 6
CONSUMABLE MATERIALS

Material	Specification	Products	Supplier
(1) Engine Oils		Refer to NOTE on previous page.	
(2) Lubricating Oil Special Preservative	VV-L-800	Brayco 300	Castrol, Inc., 16715 Von Karman Avenue Suite 230, Irvine, California 92714
		Royco 308	Royal Lubricants Company 6 Campus Drive Parsippany, New Jersey 07054
		Nox Rust 518 (Code R-62-203-1)	Daubert Chemical Company 4700 S. Central Avenue Chicago, Illinois 60638

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(3) Lubrication Oil General Purpose Low Temperature	MIL-L-7870	Caltex Low Temp Oil	Caltex Oil Products 125 E. John W. Carpenter Freeway Irving, Texas 75062
		Sinclair Aircraft Orbitlube	Sinclair Oil Corporation 550 E. South Temple Salt Lake City, Utah 84102
		1692 Low Temp Oil	Texaco, Inc. 50 W. 50th Street New York, New York 10017
(4) Lubricating Oil Heavy Duty	MIL-L-2104	Phillips 66 HDS Motor Oil (Grade 10)	Phillips Petroleum Company 758 Adams Building 4th and Keefer Bartlesville, Oklahoma 74043
		Super Lonet (Grade 10)	Sinclair Oil Corporation 550 E. South Temple Salt Lake City, Utah 84102
		PED 3342 (Grade 10)	Chevron U.S.A. 225 Bush Street San Francisco, California 94120
(5) Lubricating Oil	MIL-L-2105, Grade 75 (Supersedes MIL-L-10324 and MIL-O-6086	Mobil Lubricant SHC or RP 95-X, Formula	Mobil Oil Corporation 150 E. 42nd Street New York, New York 10017
		Gear Lubricant SZ 9285	AMOCO, Inc. 2021 Spring Road Oak Brook, Illinois 60521
		2105 Grade 75 Weight	Southwest Petro-Chemical Division of WITCO 1400 S. Harrison Olathe, Kansas 66061
(6) Oil (Air-Conditioner Compressor)		Delco Air (Viscosity 525)	Frigidaire Division, White Consolidated Industries, Inc. 6000 Perimeter Drive Dublin, Ohio 43017

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(7) Marvel Mystery Oil			Marvel Oil Company, Inc. 331-337 N. Main Street Port Chester, New York 10573
(8) Lubricating Oil		Aeroshell No. 12	Shell Oil Company 1 Shell Plaza Houston, Texas 77001
(9) Hydraulic Fluid (Brakes and Shock Struts)	MIL-H-5606	3126 Hydraulic Oil	Humble Refining Company P.O. Box 2180 Houston, Texas 77701
		3126 Hydraulic Oil	Exxon Company U.S.A. Pittsburgh Manufacturing Plant 34th Street & AVRR Pittsburgh, Pennsylvania 15201
		Aeroshell Fluid 4	Shell Oil Company 1 Shell Plaza Houston, Texas 77001
		PED 3337	Chevron U.S.A. 225 Bush Street., San Francisco, California 94120
(10) Recommended Engine Fuel	Jet A (NATO F-30, F-34) Jet A-1 (JP-5, JP-8, NATO F-42) Jet B (JP-4 NATO F-40) MIL-T-5624		Refer to Table 5 Fuel Brand Names and Type Designation.
Alternate (Limited to 150 hours during each overhaul period)	80 (80/87) 100LL Blue 100 Green (100/130) 115 (115/145)		
(11) Lubricating Grease, Gasoline and Oil	MIL-G-6032	L237	Tenneco Chemicals, Inc. P.O. Box 120 State Route 297 Chestertown, Maryland 21620
		Castrolease PV	Castrol, Inc., 16715 Von Karman Avenue Irvine, California 92714

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
		Royco 32	Royal Lubricants Company 6 Campus Drive Parsippany, New Jersey 07054
(12) Lubricating Grease		Plastilube No. 3	Thiem Corporation Subsidiary of Koppers Company, Inc. P.O. Box 93019 5151 Denison Cleveland, OH 44101
(13) Lubricating Grease		Lubriplate No.130AA	Fiske Brothers Refining Company 129 Lockwood Newark, New Jersey 07105
(14) Lubricating Grease	MIL-G-4343	Cosmolube 615	E.F. Houghton and Company Madison and Van Buren Avenue Valley Forge, Pennsylvania 19482
		Royco 43	Royal Lubricants Company 6 Campus Drive Parsippany, New Jersey 07054
(15) Lubricating Grease	MIL-G-7711 (Superseded by MIL-G-81322, See Item 17)		
(16) Lubricating Grease, Aircraft and Instruments, Low and High	MIL-G-23827	Supermil Grease No. A72832	AMOCO, Inc. 2021 Spring Road Oak Brook, Illinois 60521
		Royco 27A	Royal Lubricants Company 6 Campus Drive Parsippany, New Jersey 07054
		Aeroshell 7	Shell Oil Company 1 Shell Plaza Houston, Texas 77001

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
		RR-28	Mobil Oil Corporation 150 East 42nd Street New York, New York 10017
		Castrolease A1	Castrol Oil, Inc. Raritan Plaza 2 Edison, New Jersey 08837
(17) Lubricating Grease High Temperature	MIL-G-81322 (Replaces MIL-G-7711, See Item 15)	Mobilgrease 28	Mobil Oil Corporation 150 East 42nd Street New York, New York 10017
		Royco 22S	Royal Lubricants Company 6 Campus Drive Parsippany, New Jersey 07054
		Aeroshell Grease 22	Shell Oil Company 1 Shell Plaza Houston, Texas 77001
(18) Lubricating Grease, Molybdenum Disulfide	MIL-G-21164	Castrolease MSA (C)	Castrol Oil, Inc. Raritan Plaza 2 Edison, New Jersey 08837
		Electro-Moly/11	Electrofilm Manufacturing Company P.O. Box 55669 2395 Rye Canyon Road Valencia, California 91355
		Everlube 211-G Moly Grease	EM Lubricants 6940 Farmdale Avenue North Hollywood, California 91605
		Royco 64C	Royal Lubricants Company 6 Campus Drive Parsippany, New Jersey 07054
		Aeroshell Grease 17	Shell Oil Company 1 Shell Plaza Houston, Texas 77001

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
		Aeroshell 33MS Grease	Shell Oil Company 1 Shell Plaza Houston, Texas 77001
		Chevron Aviation Grade 44	Chevron U.S.A. 225 Bush Street San Francisco, California 94120
(19) Molybdenum Disulfide	MIL-M-7866	Molykote Z	Haskell, Inc., 100 East Graham Place Burbank, California 91502
		Moly-Paul No. 4	K.S. Paul Products, Ltd. Corneliustrasse 72-74 Postfach 622 Duesseldorf, Germany
(20) Grease	MIL-G-10924	Shell A and A Grease	Shell Oil Company 1 Shell Plaza Houston, Texas 77001
		PED 3355	Chevron U.S.A. 225 Bush Street San Francisco, California 94120
		Cosmolube 506	E.F. Houghton and Company Madison and Van Buren Avenue Valley Forge, Pennsylvania 19482
(21) Lubricant, Powdered Graphite	SS-G-659 (Superseded by MIL-M-7866, See Item 19)		
(22) Aviator's Breathing Oxygen	MIL-O-27210		Obtain locally
(23) Thread Compound, Low Pressure Oxygen System	MIL-T-5542		Obtain locally
(24) Thread Sealant, High Pressure Oxygen System	MIL-T-27730		Obtain locally

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(25) Soap Solution, Oxygen System Leak Testing	MIL-L-25567	TEC 913 Leak Detector	TEC Manufacturing Company Monterey Pass Road Monterey Park, California 91734
(26) Anti-Seize Compound, White Lead Base	TT-A-580	Armite Product	Armite Laboratories 1845 Randolph Street Los Angeles, California 90001
(27) Petrolatum, Technical	VV-P-236 or Pharmaceutical Grade Petroleum Jelly (Replaces MIL-T-5544)		Obtain locally
(28) Thread Locking Compound		Loctite Sealant, Grade A (for studs over 2 inches in dia.) Stud Loc (for studs 1/4 to 2 inches in dia.)	Loctite Corporation 705 N. Mountain Road Newington, Connecticut 06111
(29) Anti-Seize Compound	MIL-F-16232, Type A Class 2		Obtain locally
(30) Solvent, Propeller Slip Ring Cleaning		CRC2-26	CRC, Inc, 885 Louis Drive Warminster, Pennsylvania 18974
(31) Cleaning Agent		Turco 4217	Turco Products, Inc. 7300 Bolsa Avenue Westminster, California 92684
(32) Paint Stripper		Turco 4260	Turco Products, Inc. 7300 Bolsa Avenue Westminster, California 92684
(33) Solvent	TT-N-95, Type II	Naphtha, Aliphatic	Obtain locally
(34) Solvent	TT-M-261	Methyl Propyl Ketone (MPK)	Obtain locally
(35) Acetone	ASTM D-329 (Supersedes O-A-51)		Obtain locally
(36) Technical Toluene	TT-T-548		Obtain locally
(37) Vinyl Thinner		T607 Vinyl Primer	Pratt and Lambert, Inc. Coatings Division P.O. Box 2153 16116 East 13th Street Wichita, Kansas 67201

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(38) Solvent, Dry Cleaning	PD-680, Type III		Obtain locally
(39) Gasket Cement		Permatex No. 2	Permatex Company, Inc. Division of Loctite Corporation 705 N. Mountain Road Newington, Connecticut 06111
(40) Cement		A56B	B.F. Goodrich Company P.O. Box 5471 Akron, Ohio 44313
(41) Cement, General	MIL-A-1154	EC1403 (Same as EC1300L)	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(42) Adhesive		EC2141	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(43) Adhesive		EC1300L (Same as EC1403)	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(44) Adhesive Primer		Bostic 1007P Primer	USM Corporation, Subsidiary of Emhart Industries, Inc. Bostik Division Boston Street Middleton, Massachusetts 01949
(45) Sealer		EC2084	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(46) Sealer		EC801A2	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(47) Urethane Primer		AA92 Urethane Primer	The Sherwin-Williams Company 101 Prospect Avenue NW Cleveland, Ohio 44115
(48) Primer (Epoxy)	MIL-P-23377		Pratt and Lambert, Inc. Coatings Division P.O. Box 2153 16116 East 13th Street Wichita, Kansas 67201

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
			Sterling Lacquer Manufacturing Company 3150 Brannon Avenue St. Louis, Missouri 63139
			Ameritex Chemical and Coatings Company, Inc. 801 East Lee Irving, Texas 75060
(49) Wash Primer		EX2016G Wash Primer	Pratt and Lambert, Inc. Coatings Division P.O. Box 2153 16116 East 13th Street Wichita, Kansas 67201
(50) Primer		A4004	Dow Corning P.O. Box 997 3901 S. Saginaw Road Midland, Michigan 48602
(51) Acid Etching Primer	MIL-C-8514		Obtain locally
(52) Air-Conditioning Refrigerant	Dichlorodifluoro-methane	Racon 12	Racon, Inc. 6040 S. Ridge Road Wichita, Kansas 67215
		Genetron 12	Allied Chemical P.O. Box 1021 101 Columbia Road Morristown, New Jersey 07960
		Freon 12	DuPont, Inc. Freon Products Division 1007 Market Street Wilmington, Delaware 19898
(53) Synthetic Rubber Coating		PR611M	Courtaulds Aerospace, Inc. P.O. Box 1800 5454 San Fernando Road Glendale, California 91209
(54) Turco Metal-glo No. 3			Turco Products, Inc. 7300 Bolsa Avenue Westminster, California 92684

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(55) Aerodynamic Smoother		White Star	Dynatron-Bondo Corporation 3700 Atlanta Industrial Parkway NW Atlanta, Georgia 30331
(56) Laminated Glass Cloth	MIL-P-9084	Trevano	Coast Manufacturing and Supply Inc. Box 71 Livermore, California 94550
		Uniglass	United Merchants and Manufacturing, Inc. 1407 Broadway, New York, New York 10018
(57) Resin	MIL-R-7575	Laminac 4116	American Cyanamid Company Industrial Chemicals and Plastics Division South Cherry Street Wallingford, Connecticut 06492
		Glidpol 1001	The Glidden Company 925 Euclid Avenue Cleveland, Ohio 44114
(58) Door-Ease			American Grease Stick Company 2651 Hoyt Muskegon, Michigan 49443
(59) Size 1 Teflon Tape	MIL-T-27730		Obtain locally
(60) Sanding Surfacer		P900 Primer Surfacer	Ameritex Chemical and Coatings Company, Inc. 801 East Lee Irving, Texas 75060
(61) Grease		Molykote GN	Dow Corning P.O. Box 997 3901 S. Saginaw Road Midland, Michigan 48602

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(62) Gasket Cement		Permatex No. 2	Permatex Company, Inc. Division of Loctite Corporation 705 N. Mountain Road Newington, Connecticut 06111
(63) Adhesive		EC2216	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(64) Translucent Adhesive		RTV108	General Electric Company Silicone Products Department Hudson River Road Waterford, New York 12188
(65) Cement		Bostic 1009A with Bostic 1008B	USM Corporation, Subsidiary of Emhart Industries, Inc. Bostik Division Boston Street Middleton, Massachusetts 01949
(66) Epoxy Black, Gloss Finish No. 54-515			Pratt and Lambert, Inc. Coatings Division P.O. Box 2153 16116 East 13th Street Wichita, Kansas 67201
(67) Aluminum Pressure Sensitive Foil tape 3 x 0.002 inch			Owens Corning Fiberglass Corporation Fiberglass Tower Toledo, Ohio 43604
(68) Anti-Seize Compound	MIL-P-16232 Type M, Class 2		Obtain locally
(69) Industrial Oven Insulation 1 inch Thick			Owens Corning Fiberglass Corporation Fiberglass Tower Toledo, Ohio 43604:erb
(70) Jet Fuel Anti-icing Inhibitor	MIL-DTL-85470(B) (Supersedes MIL-I-27686)	Hi-Flash Prist	Prist Division PPG Industries, Inc. 5926 F.M. 1960 West Houston, Texas 77069

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(71) Lubricant	MIL-L-8937 Cancelled		Electrofilm Manufacturing Company P.O. Box 55669 25395 Rye Canyon Road Valencia, California 91355
(72) Toilet (Flush Type) Cleaner		Sana-Pak No. 2031	Celeste Industries Corporation P.O. Box 775 Clifton Industrial Park Easton, Maryland 21601
(73) Solvent		Varsol	Exxon Company U.S.A. P.O. Box 2180 800 Bell Street Houston, Texas 77252
(74) Metal Protector	MIL-C-16173, Grade 2	LPS No. 3	Holt Lloyd Corporation 4647 Hugh Howell Road Tucker, Georgia 30084
(75) Acrylic Base Adhesive	MIL-A-8576	PS-30 Cement	Cadillac Plastic Chemical Company 410 Freilenhuisen Newark, New Jersey 07114
(76) Cement		MP-1000-D-5 Cement	Cresent Engineering and Research Company 5540 N. Peck Road El Monte, California 91732
(77) Anti-Seize Compound MICA Base (Glow Plug)		Mica-Lube No. A768	B & G Service company, Inc. Emerson, New Jersey
(78) Sealant		No. 150	Carborundum Company Box 337 Niagara Falls, New York 14302
(79) Conditioning Paint		776SR	Minnesota Mining and Manufacturing Company 3M Center St. Paul, Minnesota 55144
(80) Rigidizer			Carborundum Company Box 337 Niagara Falls, New York 14302

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(81) Circuit Coat (Printed Circuit Boards)		CE-1155	Furane-Plastics, Inc., 5121 San Fernando Road, Los Angeles, California 90039
(82) Lubricating Grease		Moly-Tex Grease No.2	Texaco, Inc. 50 West 50th Street New York, New York 10017
(83) Mineral Oil		SAE 30	Obtain locally
(84) Corrosion Preventive Compound	MIL-C-16173, Grade 1	Braycote 137	Castrol, Inc. 16715 Von Karman Avenue Suite 230 Irvine, California 92714
		Petrotech 1-4	Penreco, Division of Pennzoil Company, P.O. Box 2967 700 Miliam Street Houston, Texas 77252
(85) Paint Stripper		Turco 5351 or equivalent	Turco Products, Inc. 7300 Bolsa Avenue Westminster, California 92684
(86) Grease, Aircraft, High Temperature	MIL-G-3545 (Superseded by MIL-G-81322, See Item 17)		
(87) Tire Sealant		Aero Seal (Gallon or Pint)	Hawker Beechcraft Service Centers
(88) Corrosion Preventative Compound	MIL-C-16173, Grade 2	Petrotect Grade 2 (use only products qualified under products list 16173)	Obtain locally
(89) Chemical Conversion Coating for Aluminum and Aluminum Alloys	MIL-C-81706 per MIL-C-5541	Alodine 1200, 1200S 1201, 1203, or 600	Parker & Amchem 32100 Stephenson Highway Madison Heights, Michigan 48071
(90) Isopropyl Alcohol	TT-I-735		Obtain locally
(91) Compound, Deicing		Icex	B.F. Goodrich Company Chemical Deicing Group Systems P.O. Box 1277 1555 Corporate Woods Parkway Uniontown, Ohio 44685
(92) Trichloroethane	ASTM D4080 (Supersedes O-T-634)		Obtain locally

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(93) Sealing Compound		EC1239	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(94) Sealing Compound		Pro-Seal 890B1/2	Courtaulds Aerospace, Inc. P.O. Box 1800 5454 San Fernando Road Glendale, California 91209
(95) Adhesive		EC847	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(96) Sealing Compound		EC1239	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(97) Tape		EC1202	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
(98) Desiccant	MIL-D-3464		Obtain locally
(99) Oil	SAE J1899	Grade 1100	Obtain locally
(100) Primer, Zinc Chromate	TT-P-1757		Obtain locally
(101) Preservative, Hydraulic Fluid	MIL-H-6083	Averx 904	Mobil Oil Corporation 150 E. 42nd Street New York, New York 10017
(102) Adhesive		EC2216A/B	Minnesota Mining and Manufacturing Company, St. Paul, Minnesota
		Epibond-104	Ciba-Geigy Corporation Furane Products Division 5121 San Fernanco Road West Los Angeles, California 90039
		Metal Set A4	Smooth-On, Inc. 1000 Valley Road Gillette, New Jersey 07933
(103) Sealer		*PR-1221B1/2 or PR-1221B8	Courtaulds Aerospace, Inc. P.O. Box 1800 5454 San Fernando Road Glendale, California 91209

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(104) Battery		D-Cell, Hercules HS150 (or equivalent)	Union Carbide Corporation Battery Products 25225 Detroit Road Westlake, Ohio 44145
(105) Electrical Conductive Sealant		Tecknit 72-00002	Technical Wire Products, Inc. 129 Dermody Street Cranford, New Jersey 07016
(106) Adhesive		Silastic 140	Dow Corning P.O. Box 997 3901 S. Saginaw Road Midland, Michigan 48602
(107) Leak Check Fluid		Leak-Tek 372	American Gas and Chemical Company, LTD 220 Pegasus Avenue Northvale, New Jersey 07647
(108) Lubricant		CRC 3-36	CRC, Inc, 885 Louis Drive Warminster, Pennsylvania 18974
(109) Lubricant		LPS No. 1	Holt Lloyd Corporation 4647 Hugh Howell Road Tucker, Georgia 30084
(110) Spray Lubricant		WD-40	WD-40 Company 1061 Cudahy Place P.O. Box 80607 San Diego, California 92110
(111) Lubricant		Molykote M-77	Dow Corning P.O. Box 997 3901 S. Saginaw Road Midland, Michigan 48602
			Dixie Bearing, Inc. 1408 Haines Avenue Jacksonville, Florida 32206
(112) Grease, Aircraft High Temperature		Aeroshell Grease No.5	Shell Oil Company 1 Shell Plaza Houston, Texas 77001

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(113) Primer		Locquic Primer Grade T	Loctite Corporation 705 N. Mountain Road Newington, Connecticut 06111
(114) Sealing Compound		Loctite Sealing Compound No. 277	Loctite Corporation 705 N. Mountain Road Newington, Connecticut 06111
(115) Sealing Compound		Pro-Seal 890B4	Courtaulds Aerospace, Inc. P.O. Box 1800 5454 San Fernando Road Glendale, California 91209
(116) Adhesive		EA-901NA	Dexter Hysol Aerospace, Inc. P.O. Box 312 2850 Willow Pass Road Pittsburg, California 94565
(117) Coating	MIL-I-46058	Conathane CE-1155	Conap, Inc. 1405 Buffalo Street Olean, New York 14760
(118) Coating	MIL-I-46058	Conathane CE-1164	Conap, Inc. 1405 Buffalo Street Olean, New York 14760
(119) Solvent		Product S8	Conap, Inc. 1405 Buffalo Street Olean, New York 14760
(120) Primer		Loctite Grade N	Loctite Corporation 705 N. Mountain Road Newington, Connecticut 06111
(121) Retaining Compound		Loctite No. 75	Loctite Corporation 705 N. Mountain Road Newington, Connecticut 06111
(122) Dry Spray Lubricant		Lubriplate Aero	Fiske Brothers Refininig Company 129 Lockwood Newark, New Jersey 07105
(123) Filler		Devcon 11220	Devcon Corporation 30 Endicott Street Danvers, Massachusetts 01923

Table 6
CONSUMABLE MATERIALS (Continued)

Material	Specification	Products	Supplier
(124) Filler		Devcon 11410	Devcon Corporation 30 Endicott Street Danvers, Massachusetts 01923
(125) Biocidal Agent		Biobor JF	U.S. Borax and Chemical Corporation 8600 W. Bryn Mawr Avenue Suite 710 Chicago, Illinois 60631
(126) Sealant		PR1425B1/2	PRC DeSoto International Inc. 5430 San Fernando Rd. Glendale, CA 91209
* Use PR-1221B1/2 when assembly time is within 30 minutes after sealer is mixed and use PR1221B8 when assembly time will take more than 30 minutes.			

7. OPERATING PARAMETERS

A. Power Plant (Specifications and Information)

PT6A-20 TURBOPROP ENGINE	
Operational Shaft Horsepower at 2,200 rpm	550 hp (For basic 99 without kit 99-5014 installed, serials U-1 thru U-55, except U-50)
Engine Type	Free Turbine
Type of Combustion Chamber	Annular
Compressor Ratio	6.3:1
Propeller Shaft Rotation (looking forward)	Clockwise
Propeller Shaft Gear Ratio	0.0668:1
Dry Weight	275 lbs.
PT6A-27 and PT6A-28 TURBOPROP ENGINE	
Operational Shaft Horsepower at 2,200 rpm	99A: 550 hp; A99A, B99: 680 hp 99 with kit 99-5014 installed, serials U-1 thru U-55, except U-50: 680 hp
Engine Type	Free Turbine
Type of Combustion Chamber	Annular
Compressor Ratio	7.0:1
Propeller Shaft Rotation (looking forward)	Clockwise
Propeller Shaft Gear Ratio	0.0668:1
Dry Weight	289 lbs.

PT6A-36 TURBOPROP ENGINE	
Operational Shaft Horsepower at 2,200 rpm	C99: 715 hp, serials U-50; U-165 and After
Engine Type	Free Turbine
Type of Combustion Chamber	Annular
Compressor Ratio	7.0:1
Propeller Shaft Rotation (looking forward)	Clockwise
Propeller Shaft Gear Ratio	0.0668:1
Dry Weight	311 lbs.

B. Engine Oils

PT6A-20 TURBOPROP ENGINE	
Type	Use type recommended in Consumable Materials Table 6.
Oil Tank Capacity	2.3 U S Gallons
Oil Tank Expansion Space	0.7 U S Gallons
Oil Tank Usable Quantity	1.5 U S Gallons
Oil Pressure	65 to 75 psi
Oil Consumption - Maximum Average	0.5 lb/hr
PT6A-27 and PT6A-28 TURBOPROP ENGINE	
Type	Use type recommended in Consumable Materials Table 6.
Oil Tank Capacity	2.3 U S Gallons
Oil Tank Expansion Space	0.7 U S Gallons
Oil Tank Usable Quantity	1.5 U S Gallons
Oil Pressure	80 to 100 psi
Oil Consumption - Maximum Average	0.2 lb/hr
PT6A-36 TURBOPROP ENGINE	
Type	Use type recommended in Consumable Materials Table 6.
Oil Tank Capacity	2.3 U S Gallons
Oil Tank Expansion Space	0.7 U S Gallons
Oil Tank Usable Quantity	1.5 U S Gallons
Oil Pressure	85 to 105 psi, 40 psi minimum at idle
Oil Consumption - Maximum Average	0.2 lb/hr

C. Propellers

Hartzell Propeller Co	HC-B3TN-3/T10173E-8 (U-1 thru U-164, except U-50)
	HC-B3TN-3B or 3M/T10173K8 (U-50, U-165 and After)
Type	Reversing
Deicer Brush Block Replacement	Goodrich, replace brushes when 15/32 inch maximum distance is measured on wire inserted through inspection hole in brush support block.
Deicer Brush Module Replacement	Goodrich, replace module when 3/8 inch of brush material remains. Refer to Chapter 30-60-00.
Propeller Reverse Brush Block	Replace when carbon block metal retaining clip(s) are 0.005 inch or less from the low pitch stop collar.

D. Fuel System

Engine Fuel	Jet-A (NATO F-30, F-34) Jet A-1 (JP-5, NATO F-42) Jet B (JP-4, JP-8, NATO F-40) MIL-J-5624
Alternate Fuel (Limited to 150 hours during each overhaul period)	80 (80/87) 100LL Blue 100 Green (100/130) 115 (115/145)
Fuel Capacity	99, 99A: 370 Gallons Total A99A: 254 Gallons Total B99, C99: 368 Gallons

E. Miscellaneous

Brake Fluid Reservoir	Located immediately behind aft bulkhead in forward baggage compartment, fill the reservoir to the full mark on the dipstick.
Heater Spark Plug Gap (U-1 thru U-164, except U-50)	0.187 (3/16) to 0.250 (1/4).
Brake Wear Limits (Standard Wheel and Brake)	If the thickness of the segmented carrier and lining assembly is 0.198 or less, overhaul the brakes.
Brake Wear Limits (Flotation Wheel and Brake)	The brake should be removed from service and overhauled when the end of the wear indicator pin is flush with the face of the pin bushing (with brakes applied).
Pressure Relief Valve Adjustment	Gage reading of 18 $\pm$ 1 psi with reading of 70 to 80 percent registered on the tachometer. Reading is typical for one or both engines operating, due to dual regulator valve installation.

Oxygen Cylinder(s)	Fill slowly to 1800 ± 50 psig at 70°F. Increase 3.5 psig for each degree increase. Decrease 3.5 psig for each degree decrease.

F. Service

Battery	100 hours or 30 days whichever occurs first.
Main Gear Tires	**NOTE** Do not mix 8-ply rating tires with 10-ply rating tires on the same landing gear. 99, 99A (Without Kit 99-5014 installed) with 8-ply rated tires. Inflate 88 to 92 psi. 99, 99A (With Kit 99-5014 installed) and A99A with 8-ply rated tires. Inflate 92 to 96 psi. 99, 99A (Without Kit 99-5014 installed) with 10-ply rated tires. Inflate 78 to 82 psi. 99, 99A (With Kit 99-5014 installed) and A99A with 10-ply rated tires. Inflate 82 to 86 psi. B99 and C99 With 8-ply tires, inflate to 100 to 104 psi.
Flotation Gear Tires	99, 99A (Without Kit 99-5014 installed) with 6-ply rating tires: inflate 48 to 52 psi. 99, 99A (With Kit 99-5014 installed) and A99A with 6-ply rating tires: inflate 51 to 55 psi. B99 and C99 With 6-ply (tube type) tires, inflate 55 to 59 psi.
Nose Gear Tires	99, 99A (Without Kit 99-5014 installed) inflate 50 to 55 psi. 99, 99A (With Kit 99-5014 installed) and A99A inflate 53 to 57 psi. B99 and C99 With 6-ply tires inflate 55 to 59 psi.
Oil Change	Refer to P&W S/B 1001 or subsequent.
Instrument Air Filter	400 hours or sooner in dust or heavy smoke conditions.
Main Strut Extension	4 1/4 to 4 1/2 inches.
Main Strut Extension (Flotation)	4 1/2 to 4 3/4 inches.
Nose Strut Extension	3 to 3 1/2 inches.
Heater Fuel Filters (U-1 thru U-164, except U-50)	100 hours.
Fuel System Screens and Strainers	100 hours or sooner if conditions warrant.
Pitot Drains	100 hours or sooner if conditions warrant.
Shimmy Damper	100 hours. Full when distance from end of piston rod to bottom of hole in aft floating piston is 2 5/8 to 2 7/8 inches. When the distance is 1/2 inch from full the damper should be serviced.
Stall Warning Switch	Adjust to 7 to 9 mph above a complete stall.

G. Flight Control Surface Travel

Flaps	0° Full up. 13° +1°/-2° Approach. 43° +1°/-2° Full Down.
Aileron	16° ± 1 1/2° Up from 0° position. 22° ± 1 1/2° Down from 0° position.
Aileron Tab	15° ± 1 1/2° Up. 15° ± 1 1/2° Down.
Rudder	20° to 21° Left. 26° to 27° Right.
Rudder Tab	30° ± 1 1/2° Left. 30° ± 1 1/2° Right.
Elevator	12° +1°/-0° Up. 15° +1°/-0° Down (with stabilizer in 0° position).
Stabilizer	99, 99A: 3 1/4° to 3 1/2° Leading edge UP; 99 and A99 With Kits 99-5011 and 99-5015 installed, A99A, B99, C99: 4° to 4 1/2° Leading edge UP; 3 1/4° Leading edge DOWN.

H. Flight Control Cable Tension Settings

NOTE All cable tensions given are tensiometer readings at 59°F.	
Rudder Cables	85 ± 5 lbs.
Elevator Cables	29 ± 8 lbs.
3/16 inch Dia Aileron Cables	47 +15/-0 lbs.
Aileron Cables in Fuselage	37 +5/-0 lbs.
Rudder Tab Cables	9 +3/-2 lbs.
Aileron Tab Cables	9 +3/-2 lbs.

8. TORQUES

A. Propeller

Mount Bolts	100 +0/-10 foot-pounds (1,080 to 1,200 inch-pounds).
Nuts on Low Pitch Stop Rods	12 foot-pounds.

B. Engine Components and Mounting Bolts

Mount Bolts (large bolt through center of mount)	400 to 550 inch-pounds (U-1 thru U-164, except U-50). 480 to 600 inch-pounds (U-50, U-165 and After).
Mounting Attach Bolts	180 to 225 inch-pounds (U-1 thru U-164, except U-50). 275 to 330 inch-pounds (U-50, U-165 and After).
Engine V-brace to Firewall	365 to 390 inch-pounds.
Engine Control Cable Attach Bolts	20 to 30 inch-pounds.
Upper Mount Rods to the Firewall	30 to 40 inch-pounds, safety with cotter pin.
Glow Plugs or Spark Igniters	Tighten to 300 inch-pounds, back off to zero inch-pounds, then torque to 300 to 360 inch-pounds.
Bleed Air Adapter Bolt	50 to 70 inch-pounds.
Governor, (Primary, Propeller Overspeed, Fuel Topping)	170 to 190 inch-pounds.
Exhaust Stack Attach Bolts	50 to 70 inch-pounds.

C. Wing Attach Bolts

Upper Forward Spar Attachment	1,300 +0/ -120 inch-pounds (U-1 thru U-150, except U-50). 2,500 +0/ -120 inch-pounds (U-50, U-151 and After).
Upper Rear and Lower Rear Spar Attachment	1,300 +0/ -120 inch-pounds (U-1 and After).
Lower Forward Spar Attachment	Tighten the bolt and check the preload indicating washer assembly as outlined in Chapter 57-00-00. Torque should be applied to the nut except installations with barrel nuts.

D. Landing Gear

Nose Landing Gear Drag Brace, Lower Attaching Stud	200 to 400 inch-pounds.
Nose Gear Hinge Bolts	200 to 400 inch-pounds on lubricated bolts. 360 to 720 inch-pounds on nonlubricated bolts.
Main Landing Gear Hinge Bolts	800 inch-pounds, or next higher castellation not to exceed 1,200 inch-pounds to allow cotter pin installation.

E. Heater

Heater Spark Igniter	28 foot-pounds (U-1 thru U-164, except U-50).
Spark Igniter Shielded Lead	20 foot-pounds (U-1 thru U-164, except U-50).
Upper and Lower Section of Spark Igniter	36 foot-pounds (U-1 thru U-164, except U-50).
Igniter Shielded Lead	25 foot-pounds (U-1 thru U-164, except U-50).

F. Empennage

Stabilizer Hinge Bracket Mount Bolts	50 to 70 inch-pounds.

G. Air-Conditioner

Air-Conditioner Compressor to Mount Assembly	95 to 110 inch-pounds.
Firewall Line Connection	20 to 30 inch-pounds.

Published by
Hawker Beechcraft Corporation
P.O. Box 85
Wichita, Kansas
67201-0085 USA

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Hawker Beechcraft Corporation

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL

P/N 99-590015-1, Revision C4, May 1/17

The chapters which have been revised or added are listed below with the Highlights of each change. Insert the Revision in accordance with the attached Instruction Page. Enter the Revision and the date inserted on the Record of Revisions page of this manual. The Highlights Page may be retained with the manual for future reference.

Highlights

Chapter/Section	Description
Title Page Vol 1	Revision date and number revised.
Title Page Vol 2	Revision date and number revised.
A Page	Revision date and number revised.
Chapter 06	
06-LOEP	Page dates and page numbers revised.
06-10-00, 001	Corrected page header.

1. INFORMATION

Since a wide variety of avionic components and equipment is available, avionic manufacturers normally supply parts and servicing manuals with each set/component. The manufacturer of the equipment should be contacted when additional parts or servicing information is required. Reissues and revisions are automatically provided to the subscription holders of the manual.

Additional publications are listed on the web at http://pubs.hawkerbeechcraft.com. For more information on these publications, or to check subscription status, contact the Technical Manual Distribution Center (TMDC) at 1.800.796.2665 or 316.676.8238, fax 316.671.2540, E-mail tmdc@hawkerbeechcraft.com.

The Interactive Maintenance Library (IML) contains selected Manuals in a digital format. This manual, along with others, is available on CD-ROM and Online. Optional paper copies of the manuals on the CD-ROM are available for purchase.

Help Line phone support for IML CD and Online Users is available 8:00 AM to 4:30 PM Central Time (US and Canada). During off-hours, leave a detailed voicemail message. Calls will be returned within one business day. Contact the Help Line at 1.800.240.2959 or 1.316.676.3053, E-mail iml@hawkerbeechcraft.com.

Comprehensive user guides for the IML CD and Online manuals are available on the Hawker Beechcraft Corporation (HBC) Technical Publications website http://pubs.hawkerbeechcraft.com. Illustrated and detailed procedures for using IML Features are included in the downloadable user guides.

A. General

This Maintenance Manual applies to the Hawker Beechcraft Corporation Model 99 Airliner Series (U Serials). The wording **MODEL 99 AIRLINER SERIES** appearing in the **masthead** at the top of each page is not intended to indicate applicability beyond that noted above.

In addition to this manual and its subsequent revisions, additional maintenance information is published in the form of Hawker Beechcraft Corporation service bulletins. The information contained in these service bulletins is an integral part of, and is to be used in conjunction with, the information contained in this manual.

The Model 99 Airliner Series Maintenance Manual is prepared in accordance with the requirements of the ATA Specification 2200 (Air Transport Association of America) with respect to the arrangement and content of the System/Chapters within the designated chapter-numbering system. This manual includes the maintenance information required to be available by 14 CFR Part 23. This Maintenance Manual is supplemented by the following publications:

- Model 99 Airliner Parts Catalog, P/N 99-590014 (U-1 thru U-164, except U-50)
- Model 99 Airliner Parts Catalog, P/N 99-590031-1 (U-50, U-165 and After)
- Model 99 Airliner Series Wiring Diagram and Electrical System Manual, P/N 99-590028 (U-1 thru U-164, except U-50)
- Model 99 Airliner Series Wiring Diagram and Electrical System Manual, P/N 99-590030-5 (U-50, U-165 and After)
- Model 99 Airliner Series Component Maintenance Manual, P/N 99-590015-5B

It shall be the responsibility of the owner/operator to ensure that the latest revision of the publications referenced in this manual are utilized during operation, servicing and maintenance of the airplane.

Hawker Beechcraft Corporation expressly reserves the right to supersede, cancel and/or declare obsolete any parts, part numbers, kits or publications that may be referenced in this manual without prior notice.

WARNING: Use only parts obtained from sources approved by Hawker Beechcraft Corporation, in connection with the maintenance and repair of Hawker Beechcraft Corporation airplanes.

Genuine Hawker Beechcraft Corporation parts are produced and inspected under rigorous procedures to insure airworthiness and suitability for use in Hawker Beechcraft Corporation airplane applications. Parts purchased from sources other than those approved by Hawker Beechcraft Corporation, even though outwardly identical in appearance, may not have the required tests and inspections performed, may be different in fabrication techniques and materials, and may be dangerous when installed in an airplane.

Salvaged airplane parts, reworked parts obtained from sources not approved by the Hawker Beechcraft Corporation or parts, components or structural assemblies, the service history of which is unknown or cannot be authenticated, may have been subjected to unacceptable stresses or temperatures or have other hidden damage, not discernible through routine visual or usual nondestructive testing techniques. This may render the part, component or structural assembly, even though originally manufactured by the Hawker Beechcraft Corporation, unsuitable and unsafe for airplane use.

Hawker Beechcraft Corporation expressly disclaims any responsibility for malfunctions, failures, damage or injury caused by use of parts not approved by the Hawker Beechcraft Corporation.

Any maintenance requiring the disconnection and connection of flight control cables, plumbing, electrical connectors or wiring requires identification of each side of the component being disconnected to facilitate correct reassembly. At or prior to disassembly, components should be color coded, tagged or properly identified in a way that it will be obvious how to correctly reconnect the components. After connection of any component, remove all identification tags. Check all associated systems for correct function prior to returning the airplane to service.

B. Correspondence

If a question should arise concerning the care of your airplane, it is important to include the airplane serial number in any correspondence. The serial number appears on the model designation placard. Refer to Chapter 11 for placard location.

C. Publications Change Request (PCR)

If an irregularity or missing information is noted, the user of this manual may access a PCR form at http://pubs.hawkerbeechcraft.com. Instructions on how to submit a PCR are available on the web page.

D. Normal Revisions

Normal Revisions to this manual are issued to provide changes to maintenance information.

1. Paper Revision

   That portion of text which has been revised by the addition of, or a change in, information is denoted by a solid revision bar adjacent to the text. The date printed on the bottom of each page can be compared to the “A” page to determine the revision number. Each revised page will ONLY show revision bars for text changed by the revision. There will not be a revision bar if text was deleted from the page.

   Revised illustrations will be identified by a revision bar printed on the side of the page.

2. CD-ROM Revision

   Normal revised text on the CD-ROM will be highlighted yellow across the revised passage of text. For each revision of this manual, a new CD-ROM will be issued.

   The CD-ROM may contain revised illustrations. Revisions to the illustrations are not identified.

E. Temporary Revisions

Temporary Revisions to this manual are issued to provide maintenance information in the interim between normal revisions. Each temporary revision is issued by the chapter number to which it applies, followed by a sequential number in the order of publication (Temporary Revisions 12-1, 12-2, etc.). If relevant, the information in the temporary revision should be included in the next normal revision of the manual.

1. Paper Temporary Revisions

   Temporary Revisions are printed on yellow paper and are to be inserted in the maintenance manual in accordance with the instructions provided and adjacent to applicable chapter, section, and subject matter in the manual.

2. CD-ROM Temporary Revisions

   A new CD-ROM will be issued for each Temporary Revision to this manual. This information is listed in conjunction with the applicable chapter, section, subject on the CD-ROM.

F. Revised Text

That portion of text which has been revised by the addition of, or a change in, punctuation and/or information is denoted by a solid revision bar adjacent to the textual column in the margin of this paragraph. Each page may or may not have revision bars. That date printed on the bottom of each page indicates when the information on that page was changed. Each page will ONLY show revision bars for punctuation and/or text changed by the current revision. Revised text in the IML will be denoted by yellow highlighting.

G. Revised Illustrations

When an illustration is modified or a new illustration is added, it will be noted by a solid line (revision bar) along the outside margin of the illustration.

H. Warnings, Cautions, and Notes

- **WARNING** - Brings attention to an operating procedure, inspection or maintenance practice, which if not correctly followed, could result in personal injury or loss of life.
- **CAUTION** - Brings attention to an operating procedure, inspection, repair or maintenance condition, which if not strictly observed, could result in damage or destruction of equipment.
- **NOTE** - Brings attention to an operating procedure, inspection, repair or maintenance condition, which is essential to highlight.

2. MANUAL LAYOUT

A. Title Page

A Title page is located at the beginning of the manual and provides the part number of the manual, and lists all aircraft models pertaining to this manual and their respective serial numbers. Information throughout this manual is applicable to all serial numbers listed on the title page except where specifically stated.

B. List of Effective Revisions/List of Effective Pages

The printed manual will have a List of Effective Revisions/List of Effective Pages, (“A” page) following the title page of the manual. The List of Effective Revisions page lists the revisions currently effective for the manual. The List of Effective Pages section lists the page effectivity for the Title Page(s), “A” page(s) and Introduction chapter. It will also show the effective pages for an entire manual if the manual does not have individual Chapter List of Effective Pages.

C. Record of Revisions Page

The printed manual will have a Record of Revisions page. The Record of Revisions is provided following the List of Effective Revisions/List of Effective Pages (“A” page). When a revision is inserted, the revision number, the date the revision is inserted into the manual, and the initials of the person(s) inserting the revision should be recorded on this page.

IML CD and Online Manuals do not include a Record of Revisions page. Revisions standard is avalible by a link to the catalog Title Page or accessible by clicking Help on the menu bar and selecting ”About This IML Book...”.

D. Log of Temporary Revisions Page

The printed manual will have a Log of Temporary Revisions page. The Log of Temporary Revisions Page is located following the Record of Revisions page. The Log of Temporary Revisions page provides a history of each temporary revision, including the revision number which incorporated the temporary revision into the manual.

E. Record of Temporary Revisions Page

The printed manual will have a Record of Temporary Revisions page. The Record of Temporary Revisions Page is located following the Log of Temporary Revisions page. When a temporary revision is inserted or removed from this manual, the appropriate information should be recorded on this page.

F. Introduction

This section contains general and specific information on how to use this manual.

G. Chapter List of Effective Page

The printed manual may have a Chapter List of Effective Pages. The List of Effective Pages follow the Chapter-Divider-Tab and lists the issue date of each page that is effective for that chapter.

H. Chapter Table of Contents Pages

The printed manual may have a Chapter Table of Contents Pages. The Chapter Table of Contents Pages follow the Chapter List of Effective Pages and lists the contents of the data for that chapter.

3. HOW TO USE THE MANUAL

A. ATA Subject Matter Assignment

The contents of this manual are organized into four levels. The four levels are:

(1) Group

These are the primary divisions of the manual that enable broad separation of content. Typical of this division is the separation between Airframe Systems and the Power Plant.

(2) System/Chapter

The various groups are broken down into major systems such as Environmental Systems, Electrical Power, Landing Gear, etc. The systems are arranged more or less alphabetically rather than by precedence or importance. They are assigned a number, which becomes the first element of the standardized numbering system. Thus, the element **28** of the number 28-40-01 refers to the chapter **FUEL**. Everything concerning the fuel system will be covered in this chapter.

(3) Subsystem/Section

The major systems/chapters of an airplane are broken down into subsystems. These subsystems are identified by the second element of the standard numbering system. The element **40** of the number 28-40-01 concerns itself with the indicating section of the fuel system.

(4) Unit/Subject

The individual units within a subsystem/section may be identified by the third element of the standard numbering system. The element **01** of the number 28-40-01 is a subject designator. This element is assigned at the option of the manufacturer and may or may not be used.

B. Application

Any publication conforming to the ATA format will use the same basic numbering system. Thus, whether the manual is a Model 99 Airliner Maintenance Manual, or a Model 99 Airliner Wiring Diagram Manual, the person wishing information concerning the indication portion of the fuel system, would refer to the System/Chapter Tab **28-FUEL**. The table of contents in the front of this chapter will provide a list of subsystems covered in this chapter. For example, the fuel system chapter with a full index would contain:

28-00 - General

28-10 - Storage (Tanks, cells, necks, caps, instruments, etc.).

28-20 - Distribution (Fuel lines, pumps, valves, controls, etc.).

28-30 - Dump (If in-flight dumping system is installed, it would appear here).

28-40 - Indicating (Quantity, temperature, pressure, etc., does not include engine fuel flow or pressure).

C. References to Procedures, Figures, Equipment and Materials

A system has been developed to provide a method of allowing the user to quickly locate data referred to in the other manuals. This system provides information for both the printed manual as well as a hyperlink in electronic manuals.

Here are a few examples:

When the user is directed to a procedure in another manual, the text will be as follows: Refer to SERVICING THE AIR CONDITIONER SYSTEM in Chapter 12-10-00. The procedure is found in Chapter 12-10-00 of the Maintenance Manual.

When the user is in a procedure and is directed to information about a tool, piece of equipment or material used, the text will be as follows: (104, Table 6, 91-00-00), which contains a listing of all the special tools and equipment used to maintain the Model 99 airplane. Item 104 in Table 6 in Chapter 91-00-00 is a Battery.

4. CHAPTER/SYSTEM INDEX GUIDE TABLE

The following System/Chapter, Subsystem/Section Index Guide is prepared in accordance with ATA Specification No. 2200. This outline in general is used in Maintenance Manuals, Parts Catalogs and Wiring Diagram Manuals. The organization of information will follow this outline, however the Subsystem/Sections may vary slightly to accommodate accurate coverage of the specific aircraft.

The following chapters are not applicable to this Maintenance Manual: 29, 49, 53, 54, 60, 65, 70, 73, 75, 78, 80, 81, and 83.

System/Chapter	Subsystem/Section	Title
INTRODUCTION		
AIRCRAFT GENERAL		
05	TIME LIMITS/MAINTENANCE CHECKS	
	10	Time Limits
	20	Scheduled Maintenance Checks
	21	Continuous Inspection Procedures
	50	Unscheduled Maintenance Checks
06	DIMENSIONS AND AREAS	
	10	Airplane Dimensions
	30	Airplane Stations
	50	Airplane Access Panels
07	LIFTING AND SHORING	
	00	General
	10	Jacking
	20	Shoring
08	LEVELING AND WEIGHING	
	00	General
09	TOWING AND TAXIING	
	10	General
10	PARKING AND MOORING	
	10	Parking
	20	Mooring

System/Chapter	Subsystem/Section	Title
11	PLACARDS AND MARKINGS	
	00	General
	20	Exterior Placards and Markings
12	SERVICING	
	00	General
	10	Replenishing
	20	Scheduled Servicing
	30	Unscheduled Servicing
AIRFRAME SYSTEMS		
20	STANDARD PRACTICES - AIRFRAME	
	00-00	Standard Practices - Airframe
	07	Locking Devices
	08	Airplane Finish Care
21	AIR CONDITIONING	
	40	Heating
	41	Bleed Air Heat
	50	Cooling
	60	Temperature Control
22	AUTO FLIGHT	
	10	Autopilot
23	COMMUNICATIONS	
	60	Static Discharging
24	ELECTRICAL POWER	
	00	Electrical System
	30	DC Generation
	31	Battery Power

System/Chapter	Subsystem/Section	Title
24 (Cont'd)	32	Battery Monitor
	40	External Power
	50	Electrical Load Distribution
25	EQUIPMENT/FURNISHINGS	
	10	Flight Compartment
	20	Passenger Seats
	60	Emergency
26	FIRE PROTECTION	
	10	Fire and Smoke Detection
	11	Engine Bleed Air Warning System
	20	Engine Fire Extinguisher
27	FLIGHT CONTROLS	
	00	Flight Controls
	10	Aileron Control System
	20	Rudder Control System
	30	Elevator Control System
	31	Stall Warning System
	40	Horizontal Stabilizer Control System
	50	Flaps
	70	Gust Lock and Damper
28	FUEL	
	00	Fuel System
	10	Storage
	20	Distribution
	21	Fuel System Valves
	22	Fuel Heaters
	40	Indicating
	41	Indicating (99, A99, B99)
	42	Indicating (C99)

System/Chapter	Subsystem/Section	Title
30	ICE AND RAIN PROTECTION	
	10	Airfoil
	20	Air Intakes
	21	Inertial Anti-Icer
	22	Brake Deicer
	30	Pitot System Deicer
	40	Windows and Windshields
	41	Windshield Wiper
	60	Propellers
31	INSTRUMENTS	
	10	Instruments
32	LANDING GEAR	
	00	General (Mechanical)
	01	General (Hydraulic)
	10	Main Gear and Doors
	20	Nose Gear and Doors
	30	Extension and Retraction (Mechanical)
	31	Extension and Retraction (Hydraulic)
	40	Wheels and Brakes
	41	Brake Deice
	50	Steering
	60	Position and Warning
33	LIGHTS	
	00	General
	01	Annunciator System
	10	Flight Compartment
	11	Edgelighted Panels
	20	Passenger Compartment
	30	Baggage and Cargo Compartment
	40	Exterior

System/Chapter	Subsystem/Section	Title
33 (Cont'd)	50	Emergency Lighting
34	NAVIGATION AND PITOT/STATIC	
	10	Flight Environmental Data
35	OXYGEN	
	00	General
36	PNEUMATIC	
	00	General
37	VACUUM	
	00	General
38	WATER/WASTE	
	30	Chemical Toilet
39	ELECTRIC PANELS, PARTS & INSTRUMENTS	
	00	General
	10	Instrument & Control Panels
	20	Electrical & Electronic Equipment Racks
STRUCTURES		
50	CARGO AND ACCESSORY COMPARTMENTS	
	00	General
51	STRUCTURES	
	00	General
52	DOORS	
	10	Passenger/Crew
	20	Emergency Exit
	21	Emergency Exit Signs

System/Chapter	Subsystem/Section	Title
52 (Cont'd)	30	Cargo
55	STABILIZERS	
	00	Empennage
	10	Horizontal Stabilizers
	20	Elevator
	30	Vertical Stabilizer
	40	Rudder
56	WINDOWS	
	00	General
	10	Flight Compartment
	20	Cabin
57	WINGS	
	00	General
	30	Wing Section
	50	Flight Surfaces
PROPELLER		
61	PROPELLERS	
	10	Propeller Assembly
	20	Controlling
	21	Propeller Autofeathering
	22	Propeller Synchrophaser
	23	Indicating
	40	Testing
POWER PLANT		
71	POWER PLANT	
	00	General

System/Chapter	Subsystem/Section	Title
71 (Cont'd)	10	Cowling
	20	Mounts
	30	Fireseals
	50	Electrical Harness
	70	Engine Drains
72	ENGINE	
	00	General
74	IGNITION	
	00	General
76	ENGINE CONTROLS	
	00	General
77	ENGINE INDICATING	
	00	General (U-1 thru U-164, except U-50)
	01	General (U-50, U-165 and After)
	10	Power (U-1 thru U-164, except U-50)
	11	Power (U-50, U-165 and After)
	20	Temperature (U-1 thru U-164, except U-50)
	21	Temperature (U-50, U-165 and After)
79	OIL	
	00	General
82	WATER INJECTION	
	00	General
91	CHARTS	
	00	Charts

List of Effective Pages

Section	PAGE	DATE
	Title Page (Vol 1)	May 1/17
	Logo Page	
	Title Page (Vol 2)	May 1/17
	Logo Page	

Section	PAGE	DATE
	"A" Page	C4
	"B" Page	C4
Introduction	1 thru 13	Nov 1/10

NOTE - The chapter List of Effective Pages is located in the front of each chapter.

LIST OF EFFECTIVE REVISIONS

Part Number	Date	Chapters Affected
99-590015-1C	November 1, 2010	Reissue
99-590015-1C1	November 1, 2011	05, 27, 32, and 56
99-590015-1C2	August 1, 2012	27, 32, and 33
99-590015-1C3	May 1, 2013	30
99-590015-1C4	May 1, 2017	06

Beechcraft

MODEL 99 AIRLINER SERIES MAINTENANCE MANUAL
P/N 99-590015-1

Date	Reason for Update
May 2013	Release of Revision C3

Record of Temporary Revisions

Note: Insert this Record of Temporary Revisions after the Log of Temporary Revisions page(s). When a Temporary Revision is inserted, the temporary revision number, the affected chapter, the date the revision is inserted into the manual, and the initials of the person(s) inserting the revision should be recorded on this page. When a Temporary Revision is removed, enter the manual revision number that incorporated the Temporary Revision and the date the Temporary Revision was removed from the manual.

Temporary Revision No.	Affected Chapter	Inserted		Removed	
		Init	Date	By Revision No.	Date

Hawker Beechcraft

Model 99 Airliner Series

99	(U-1 thru U-35, U-37 thru U-49, U-51 thru U-79, U-86 thru U-88, U-90 thru U-92, U-94, U-95, U-98 thru U-100, U-102, U-103, U-106 thru U-109, U-114, U-119 thru U-122, U-124, U-136)
99A	(U-36, U-80 thru U-85, U-89, U-93, U-96, U-97, U-101, U-104, U-105, U-110 thru U-113, U-115 thru U-118, U-123, U-125 thru U-131, U-133 thru U-135, U-137 thru U-145, U-147)
A99A	(U-132)
B99	(U-146, U-148 thru U-164)
C99	(U-50, U-165 and After)

Maintenance Manual

This manual was formerly called the
Beechcraft 99 Airliner Series Maintenance Manual

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P/N 99-590015-1
Issued: January 1, 1973

P/N 99-590015-1C4
Revised: May 1, 2017

Beechcraft