GENERAL - MAINTENANCE PRACTICES

FUEL HANDLING PRACTICES

When filling the aircraft fuel cells, always observe the following:

a. Service the fuel cells with 100/130 octane fuel or if not available, use 115/145 octane fuel (1, Chart 207, 91-00-00).

b. Make sure the aircraft is statically grounded to the servicing unit.

c. Do not fill fuel cells near open flame or within 100 feet of any open energized electrical equipment capable of producing sparks.

d. Do not insert the fuel nozzle more than 3 inches into the filler neck; to do so may cause damage to the rubber fuel cell.

Most fuel injection system malfunctions can be attributed to contaminated fuel. Inspecting and cleaning the fuel strainers should be considered to be of the utmost importance as a regular part of preventive maintenance.

Normally the fuel strainers should be inspected and cleaned every 100 hours. However, the strainers should be inspected and cleaned at more frequent intervals depending on service conditions, fuel handling equipment and when operating in localities where there is an excessive amount of sand or dust.

Open each of the seven snap-type fuel drains daily to allow condensed moisture to drain from the system.

NOTE

If the cells are to remain unfilled for 10 days or more, apply a thin coating of light engine oil to the inside surface of the cell to prevent deterioration and cracking.

AIRCRAFT DEFUELING

To insure that all fuel is removed from the system, the fuel should be drained through the boost pumps. To expedite the defueling operation, the boost pumps may be used to pump the fuel out of the system. The following steps must be accomplished before energizing the pumps:

a. Apply external power to the aircraft electrical system.

b. Place the fuel selector valve in the "ON" position and the mixture lever in "IDLE CUT-OFF".

c. Remove the filler caps to vent the system.

d. Disconnect the fuel line at the firewall and attach a drain hose. Provide a suitable container for the fuel.

e. Energize the boost pumps.

f. When fuel is no longer pumped from the aircraft, open the sump drains to complete the defueling operation.

"END"

STORAGE - MAINTENANCE PRACTICES

NACELLE FUEL CELL REMOVAL
(Figure 201)

a. Drain and purge the fuel cell.

b. Remove the fuel cell access plate (1), the forward access plate (2) and the vent line access plate (3).

c. Remove clamp from 3-inch interconnect tube in bottom of cell and the vent nipple clamp and interconnect line in wheel well.

d. Remove the fuel transmitter. (Refer to 28-40-00).

e. Unsnap the fuel cell and remove it from the nacelle cavity through the access hole (1).

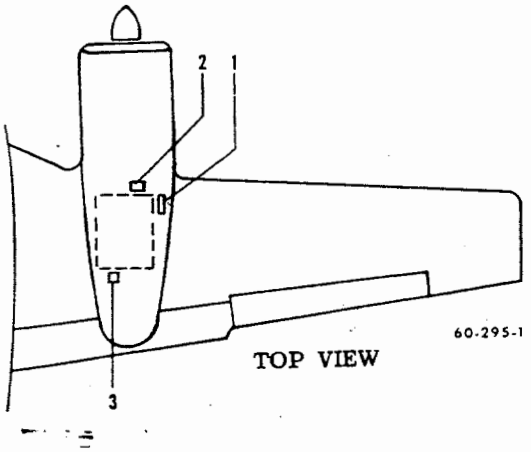
NOTE

Tape edge of access hole to protect cell from damage during removal.

If the fuel cell is to be stored for a period of 10 days or longer, coat the inside of the cell with light engine oil to prevent cracking or deterioration.

NACELLE FUEL CELL INSTALLATION

a. Carefully insert the fuel cell into the nacelle cavity, through access hole (1), and snap in place.



1. Fuel Cell Access and Transmitter
2. Forward Access Plate
3. Nacelle Vent Line Access Plate

Nacelle Fuel Cell Access Openings
Figure 201

b. Install the fuel transmitter. (Refer to 28-40-00).

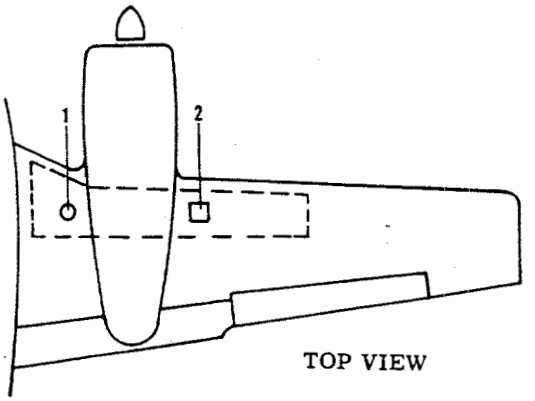
c. Install fuel cell access plate (1).

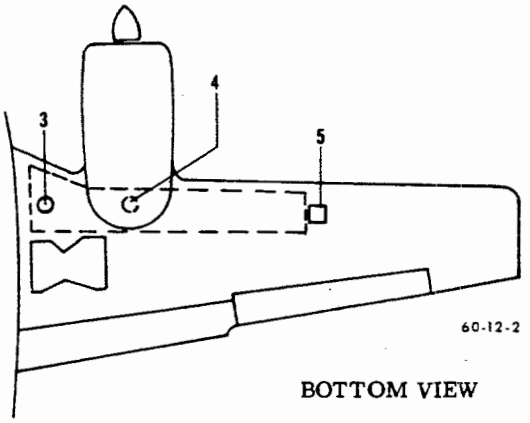
d. Connect 3-inch interconnect tube at bottom of fuel cell, and vent nipple and interconnect line in wheel well. Torque rubber fuel nipples and interconnect clamps to 25± 5 inch-pounds.

e. Install forward access plate (2) and vent line access plate (3).

INBOARD LEADING EDGE FUEL CELL REMOVAL
(Figure 202)

a. Drain and purge the fuel cell.





1. Access Plate and Transmitter.
2. Fuel Cell Access and Transmitter
3. Fuel Pump
4. Fuel Cell Access Plate (Under Removable Aft Nacelle Section)
5. Fuel Cell Access Plate

Inboard Leading Edge Fuel Cell Access Openings
Figure 202

b. Remove the inboard and outboard fuel transmitters. (Refer to 28-40-00).

c. Remove the fuel boost pump. (Refer to 28-20-00).

d. Remove the clamp and the fuel cell drain valve.

NOTE

On serials P-231 and after, a spacer is installed between the drain valve clamp and the fuel cell liner washer to prevent the drain valve from being pushed into the tank.

e. Remove the lower aft nacelle section and the fuel cell access plate (4).

f. Remove the cotter pin securing the flapper check valve assembly to the internal inboard interconnect. Remove the interconnect clamp.

g. Remove the outboard internal interconnect clamp through the fuel access hole (2).

h. Remove the fuel access plate (5).

i. Disconnect all fuel and vent plumbing.

j. Remove screws and bolts attaching the fuel cell outlet plate to the fuel cell.

k. Unsnap the fuel cell and remove the cell through the fuel access hole (2).

NOTE

Tape edge of fuel cell liner and access hole to prevent damage to the fuel cell.

l. Check the flapper check valve collar and valve hinge strap for parallelism within .03 inch. Check the valve for an opening of 35 ± 10 degrees.

NOTE

If the fuel cell is to be stored for a period of 10 days or longer, coat the inside of the cell with light engine oil to prevent cracking or deterioration.

INBOARD LEADING EDGE FUEL CELL INSTALLATION

CAUTION

Exercise caution when installing baffled fuel cells to prevent damage to the flapper valve.

a. Carefully insert the fuel cell through the fuel cell access hole (2) and snap the cell in place.

b. Install the fuel cell outlet plate with screws and bolts. Torque to 20 to 30 inch-pounds. Safety wire the bolts.

c. Connect all fuel and vent plumbing. Torque the rubber fuel fitting nipples to 25 ± 5 inch-pounds.

d. Install the outboard internal interconnect clamp through fuel access hole (2).

NOTE

Torque interconnect clamps to 25 ± 5 inch pounds.

e. Install the inboard internal interconnect clamp.

f. Install the flapper check valve with a new cotter pin. Check the valve for freedom of movement and for proper seating.

NOTE

The flapper check valve must hinge downward.

g. Install the fuel cell access plate (4). Torque to 45 to 55 inch-pounds and safety wire.

h. Install the lower aft nacelle section.

i. Install the fuel cell drain valve and clamp.

j. Install the fuel boost pump. (Refer to 28-20-00).

k. Install the inboard and outboard fuel transmitters. (Refer to 28-40-00).

INSTALLATION OF VELCRO TAPE WITH REPLACEMENT FUEL CELLS (P-4 through P-195)

(Figure 203)

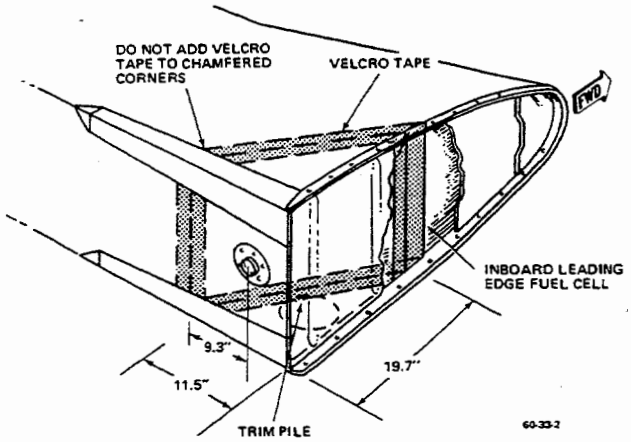
Aircraft that have or will have the existing inboard leading edge fuel cells replaced with spare fuel cells P/N 58-380030-7, 58-380030-8, 58-380030-9, and 58-380030-10 should install Kit 60-9002-3 S in the fuel cell liner top, bottom, root rib, and spar as described below and in Service Instructions No. 0585-281. If only one fuel cell is to be replaced, use Kit 60-9002-1 S for left hand installation and Kit 60-9002-2 S for right hand installation. Serials P-196 and after are delivered from the factory with the equivalent of Kit 60-9002-3 S installed.

a. Remove the fuel cell as described in INBOARD LEADING EDGE FUEL CELL REMOVAL.

NOTE

Aircraft that have had any of the above fuel cells installed do not require complete removal of the fuel cell. Access covers and inboard fittings should be removed and the inboard end

of the fuel cell pulled back far enough to allow installation of the velcro tape.



Installation of Velcro Tape
Figure 203

b. Lightly sand the surface that the velcro tape will be bonded to as shown in Figure 203 and thoroughly clean the sanded surface with toluol (22, Chart 207, 91-00-00).

c. Apply a coat of cement (39, Chart 207, 91-00-00) to the cleaned surface.

d. Activate the velcro tape by dipping into a bath of MIL-M-13999 methyl ethyl ketone (21, Chart 207, 91-00-00) and press the backing of the velcro tape to the cemented surfaces. Allow 15 minutes (minimum) for drying to ensure a good bond.

e. Position the fuel cell in place and press the velcro hook and pile together by pressing outward in the area of the velcro tape.

f. Inspect the flapper valve for freedom of movement under its own weight. If the flapper valve binds, refer to INBOARD LEADING EDGE FUEL CELL FLAPPER VALVE INSPECTION.

NOTE

Before closing the zipper, inspect the cavity inboard of the baffle for foreign material.

g. Close the zipper and refer to INBOARD LEADING EDGE FUEL CELL INSTALLATION for further instructions on installing the fuel cell.

NOTE

Installation of Kit 60-9002-3 S is required on a first time basis only. Repeat installations of the kit are unnecessary when new fuel cells are installed.

h. Make the appropriate log book entry.

i. After installation of baffled fuel cells, change the usable fuel placards on the filler caps and fuel selector valve as instructed in Service Instructions No. 0559-281.

INBOARD LEADING EDGE BAFFLED FUEL CELL - FLAPPER VALVE INSPECTION

On aircraft that are equipped with baffled fuel cells, the flapper valves should be inspected periodically (Beech Aircraft recommends that the inspection be accomplished at each annual inspection) for freedom of operation and proper seating. The inspection may be accomplished as follows:

a. Remove the fuel boost pump (refer to 28-20-00).

b. Open the zipper in the fuel cell baffle.

c. Locate the flapper valve in the lower aft portion of the baffle and determine whether the valve element is metal or phenolic.

d. If the flapper valve element is metal, it should be inspected and repaired, if necessary, as described below:

1. Move the flapper valve element through its full travel. There should be no binding and the element should seat flush against the valve plate.

2. If the flapper valve element binds and/or does not seat properly, the element arm could be bent. The arm can be straightened by placing a screwdriver between the arm and element and pressing the element toward the closed position.

3. If after straightening the arm, the element still binds and/or does not seat properly, the element should be removed and replaced with a new element. The element may be replaced by removing the two bolts from the upper part of the flapper valve assembly. The same attaching parts should be used to install the new element. The new flapper element should be inspected after installation, to determine that it was not damaged during installation which could cause it to bind and/or not seat properly.

e. If the flapper valve element is phenolic, it should be inspected and reworked, if necessary, as described below:

1. Move the flapper valve element through its full travel. There should be no binding and the element should seat flush against the valve plate.

2. If the flapper valve element binds and/or does not seat properly, the upper rear side of the element may be binding against the valve plate.

3. The flapper valve element may be relieved from binding by filing a small radius on the upper rear side of the element.

NOTE

A shop towel saturated with light oil may be placed directly below the flapper valve to absorb the phenolic dust during rework.

4. After determining that the flapper valve is functioning properly, thoroughly wipe the area in the vicinity of the flapper valve with an oil saturated shop towel.

f. Close the zipper in the main fuel cell baffle.
g. Clean the gasket contact areas on the fuel cell and the fuel boost pump.
h. Install the fuel boost pump (refer to 28-20-00).

BOX SECTION FUEL CELL REMOVAL
(Figure 204)

a. Drain and purge the fuel cell.
b. Remove the access plate (1).
c. Remove the inboard (2) and outboard (3) access plates on the underside of the wing.
d. Remove the fuel cell plates and remove the internal fuel cell interconnect clamps.

NOTE

On later Duke A60's, beaded interconnects are incorporated in the fuel system so that fuel cells aft of the main spar can be attached without opening the leading edge cavities.

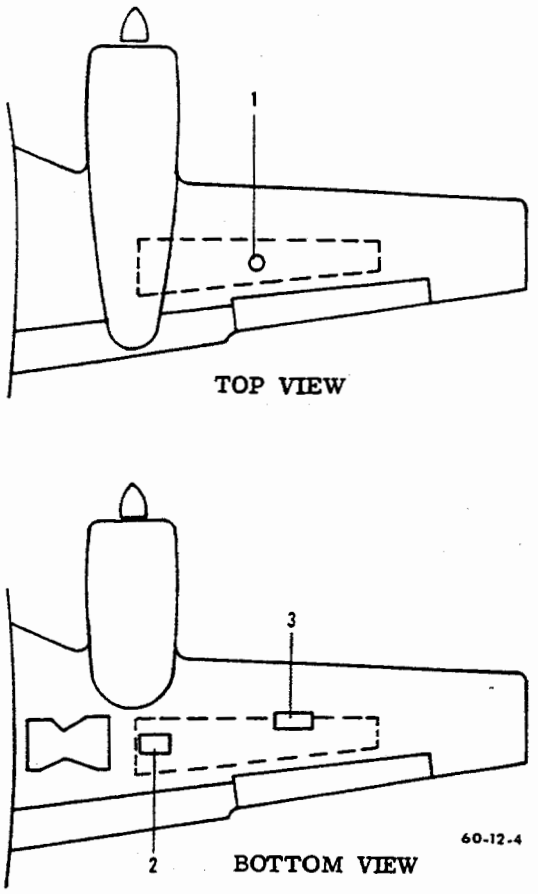
e. Disconnect the drain and vent plumbing.
f. Unsnap the fuel cell and remove it from the wing cavity through the outboard access hole (3).

NOTE

If the fuel cell is to be stored for a period of 10 days or longer, coat the inside of the cell with light engine oil to prevent cracking or deterioration.

BOX SECTION FUEL CELL INSTALLATION

a. Carefully insert the fuel cell into the wing cavity through the outboard access hole (3), and snap in place.
b. Connect all fuel and vent plumbing. Torque the rubber fuel fitting nipples to 25± 5 inch-pounds.
c. Install the internal fuel cell interconnect clamps. Torque clamps to 25 ± 5 inch-pounds.
d. Install the inboard (2) and outboard (3) access plates on the underside of the wing. Torque access plates to 45 to 55 inch-pounds.
e. Install access plate (1).



1. Access Plate
2. Box Section Cell Inboard Access
3. Box Section Cell Outboard Access

Box Section Fuel Cell Access Openings
Figure 204

OUTBOARD LEADING EDGE FUEL CELL REMOVAL
(Figure 205)

a. Drain and purge the fuel cell.
b. Remove the screws securing the filler neck (3) to the wing skin.
c. Remove the access plates (2) and the fuel and vent plumbing access plate (1) on the underside of the wing.
d. Disconnect the fuel and vent plumbing.
e. Remove the internal fuel cell interconnect clamps.

g. Install and secure access plates.

h. Pressurize the fuel system using 0.50 + 0.25 - 0.00 psig. There should be no pressure loss in 15 minutes.

i. Fuel and restore electrical power to the airplane as required.

LEAK TEST
(Figure 207)

a. Ensure that all repairs to the wet wing tip are completed and sealed.

b. Connect pressure source to the wet wing tip as shown in Figure 207.

CAUTION

Ensure that the filler cap and access plate are secure. This test should be done when the wet wing tip is removed from the fuel system of the airplane.

c. Apply 3.73 ± 0.25 psig to the wet wing tip for 5 minutes.

d. Apply leak detector compound MIL-L-25567 (14, Chart 207, 91-00-00) to the outside of the wet wing tip.

e. Mark all leaks as indicated by bubbles.

f. Depressurize the wet wing tip, locate and repair leaks (see Figure 206).

g. Repeat the pressurization test procedure and repair leaks until there is no leakage.

h. Rinse leak detector compound off wet wing tip with clean water and wipe dry with clean cloths.

i. Install wet wing tip on airplane.

LEAKAGE CHECKS AND REPAIRS (WET WING TIP)
(Figure 208)

To classify the degree of leakage in a wet wing tip fuel cell, measure the size of the wet area around the leak. A more accurate measurement may be obtained by wiping the leakage clean and applying talcum powder in the area of the leak. After 30 minutes, recheck the area to determine if the leak classifies as a stain, seep, heavy seep or running leak as indicated in Figure 208. Fuel leaks must also be classified as to whether they occur in an open area or in an enclosed area to differentiate between those that require immediate repair and those not considered potential flight hazards.

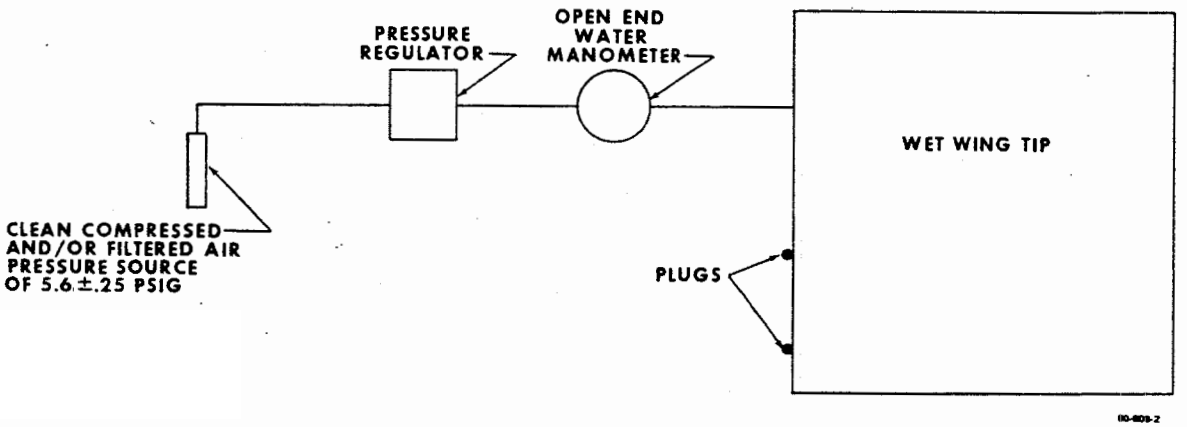
WARNING

Any leakage in an enclosed area, such as the wheel well, or in an area where the fuel will blow into the fuselage, requires grounding until repair is made.

a. Repair of heavy seeps or smaller leaks in an open area may be delayed until the airplane is down for other maintenance.

b. Any leakage in an enclosed area requires immediate grounding and repair.

c. Remove any sealant around the leak with a sharp, nonmetallic tool, such as a tool of chisel-shaped Formica.



Leak Test Set-up
Figure 207

Scarf the ends of the existing fillet so that a new sealant can form a continuous and smooth tie-in. PR-890B-1/2 (19, Chart 205, 91-00-00) or EC-1675B-1/2 (20, Chart 205, 91-00-00) is recommended for the sealing process. Thoroughly clean the area to be repaired with methyl ethyl ketone or naphtha prior to sealing. The following repairs are permissible:

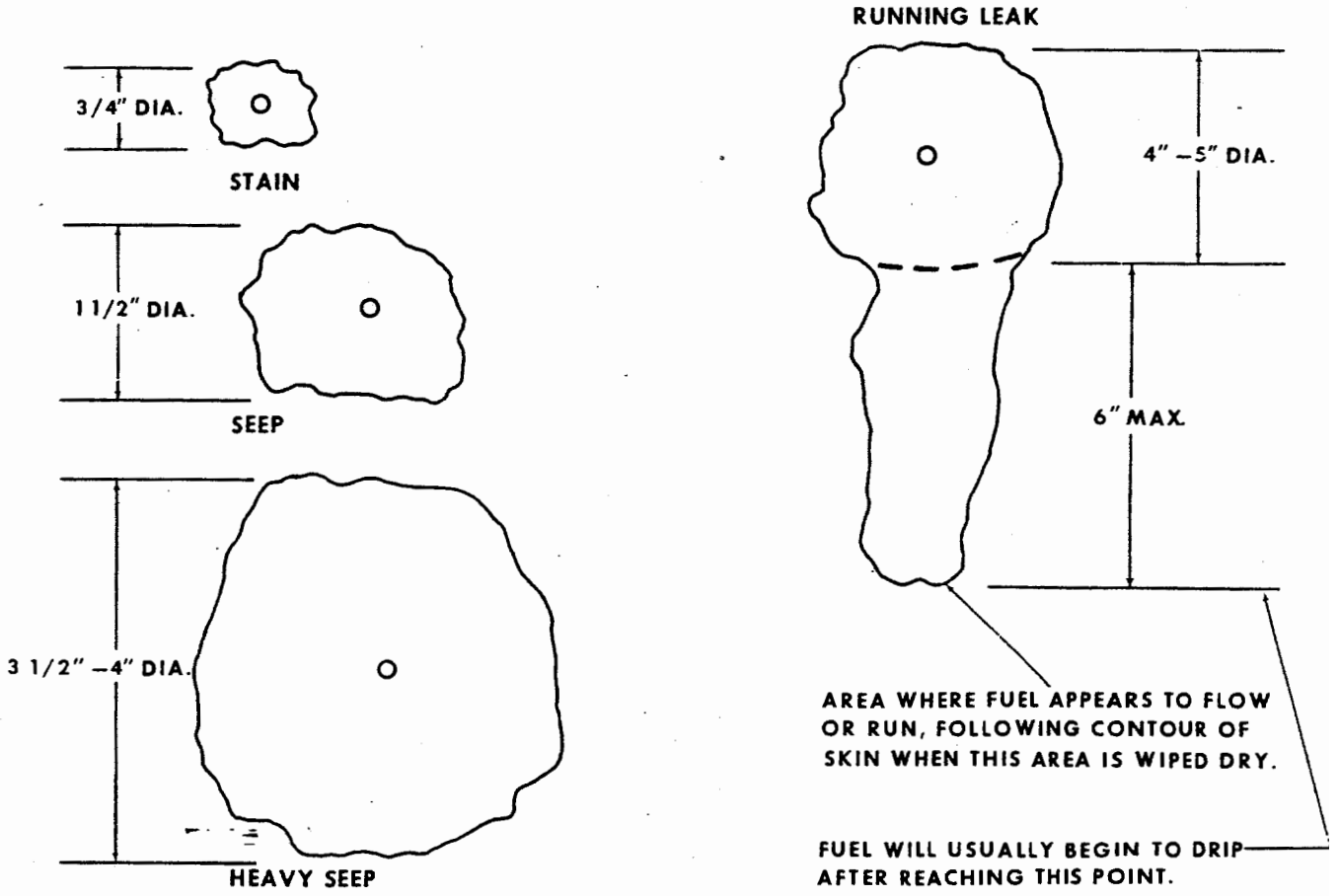
1. If the leakage is around a rivet, restrike the rivet. This can only be done once. If the leak persists, replace the rivet.
2. If the leakage is around a bolt with a gasket type seal, retorque the bolt. If the leak persists, replace the seal or the bolt.
3. If the leakage is at the gasket around an access opening or fitting, retorque the attaching hardware. If the leak persists, replace the gasket.

FUEL CELL LEAKAGE CHECK

Although the chemical test is more sensitive, either of the following test procedures may be used to detect leaks in the bladder cells.

a. Soapsuds Test.

1. Attach test plates to all fittings.
2. Inflate the cell with air to a pressure of 1/4 psi maximum.



Leakage of Wet Wing Tip Fuel Cell
Figure 208

3. Apply a soap and water solution to all repaired areas and any areas suspected of leakage. Bubbles will appear at any point where leakage occurs.

4. After test, remove all plates and wipe soap residue from the exterior of the cell.

b. Chemical Test

1. Attach test plates to all fitting openings except one.

2. Pour ammonia on the absorbent cloth in the ratio of 3 cc per cubic foot of cell capacity. Place the saturated cloth inside the cell and install the remaining test plate.

3. Make up a phenolphthalein solution as follows: add 40 grams phenolphthalein crystals to 1/2 gallon of ethyl alcohol, mix, then add 1/2 gallon of water.

4. Inflate the cell with air to a pressure of 1/4 psi maximum.

5. Soak a large white cloth in the phenolphthalein solution, then wring it out thoroughly and spread it smoothly on the outer surface of the cell. Press the cloth down to insure detection of minute leaks.

6. Check the cloth for red spots which will indicate a leak. Mark any leaks found and move the cloth to a new location. Repeat this procedure until the entire exterior surface of the cell has been covered. If red spots appear on the cloth, they may be removed by resoaking the cloth in the solution.

7. The solution and test cloth are satisfactory only as long as they remain clean. Indicator solution that is not in immediate use should be stored in a closed container to prevent evaporation and deterioration.

After the test, remove all plates and test equipment. Allow the cell to air out.

NOTE

In conducting the tests outlined above, the cell need not be confined by a cage or jig, providing the 1/4 psi pressure is not exceeded.

FUEL CELL REPAIR

GOODYEAR FUEL CELLS

The following items for field repairable injuries (inside or outside) are permissible. Damaged cells, not covered by these items should be returned to the Goodyear Tire And Rubber Company, Rockmart, Georgia, for repair.

a. Punctures.
b. Slits - to maximum 3 inch length.
c. Abraided holes.
d. Loose hangers and glove snaps (hot repair only).

Repair the fuel cell as follows:

a. Thoroughly clean the damaged area, (at least one square foot surrounding the injury) with methyl ethyl ketone solvent. Three washings are recommended to assure cleanliness.

b. Cut a patch from repair material furnished in repair Kit No. 2F1-2-31853, large enough to extend beyond damaged area by 2 inches in all directions. The patch should be thinned toward the edges.

c. Place the dull or gum stock side of the patch next to the cell. Wash the patch thoroughly in methyl ethyl ketone.

NOTE

No patch is required for loose hanger or glove snap repair, but heat must be used for curing.

d. Mix the cement in the following proportions and sequence. Use one quart can containing 272 grams of 2342C, heat if necessary to liquify. To this, add one can of 2233C (185cc) and stir until smooth (a minimum of 5 minutes is required).

e. Apply two evenly brushed coats of cement to the cell and patch surface. Allow 30 minutes drying time between the second coat and the application of patch (do not use cement after it has been mixed more than two hours).

f. Center repair patch over repair area and roll down firmly using the 1 inch stitcher. Start rolling from center of patch to the outside edge. This will remove any trapped air. Hanger or glove snap repairs use the same cementing and drying time as regular repair but will require heat for curing.

g. Place cellophane over repair and on inside of cell under repair. It is very important that the cellophane on the inside of the cell be placed under the repair area, thus preventing the two inside surfaces of the cell from being cemented together. Over the outside cellophane place the 1/4 inch cloth-backed foam rubber, cloth side up. Over the foam rubber place a 1/4 x 6 x 6 aluminum plate and place a "C" clamp (8 inch min. clamp) over the metal plate and underneath the work bench top. With patch and plate centered over repair area, tighten the "C" clamp until cement is forced out under edges of the repair (let cure for 72 hours).

NOTE

Air cured repairs are to be made at room temperature of approximately 75°F. Add 25% to the cure time for each 10° drop in temperature.

If heat is used for curing, insert element from No. 2F1-3-25721 cure iron (contained in Kit 2F1-2-31234) between "C" clamp and metal plate, then tighten "C" clamp until cement is forced out from edges of the patch. Cure for 2 hours at 240°F. Allow iron to cool for 15 to 20 minutes before removing. Remove iron, metal plate, foam rubber and cellophane. Dampen cellophane with water, using a sponge, and remove by peeling off. Loose edges of patch up to 1/4 inch maximum may be trimmed off and buffed smooth. Protect area of tank around buffed area with masking tape. Loose edges exceeding 1/4 inch may be recemented using the same cure procedure as previously used.

Storage and Handling: Prior to storage, clean the cell with warm water and soap. Dry and wrap in as small a package as possible and place in a cardboard box. Store in a cool dry room away from any electric motor that might be in operation.

Materials and Equipment needed for repair:

a. Air Cure: (Kit 2F1-2-31853)

QUANTITY	*NOMENCLATURE*	*GOODYEAR PART NO.*
2Qts.	Cement	2342C
2-1/2 Pts.	Mixture	2333C
2 Bottles	Mixture	2315C
2 (1 Pts.)	Methyl Ethyl Ketone	
1 Sheet 12" x 12"	Patch	BTC39
1 Sheet 12" x 24"	Cellophane	
1 Sheet 1/4 x 12" x 12"	Foam Rubber Cloth Back	

b. Heat Cure: (Kit No. 2F1-2-31234). Kit consists of all the above items listed in the air cure repair kit, plus the following items:

QUANTITY	*NOMENCLATURE*	*GOODYEAR PART NO.*
1	1" Paint Brush	
1	1" Stitcher	
1	Cure Iron (240°F)	2F1-3-25721
2	1/4" x 6" x 6" Aluminum Plate	

UNIROYAL FUEL CELLS

For repairs of Uniroyal fuel cells, refer to Uniroyal Handbook, Recommended Handling and Storage Procedures for Bladder Type Fuel and Oil Cells P/N FC 1473-73.

"END"

DISTRIBUTION - MAINTENANCE PRACTICES

FUEL BOOST PUMP REMOVAL

a. Drain and purge the fuel system for the appropriate wing.
b. Make sure the electrical power to the boost pump is off.
c. Remove the pump access cover on the underside of the wing.
d. Disconnect the electrical leads to the pump.
e. Cut the safety wire from around the retaining bolts and remove the bolts.
f. Pull the pump down far enough to gain access to the pump outlet line.
g. Disconnect the outlet line and remove the pump.

FUEL BOOST PUMP INSTALLATION

a. Clean the gasket contact areas on the fuel cell and the fuel pump.
b. Connect the fuel outlet line to the fuel boost pump.
c. Install the boost pump with new gaskets. Torque the bolts to 45 to 55 inch-pounds and safety wire.
d. Connect electrical leads to the pump.
e. Install the pump access plate on the underside of the wing.

ENGINE DRIVEN FUEL PUMP REMOVAL

a. Access to the engine-driven fuel pump is gained through the right cowl door on each nacelle.
b. The fuel pump is located at the rear, on the lower right side of the engine.
c. Disconnect the fuel inlet, outlet and drain plumbing from the pump. Remove the fuel pump heat shield.

NOTE

The 60-910031-3 elbow in the inlet port of the fuel pump is not a reusable part. Anytime the elbow is loosened or removed, it should be replaced with a new part.

d. Remove the pump retaining nuts and remove the pump.

ENGINE DRIVEN FUEL PUMP INSTALLATION

a. Install the fuel pump, on the lower rear right side of the engine, with a new gasket.
b. Connect fuel inlet, outlet and drain plumbing on the pump.
c. Install the fuel pump heat shield.
d. Close the right cowl door.

ENGINE DRIVEN FUEL PUMP ADJUSTMENT

a. The fuel pump is located at the rear, on the lower right side of the engine. Access is the right cowl door on each nacelle.
b. Install a fuel pressure gage (0-30 psi range) and a "T" fitting on the pressure side of the pump.
c. Break the safety wire and loosen the lock nut of the adjusting screw.
d. Disconnect and plug the air reference line from the engine pump.

NOTE

Allow the air reference line fitting on the fuel pump to remain open while adjusting the pressure on the pump.

e. Operate the engine at 2900 rpm and set the engine pump to 23 psi with the boost pump off.
f. Reinstall the air reference line to the pump.
g. Tighten the lock nut and safety wire.

FUEL SELECTOR VALVE REMOVAL

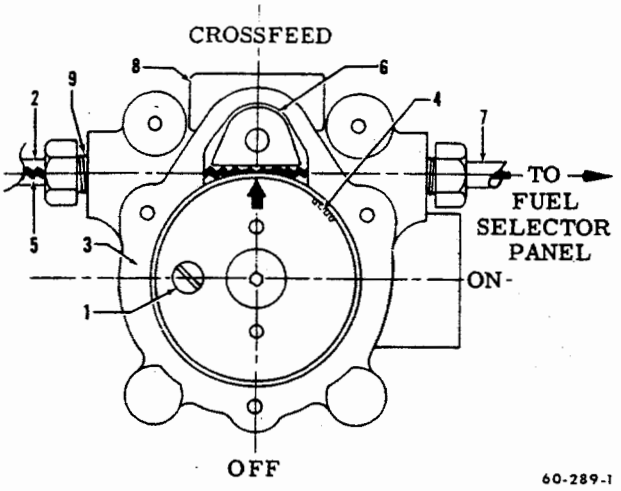
a. Drain the fuel system.
b. Place the aircraft on jacks and partially retract the gear until the inboard main gear door is fully extended.
c. Remove the selector control cable. (Refer to FUEL SELECTOR VALVE CONTROL CABLE REMOVAL).
d. Remove the fuel selector valve plumbing.
e. Remove the bolts securing the selector valve to the mounting bracket.

FUEL SELECTOR VALVE INSTALLATION

a. Position selector valve on the mounting bracket and install attaching bolts.
b. Lubricate threaded fittings with VV-P-236 petrolatum (Chart 208, 91-00-00).
c. Connect the fuel selector plumbing.
d. Install the selector control cable. (Refer to FUEL SELECTOR VALVE CONTROL CABLE INSTALLATION.)
e. Rig the selector control cable. (Refer to FUEL SELECTOR VALVE CONTROL CABLE RIGGING.)

FUEL SELECTOR VALVE CONTROL CABLE REMOVAL
(Figure 201)

a. Remove the cover plate (not shown), stop screw (1) and overtravel tube (2) from the valve gearbox (3) located in the wheel well.



Fuel Selector Valve
(LH Shown, Cover Plate Removed)

1. Stop Screw
2. Over Travel Tube
3. Selector Valve Gearbox
4. Selector Gear
5. Control Cable
6. Slider
7. Cable Housing
8. Cross Feed Port
9. Over Travel Port

Fuel Selector Valve
Figure 201

b. The cable may be removed through the overtravel port by rotating the selector gear.

FUEL SELECTOR VALVE CONTROL CABLE INSTALLATION

Refer to FUEL SELECTOR VALVE CONTROL CABLE RIGGING for installation procedures.

FUEL SELECTOR VALVE CONTROL CABLE RIGGING

(Figure 201 and 202)

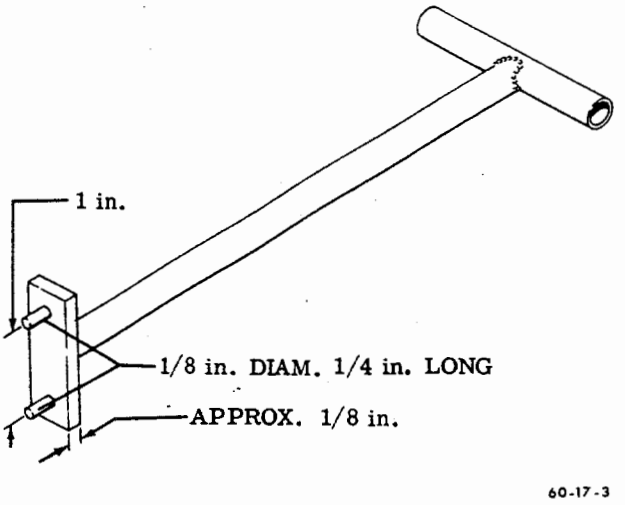
To aid in the rigging procedure, a locally manufactured "rigging tool" (see Figure 202) may be constructed for turning the selector gear. Tubing of 1/4 to 3/8 inch diameter is used for the handle and 1/8 inch steel pins are used for the protrusions which contact the selector gear.

a. Remove the cover plate (not shown), stop screw (1) and overtravel tube (2) from the fuel selector valve gearbox (3) located in the wheel well. (See Figure 201).

b. Using the rigging tool, set the selector gear (4) in the CROSSFEED position. The arrow on the selector gear should be positioned at 12 o'clock.

c. Set the selector handle pointer on the fuel selector panel 180° from the ON position and hold firmly in this position.

d. Insert the control cable (5) through the overtravel



Rigging Tool
Figure 202

port and rotate the cable (LH thread) to engage two threads in the selector valve gear.

e. Rotate the selector gear 6-3/4 revolutions to feed the control cable through its housing up to the fuel selector panel gearbox in the pilot's compartment.

f. Screw the cable in (LH thread) until the cable end is 4.3 inches minimum to 4.5 inches maximum from the face of the overtravel port. It will be necessary to have someone hold the selector handle pointer in position until the control cable is engaged with the gears in the fuel selector panel.

g. Move the fuel selector handle pointer to the CROSSFEED position. The arrow on the selector gear should now be at the 12 o'clock position. The cable end should measure 2.7 inches minimum to 3.3 inches maximum from the overtravel port.

h. Install the stop screw and install and safety the overtravel tube. Place the selector handle in all positions to insure proper selection and operation.

i. Install and safety the cover plate. No lubricant is used on the fuel selector valve.

VENT FLOAT VALVE REMOVAL

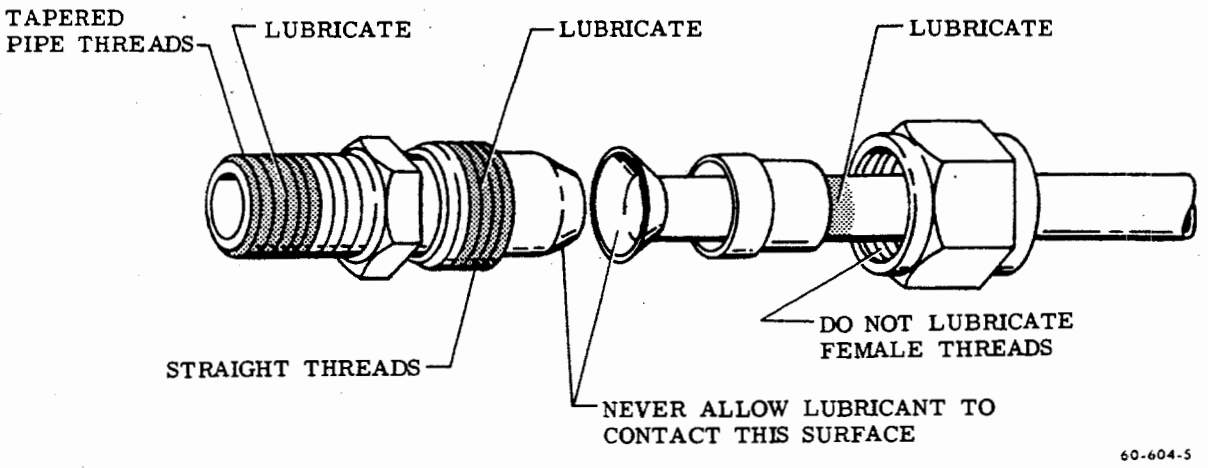
a. Remove the two access plates on the lower side of the wing tip.

b. Loosen the clamps and disconnect the three vent lines from the float valve.

c. Loosen the clamp securing the float valve in position.

NOTE

Mark the position of the float valve in the clamp. The float valve must be reinstalled in the same position to enable the float to function properly.



Lubrication of Flared Fittings
Figure 203

d. Remove the float valve from the clamp.

VENT FLOAT VALVE INSTALLATION

a. Position the float valve in the clamp in the same position as noted during removal. Torque the clamp to 25± 5 inch- pounds.

b. Connect the three vent lines to the float valve and torque the hose clamps to 25 ± 5 inch-pounds.

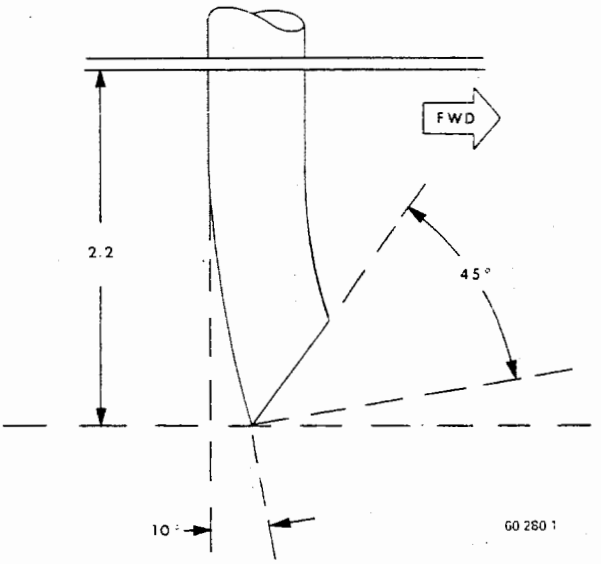
c. Install the two access plates on the lower side of the wing tip.

FLARED FITTINGS
(Figue 203)

When installing flared fittings and hoses, make sure the threads are properly lubricated with VV-P-236 petrolatum (Chart 208, 91-00-00). When previously installed fittings are removed, they should be wiped clean and relubricated before they are reinstalled. Torque all fittings in accordance with (Chart 204, 91-00-00).

EXTERNAL FUEL CELL VENT LINE
(Figure 204)

The end of the fuel vent lines should extend 10 degrees forward from vertical for a distance of 2.2 inches below the lower surface of the wing. The end of the line is scarfed at a 45 degree angle facing forward to ensure a positive vent pressure, for any other configuration would create a negative pressure that would pull the air, or air and fuel from the fuel cell.



Fuel Vent Line
Figure 204

"END"

INDICATING - MAINTENANCE PRACTICES

FUEL QUANTITY INDICATORS

Fuel quantity is measured by float type transmitter units which transmit the common level indication to a single indicator for each respective wing. Two transmitters are located in each inboard leading edge wing cell, one outboard and one inboard. One transmitter is located in each nacelle fuel cell.

FUEL TRANSMITTER REMOVAL

a. Remove fuel cell access plate. (See Figure 201 or 202, 28-10-00.)
b. Disconnect electrical wire at the transmitter.
c. Remove the bolts attaching the fuel transmitter to the fuel cell.
d. Cover the open fuel cell port to prevent entry of foreign materials.

FUEL TRANSMITTER INSTALLATION

a. Remove the cover from the fuel cell port. Clean the surfaces of the fuel cell and transmitter.
b. Install the fuel transmitter with a new gasket. Torque the attaching bolts to 25 ± 5 inch-pounds and safety wire.
c. Connect the electrical wire to the transmitter.
d. Install the fuel cell access plate.

NOTE

On airplanes serials prior to P-186, when AC transmitters are removed for replacement, install Rochester transmitters. When this is done, the remaining transmitters in the applicable wing and the printed circuit board must be replaced.

ADJUSTING AND TESTING FUEL GAGING SYSTEM

An external power unit regulated at 28.25 VDC should be connected to the airplane when checking and adjusting the fuel gages for proper indication in the full and empty positions.

a. Remove the access plates covering the fuel quantity transmitters.
b. To check the fuel gage for full readings it is necessary to substitute a resistor for the three transmitters on each side. On airplanes equipped with AC transmitters, a 90 ohm resistor is required; on airplanes equipped with Rochester transmitters, a 270 ohm resistor is required. The resistor may be connected as follows:

1. Connect the resistor to wire E13A20 (at inboard leading edge transmitter) and to E11B20 (at nacelle transmitter) for the right wing system.

2. Connect the resistor to wire E17A20 (at inboard leading edge transmitter) and to E15B20 (at nacelle transmitter case) for the left wing system.

Actual fuel level for this test is irrevalent.

NOTE

A potentiometer, located on the printed circuit board behind each fuel quantity gage, may be adjusted to obtain a full reading on the gage.

c. The empty reading on the fuel gage may be checked by jumping a 20 gage wire between the inboard leading edge transmitter and the nacelle transmitter. Connect wire E13A20 to E11B20 (right wing) and E17A20 to E15B20 (left wing). Fuel level for this test is irrelevant.
d. To check the transmitters for the proper resistance in the empty position, it is necessary to defuel the airplane. If the fuel gages do not read empty, measure the resistance of the transmitters in series.

1. Connect the lead of an ohmeter to the number 2 terminal of the inboard leading edge transmitter and the other lead to the number one terminal on the nacelle transmitter (right wing).
2. Connect the lead of an ohmeter to the number 2 terminal of the inboard leading edge transmitter and the other lead to the case terminal of the nacelle transmitter (left wing).

The total resistance for each side should be 0 to 0.5 ohms. If the total resistance is excessive, check each transmitter and/or associated wiring for the cause of high resistance.

e. To check the transmitters for the proper resistance in the full position, visually check to determine all tanks are full. If the fuel gage does not give a full indication, check the resistance total for each wing as in the preceding step.
f. With AC transmitters installed, the total resistance should be 90 ohms per wing; individual transmitters should register 30 ± 2 ohms. With Rochester transmitters installed, the total resistance should be 270 ohms per wing; individual transmitters should register 90 ± 2 ohms.
g. Replace the access covers.

FUEL FLOW INDICATOR

For fuel flow indicator refer to Chapter 73.

FUEL SIGHT GAGE (P-402 and after)

For conveniance during fueling operations there is a fuel sight gage installed in each wing. The gage is installed in the upper leading edge just outboard of each nacelle. The gage is of the mechanical float type and the indicated fuel quantity is in U.S. gallons.

FUEL SIGHT GAGE REMOVAL (P-402 and after)

a. Remove the screws in the access plate and remove the plate.

b. Remove the screws around the sight gage and remove the gage.

FUEL SIGHT GAGE INSTALLATION (P-402 and after)

a. Position the gasket and gage in place (the gage will fit only one way).

b. Install the attachment screws and torque to 25 inch-pounds.

c. Safety wire the attachment screws.

d. Install the access cover plate and secure with screws.

"END"

CHAPTER 30

LIST OF PAGE EFFECTIVITY

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"END"

CHAPTER 30 - ICE AND RAIN PROTECTION

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CHAPTER 30 - ICE AND RAIN PROTECTION

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"END"

GENERAL - DESCRIPTION AND OPERATION

The Duke 60 series aircraft utilizes heated pitot, heated stall warning, and heated fuel vents for standard ice protection equipment. Optional icing equipment includes: pneumatically operated surface deice boots and electrically heated propellers, heated windshield and heated ventilation ram air inlet scoop. In addition, an alternate static air source backs up the fuselage mounted static air source buttons.

"END"

AIRFOIL - DESCRIPTION AND OPERATION (Figure 1)

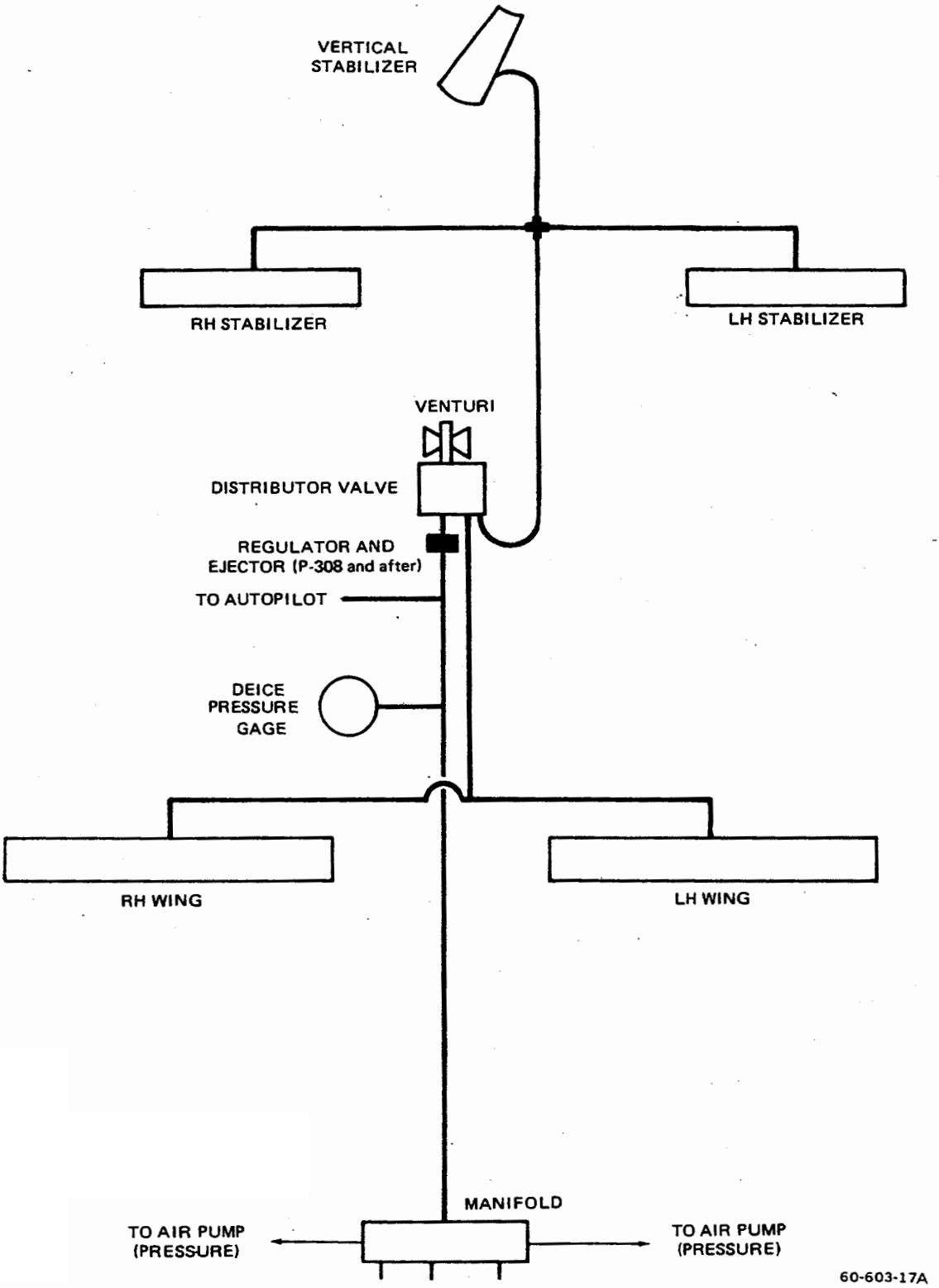
Deice boots on the wing and empennage leading edges are inflated by the two engine-driven pressure pumps. A venturi, operated from the pressure pumps, supplies vacuum for boot hold down at all times except during the inflation mode. Through an electric timer, solenoid-operated control valves cause all the boots to be inflated simultaneously. The timer is controlled by a three-position switch: SURFACE ONE CYCLE, and MANUAL with OFF position centered. This switch is located on the left subpanel. ONE CYCLE and MANUAL switch positions are momentary. A gage is provided to indicate system pressure. Momentary engagement of the ONE CYCLE position will cause the boots to inflate for five to eight seconds, then deflate to the vacuum hold-down condition. The MANUAL position will inflate the boots only as long as the switch is held in engagement; when the switch is released, the boots deflate. Leave the deicing system off until 1/2 to 1 inch of ice is accumulated. During inflation, the deice system pressure gage should register approximately 15 to 18 psi. Sufficient pressure for proper operation of the system is available with one engine inoperative.

On airplane serials P-3 through P-307, when the surface deice system is operated with the cabin pressure switch in the DUMP position, cabin pressure oscillations will occur. This is caused by a momentary loss of vacuum to the outflow valve while the boots are pressurizing. This vacuum loss allows the outflow valve to close and create a small residual cabin pressure. After a small increase, this pressure is then dumped by the safety valve.

The cabin pressurization shut off controls should be pulled during this mode to divert cabin pressurizing air overboard and prevent excessive cabin pressure oscillations. Cabin ventilation may be obtained by pulling out the cabin air control. In this mode pressure oscillations will be small.

On airplane serials P-308 and after, the vacuum used by the cabin pressurization system is developed in an ejector installed in the supply line for the deice distribution valve. Thus, operation of the deice system does not cause fluctuations of cabin pressurization.

For night operation, a wing ice light is provided on the outboard side of the left nacelle. The switch, placarded WING ICE, is on the left subpanel.



Airfoil Deice System Schematic
Figure 1

"END"

TROUBLESHOOTING
AIRFOIL DEICER SYSTEM

The following troubleshooting procedures are based on the assumption that the engine-driven dry air pumps are operational.

	TROUBLE	*PROBABLE CAUSE*	*REMARKS*
1.	Deicer boots do not inflate (either or both engines operating at a minimum cruise RPM for a period of eight seconds).	a. Open circuit breaker.	a. Push deicer circuit breaker to reset.
		b. Loose electrical connection or broken wire.	b. Tighten or repair as required.
		c. Time delay relay not functioning.	c. Replace time delay relay.
		d. Control relay not functioning	d. Replace control relay.
		e. Deicer boot punctured.	e. Repair as described in this chapter or replace.
		f. Distributor valve not functioning.	f. See steps 4 and 5.
		g. Piping lines kinked, blocked, or not connected.	g. Inspect and repair or replace as required.
		h. Leak in system.	h. Locate and repair.
2.	Deicer boots inflate too slowly (either or both engine operating at minimum cruise RPM for a period of eight seconds)	a. Piping lines kinked, partially blocked, or not securely connected.	a. Inspect and repair or replace as required.
		b. Leak in system.	b. Locate and repair.
		c. Deicer boot punctured.	c. Repair as described in this chapter or replace.
		d. Distributor valve not functioning.	d. See steps 4 and 5.
3.	Deicer boots deflate too slowly.	a. Piping lines kinked, partially blocked, or not securely connected.	a. Inspect and repair or replace as required.
		b. Overboard line from distributor valve partially blocked.	b. Inspect and repair or replace as required.
		c. Distributor valve not operating properly.	c. Overhaul or replace.
		d. Electrical circuit malfunctioning.	d. See Wiring Diagram Manual, P/N 60-590001-29.
		e. Vacuum ejector on distributor valve plugged.	e. Remove obstruction or replace.

TROUBLESHOOTING
AIRFOIL DEICER SYSTEM (Cont'd)

TROUBLE	*PROBABLE CAUSE*	*REMARKS*
	NOTE	
	The following items might aid in ascertaining whether or not the distributor valve is functioning properly.	
4. One or more boots do not inflate -- with pressure gage at normal reading with switch held in MANUAL or momentarily placed in SINGLE position.	a. Defective wiring in external circuit or other units.	a. See Wiring Diagram Manual P/N 60-590001-29 and disconnect plug at distributor valve. Voltage should be approximately 28 VDC between A-C and B-C. Make sure C is well grounded. On airplane P-390 and after, voltage should cycle at approximately 28 VDC between the blue-white wires. Make sure the white wire is well grounded
	b. Faulty solenoids in distributor valve.	b. Measure resistance of solenoids. Reading should be 17.5 ohms + 5% through the receptacle pins A-C and B-C. On airplanes P-390 and after reading should be 127 ohms ± 5% between the blue-white wires. Replace the distributor valve if readings do not check.
	c. Mechanical failure in distributor valve.	c. Disconnect lines at the outlet ports of the distributor valve and check valve operation with a gage. If trouble is not found in the distributor valve, inspect boots and lines for leaks or blockage.
	d. Piping lines kinked, blocked, or not connected.	d. Inspect and repair or replace as required.
5. One or more boots inflate but do not deflate readily -- with pressure gage at normal reading.	a. Defective boots.	a. Repair as described in this chapter or replace.
	b. Obstruction of lines.	b. Disconnect line from exhaust port of distributor valve and see if line is clear to low pressure area.
	c. Mechanical failure in distributor valve.	c. With line disconnected, see if exhaust port is discharging; if not replace distributor valve.

"END"

AIRFOIL DEICER - MAINTENANCE PRACTICES

SERVICING

Since the deicer boots and related components operate on clean air supplied from the pressure manifold, little is required in the form of servicing the system. The boots should be checked for engine oil after servicing and at the end of each flight, and any oil found should be removed. This can be accomplished by the use of a neutral soap and water solution. Care should be exercised to avoid scrubbing the surface of the boot as this will tend to remove the special conductive surfacing.

NOTE

Because the deicer boots are made of soft flexible stock, care must be exercised against dragging gasoline hoses over them or resting ladders or platforms against the surface of the boots.

SURFACE DEICER BOOT REMOVAL

To loosen or remove an installed deicer boot, use toluol (22, Chart 207, 91-00-00) to soften the "adhesion" line where the boot is joined to the metal surface. The solvent should be applied sparingly with a brush or trigger type oil can with a spout. Slowly peel the boot back, allowing the solvent time to undercut the boot. Exercise care not to injure the boot during removal.

SURFACE DEICER BOOT INSTALLATION

PREPARATION OF METAL SURFACES

Solvent Cleaning: The metal surface should be completely clean to prevent adhesion failure. Using a grease-free cloth dampened in MIL-M-13999 methyl ethyl ketone, (21, Chart 207, 91-00-00) go over the area to be covered by the boot. Change the cloths frequently, to avoid contaminating a previously cleaned area. Do not contaminate the clean supply of methyl ethyl ketone, by dipping a used cloth into it. Repeat the process. Using a clean, damp cloth and a clean dry cloth, go over the area again; use the dry cloth (following the damp cloth) to wipe the surface dry, rather than letting it air dry.

Chemical Cleaning: Follow the solvent cleaning, with a grease-free cloth wetted with an acid cleaner (41, Chart 207, 91-00-00). Vigorously scrub surface.

CAUTION

Although the acid cleaner is a mild acid solution, protective rubber gloves should be worn and contact with the skin should be avoided.

After the acid cleaner has had one minute's contact, wipe dry with a clean cloth. Allow a minimum of one hour dry-time before applying cement. At the end of the dry-time, wipe the surface with a clean cloth and inspect the cloth for dirt. If dirt is present, reclean with methyl ethyl ketone; if not, cover the clean surface with paper until the cementing operations are begun.

PREPARATION OF RUBBER SURFACES

If the deicer boot has a smooth back finish, roughen it slightly with sandpaper before beginning the cleaning operation. Wet a clean cloth with toluol (22, Chart 207, 91-00-00) and carefully clean the rough back surface of the boot. Change cloths frequently to avoid contamination of the cleaned areas. Clean the boot a minimum of two times; if the area still seems dirty, reclean the surface in the same manner.

Application of Adhesive: The drying of the cement is a function of time and temperature, and the table below should be used as a shop guide when applying the cement:

Temperature - °F	Minutes of Dry Time
Above 80	30
60-80	45
Below 60	60

Do not apply cement under dusty conditions or in high humidity (80% relative humidity or above). Prior to cementing, mask off the boot area on the metal surfaces, allowing 1/2 to 3/4 inch margin.

SPRAY COAT METHOD:

If the EC-1300L adhesive (12, Chart 205, 91-00-00) is applied by spray, the first coat on the back surface of the boot and on the metal surface should dry a minimum of 30 minutes. The second cross coat on each surface should be allowed to dry a minimum of 30 minutes, preferably one hour.

BRUSH COAT METHOD:

Apply an even brush coat of EC-1300L adhesive (12, Chart 205, 91-00-00) to the back surface of the boot and the metal surface of the airplane. Allow a minimum of 30 minutes to dry. Apply a second coat to each surface in a smooth, even layer. Brushing in one area too long tends to soften the first coat and "rolling" and "balling up" will result. Allow the coating to thoroughly dry a minimum of 30 minutes, preferably one hour before installation. Excess drying time (not to exceed 7 days) is not critical as long as the surfaces are not contaminated.

INSTALLATION OF THE BOOT

Using a chalk line, snap a line centrally located on the leading edge of the surface. Snap a line, centrally located cordwise, on the cemented side of the boot.

Securely attach hoses to the deicer connection, being careful to handle the boot section without getting finger marks on the adhesive. Using a lint-free cloth, heavily moistened (not dripping) with toluol, (22, Chart 207, 91-00-00) reactivate the surface of the leading edge and boot about 3 inches on either side of the chalk line. Position the boot chalk line directly on the leading edge chalk line and hand roll the boot surface onto the leading edge. Moving along the center line of the leading edge, continue reactivating the adhesive in strips 6 inches wide by 24 inches long. Avoid excessive rubbing of the adhesive surface as some of the adhesive may be removed. Hand roll the joined surfaces to ensure complete contact of the adhesive and elimination of air pockets. If the boot does not follow the chalk line on the leading edge, pull it up immediately with a quick motion and reposition properly. Now complete the installation by activating the adhesive surfaces and rolling on the top and lower half of the boot in sequence. Finally roll the entire boot (applying pressure) moving in a direction parallel with the inflatable tubes. Use a narrow stitch roller between tubes to eliminate air entrapment. If an air pocket or blister is noted immediately after boot installation, the air may be removed by inserting a hypodermic needle into the blister and allowing the air to escape. The surfaces may then be pressed down, permitting the surfaces to adhere.

NOTE

When removing entrapped air from the boot by use of a needle, be extremely careful not to puncture one of the inflatable tubes.

SEALING EDGES

Fair in all around cut edges and trailing edges of the boot with EC-801 sealer (11, Chart 205, 91-00-00) and cover all exposed adhesive. Never try to remove excess adhesive closer than 1/4 inch from the boot edge. After all adhesives and sealing compounds have dried and cured, remove masking tape and clean adjacent areas with solvent.

STALL STRIP INSTALLATION (RUBBER)
(Figure 201)

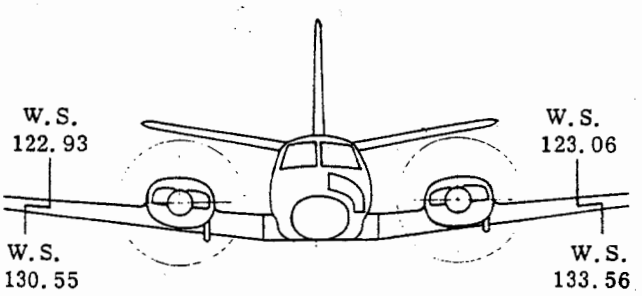
a. The stall strips are 7.62 inches long for the right wing and 10.50 inches long for the left. The right stall strip is installed with its inboard end at Wing Station 122.93 and the left stall strip is installed with its inboard end at Wing Station 123.06.

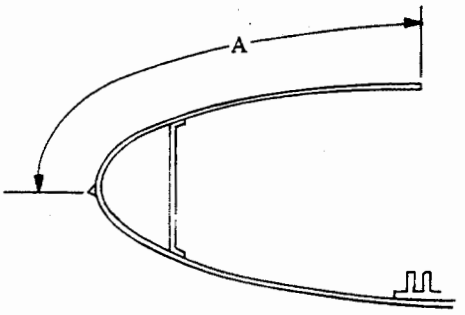
b. Clean boot surface thoroughly, removing all old cement. Mask off the area where the new strip is to be installed and wipe with MIL-M-13999 methyl ethyl ketone (21, Chart 207, 91-00-00).

c. Bostic 1008, a two part cement, (18, Chart 205, 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).

d. 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 shown in the illustration. The cement will set in about 6 hours.

e. When dry, coat the area with A56B cement (1, Chart 205, 91-00-00) to replace the conductivity of the boot.





60-34-1

Right Wing	Left Wing
Dimension "A" is 15.69 at Wing Station 122.93 and 15.06 at Wing Station 130.55.	Dimension "A" is 15.69 at Wing Station 123.06 and 15.00 at Wing Station 133.56.

Stall Strip Installation
Figure 201

RESURFACING DEICER BOOTS

Static electric charges, if allowed to accumulate, would eventually discharge through the boot to the metal skin beneath,

causing static interference with radio equipment and possibly puncturing the rubber. Also, such static changes are a temporary fire hazard after each flight. To dissipate static electric charges, a thin coating of conductive cement is applied over the neoprene of the boot. From time to time it may be necessary to restore the conductivity to efficiently dissipate such charges. When resurfacing seems advisable, the principal factors involved are:

a. If the surfacing material has abraded.
b. If the surfacing has developed cracks.
c. If the conductivity is low.

The following procedures should be accomplished when resurfacing deicer boots.

a. Clean the deicer boots thoroughly with toluol (22, Chart 207, 91-00-00).
b. Roughen the entire surface of the boot with fine sandpaper.
c. Clean the surface again with a clean lint-free cloth moistened with toluol.
d. Apply masking tape beyond the upper and lower trailing edges, leaving a 1/4 inch gap of bare metal.
e. Brush one coat of Goodrich A56B cement (1, Chart 205, 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 A56B cement has aged three months or more, it may be necessary to dilute it with toluol to obtain the proper brushing consistency. Mix thoroughly, approximately five parts cement to one part toluol.

ADJUSTMENT

Adjustment of the pneumatic pressure system is performed by adjusting the various regulators in a specified sequence. A PRESSURIZATION SYSTEM ADJUSTMENT CHART corresponding to applicable illustrations and a general adjustment procedure for each individual regulator are provided in Chapter 36-00-00.

COMPONENT LOCATION

TIME DELAY RELAY

The time delay relay is located forward of the instrument panel in the upper LH corner of the forward pressure bulkhead.

CONTROL RELAY

The control relay is located forward of the instrument panel in the upper LH corner of the forward pressure bulkhead.

DISTRIBUTOR VALVE

The distributor valve is located just forward of the access opening on the RH side of the aft fuselage. The distributor valve is accessible, for removal and installation, through the access openings on the lower aft fuselage and the RH side of the aft fuselage. The valve may be removed by disconnecting the hoses and removing the attaching screws.

COMPONENT REPLACEMENT

No maintenance on these components is recommended. Repair or replacement of parts should be made through the Beech Aircraft Corporation overhaul, and exchange program.

"END"

AIR INTAKES - DESCRIPTION AND OPERATION

The possibility of induction system icing is reduced by the non-icing characteristics of fuel injection engines and is backed up by an automatic alternate air source. Should the ram air scoop or filter become clogged with ice, a spring-loaded door on the firewall will open automatically, and the induction system will operate on alternate air. When operating on alternate air above the critical altitude, approximately 8 to 10 inches of manifold pressure will be lost.

On airplanes P-3 thru P-266, an optional equipment ram air inlet electrothermal lip boot is utilized. The boot is activated by a separate switch placarded RAM AIR INLET-OFF.

"END"

PITOT AND STATIC - DESCRIPTION AND OPERATION

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.

"END"

WINDOWS AND WINDSHIELD - DESCRIPTION AND OPERATION

The pilot's and copilot's windshields are electrically heated to protect against icing. An inverter, also used as a standby for the avionics inverter, is installed for the operation of the pilot's windshield heat and is activated by a switch on the pilot's subpanel marked L WSHLD - OFF. The copilot's windshield is activated by a switch on the pilot's subpanel marked R WSHLD - OFF - L WSHLD. Each switch is protected by a 3/4-ampere circuit breaker located on the right subpanel. The 45-ampere circuit breaker that protects the other components of the system is located between the two LH bus isolation circuit breakers in the LH nacelle electrical equipment compartment. For equipment requiring AC current, a three position switch marked MN INV - OFF - STBY INV must be placed in the MN INV position. Should a failure occur in the main inverter, the switch can be placed in the STBY INV position. This opens a relay to direct the current from the windshield heat inverter to the avionics provided the L WSHLD switch is on. Power for the operation of both systems cannot be supplied by this inverter at the same time.

On airplane serials P-459 and after and prior airplane serials with Kit No. 60-3008-1S installed, a windshield voltage indicator is provided on the instrument panel which enables the operator and/or maintenance to monitor the voltage from the inverter to the windshield heater.

In operation a sensing element installed in the windshield sends a signal to the temperature control box located on the aft side of the forward pressure bulkhead at the base of the pilot's windshield, closing a relay permitting current to flow through the heating elements. The control box is factory adjusted between 90°F and 110°F to maintain the desired mean temperature. The control box operates in an ambient temperature range of −65°F to 160°F.

"END"

TROUBLESHOOTING
WINDOWS AND WINDSHIELD DEICER SYSTEM

TROUBLE	*PROBABLE CAUSE*	*REMARKS*
1. Windshield fails to heat.	a. Circuit breaker on RH subpanel tripped.	a. If circuit breaker persists in tripping, check for short and correct.
	b. Switch faulty.	b. If no voltage at switch output with correct voltage at switch input, replace switch.
	c. No input or output voltage to inverter circuit breaker.	c. Check for short and correct.
	d. Sensing element faulty.	d. Check for circuit continuity and replace windshield.
	e. No AC output voltage from control relay.	e. Check control relay, sensing element and control box and replace as required.
	f. Damaged heater circuit.	f. Replace windshield.

"END"

WINDOWS AND WINDSHIELD - MAINTENANCE PRACTICES

ELECTRICALLY HEATED WINDSHIELD RESISTANCE CHECK

To check for incorrect resistance or the presence of a short or open circuit in the heating elements of the windshield, the following procedure may be used:

a. With the windshield deicing system turned OFF, disconnect the leads to the heating element at the lower end of each bus.

b. Using an ohmmeter, determine the resistance of the heating element by placing the leads of the ohmmeter across the heating element leads. The resistance should measure 97.0 ± 9.7 ohms.

c. Reconnect the leads of the heating element.

SENSING ELEMENT RESISTANCE CHECK (Figure 201)

The resistance of the sensing element of the windshield varies with temperature changes. Figure 201 provides the acceptable range of resistance values at various ambient temperatures. The resistance of the sensing element may be checked as follows:

a. With the windshield deicing system turned OFF, disconnect the wire connected to one of the terminals of the sensing element.

b. Use an ohmmeter to determine the resistance of the sensing element.

c. Determine the temperature of the windshield.

d. Determine if the resistance measured falls within the tolerance shown in Figure 201.

e. Reconnect the windshield sensing element wire.

ELECTRICALLY HEATED WINDSHIELD VOLTAGE CHECK

CAUTION

Ground use of windshield heat is limited to 10 minutes.

a. Connect a precision voltmeter between the windshield heater control switch and the windshield heater.

b. Start the engines in accordance with the applicable Pilot's Operating Manual.

c. Set propeller speed at 1200 to 1500 rpm.

d. Turn the LH windshield switch ON. Note the increase on the voltmeter (minimum reading of 220 vac). A voltmeter reading of less than 220 vac indicates a malfunction of windshield heat.

e. Repeat steps "a" through "d" with the RH windshield heat switch ON.

f. Shut-down airplane engines in accordance with the applicable Pilot's Operating Manual.

g. Remove voltmeter from the airplane and restore wiring to the windshield heater circuit.

ELECTRICALLY HEATED WINDSHIELD FUNCTIONAL TEST

After completing the preceeding resistance checks, determine that the ambient temperature is 90° or less and perform the following functional check:

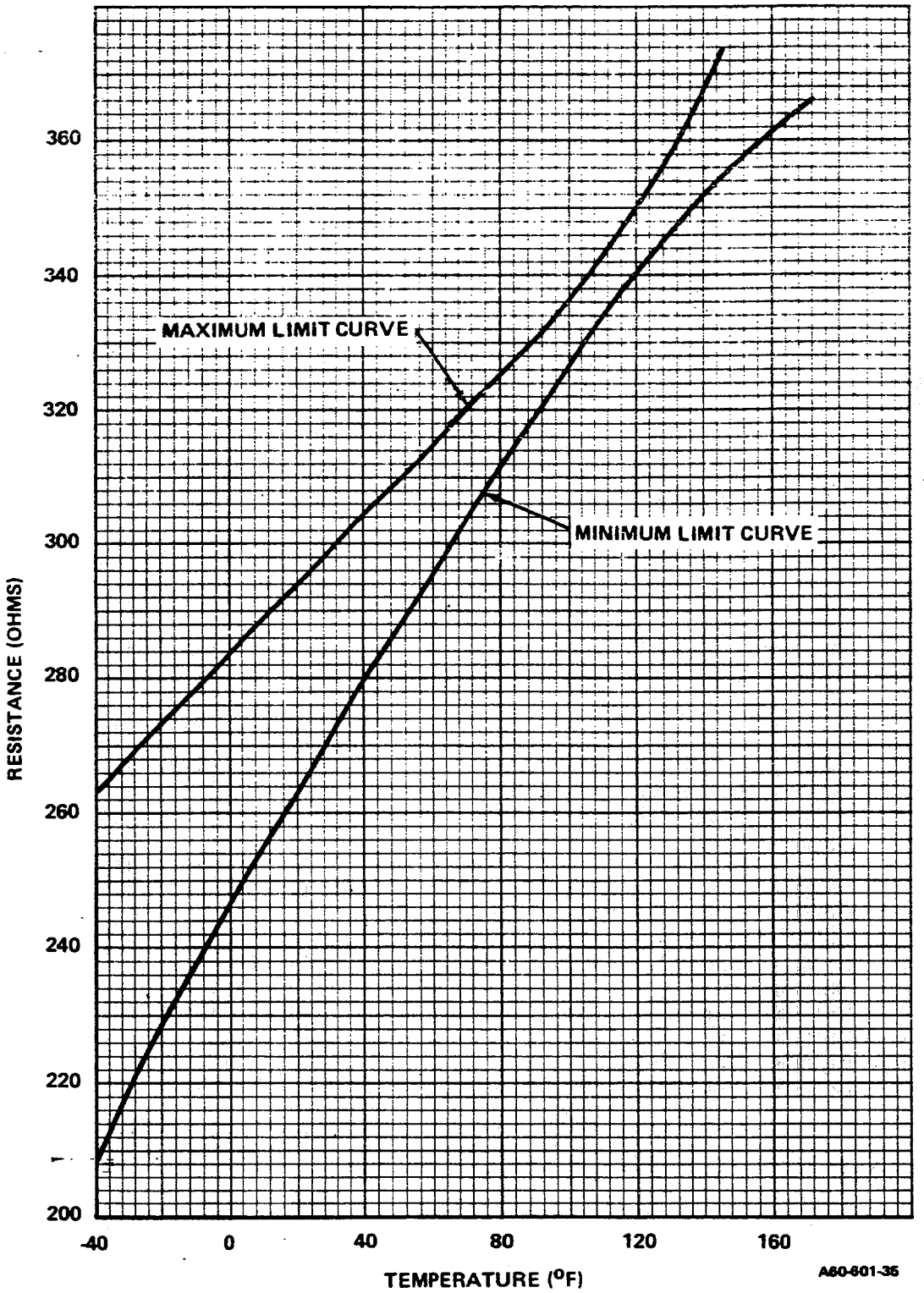
CAUTION

Ground use of windshield heat is limited to 10 minutes to prevent damage to the inverter.

a. Place the windshield DEICE control switch in the L WSHLD position.

b. Determine that the windshield should immediately begin to heat. Presence of heat may be determined by holding the hand against the heated portion of the windshield.

c. Place the windshield DEICE control switch in the OFF position.



Sensing Element Resistance Graph
Figure 201

"END"

PROPELLERS - DESCRIPTION AND OPERATION

ELECTRIC PROPELLER DEICING (Prior to P-579, FIGURE 1)

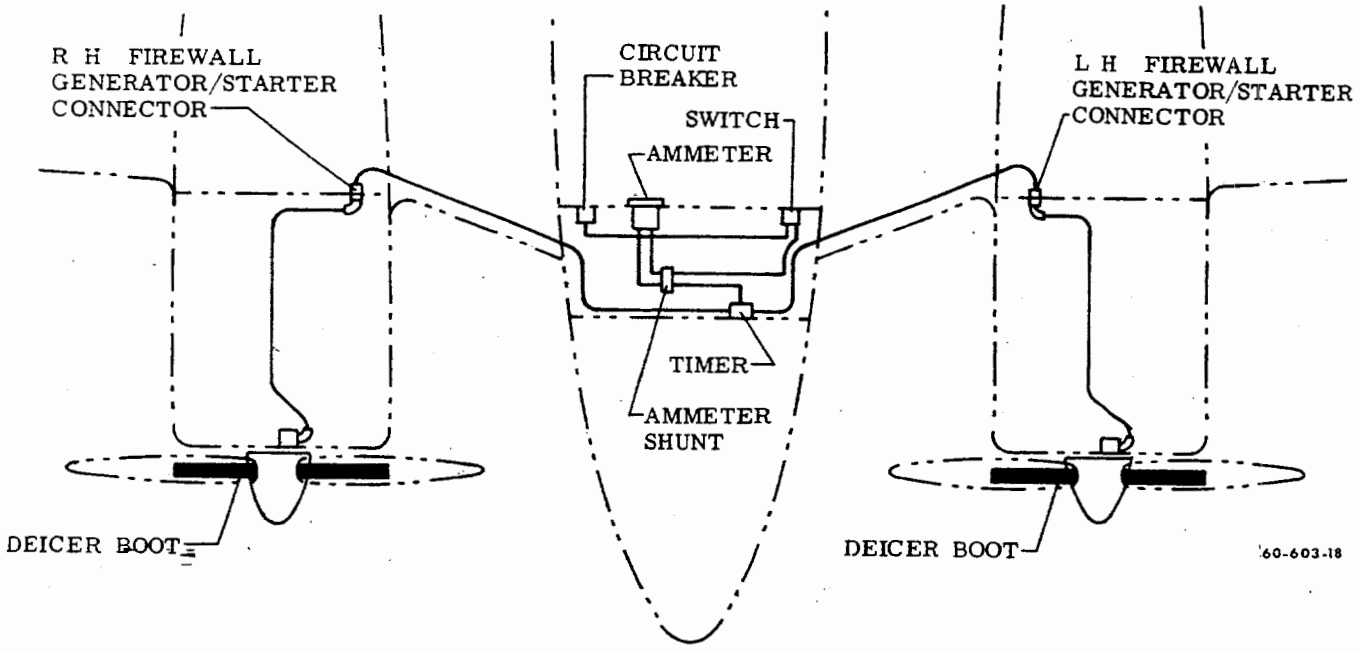
The electric propeller deicer system includes an electrically heated boot for each propeller blade, brush assemblies, slip rings, an ammeter, a control switch and a circuit breaker. When the switch is turned on, the ammeter registers the amount of current (14 to 18 amps) passing through the system. If a short develops in the system the circuit breaker will cut off the power to the timer. The current flows from the timer 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 starter ring gear. 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 rotation and the blast of the airstream. 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 will register a momentary deflection.

ELECTRIC PROPELLER DEICING (P-579 and after)

On airplanes P-579 and after, the electrothermal deice boots mounted on the propeller blades are electrically heated. Direct current for deice boot heating is supplied through a system of controls by two brushes which ride 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. Maintenance of the modular brush blocks and slip ring assemblies are covered elsewhere in this chapter under respective headings.

Current for operation of the deice timing control and the deice boots is supplied through a switch, located on the pilot's LH subpanel, and a 20-ampere circuit breaker. When the switch is placed in the ON position, the deice timer begins to run, initiating automatic cycling of electrical power to the deice boots. At intervals of approximately 90 seconds in duration, the timer alternately cycles power to the LH then RH propeller deice boots. Current for operation of the deice system is indicated by an ammeter located on the RH instrument panel.



Propeller Deicer System
Figure 1

"END"

PROPELLERS - TROUBLESHOOTING

The ammeter of the deicer system can be used to indicate the general nature of most electrical problems. Consequently, it is recommended that troubleshooting be preceded by the ammeter test as outlined under ELECTRIC PROPELLER DEICER in Chapter 5-20-00 of this manual, and the HEAT TEST described in this chapter to determine which circuits are involved. A reading of two-thirds the normal amount of current is an indication that one of the circuits is open between the slip ring and deicer heater. If the ammeter registers excess current, the power lead is shorted to ground. It may be possible that the excess current has welded the timer contacts in one phase. Under these circumstances, the timer will either feed current to the welded contacts continuously or will not cycle. If the former is true, the heat test will show two phases heating simultaneously throughout three of the four phases. Unless the grounded power lead is located and corrected, any new timer that is installed may suffer the same internal damage during the first use of the system.

TROUBLESHOOTING
PROPELLER DEICING SYSTEM (PRIOR TO P-579)

	TROUBLE		*PROBABLE CAUSE*		*REMARKS*
1.	Ammeter shows zero current. (All 4 phases of the 2-minute 16-second cycle.)	a.	Circuit breaker tripped.	a.	Locate and correct short before resetting circuit breaker.
		b.	Switch faulty.	b.	If no voltage at switch output with voltage at switch input, replace the switch. If voltage is OK at switch output, go to step "d".
		c.	No power from aircraft.	c.	If no voltage into switch, locate and correct open circuit.
		d.	Ammeter faulty. (If some or all deicers heat with ammeter at zero, replace the ammeter.)	d.	Test for voltage up to and out of ammeter. If low or zero output but proper input, replace ammeter. If no voltage to ammeter, locate and fix open between switch and ammeter.
		e.	Open circuit between ammeter and timer.	e.	Disconnect harness at timer and check voltage pin B (of harness) to ground. If none, locate and correct open circuit.
2.	Ammeter shows normal current part of cycle, zero current rest of cycle.	a.	Open in wiring between timer and firewall connector.	a.	Refer to HEAT TEST in this chapter to find deicers not heating and test for voltage on that pin of firewall connector. If zero over 2 minutes, locate and fix open in wiring from timer to firewall.
		b.	Open between firewall and deicer lead straps.	b.	If voltage to firewall plug, try voltage at junction of deicer lead and slip ring lead. If no voltage, find and correct open in wiring to brush block, open within brush block, or no contact brush to slip ring.
		c.	No ground circuit, one engine.	c.	If voltage at deicer leads, locate and fix open from deicer to ground.
3.	Ammeter shows normal current part of cycle, low current rest of cycle.	a.	Inner and outer deicers heating same phase.	a.	Locate and repair incorrect connections.

TROUBLESHOOTING
PROPELLER DEICING SYSTEM (Cont'd)
(PRIOR TO P-579)

TROUBLE	PROBABLE CAUSE	REMARKS
	b. Open in deicer or slip ring assembly.	b. Disconnect deicer straps to check heate resistance. If resistance is within specifie limits, locate and fix open in slip rin leads. If not, replace deicer with oper circuit.
	c. High resistance in circuit with low current.	c. If not in contact of brush to slip rin (including ground brush), trace wiring t deicer and to timer to fix partially broke wire, loose or corroded connection.
4. Ammeter shows low current over entire cycle.	a. Aircraft voltage low.	a. Check voltage into switch.
	b. Ammeter faulty.	b. Refer to step "1d".
	c. High resistance up to timer.	c. Check for partially broken wire, loose o corroded connection in wiring from aircra supply to timer input.
5. Ammeter shows excess current over entire cycle.	a. Ammeter faulty.	a. Refer to step "1d".
	b. Ground between ammeter and timer.	b. Disconnect harness at timer and, wit ohmmeter, check from pin B (of harness to ground. If ground is indicated, locat and correct.
6. Ammeter shows normal current part of cycle, excess current rest of cycle.	a. Ground between timer and brush block.	a. Disconnect leads at brush block an check from power leads to ground wit ohmmeter. If ground is indicated, locat and correct.
	b. Ground between brush block and deicers. (Excluding ground brush circuit.)	b. If no short exists at brush-slip rin contact, check for ground from slip rin lead to bare prop while flexing slip rin and deicer leads. If a ground is indicatec locate and correct.
	c. Short between two adjacent circuits.	c. Check for shorts or low resistanc between circuits, if any, locate an correct.
	d. Timer faulty.	d. Test timer as indicated in DEICER TIME CHECK in this chapter.
7. Ammeter does not "flick" each 34 seconds.	a. Timer ground open.	a. Disconnect harness at timer check wit ohmmeter from pin G (of harness) t ground. If no circuit, refer to Wirin Diagram Manual (P/N 60-590001-29) t fix open circuit.

TROUBLESHOOTING
PROPELLER DEICING SYSTEM (Cont'd)
(PRIOR TO P-579)

TROUBLE	PROBABLE CAUSE	REMARKS
	b. Timer contacts are welded (caused by short circuit in system).	b. Test timer as in DEICER TIMER CHECK in this chapter. If timer does not cycle with voltage at pin B, replace timer but be sure short causing original problem has been located and corrected.
8. Ammeter flicks between 34 second phase periods.	a. Loose connection between aircraft power supply and timer input.	a. If trouble occurs over entire cycle, trace wiring from power source to timer input to locate and tighten loose connection.
	b. Loose or poor connection timer to deicers.	b. If trouble occurs in part of cycle, find which deicers are affected and check for rough to dirty slip rings causing brush to "skip". If not this, trace circuits to locate and fix loose or poor connection. (If all deicers on one prop are affected, check the ground circuit.)
	c. Timer cycles erratically.	c. Test timer as indicated in DEICER TIMER CHECK in this chapter.
9. Radio noise or interference with deicers on.	a. Brushes "arcing".	a. Check brush alignment as outlined under ELECTRICAL PROPELLER DEICER in Chapter 5-20-00 of this manual. Look for rough or dirty slip rings. If this is the cause, clean, machine or replace slip ring assembly. Check for slip ring alignment.
	b. Loose connection.	b. Refer to step 8 above.
	c. Switch faulty.	c. Try jumper wire across switch. If radio noise disappears, replace the switch.
	d. Wiring located less than 8 inches from radio equipment wiring.	d. Replace at least 8 inches from input wiring to radio equipment.
10. Cycling sequence not correct.	a. Crossed connections.	a. Check Wiring Diagram Manual (P/N 60-590001-29) for improper connections.
11. Rapid brush wear or frequent breakage.	a. Brush block out of alignment.	a. Check brush alignment as outlined under ELECTRIC PROPELLER DEICER in Chapter 5-20-00 of this manual.
	b. Slip ring wobbles.	b. Check slip ring alignment with dial indicator.

"END"

TROUBLESHOOTING ELECTROTHERMAL PROPELLER DEICE SYSTEM (P-579 and After)

Propeller deice ammeter reading outside the shaded area of the meter (14-18 amperes) is an indication that a fault may exist in the deice system. It should be noted, however, that current readings above or below the shaded areas of the deice ammeter may indicate an output voltage outside the normal operating range (28.25 ± .25 vdc) rather than a defect in the deice system itself. Excessively high operating voltage could conceivably damage the deice system and create a multiple fault condition; therefore, operation should not be instituted until the fault or faults have been corrected. Use of battery power alone, during operation of the deice system, should be avoided because the battery output voltage will be lower than normal operating voltage and may produce ammeter readings below the shaded area of the meter.

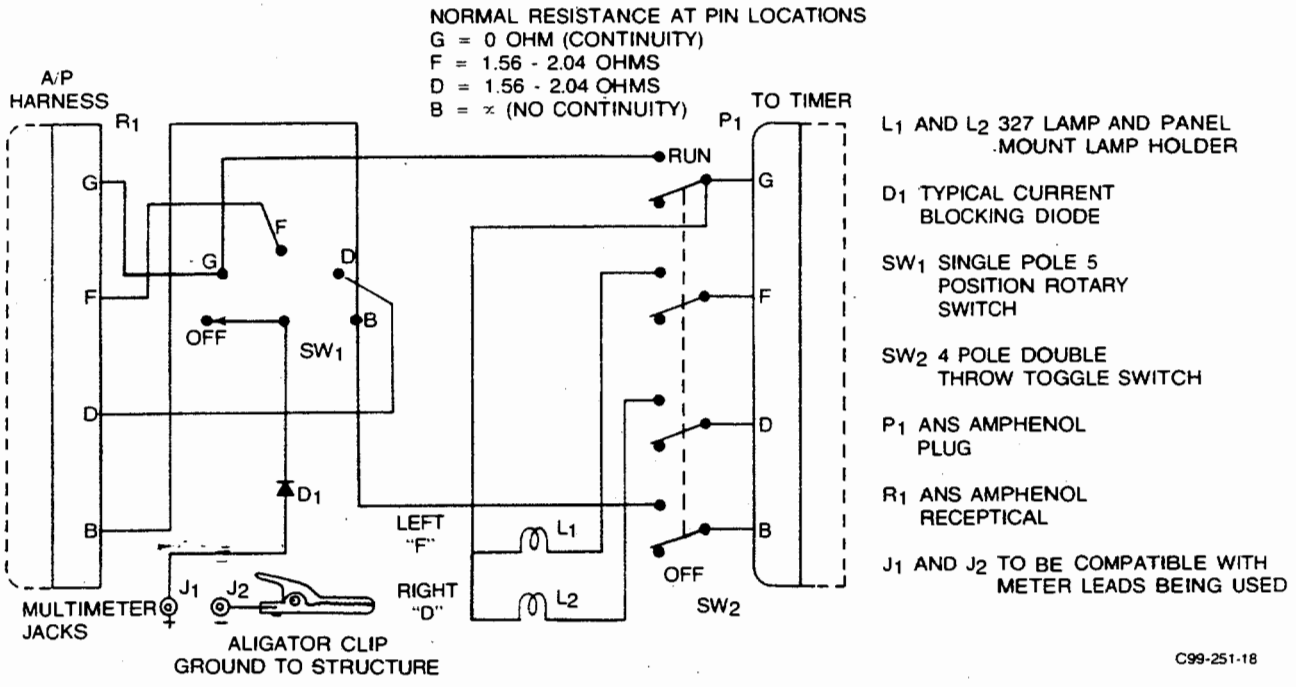
All resistance and continuity checks are made with the engines off, battery off and the timer disconnected. Resistance values specified in the troubleshooting chart may not be exact, as small variances may occur from one installation to another and will be subject to the accuracy of the particular resistance measuring instrument being used. For this reason it is recommended that a sensitive multimeter be used known to be accurate with ±1% (digital being preferred).

The test unit (Figure 101) was designed to be used in conjunction with these troubleshooting procedures and can be built with standard parts normally found in the shop. Operation and use of the test unit as outlined in the troubleshooting procedure makes it possible to positively check the timer in the airplane during system operation.

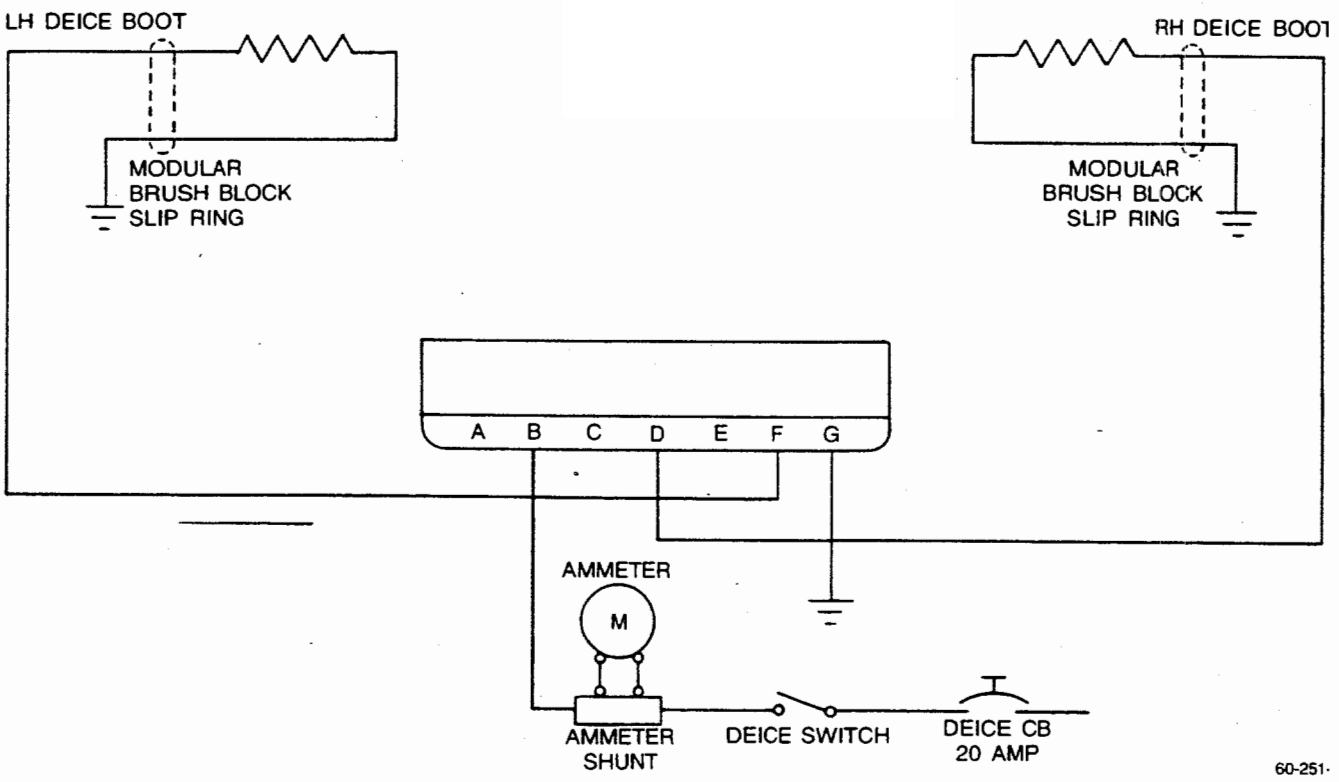
The troubleshooting chart which follows allows for an orderly flow of checks in a sequence consistent with the most convenient order of activity for the technician. The numbers in parenthesis, preceding some steps of the troubleshooting sequence, refer to notes found at the bottom of the chart. An electrothermal propeller deice control schematic (Figure 102) should be used for reference during system troubleshooting.

CAUTION

Propeller deice system must not be ground operated for extended periods of time as damage to the deice boots and pitting of the slip rings may occur.

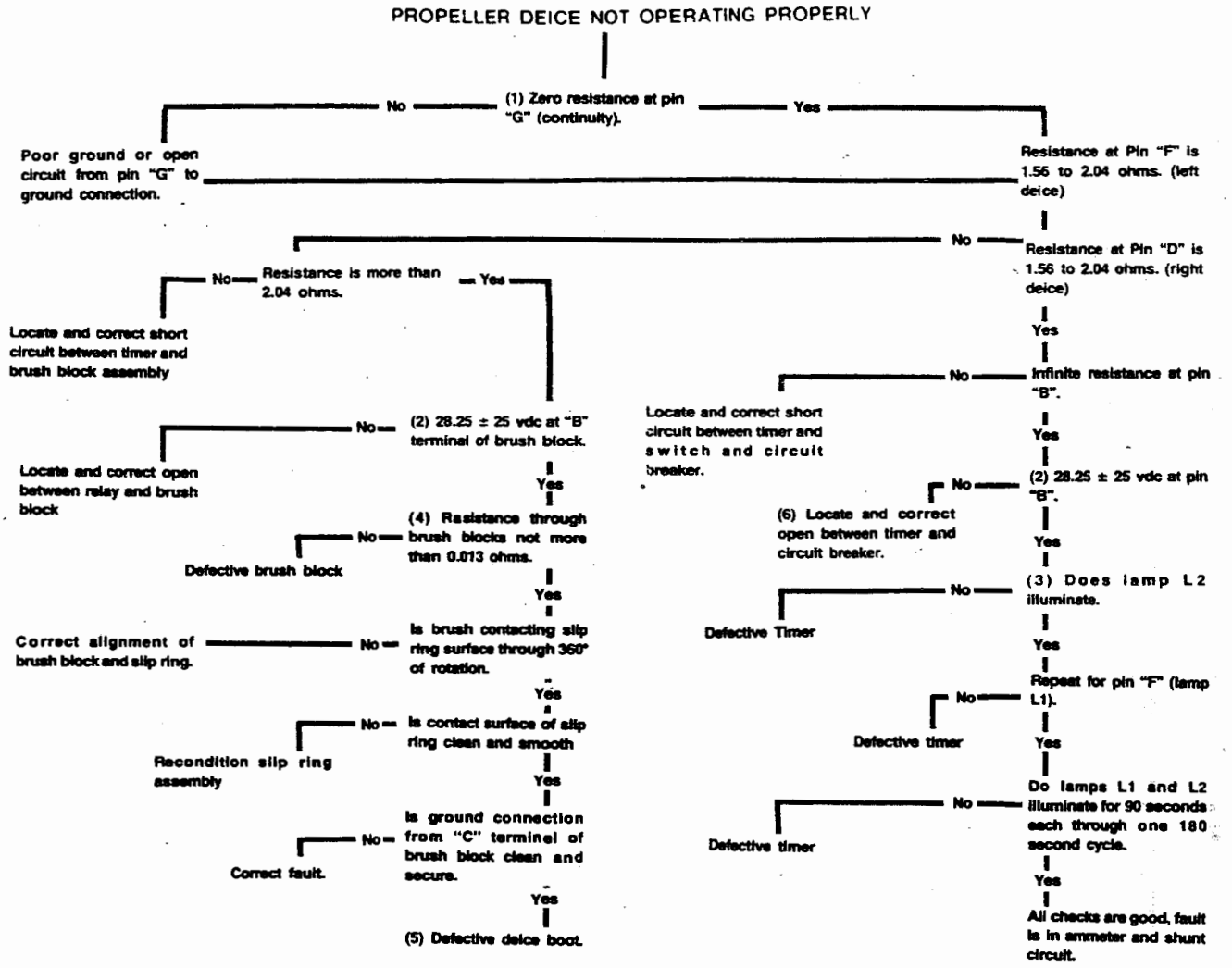


Electrothermal Propeller Deice Test Unit
Figure 101



Propeller Deice Control Schematic
Figure 102

TROUBLESHOOTING
ELECTROTHERMAL PROPELLER
DEICE (P-579 and after)



(1) All resistance and continuity checks are made with the battery switch OFF and the timer disconnected (if using test unit in Figure 101, placing SW 2 in the OFF position will disconnect timer)

(2) Voltage measurements are made with the propeller deice switch ON, circuit breaker in, and an auxiliary power unit connected or at least one generator on the line. Deice ammeter readings will vary directly with voltage.

(3) Reconnect timer (if using test unit Figure 101, place SW 2 in run position) advance timer by turning automatic propeller deice switch off then on again till lamp L2 illuminates. At no time during this check should both lamps (L1 and L2) be illuminated.

(4) Resistance is measured from face of brush block to terminal and must not exceed 0.013 ohms.

(5) Resistance measured through individual deiced boots should not exceed 5.9 ohm.

(6) Check switch and circuit breaker for continuity through switch in on position.

"END"

PROPELLER - MAINTENANCE PRACTICES

PROPELLER DEICER BOOT REMOVAL

a. Remove the propeller spinner.

b. On airplanes P-3 thru P-309, disconnect the deicer boot leads from the spinner bulkhead. Remove the clip securing the strap to the spinner bulkhead and the clamp securing it to the propeller hub.

c. On airplanes P-310 and after, disconnect the deicer boot leads from the clamp on the propeller hub.

d. Use MIL-M-13999 methyl ethyl ketone (21, Chart 207, 91-00-00) or toluol (22, Chart 207, 91-00-00) to soften the adhesion line between the boot and the blade, loosen one corner 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.

e. Apply a slow, steady pull on the boot to pull it off the propeller surface while continuing to use the solvent to soften the adhesive.

f. Remove the remaining adhesive from the boot and propeller blade with toluol or methyl ethyl ketone.

PROPELLER DEICER BOOT INSTALLATION (Figure 201)

a. Position the deicer boot on the propeller blade so that its center line at the inboard end is adjacent to the split in the propeller blade clamp and 2 ± 1/16 inch outboard of the clamp, and the center line at the outboard end falls on the blade leading edge. Be sure the lead strap is in the proper position to be clamped to the blade retaining clamp.

b. Mask off an area approximately 1/2 inch from the end and each side of the boot.

c. Remove the deice boot and strip any paint in the masked area from the retaining clamp outboard. On propeller blades coated with urethane, sand lightly, using 320 grit sandpaper, to remove all glaze from the urethane coating. Clean the area thoroughly with MIL-M-13999 methyl ethyl ketone (21, Chart 207, 91-00-00). For final cleaning, wipe the solvent off quickly with a clean, dry, lint-free cloth to avoid leaving a film.

CAUTION

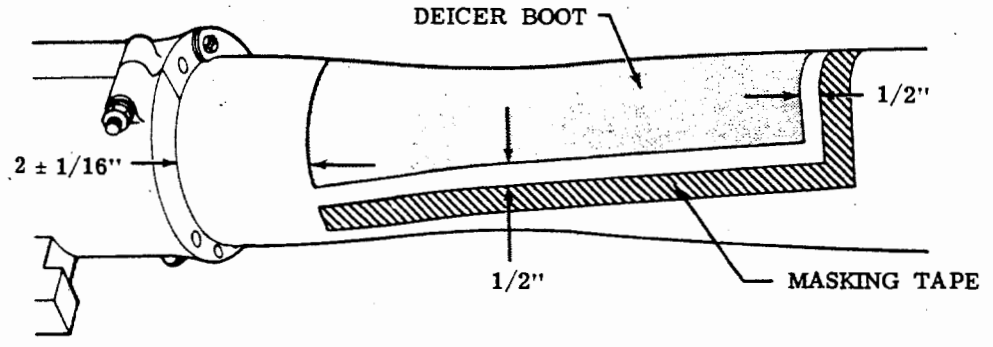
The metal and rubber parts must be thoroughly clean to assure maximum adhesion.

d. Moisten a clean cloth with methyl ethyl ketone or toluol and clean the unglazed surface of the deicer boot, changing the cloth frequently to avoid contamination of the clean area.

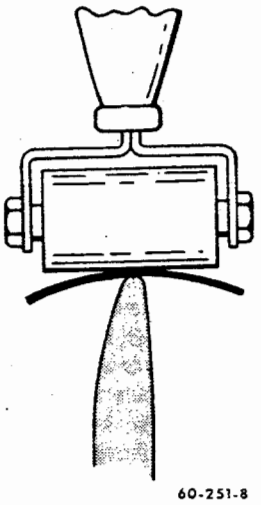
NOTE

To prevent the edges of the deicer boots from curling while applying the cement, place masking tape around the edges of the glazed side of the boot. Remove the masking tape before installing the boot.

e. Apply one even brush coat of EC-1300L cement (12, Chart 205, 91-00-00) to the propeller blade. 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



Deicer Boot Installation
Figure 201

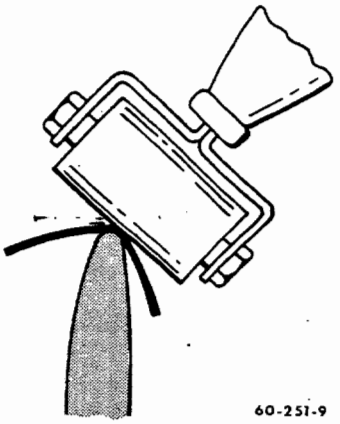


Center Rolling
Figure 202

humidity is between 75% and 90%. Do not apply the cement if the relative humidity is higher than 90%.

f. After allowing sufficient drying time, apply a second brush coat of cement to the propeller and one coat of cement to the unglazed surface of the deicer boot. Do not apply cement to more than 1/2 inch of the deicer lead strap. Allow the cement to dry.

g. Position the deicer boot on the propeller, starting 2 ± 1/16 inch from the blade retaining clamp, making sure the lead strap is in position to clamp to the blade retaining clamp. Moisten the cement lightly with methyl ethyl ketone or toluol and tack the boot center line to the blade leading edge. If the center line of the boot deviates from the blade leading edge, pull up with a quick motion and replace properly. Roll firmly along the center line with a rubber roller. (See Figure 202.)



Side Rolling
Figure 203

CAUTION

Never use a metal or wooden roller for this purpose, for they would damage the heating elements in the deicer boot.

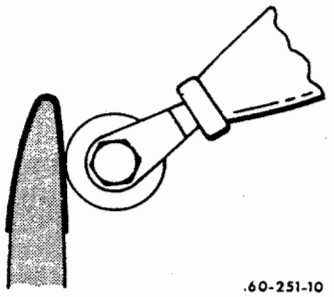
h. Gradually tilting the roller, work the boot carefully over each side of the blade contour. Avoid trapping air pockets under the boot. (See Figure 203.)

i. Roll outwardly from the center line to the edges of the boot. (See Figure 204.) If excess material at the edges tends to form puckers, work them out smoothly and carefully with the fingers.

j. Roll the tapered edges of the boot with a narrow steel stitcher roller.

k. Clean the blade with a clean cloth moistened with toluol or methyl ethyl ketone. Be careful not to let solvent run into the edge of the boot.

l. Apply one even brush coat or EC-801 sealer (11, Chart 205, 91-00-00) behind the lead strap where the boot and the blade meet.



Edge Rolling
Figure 204

NOTE

The EC-801 sealer is a two part sealer and must be thoroughly mixed. Mix the EC-801A and EC-801B combination as directed on the containers.

m. Apply one even brush coat of EC-801 sealer around the edges of the boot, allowing 1/16 to 1/8 inch overlap on the boot but extended to the masking tape. Remove the masking tape immediately after applying the sealer to obtain a neat border.

n. Allow sufficient time for the EC-801 to dry (from 24 to 72 hours, depending on conditions).

o. Apply satin finish black urethane paint to an area around the boot so that it covers all of the sealer and overlaps the edge of the boot and the blade by a minimum of 1/8 inch.

NOTE

To prevent propeller blade bearing grease from causing the boot to peel back or deteriorate, the urethane paint should be applied in a uniform coat so that grease cannot get to the boot cement or sealer.

p. Allow the urethane paint to dry as specified by the manufacturer.

q. On airplanes P-3 thru P-309, install the clamp securing the lead strap to the propeller blade retaining clamps. Connect the lead terminals and install the clip on the spinner bulkhead. There must be no slack between the terminal and the clip to assure enough slack between the clip and the clamp on the blade to allow propeller feathering.

r. On airplanes P-310 and after, connect the lead terminals to the clamp at the propeller hub.

DEICER TIMER CHECK (PRIOR TO P-579)

Experience in the field has indicated that often the timer is considered defective when the source of the trouble lies elsewhere. For this reason, the following test should be performed before the timer is removed as defective.

a. With the wiring harness disconnected at the timer and the deicer switch in the ON position, check the voltage from pin B of the harness plug to ground. If no voltage is present, the timer is NOT at fault; however, if system voltage is present at pin B, check the circuit from harness plug pin G to ground with an ohmmeter. If no circuit is indicated, the fault is in the ground lead rather than the timer. If ground connection is open, the timer step switch will not change position.

b. After the ground and power circuits have been checked, connect a jumper wire between pin B of the timer receptacle and terminal B of the connector plug and from pin G of the timer receptacle to ground. With the deicing system switch ON, check the voltage to ground from pin B of the timer. The voltmeter should indicate approximately 24 volts when the airplane battery supply is being used. Next, check the DC voltage to ground from pins C, D, E, and F, the points at which the system voltage is impressed in sequence to cycle power to the propeller deicers. Each of the plugs should read 24 volts in the following sequence.

Timing Sequence	Time ON	Areas of Prop Deicers Heated
Pin C	30 sec.	Right engine prop, Outb'd. halves
Pin D	30 sec.	Right engine prop, Inb'd. halves
Pin E	30 sec.	Left engine prop, Outb'd. halves
Pin F	30 sec.	Left engine prop, Inb'd. halves

NOTE

The timer does not reposition itself to start at pin C when the system is turned off, but will begin its cycling at the same position in which it was when last turned off. Cycling will then proceed in the order of C, D, E, and F as before.

After a voltage reading of 24 volts DC is obtained, hold the voltmeter probe on the pin until the voltage drops to zero before moving the probe on to the next pin in the sequence noted above. After the correctness of the cycling sequence has been established, turn the deicing system switch OFF at the beginning of one of the "on-time" periods and record the letter of the pin at which the voltage supply is present to facilitate performance of the following test.

DEICER TIMER CHECK (P-579 AND AFTER)

Experience in the field has indicated that often the timer is considered inoperable when the source of the trouble is elsewhere. For this reason, the following test should be performed before the timer is judged to be inoperable.

a. With the timer harness plug disconnected and the deicer switch in the ON position, check for voltage from pin B of the plug to ground. If no voltage is present, the timer is not defective; check the circuit breaker switch or the power supply. However, if system voltage is present at pin B, check the circuit from the harness to ground with an ohmmeter. If there is no continuity, the fault is in the ground circuit rather than the timer. If the ground circuit is open, the timer will not cycle.

b. After the ground and power circuits have been checked, connect a jumper wire between pin B of the timer receptacle and terminal B of the connector plug, and from pin G on the timer receptacle to ground. With the deicing system switch ON, check the voltage to ground from pin B of the timer. The voltmeter should indicate approximately 24 volts dc when the airplane battery supply is being used. Next, check the voltage to ground from pins D and F, the points at which the system voltage is impressed in sequence to cycle power to the LH and RH propeller deicers. The presence of 24 vdc system voltage should alternate at pins D and F for 90 seconds in duration as the timer cycles.

HEAT TEST (PRIOR TO P-579)

Before this test can be performed, the jumper wire installed for the timer test must be removed so that the connector plug can be replaced in the timer receptacle. Two men are required to perform this test, one in the pilot's compartment

to monitor the ammeter while the other checks the deicer boots. The man in the pilot's compartment turns the deicer system switch ON while the man outside feels the deicer boots to see if they are heating properly. The man in the pilot's compartment observes the ammeter for the proper readings (14 to 18 amperes) throughout the timing sequence. The ammeter needle should deflect every 30 seconds in response to the switching action of the timer. Each time this occurs, the man in the pilot's compartment must notify the man inspecting the propeller deicer boots so that the latter can change the position of his hands to check the proper heating sequence of the propeller deicer areas. If any irregularities are detected, a continuity check should be performed on the wiring from the timer to the brush block holders and the propeller deicer terminal connections.

HEAT TEST (P-579 AND AFTER)

Remove the jumper wires that were installed for the timer test and reconnect the timer receptacle. To perform this test, two people are required - one person in the flight compartment to operate the propeller deice switch and observe the propeller deice ammeter, the other on the ground checking the deice boots for proper heating. While the person in the flight compartment observes the ammeter for a reading of 14 to 18 amps, the person on the ground checks for a rise in heat on each propeller deice boot for approximately 90 seconds on each side. If either boot fails to heat, check the circuit between the timer and the propeller deice boot for continuity.

CAUTION

While following the instructions of the above "Heat Test" section, rotate the propeller back and forth to prevent arcing between the brushes and slip ring.

WARNING

Before moving the propeller, ensure that the ignition switch is OFF and that the engine has completely cooled, as there is always the danger of a cylinder firing when the propeller is moved.

CONTINUITY TEST (PRIOR TO P-579)

After removing the plug from the timer, use an ohmmeter to check continuity from:

a. Pin C of the plug to the outboard terminal of one prop boot on the right engine.
b. Pin D of the plug to the inboard terminal of one prop boot on the right engine.
c. Pin E of the plug to the outboard terminal of one prop boot on the left engine.
d. Pin F of the plug to the inboard terminal of one prop boot on the left engine.
e. Pin G of the plug to ground.
f. Ground terminal of one prop boot on the right engine to ground.
g. Ground terminal of one prop boot on the left engine to ground.

CONTINUITY TEST (P-579 AND AFTER)

After removing the plug from the timer, use an ohmmeter to check continuity from:

a. Pin D of the plug to the terminal of the propeller deice boot on the right engine.
b. Pin F of the plug to the terminal of the propeller deice boot on the left engine.
c. Pin G of the plug to ground.
d. Ground terminal of the propeller boot on the right engine to ground.
e. Ground terminal of the propeller boot on the left engine to ground.

BRUSH TO SLIP RING RESISTANCE TEST

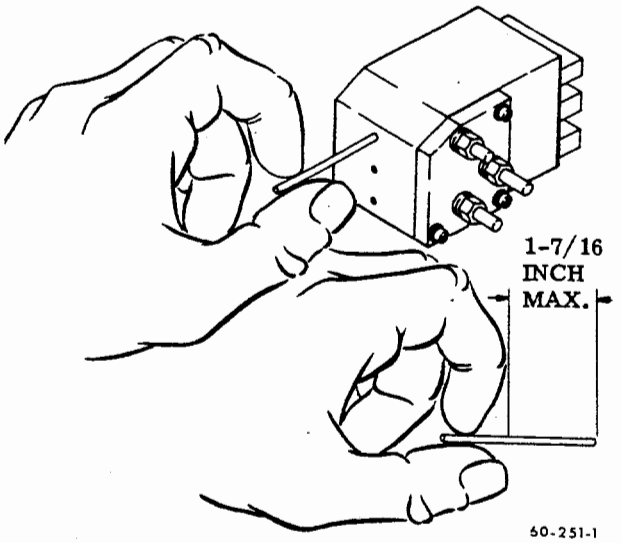
To check for incorrect resistance or the presence of a short or open circuit at the brush-to-slip ring contact, disconnect the harness at the timer and check the resistance from each deicer circuit lead (pins C, D, E, and F of the harness plug) to ground with a low range ohmmeter. If the resultant readings are not 1.55 to 1.78 ohms, disconnect the deicer lead straps to measure heater resistance individually. Individual boot resistance should measure between 4.58 and 5.26 ohms. If the readings in the first check are not within the accepted limits but those in the second check are, the trouble is probably in the brush-to-slip ring area. If the readings in the second check are also off, the deicer concerned is damaged and must be replaced.

BRUSH BLOCK RESISTANCE CHECK (PRIOR TO P-579)

To check for an open circuit, a short, or high resistance in the brush block, measure the resistance from the face of the brush to its terminal studs with a low range ohmmeter. If this resistance measures over 0.013 ohms, locate and repair the cause of excessive resistance. If the resistance is infinite, locate and correct the open circuit or ground, or else replace the brush. Check the resistance between the three terminal studs. This resistance should not be less than 5 megohms.

BRUSH BLOCK RESISTANCE CHECK (P-579 AND AFTER)

To determine if an open or short circuit or high resistance is present in the brush block, measure the resistance from the face of the brush to its terminal stud or receptacle pin with a low-range ohmmeter. The resistance reading should not exceed 0.013 ohms. If this ohm value is exceeded, locate and repair the problem area. If the resistance reading is infinite, locate and repair the open circuit or replace the brush. Check the resistance between the receptacle pins and the terminal studs. (The resistance reading should be less than 0.5 megohms).



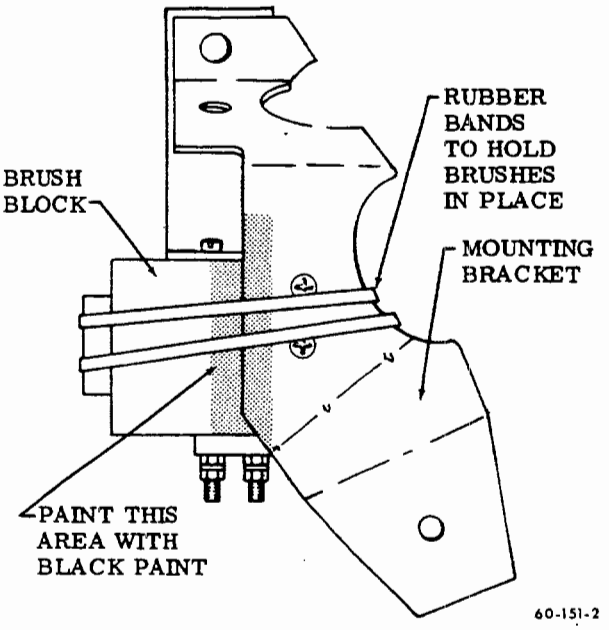
Determining Deicer Brush Wear
Figure 205

BRUSH REPLACEMENT (PRIOR TO P-579)

The propeller deicer brushes should be replaced when a minimum of 1/4 inch of brush material remains. It is good practice, however, to replace the brushes when 3/8 inch of the brush material still remains. Brush length may be determined by inserting a piece of safety wire into the holes at the back of the brush block assembly (Figure 205). When 1-7/16 inch dimension is measured, there is approximately 1/4 inch of brush material left. Replace the brushes as follows:

NOTE

The brush block and mounting bracket should be removed intact so the brush block's exact location with respect to the mounting bracket can be marked. This will facilitate alignment of the brush block during reinstallation.



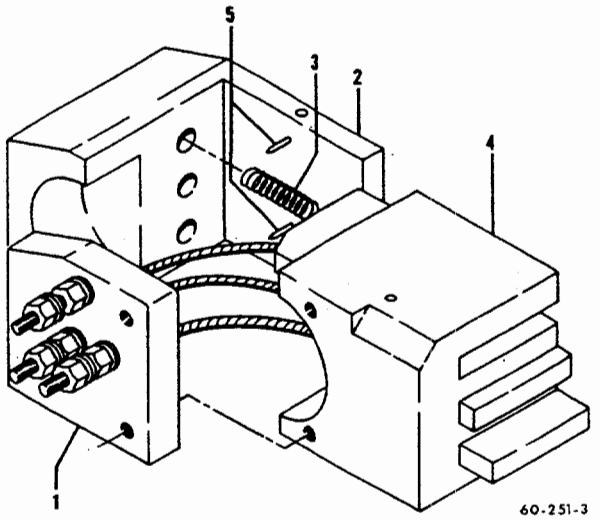
Marking Location of Brush Block
Figure 206

a. Loosen both ends of the brush block mounting bracket and remove the complete assembly.

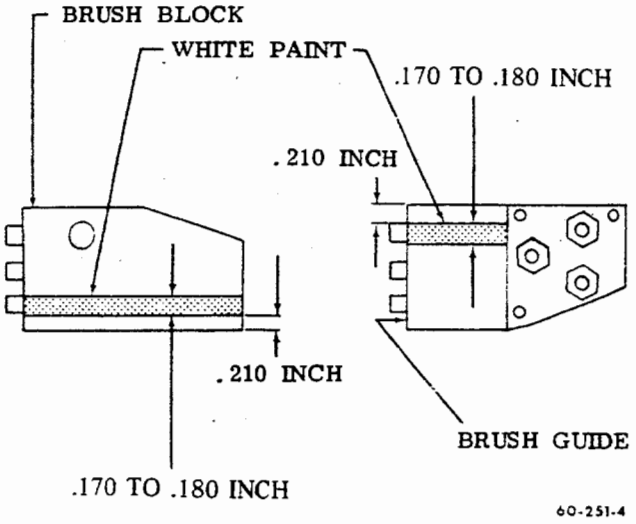
b. Tag the lead wires attached to the terminals on the brush block and disconnect them.

c. Clean the brush block and mounting bracket. Paint the side of the brush block and the edge of the mounting bracket as shown in Figure 206 with black paint.

d. Remove the brush block from the mounting bracket.



Deicer Brush Block Assembly
Figure 207



Deicer Brush Alignment Strip
Figure 208

e. Disassemble the brush block (Figure 207) by removing the screws attaching the terminal plate to the brush block, then separate the brush block by pulling the guide block approximately 1/4 inch toward the terminals to disengage the guide pins.

f. Mask off and paint a white stripe .170 to .180 inch wide on the brush block and brush guide. Locate as shown in Figure 208. The white stripes should be in line with the outer brush and will be used to align the brushes with the slip ring during reinstallation.

g. Remove the terminal plate, brushes and springs from the brush block.

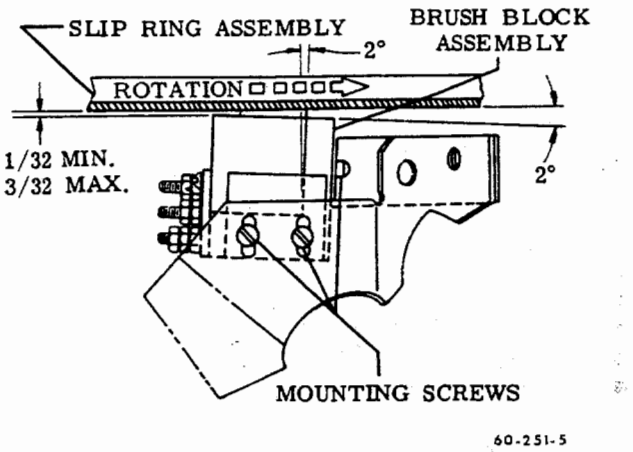
h. Disconnect the wires from the brushes being replaced, noting which terminal they correspond to.

i. Solder the wires from the new brushes to the appropriate terminals, holding the "wicking" to 1/8 inch maximum.

j. Install each brush in its correct groove in the guide block. Insert new springs into the guide block behind the brushes. Taking care not to apply a side load on the brushes or damage or pinch the brush leads, bring the support block into position. Install the opposite end of the springs into their corresponding holes in the support block. Press the two blocks together until the guide pins in the support block slip into the holes in the guide block. Install the screws which hold the terminal plate to both blocks.

NOTE

When replacing brushes or brush retainer assemblies, always install new brush springs.



Deicer Brush Block Installation
Figure 209

k. Check the amount of brush protrusion from the block. If the brushes protrude less than 9/16 inch, the brush leads should be untwisted to give more length. If this distance is more than 5/8 inch, the lead should be twisted to shorten the effective length until the brushes protrude from 9/16 to 5/8 inch. The brushes should then be checked for free sliding action.

l. Reinstall the brush block assembly on the mounting bracket, using the black paint (see step "c") as a guide for correct alignment.

m. Carefully push the brushes back into the brush block and secure them in place with a rubber band. (See Figure 206).

n. Using care not to bend or distort the mounting brackets, reinstall the brush block and bracket assembly as originally removed from the engine in step "a". Cut the rubber band and rotate the propeller to remove the broken pieces of rubber band.

NOTE

The white stripes on the top and bottom of the brush block should align with the outer slip ring. If the white stripes do not align with the outer slip ring it may be necessary to add or remove, all or part of the laminated shim. Removing shim material will move the brush block out. Each laminate in the shim is approximately .003 inch.

o. Check for proper clearance between the slip rings and the brush block (see Figure 209). The clearance should be 1/16 ± 1/32 inch with an angle of approximately two degrees from perpendicular, as measured toward the direction of slip ring rotation. If not correct, loosen the brush

block mounting screws and move in the elongated holes to correct the brush block position.

p. To preclude arcing caused by the rough surfaces of the new brushes, the engine should be operated for at least five hours before the deicer system is turned on. This does not apply to ground checks of the system performed while the engine is not running.

BRUSH REPLACEMENT (P-579 AND AFTER)

The modular brush assembly (P/N 3E2071) is made up of two modules (P/N 3E2011-1 and P/N 3E2011-2), each consisting of a plastic housing with an integral brush and spring. These modular units are stacked with a spacer and held together by screws to produce the modular brush assembly. When a brush wears out, the module containing it must be replaced since individual brush replacements are not available. Replace the entire brush module when only 3/8 inch of the brush material remains.

NOTE

During measurement, only 1/16 inch of brush should protrude from the brush module, with this being the normal protrusion when the brush is installed on the airplane.

Brush wear is determined by inserting a pin into a hole in the back of the brush module as shown in Figure 205. On all modules having brushes with rods, the brush module should be replaced when the dimension shown in Figure 205 is 17/34 ± 1/32 inch. On all rodless brushes, the module should be replaced when this dimension is 1 7/64 ± 1/32 inch. Use the following procedures when replacing the brushes:

a. Disconnect the wire harness terminals at the modular unit terminals.

b. Remove the screws, nuts, and washers that secure the modular unit to the mounting bracket.

c. Remove the assembly retaining screws and separate the modules and spacers.

d. Replace each module with another of the same part number. (The part number is etched into the plastic housing.)

e. Restack the modules and spacers as they were unstacked in step "c". (Stacking arrangement may be changed if there is interference with any other engine or propeller component.)

f. Install the assembly screws so that the screw head fits in the recess in the spacer. Place the flat washer between the star washer and the modular housing and install the retaining nut. Ensure that the assembly is square before tightening the assembly screws.

g. Place the modular brush assembly on the mounting bracket and insert the mounting screw through both the bracket and the brush block assembly. Place one washer under the head of the screw and one under the nut.

h. Before installing the retaining nuts, ensure that the brushes are aligned with the slip rings so that the entire face of the brush is in contact with the copper rings. If the brushes do not align with the slip rings throughout the entire 360 degree rotation of the slip ring, add or remove spacers (P/N 4E2218-3) between the modules until the brushes are properly aligned with the approximate center of the copper ring.

i. Install the retaining nut and washers, ensuring that 1/16 ± 1/32 inch is maintained between the brush module and the slip ring surface. To prevent damage to the brushes, the modular brush assembly should be angled so that the brushes contact the slip ring at an angle of approximately 2 degrees from perpendicular as measured toward the direction of slip ring rotation.

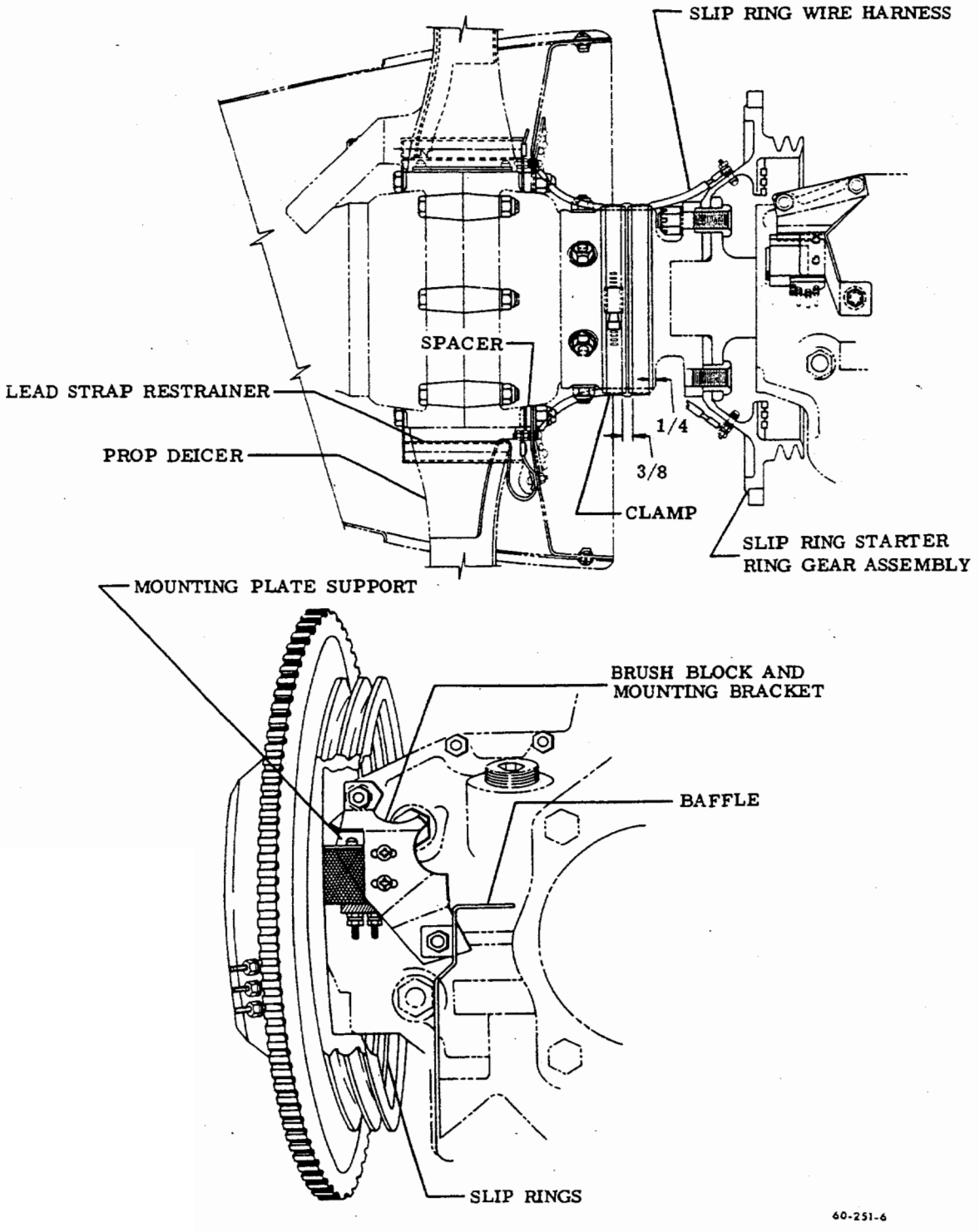
j. Reconnect the wire harness terminals to the modular unit terminals.

SLIP RING MACHINING

Slip rings which have roughened or damaged surfaces, but which are structurally sound, can be machined and restored to serviceability. Remove the slip ring assembly from the aircraft and mount it in a lathe. Position is concentrically in the lathe, with not over 0.002 inch wobble or run-out over 360 degree rotation. Take light cuts for a smooth finish and cut no deeper than required to remove surface damage. The contact surfaces of the three slip rings must be parallel within 0.005 inch, and flat within 0.005 inch overall. Deviation from flat is not to exceed 0.002 inch over a 4 inch arc. If necessary, undercut the insulation between the slip rings to a depth of 0.020 to 0.030 inches below the contact surface of the slip rings. In this operation, width of the slip ring MUST NOT be reduced more than 0.005 inch. Contact surfaces of the slip rings must have a finish of 29-35 micro inches. Deburr the slip ring edges and reinstall in the aircraft and align.

NOTE

If, in machining, the solder or braze connection on the underside of the slip ring is exposed, replacement of the slip ring assembly will be necessary.



Propeller Deicer Installation
Figure 210

"END"

CHAPTER 32

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"END"

BEECHCRAFT
DUKE 60 SERIES
MAINTENANCE MANUAL

CHAPTER 32 - LANDING GEAR

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TABLE OF CONTENTS (Cont'd)

"END"

GENERAL - DESCRIPTION AND OPERATION

The landing gear is operated by a split-field series-wound motor and an actuator located under the cabin floorboards aft of the main spar. One field is used to drive the motor in each direction. To prevent overtravel of the landing gear, a dynamic braking 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 landing gear almost instantly.

The landing gear motor is controlled by the gear extension switch located on the left subpanel. The larger upper arms and the lower arm of the actuator, in conjunction with rod assemblies and linkage, control extension and retraction of the main and nose landing gear. Rod assemblies attached to the smaller upper actuator arms operate the inboard main landing gear doors.

Landing gear limit switches, located adjacent to the actuator, limit the gear travel during the extend and retract cycle. These switches, when actuated, terminate the landing gear travel.

To prevent accidental landing gear retraction on the ground, a safety switch on the left main landing gear breaks the control circuit whenever the strut is compressed.

CAUTION

Never rely on the safety switch to keep the gear down while taxiing, landing, or on the takeoff roll. Always check the position of the landing gear switch.

The Duke 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.

SAFETY SYSTEM

The optional landing gear safety system functions through the action of a solenoid in the landing gear position switch in conjunction with a three-position safety system switch, two pressure switches mounted on the inboard side of the left main landing gear wheel well and two microswitches located adjacent to the existing throttle position warning switches.

Each pressure switch is connected into the pitot and static system. The pressure switch in the gear-up circuit is actuated by the pressure differential that exists between the pitot and static systems and will close with increasing pressure at approximately 85 ± 2 knots. The pressure switch in the gear-down circuit will close with decreasing pressure at 120 ± 2 knots. When the landing gear position switch is in the UP position and an airspeed of 85 ± 2 knots. has been attained, the pressure switch in the gear-up circuit closes and actuates a relay mounted on the front spar, thus completing the circuit and retracting the landing gear. A diode locks the relay in the closed position until the retraction cycle is completed. For the preceding to occur, however, the microswitch adjacent to the throttle position switch must also be in the open position. This microswitch is actuated by the throttle control when the throttles are advanced sufficiently for the manifold pressure gage to register approximately 17 ± 1 in. Hg. Conversely, if the throttles are retarded beyond the position corresponding to approximately 17 ± 1 in. Hg of manifold pressure, the microswitch will close. If at the same time the microswitch closes, the airspeed has dropped below 120 ± 2 knots, the resultant pressure differential between the pitot and static systems will actuate the pressure switch in the gear-down circuit. With both the microswitch and pressure switch closed, the current flow through the solenoid will cause the landing gear position switch to drop into the DOWN position, thus completing the gear-down circuit.

If the landing gear position switch is placed in the UP position while the landing gear safety system is in the ON position, the landing gears will retract when the following conditions are mutually fulfilled.

a. The airplane must have attained an airspeed of at least 85 ± 2 knots.
b. The throttle setting must have been advanced sufficiently to have produced a manifold pressure of approximately 17 ± 1 in. Hg.

By the same token, the landing gear automatically extends under the following conditions:

a. The airspeed must have dropped below 120 ± 2 knots.
b. The throttle setting must have been retarded enough for manifold pressure to have dropped below approximately 17 ± 1 in. Hg.

The safety system switch is a three position switch, with normally ON or OFF positions. The switch also contains a momentary or test position for checking that the system is functioning properly. When released from the test position, the switch returns to the ON position.

MANUAL LANDING GEAR EXTENSION SYSTEM

In the event of landing gear malfunction in flight, the gear may be manually extended, but not retracted, by a hand crank located below copilot's seat.

WARNING

If the gear has been extended manually for emergency reasons, the airplane must be put on jacks and inspected before the gear controls are returned to their normal position.

"END"

TROUBLESHOOTING
LANDING GEAR ELECTRICAL SYSTEM

	TROUBLE		*PROBABLE CAUSE*		*REMARKS*
1.	Landing gear motor fails to shut off when gear is retracted.	a.	Up limit switch out of adjustment.	a.	Readjust switch.
		b.	Defective switch.	b.	Replace switch.
2.	Landing gear fails to retract.	a.	Safety switch not closing.	a.	Readjust.
		b.	Up limit switch remaining open.	b.	Replace limit switch.
3.	Landing gear motor fails to shut off when gear is extended.	a.	Down limit switch does not open.	a.	Readjust limit switch.
		b.	Defective down limit switch.	b.	Replace limit switch.
4.	Landing gear actuator is hitting internal stops.	a.	Limit switch out of adjustment.	a.	Readjust limit switch.
		b.	Dynamic brake switch defective.	b.	Replace switch.
5.	Warning horn in-operative or malfunctioning.	a.	Open or grounded circuit.	a.	Check continuity.
		b.	Throttle switches inoperative.	b.	Check and adjust as necessary.
6.	Landing gear fails to extend.	a.	Tripped circuit breaker.	a.	Reset circuit breaker.
		b.	Down limit switches open.	b.	Check down limit switch. With the gear retracted the down limit switch should be closed.
		c.	Open circuit.	c.	Run a continuity check on the down limit switch.
7.	Landing gear will not retract or extend.	a.	Bad electrical connections.	a.	Run a continuity check from circuit breaker to switch. Inspect the dynamic brake relay.
		b.	Landing gear motor not grounded.	b.	Check motor ground.
		c.	Defective control circuit.	c.	Check items 1 through 3.

"END"

MAIN GEAR AND DOORS - MAINTENANCE PRACTICES

MAIN GEAR SHOCK ABSORBERS

To check the fluid level in the landing gear shock absorbers, deflate the strut by releasing the air through the valve and permit the strut to fully compress, then remove the filler valve.

WARNING

Do not remove the filler valve until all air pressure has been released or it may be blown off with considerable force, causing injury to personnel and property damage.

If the fluid level is low, add hydraulic fluid MIL-H-5606 (13, Chart 207, 91-00-00) until the fluid overflows slightly. Slowly cycle (compress and extend) the strut to expel any trapped air. Add fluid, as necessary, and install the filler valve.

With the airplane resting on the ground and the fuel cells full, inflate the main strut until 3 inches of the piston is exposed. Rock the airplane gently to prevent possible binding of the piston in the barrel when inflating.

CAUTION

Do not inflate the struts while the airplane is on jacks, since sudden extension or over-inflation of the struts may bend the torque knee.

LUBRICATION

MAIN WHEEL BEARINGS AND GREASE FITTINGS

Lubricate the main wheel bearings and grease fittings as detailed in the Lubrication Chart, Chapter 12-20-00.

UPLOCK ROLLERS

AIRPLANE SERIALS P-3 THROUGH P-154, P-156 THROUGH P-162, P-167 THROUGH P-171 AND P-181 PRIOR TO COMPLIANCE WITH SERVICE INSTRUCTIONS No. 0482-211.

The uplock rollers on these serials should be lubricated after 50 hours with SAE 10W-30 oil, and removed from the airplane and packed with MIL-G-23827 grease (11, Chart 207, 91-00-00), every 100 hours or any time that, while cleaning the wheel well, the bearings are subjected to degreasing with solvent under pressure.

NOTE

The grease fitting on the drag leg, directly above the uplock roller bearing, does not supply lubrication for the uplock roller bearing.

The uplock roller bearing may be lubricated as follows:

a. Place the airplane on jacks.
b. Partially retract the landing gear.
c. Remove the bolt attaching the uplock roller and the "V" brace drag leg center hinge point.
d. Remove the uplock roller bearing from the bolt.
e. Hold a finger over one end of the uplock roller center bearing race and place a grease gun against the opposite side of the bearing. As grease is pumped into the inner bearing race, it will be forced through the hole in the inner race and into the bearing cavity. Completely fill the bearing with grease.
f. Reinstall roller bearings and attaching bolt.

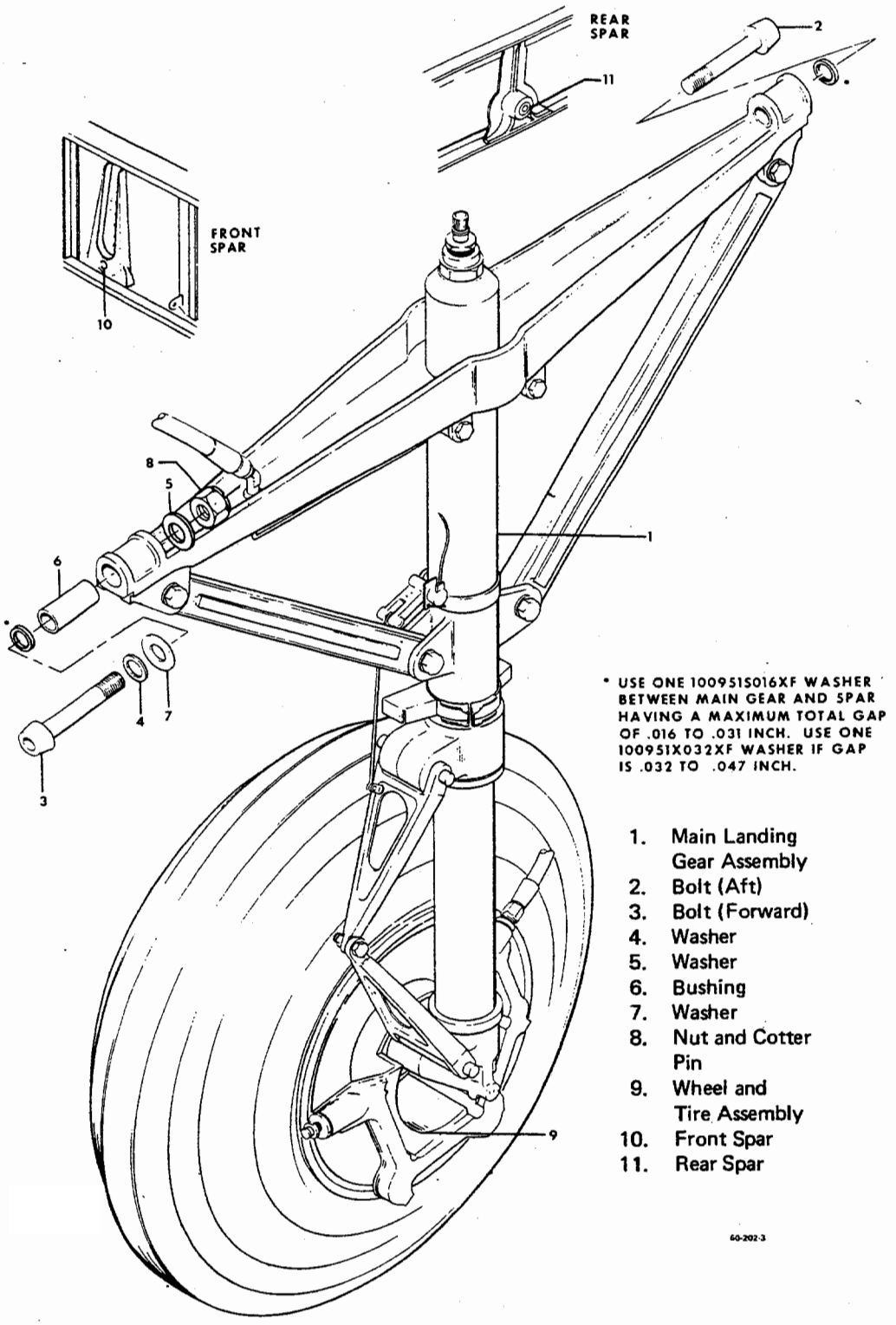
AIRPLANE SERIALS P-155, P-163 THROUGH P-166, P-172 AND AFTER, EXCEPT P-181, AND PRIOR AIRPLANE SERIALS IN COMPLIANCE WITH SERVICE INSTRUCTIONS No. 0482-211.

The uplock rollers on these serials are equipped with a grease fitting installed in the uplock roller bearing bolt and may be lubricated using a pressure grease gun. The uplock rollers should be lubricated using MIL-G-23827 grease (11, Chart 207, 91-00-00), each 100 hours, or any time that while cleaning the wheel well, the bearings are subjected to degreasing with solvent under pressure.

MAIN GEAR REMOVAL
(Figure 201)

When removing the landing gear, take care to preserve the original adjustments at the rod end fittings to facilitate reassembly.

a. With the airplane on a jack, retract the gear until the inboard landing gear door is in the fully open position.
b. Remove the outboard landing gear door from the landing gear strut.
c. Disconnect the inboard landing gear door actuating rod at the forward door hinge.
d. Unsnap the canvas cover and disconnect the uplock assembly from the strut.



Main Gear Installation
Figure 201

e. Open the brake cylinder bleed ports and pump all fluid from the system.

f. Disconnect the hydraulic line where the flexible hose couples to the tubing on the landing gear.

g. Disconnect the safety switch wire. (Left gear only).

h. Remove the bolt attaching the lift leg to the strut.

i. Remove the access door in the lower surface of the wing leading edge for access to the forward hinge bolt retaining nut and remove the nut. The rear strut brace hinge bolt is accessible by lowering the flap.

j. Remove cotter pins and nuts (8), washer (5), bolts (2 and 3), washers (4 and 7), and bushings (6) from the front and rear spars (10 and 11).

k. Lower the main gear assembly away from the airplane, being careful not to bend the skin at the wheel well edge.

MAIN GEAR INSTALLATION
(Figure 201)

a. Carefully position the main gear assembly in place against the front and rear spars (10 and 11).

b. Align bolt holes and install bushing (6), bolts (2 and 3), washers (4, 5, and 7), and nuts (8). Torque to 250 to 400 inch-pounds. Install new cotter pins.

NOTE

Use one 100951S016XF washer between main gear and spar having a maximum total gap of .016 to .031 inch. Use one 100951X032XF washer if gap is .032 to .047-inch.

c. Install the access door in the lower wing leading edge.

d. Install the bolt attaching the lift leg to the strut.

e. Connect landing gear safety switch wire. (Left gear only).

f. Connect the brake hydraulic line.

g. Connect the uplock assembly to the strut and snap the canvas cover in place.

h. Connect the inboard landing gear door actuating rod to the forward door hinge.

i. Install the outboard landing gear door to the landing gear strut.

j. Bleed the brake system.

k. Operate the landing gear and check for proper rigging of the uplock and doors.

100-HOUR INSPECTION

ACTUATOR AND MOTOR - Check the actuator and motor for security, visible damage and condition. Check motor wiring for breaks and chafed or deteriorated insulation. Check all actuating rods for cracks, evidence of bending and security.

LANDING GEAR STRUT - Inspect the strut and attaching components for cracks, security, condition, and leakage at the air filler valve and piston area.

STRUT FLUID LEVEL - Inspect the strut for proper inflation. If leakage is detected, deflate the strut and check the fluid level.

"END"

NOSE GEAR AND DOORS - MAINTENANCE PRACTICES

NOSE GEAR SHOCK ABSORBER

To check the fluid level in the landing gear shock absorber, deflate the strut by releasing the air through the valve and permit the strut to fully compress, then remove the filler valve.

WARNING

Do not remove the filler valve until all air pressure has been released or it may be blown off with considerable force, causing injury to personnel and property damage.

If the fluid level is low, add MIL-H-5606 hydraulic fluid (13, Chart 207, 91-00-00) until the fluid overflows slightly. Slowly cycle (compress and extend) the strut to expel any trapped air. Add fluid, as necessary, and install the filler valve.

With the aircraft resting on the ground and the fuel cells full, inflate the nose strut until 4-1/16 to 4-5/16 inch of the piston is exposed. Rock the aircraft gently to prevent possible binding of the piston in the barrel when inflating.

CAUTION

Do not inflate the strut while the aircraft is on jacks, since sudden extension or over-inflation of the strut may bend the torque knees.

LUBRICATION

Lubricate the nose wheel bearings and grease fittings as detailed in the Lubrication Chart in Chapter 12-20-00.

NOSE GEAR REMOVAL (Figure 201)

When removing the nose gear, take care to retain the original adjustments at the rod end fittings to facilitate reassembly.

a. With the aircraft on a jack, partially retract the landing gear to relieve the load on the retract rod compression spring.

b. Disconnect the drag leg at its fitting on the nose gear brace assembly.

c. Disconnect the shimmy dampener at the nose gear.

d. Disconnect the steering mechanism at the nose gear.

e. Disconnect the landing light wiring.

f. Remove the nose baggage compartment flooring that covers the area on either side of the wheel well to gain access to the hinge bolts.

g. Remove cotter pins (5), nuts (4), washers (2), bolts (1) and bushings (3).

h. Lower the nose gear assembly from the nose wheel well.

NOSE GEAR INSTALLATION (Figure 201)

a. Carefully position the nose gear assembly against the nose wheel well structure.

b. Align bolt holes and install bushings (3), bolts (1), washers (2), nuts (4) and cotter pins (5).

NOTE

Use 100951X016YX and 100951X032YX washers, as required, to obtain total end play between the nose gear assembly and supports, of .000 to .016-inch.

c. Install the nose baggage compartment flooring.

d. Connect the landing light wire.

e. Connect the drag leg on the nose gear brace assembly.

f. Connect the steering mechanism to the nose gear.

g. Check the shimmy dampener adjustment, then connect the shimmy dampener to the nose gear.

h. Operate the landing gear and check for proper rigging and nose gear adjustment.

SHIMMY DAMPENER REMOVAL

a. Remove the nut, washers and bolt attaching the shimmy dampener to the nose landing gear.

b. Remove the nut, washer, bolt and spacer attaching the shimmy dampener to the nose wheel well structure.

SHIMMY DAMPENER INSTALLATION

a. Position the aft end of the shimmy dampener in the bracket on the nose wheel structure and install the spacer, bolt, washer and nut.

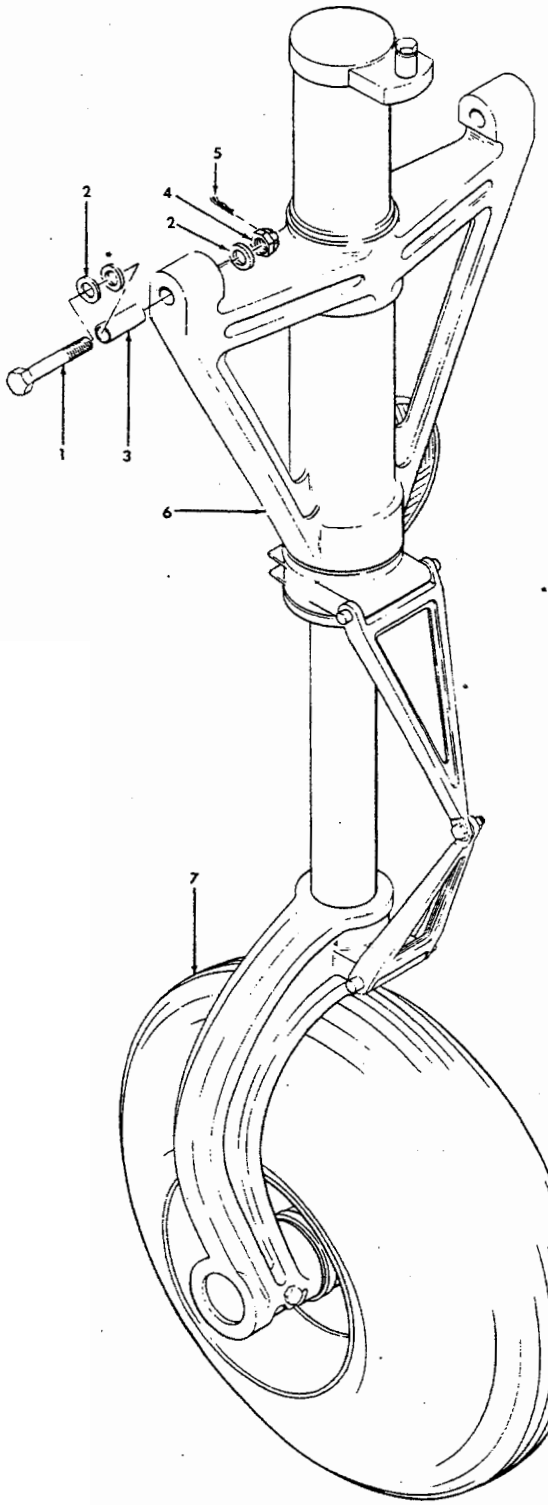
b. Adjust the shimmy dampener with the nose wheel in the extreme left position. The piston travel of the dampener should be adjusted to extend 1/16 to 1/4 inch beyond the attach point on the steering yoke.

c. Return the nose wheel to neutral and install the bolt, nut and washer.

100-HOUR INSPECTION

LANDING GEAR STRUT - Inspect the strut and attaching components for cracks, security, condition, and leakage at the air filler valve and piston area.

STRUT FLUID LEVEL - Inspect the strut for proper inflation. If leakage is detected, deflate the strut and check the fluid level.



* USE 100951X016YX AND 100951X032YX WASHERS, AS REQUIRED, TO OBTAIN TOTAL END PLAY BETWEEN NOSE GEAR AND SUPPORTS OF .000 TO .016 INCH.

1. Bolt
2. Washer
3. Bushing
4. Nut
5. Cotter pin
6. Nose Landing Gear Assembly
7. Wheel and Tire Assembly

Nose Gear Installation
Figure 201

"END"

cycling the gear listen for unusual noises at the motor and actuator. Cycle the gear a minimum of six times.

k. Install the access covers, carpet and the front seat.

l. Install the access door on the bottom of the fuselage.

LANDING GEAR DYNAMIC BRAKE RELAY REMOVAL

a. Remove the cabin right front seat.

b. Remove the access cover on top and directly behind the forward carry through structure.

c. Disconnect the electrical wiring at the dynamic brake relay.

d. Remove the two attaching screws and the relay.

LANDING GEAR DYNAMIC BRAKE RELAY INSTALLATION

a. Install the dynamic brake relay with the two attaching screws.

b. Connect the electrical wiring to the relay.

c. Install the access cover and the right front seat.

d. Cycle the landing gear and check for proper braking action. The gear should stop almost instantly.

CAUTION

When landing gear is approaching its extreme of travel, intermittently actuate the landing gear relay circuit breaker to prevent damage to the wing and fuselage structure and the actuating rods, linkage and the landing gear components.

LANDING GEAR RIGGING (Figure 201)

NOTE

Read the entire rigging procedure before attempting to rig the landing gear system. Physically locate each item as you read the procedure through.

CAUTION

Do not reverse the landing gear direction of travel without bringing the gear to a complete stop. If the direction of travel is reversed using the landing gear selector switch, damage to the sector gear may result. Use the landing gear relay circuit breaker to start or stop the landing gear during rigging.

Battery voltage is not sufficient to properly cycle the landing gear during rigging. A 28.25 (±0.25) volt auxiliary power unit capable of maintaining the initial setting within 0.25 volt during the extension and retraction cycles is recommended.

CAUTION

Excessive operation of the landing gear motor without proper cooling may cause damage to the motor. Allow five minutes cooling time after each extension and retraction cycle.

Whenever the landing gear mechanism is removed or disconnected, the landing gear should be checked to see that the system is properly rigged.

CAUTION

After making an adjustment to the gear, operate the landing gear intermittently as the system nears the limits of the retraction/extension cycle to prevent damage due to overtravel.

Prior to jacking the airplane, ensure that an unbalanced condition does not exist. Fuel should be distributed evenly in both wings to prevent an unbalanced condition which could cause the airplane to be unstable while on jacks.

a. With the airplane on jacks, allow adequate floor clearance for wheels during retraction or extension.

CAUTION

Care should be taken to ensure that the main gear retract rods are not lengthened far enough to damage the inboard landing gear doors.

b. If it is determined that the entire landing gear is out of rig, take the following precautions to prevent damage to the gear and airplane: Lengthen the main and nose gear retract rods sufficiently to eliminate the danger of the main gear V-brace damaging the wing skin when the gear is retracted. Place the uplock block in the lower position.

c. Disconnect the nose wheel door linkage. Secure the nose wheel door out of the way with tape. Disconnect the main wheel outboard door linkage at the strut.

d. Screw the stop bolts (on the main gear V-brace assembly) in until four or five threads are showing.

NOTE

The actuators (used on serials P-577 and after) may be installed on eariler airplanes. The new actuator is identified by part number 60-810030-5 or -7 and white epoxy paint on the upper actuator arm and on the upper and lower actuator housing.

e. Retract the landing gear to its 2/3 up position, then stop and inch the gear the remaining distance to the uplimit switch by intermittent operation of the landing gear relay circuit breaker. Check the emergency handcrank for 1/8 to 1/4 turn (or 5/8 to 3/4 turn on airplanes using 60-810030-5 or 60-810030-7 white actuator) free counterclockwise movement before the sector gear contacts the gearbox internal stop. If proper internal clearance is not obtained, adjust the landing gear uplimit switch. Locate the landing gear uplimit switch adjustment screw on the actuator arm and adjust the screw to stop the landing gear on its up cycle at the point where proper internal clearance is maintained.

CAUTION

The gearbox may be damaged if allowed to run full cycle into the internal stops. To preclude this possibility, the limit switches should initially be adjusted aft, for an early shutdown, if an out-of-adjustment condition is suspected.

NOTE

All adjustment and rigging of limit switches should be accomplished using an external power source adjusted to 28.25 (±0.25) volts. Due to overtravel, the landing gear will not stop at the same position that the limit switches are actuated.

On serials P-139 and after, there are two switch assemblies, one on each side of the actuator.

f. Extend the landing gear to its 2/3 down positic then stop and inch the gear the remaining distance to t downlimit switch by intermittent operation of the landi gear relay circuit breaker. Check the emergency handcra for 1/8 to 1/4 turn (or 5/8 to 3/4 turn on airplanes using 6 810030-5 or 60-810030-7 white actuators) free clockwi movement before the sector gear contacts the gearb internal stop. If proper internal clearance is not obtaine locate the landing gear downlimit switch adjustment scr on the upper arm of the landing gear actuator and adjust t screw to stop the landing gear on its down cycle at the po where proper internal clearance is achieved.

WARNING

Do not operate the handcrank with the power on.

g. Extend and retract the gear two or three times assure that the switches are correctly set. Check t handcrank each time to assure a free movement of 1/8 1/4 turn (or 5/8 to 3/4 turn on airplanes using 60-81003C or 60-810030-7 white actuators) before the sector ge contacts the internal stops.

h. Adjust both the right and left main retract rods maintain a minimum clearance of 0.12 inch between t knee joint of the V-brace and wing skin with the landing ge retracted. To determine V-brace and wing skin clearan retract the landing gear and slide a 0.12-inch feeler gau between the landing gear knee joint V-brace lift leg and t top wing skin. The main gear should retract only far enou to clear the inboard door (see Step k) in addition maintaining the minimum 0.12-inch clearance. To decrea the clearance between knee and top wing skin, short retract rod.

i. To provide an attachment point for a spring sca when rigging the door tension, fabricate a hook which c be screwed into the hole provided in the inboard gear do

j. With the gear retracted, rig the inboard landi gear door linkage so that a force of 28 to 40 pounds required to deflect the forward outboard corner of the dc 1/8 inch. With the gear down, 25 to 40 pounds should required to deflect the forward outboard corner 1/8 inch. T doors are adjusted by disconnecting the linkage rods at t clevis fitting and screwing the rods in or out to vary th length.

k. There shall be a clearance of 0.19 to 0.30 in between the main landing gear axle and the inside surfa of the inboard door, at its closest point, with the gear fu retracted.

NOTE

To measure the clearance between the main landing gear axle and the inside surface of the inboard door, place a 1/2-inch thickness or more of artists' clay or equivalent on the axle. Place one thickness of paper over the clay to prevent the clay from sticking to the door. Retract the landing gear and leave in fully retracted position long enough for the clay to remain in the flattened position, then extend the gear. Remove the paper and measure the depth of the clay. The depth should be between 0.19 to 0.30 inch.

CAUTION

Install the main landing gear door pushrod attaching bolt in the door linkage bracket with the head to rear. If installed wrong, the bolt may catch in the fuselage skin and root rib of the wing, causing damage to the landing gear retract mechanism or preventing the gear from retracting.

l. When the proper setting is obtained, leave the gears in the fully retracted position and screw the stopbolt down against the strut. To assure a firm setting, insert a 0.003-inch feeler gage under the bolt head and adjust the bolt until a firm, steady effort is required to pull the feeler gage out. With the feeler gage removed, screw the bolt 3/4 turn (counterclockwise, when viewing bolt head) from initial contact with no clearance. Tighten the locknut securely.

m. Fabricate a spring feeler gage from a piece of metal 0.002 inch thick by 1/2 inch wide by 1-1/2 inches long, a rubber band, and a hook made of a heavier piece of metal (see Figure 202). The rubber band must stretch tight enough to pull the 0.002-inch material from the torque knee contact surface when sufficient force is applied.

NOTE

On those airplanes prior to P-183, except P-163 through P-166, P-177, and P-181 which have not incorporated Kit No. 60-8006-1S, accomplish Step "n". Airplanes P-163 through P-166, P-177, P-181, and P-183 and after and earlier airplanes which have incorporated Kit No. 60-8006-1S, accomplish Step "o".

n. With the main landing gear fully extended, insert a 0.002-inch feeler gage, as shown in Figure 202 in the knee contact surface of the gear lift leg. Apply a force as shown in Detail "F" of Figure 201 until the knee joint deflects just enough to allow movement of the feeler gage. The knee should deflect when a force of 65 to 75 pounds is applied. To increase the tension, add washers (Part Number 100951S063XP) between the inboard end of retract rod spring and the end of retract rod. To decrease tension, remove washers.

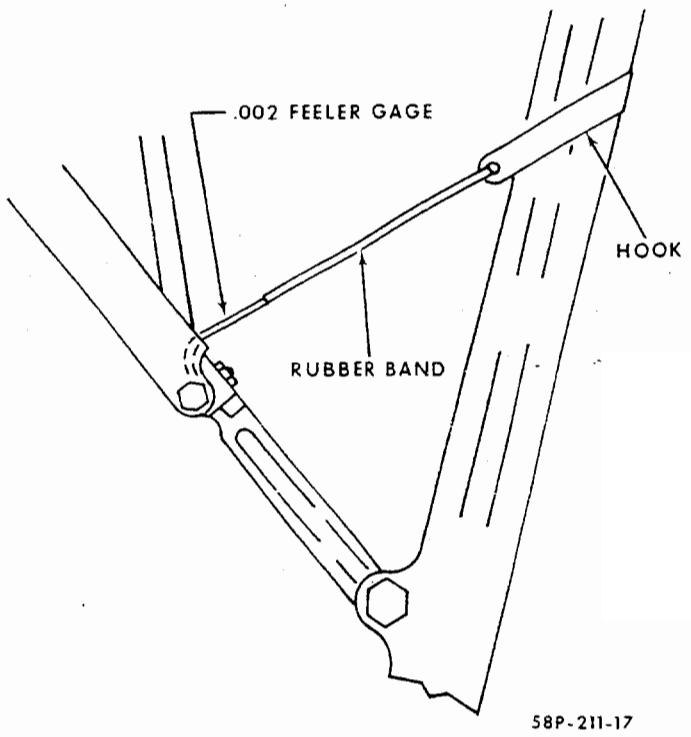
NOTE

If proper downlock tension cannot be established install a new spring. Check for a total minimum gap of 0.060 inch between the spring coils. The total spring gap is the sum of all gaps measured between the coils. Make certain the spring coils do not stack during gear operation.

o. With the main landing gear fully extended and the downlock disconnected, check the main gear retract rod end spring for minimum compression of 0.08 inch. Insert the spring feeler gage fabricated in Step "m", as shown in Figure 202, in the knee contact surface of the main gear lift leg. Apply a force, as shown in Detail "F" of Figure 201, using a push-pull scale. Apply the force until the torque knee contact surface of the lift leg releases the 0.002-inch spring feeler gage. The reading on the push-pull scale should be 55 to 65 pounds. To increase tension, add washer (Part Number 100951S063XP) as required, between the inboard end of retract rod spring and end of retract rod.

p. With the main landing gear fully extended and the downlock in the locked position, check the rollers for free movement of 0.01 to 0.02 inch between the rollers and downlock block. If this clearance is not obtained, loosen the block retaining bolts and adjust to proper clearance. Torque the bolts to 90 to 100 inch-pounds. With the main landing gear fully retracted and the uplock in the locked position, repeat this procedure to check and adjust the clearance between the uplock block and rollers as shown in Detail "E" of Figure 201.

q. With main landing gear fully extended, adjust the downlock cable to a tension of 52-1/2 (+10 −0) pounds using tensiometer in the wheel well. Tension is adjusted at the outboard end of the cable. If sufficient adjustment is not obtained at the cable eye, additional adjustment may be made at No. 3 wing rib by moving the cable housing inboard or outboard.



Main Gear Deflection
Figure 202

NOTE

Clean the clamp threads (bolt and nut).

Apply a light coat of thread locking compound (36, Chart 207, 91-00-00) on the cable clamp threads. Immediately after applying thread locking compound, install downlock cable and cable clamp on uplock cable. The cable clamp must maintain a clearance of 0.10 to 0.15 inch between the cable clamp and cable pulley. Torque cable clamp assembly to 80 to 90 inch-pounds. After five to seven minutes, retorque cable clamp assembly to 80 to 90 inch-pounds.

s. With the landing gear fully retracted, the uplock in the locked position and the uplock block adjusted as in step "p" there shall be a minimum of 0.11 inch overlap of the uplock stop angle with the lift leg of the brace assembly (Figure 203).

t. With the main landing gear fully retracted and the uplock in the locked position, apply a force of 58 to 68 pounds to the uplock assembly as shown in Figure 204. Using a feeler gage, check that the uplock stop angle deflects a maximum of 0.002 inch away from the landing gear lift leg.

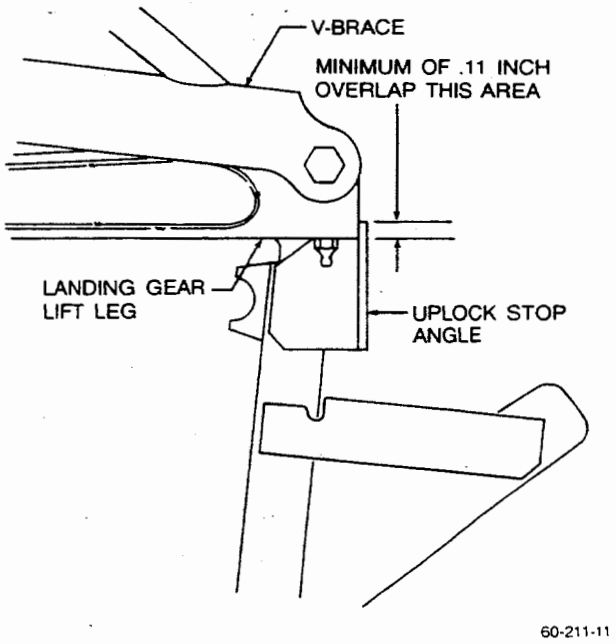
CAUTION

The uplock cable attach bolt must be installed with the bolt head aft. The bolt must be installed in this position to avoid interference between the bolt and the 49% stringer when the gear is fully retracted.

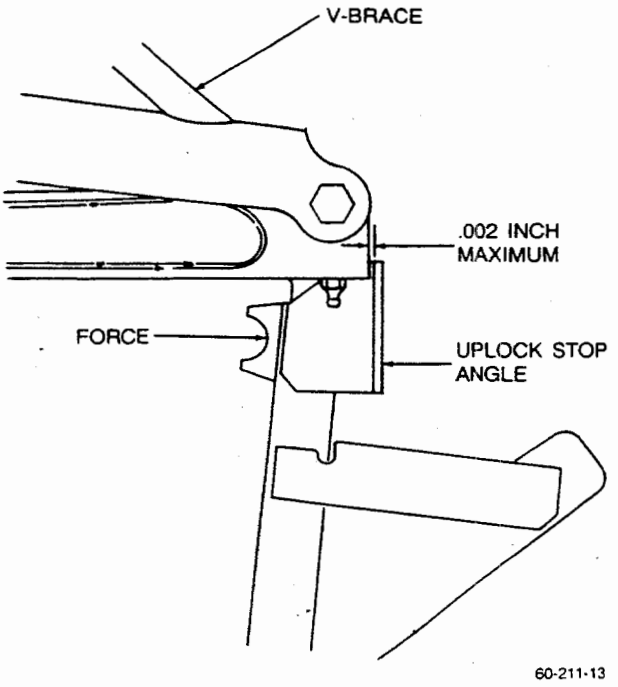
u. Extend the gear to approximately the 3/4 dow position and check for a minimum clearance of 0.50 inc between the tire and the inboard door at its closest poin while holding the door toward the tire.

v. Connect the outboard main gear door linkage an retract the gear slowly, checking to see that clearance i maintained between the door and gear. After checking t see that the door is not too tight, run the gear down an adjust the linkage as required. Continue this procedure unt a snug, firm fit is obtained when the door is completel closed.

w. Swivel the nose wheel to check the turning radiu of the strut for freedom of movement without binding c rough spots. Check the nose gear steering roller fc freedom of movement and the nose gear steering roller bo for security. Extend the gear to the full down position an



Uplock Stop Overlap
Figure 203



Uplock Deflection
Figure 204

make a preliminary adjustment on the down spring retaining nut located on the forward retract link to obtain a down spring length of 4.59 ± .030 inches (measure the forward spring retaining cap and the spring).

NOTE

Maintain a minimum spring length dimension of 4.56 inches (measure the forward retaining cap and the spring). Check that the spring does not stack or bottom out. Check to ascertain that the down spring is compressed a minimum of 1/8 inch when the gear is extended. Check that the bolts; or pins (serials P-153, P-155, P-163 through P-166, P-170 and after, and those serials P-4 through P-152, P-154, P-156 through P-162, P-167 through P-169 which have complied with Service Instructions No. 0460-212), through the slide tube are adjusted properly to allow the sleeve to slide on the tube.

CAUTION

If the nose gear down and lock light does not come on at landing gear extension at high air speeds, the length of the nose gear down spring should be readjusted to maintain a minimum compressed spring length of 4.59 ± .030 inches (measure the forward spring retaining cap and the spring). If unable to maintain proper down tension, the spring should be replaced.

x. Remove the access plate on the bottom side of the baggage compartment floor and disconnect the nose gear door linkage. Shorten the uplock linkage so the uplock will not interfere with nose gear stowage checks. Run the gear up slowly and check for clearance of the upper drag brace assembly to the baggage floor structure. Check the tire for proper clearance in the gear "up" position. The tire stows 1.0 inch above the closed gear door position. Adjust the front rod end on the forward retract rod as required to increase or decrease the tire clearance in the stowed position. Check for positive engagement of the uplock. Adjust the uplock for proper engagement. Extend the nose gear full down and check for compression of the forward retract rod plunger. Cycle the gear several times to ensure proper position. Check the down tension as follows:

1. Extend the gear to the full down position.
2. Move the lift leg joint just enough to insert a .002 inch feeler gage as shown in Figure 205 and return the gear to the full down position.

3. Apply a tension force downward at a 90° angle to the lift leg joint until the knee joint deflects just enough to allow movement of the feeler gage. The joint should deflect when a force of 60 ± 5 pounds is applied. If the down tension is not 60 ± 5 pounds, readjust the large spring retaining nut to obtain the correct down tension.

NOTE

Maintain a minimum spring length of 4.56 inches (measure the forward spring retaining cap and the spring).

y. Check that a clearance of .90 inch has been maintained between the baggage compartment floor and the nose gear in the retracted position.

NOTE

A one inch thickness of artist clay may be used to check this dimension (see the NOTE following step "k").

z. Connect the nose gear door linkage. On airplanes prior to P-281, adjust the nose gear door linkage rods to maintain a slight tension on the doors in the retracted position. On airplane serials P-281 and after, a force of 10 to 16 pounds should be required to deflect the forward, inboard corner of each door .125 inch.

NOTE

To provide an attachment point for a spring scale, fabricate a hook which can be screwed into the holes provided in the nose gear doors, as shown in Detail "I" of Figure 201. These holes are located at the inboard end of the forward hinges.

aa. Check that the nose gear tire clears the right door by a minimum of .50 inch at its closest point.

NOTE

If the tire does not clear the RH gear door by a minimum of .50 inch, the center retract rod may be shortened to increase the clearance. However, a minimum of .20 inch clearance must be maintained between the forward retract rod arm and the baggage floor when the nose landing gear is full "up". This dimension may be checked using clay.

Replace the access plate on the bottom side of the baggage compartment. With the nose gear system rigged, a final check should be made of the main gear down tension, the uplock spring tension, door tension, and the up stowage clearance.

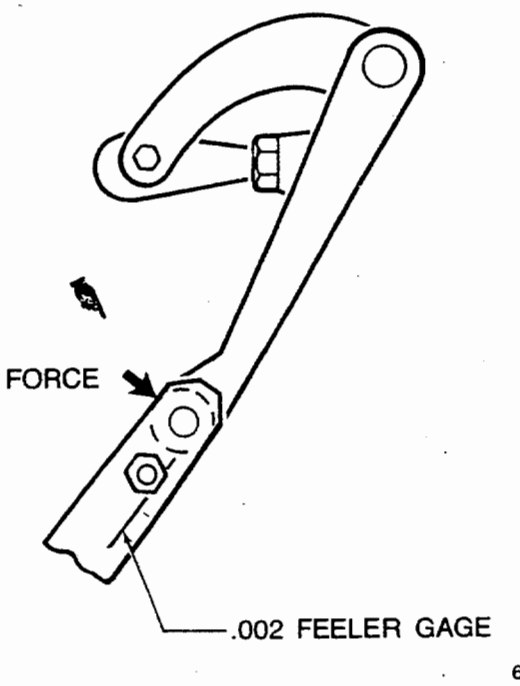
ab. Remove the wheel well access cover on the nose baggage compartment floor. With the landing gear retracted, adjust the uplock linkage until the uplock hook assembly has full engagement with the pin on the nose gear drag brace without deflecting the spring. After adjustment, extend the gear and safety the uplock link.

ac. With the gears down, check the adjustment of the safety switch. Refer to the Landing Gear Illustration, Figure 201, while adjusting the switch.

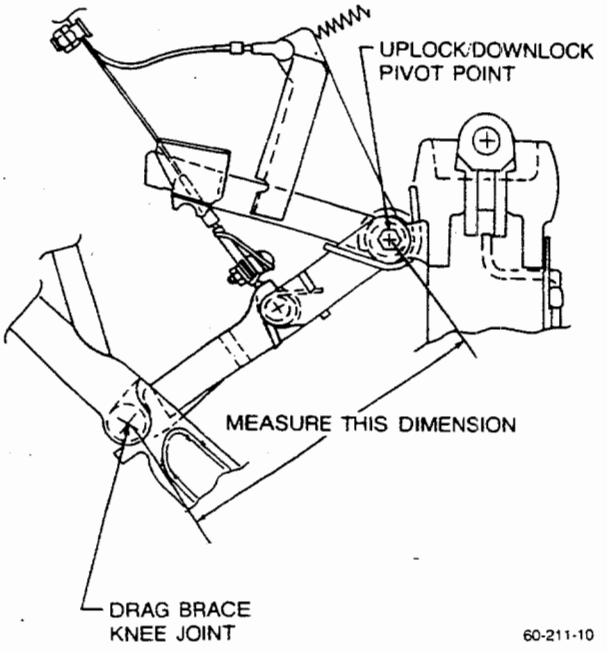
1. Remove the safety switch actuator rod (1) from the attaching bracket on the upper torque knee, then remove the retaining nut (2) and switch arm (3) from the switch shaft.
2. Jack the landing gear so the shock strut is compressed to .50 ± .12 inch from the extended position.
3. Connect the wire leads from a test light to pins inserted into the splices at wires number 1 and 3 (about 10 inches up from the safety switch).
4. Rotate the switch shaft clockwise until the test light comes on.
5. Remove the safety wire from the locking screw (4) on the switch arm (3) and back off the locking screw (4).
6. Install the switch arm (3) on the switch shaft in a position parallel to the upper torque knee and adjust the actuating rod end (1) to align with the attaching bracket on the torque knee. Install the actuating rod connecting bolt.
7. Position the shock strut to .87 ± .12 inch from the fully extended position and adjust the switch shaft counterclockwise at the adjusting screw (5) until the light goes out. When satisfactory adjustment is obtained, tighten the locking screw (4) and the retaining nut (2).
8. Recheck the landing gear travel to both dimensions described above before safety wiring the locking screw (4) to the switch arm.

ad. The landing gear position lights on the instrument panel are operated by uplock and downlock switches on each gear.

Main Gear: With the main landing gear fully extended, measure the distance between the uplock/downlock pivot



Nose Gear Deflection
Figure 205



Main Gear Downlock Switch Adjustment
Figure 206

points and the lift leg knee joint on the main landing gear as shown in Figure 206. Adjust the downlock switch (outboard side of gear) to actuate the in-transit light when the lift leg knee joint overcenter dimension is reduced by .15 ± .10 inch. With the landing gear fully retracted, adjust the uplock switches (inboard side of gear) to actuate the in-transit light when the actuator lacks five handcrank revolutions of being hard against the internal stop of the actuator. Adjust the switch as follows:

1. Handcrank the actuator until the sector gear contacts the internal stop.
2. Back the sector gear off the internal stop, by five revolutions of the handcrank.
3. Adjust the in-transit light switch to actuate in this position.

Nose Gear: With the nose gear fully extended adjust the downlock switch, located on the nose gear drag brace, so that the overtravel of the switch plunger is .050 to .125 inch after the actuation of the switch. Remove the wheel well access cover on the nose baggage compartment floor to gain access to the uplock switch. With the nose gear fully retracted, adjust the uplock switch (adjacent to the uplock hook) to actuate when the landing gear actuator lacks seven handcrank turns of being hard against the internal stop of the actuator. Adjust the switch as follows:

1. Handcrank the actuator until the sector gear is against the internal stop.
2. Back the sector gear off of the internal stop, by seven revolutions of the handcrank.
3. Adjust the uplock switch to actuate at this position.

ae. Recheck the limit switch adjustment and remove the airplane from the jack.

100-HOUR INSPECTION

RETRACT MECHANISM - Check the retraction system for proper operation of all components through at least two complete cycles. Check for unusual noises and evidence of binding.

DOORS AND LINKAGE - Check door operation, fit, rigging, and security.

POSITION INDICATORS - Check for security and adjustment of switches; wiring for breaks, condition of insulation and loose connections; indicators for proper indication.

WARNING HORN - Check for proper operation.

DOWNLOCKS - Check the main and nose gear deflection, the downlock switch for security and adjustment.

SAFETY SWITCH - Check for security and proper operation.

ACTUATOR - Check for unusual noises, binding, proper rigging.

LIMIT SWITCHES - Check for security and proper adjustment.

EMERGENCY EXTENSION - Check the system for freedom of operation and positive engagement of the downlocks.

"END"

WHEELS AND BRAKES - DESCRIPTION AND OPERATION

MAIN WHEEL ASSEMBLIES

The Duke is supplied with either Goodrich or Goodyear 6.50 x 8 wheel assemblies. (See TIRES for sizes, types and serial efeectivity.)

The wheel consists of an inner and outer magnesium wheel half held together with bolts, washers and nuts. The washers are used beneath the nuts and bolt heads to prevent galling and stress concentration.

A packing in the groove of the wheel halves provides a seal against air pressure. Bearing cups, cone bearings and seals are installed in the hub area. The Goodrich wheel assemblies are provided with balance weights; while both halves of the Goodyear wheel assembly are dynamically balanced. Identification and instruction plates are attached to both wheel halves.

Screws and safety wire retain sixteen torque keys in the slots of the flange area of the Goodrich inner wheel half; while fourteen keyway liners are riveted in place on the Goodyear inner wheel half. The torque keys/keyway liners retain the lugs of the brake disc which rotate with the wheel when the brake and wheel are mounted on the landing gear axle.

The wheel assemblies are secured to the axles with bushings, washers, nuts and cotter pins.

NOSE WHEEL ASSEMBLY

The Duke is supplied with either a Goodrich or Goodyear 6.00 x 6 wheel assembly mounting a 15.00 x 6.00 x 6, 4 ply, type VI, tube type tire. The wheel consists of an inner and outer wheel half. Goodrich wheels are manufactured of aluminum alloy; while Goodyear wheels are of magnesium. Both halves are held together with bolts, washers and nuts. Washers are used beneath the nuts and bolt heads to prevent galling and stress concentration.

Bearing cups, cone bearings and seals are installed in the hub area. Identification and instruction plates are installed on each Goodrich wheel half; while adhesive data and warning plates are installed on the outer Goodyear wheel half.

The wheel assembly is secured to the axle with a bushing, washer, nut and cotter pin.

TIRES

The main wheel tires supplied on the Duke, serials P-4 through P-189 and P-191, were 6.50 x 8, 8 ply tubeless, rim inflated tires. Later serials P-190, P-192 and after, and the earlier serials which have complied with Service Instructions No. 0536-203, are supplied with 19.50 x 6.75 x 8, 10 ply tube type tires.

The nose wheel tire supplied on the Duke is a 15.00 x 6.00 x 6, 4 ply, type VI, tube type tire.

BRAKE ASSEMBLY

Goodyear wheel and brake assemblies were supplied on serials P-4 through P-168. The later serials, P-169 and after, are supplied with Goodrich wheel and brake assemblies.

Both brake assemblies are designed for use with MIL-H-5606 hydraulic fluid (13, Chart 207, 91-00-00) to withstand 550 to 600 psi operational pressure with zero psi back pressure.

The Goodyear brake contains two rotating brake discs, which are keyed to rotate with the wheel, a stationary disc, pressure plate and back plate which are attached to the brake housing. Braking action occurs when hydraulic pressure is applied to the five small pistsons in the brake housing which force the disc stack together, creating friction between the rotating discs and the stationary parts. The pistons are sealed against leakage with packings. Indicator pins are used to determine brake wear. The brake assemblies are interchangeable between right and left by changing the location of the bleeder adapter.

The Goodrich brake contains one rotating brake disc, which is keyed to rotate with the wheel, two carrier linings, a torque plate and three piston insulators which are attached to the piston housing. Braking action occurs when hydraulic pressure is applied to the three small pistons in the piston housing which force the carrier linings together, creating friction between the rotating disc and stationary parts. The pistons are sealed against leakage with preformed packings. The brake assemblies are interchangeable between right and left by changing the location of the bleeder adapter.

HYDRAULIC BRAKE SYSTEM

Dual braking (pilot and copilot) is standard on serials P-4 through P-45. On serials P-46 and after, dual brakes are optional. The following description and operation discusses the dual braking system.

The dual hydraulic brakes are operated by either depressing the pilots or copilots rudder pedal. A shuttle valve adjacent to each set of pedals permits changing of braking action from one set of pedals to the other. The depression of either set of pedals compresses the piston rod of the attached master cylinder. The hydraulic pressure resulting from the movement of the pistons in the master cylinder is transmitted through flexible hoses and fixed aluminum tubing to the brake disc assemblies mounted on each main landing gear. This pressure forces the brake pistons to press against the linings and disc(s) of the brake assembly. Upon release of pressure against the pistons, the brake disc(s) will

have a tendency to drag against the stationary liners.

Dual parking brake valves are installed adjacent to the rudder pedals between the master cylinders of the pilot's rudder pedals and the wheel brake assemblies. After the pilot's 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. This closes the valves and retains the pressure in the brake lines. The parking brake is released when the parking brake handle is pushed in and the pedals are depressed briefly to equalize the pressure on both sides of the valves, allowing them to open.

"END"

TROUBLESHOOTING
BRAKE SYSTEM

	TROUBLE		*PROBABLE CAUSE*		*REMARKS*
1.	Solid pedal and no brakes.	a.	Brake lining worn beyond allowable limit.	a.	Replace lining.
2.	Spongy brake.	a.	Air in system.	a.	Bleed brake 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.	a.	Bleed brake system.
		b.	Defective parking brake valve.	b.	Replace the valve.
5.	Brakes grab.	a.	Stones or foreign matter locking brake disc.	a.	Clean brake disc and lining.
		b.	Warped or bent disc.	b.	Replace disc.

"END"

WHEELS AND BRAKES - MAINTENANCE PRACTICES

MAIN WHEEL AND TIRE

The wheel and tire assembly is shipped from the factory completely assembled. The bearings are packed with the proper grease and may be installed as received.

NOTE

Extended storage of the lubricated bearings may require relubrication. If this is necessary, refer to the applicable Wheel and Brake Manual (P/N 98-32751 or 98-34998) for instructions.

Install the main wheel and tire assembly on the axle as follows:

a. Visually check the nut and axle for burrs and rough threads.

b. Apply Aeroshell 5 grease (38, Chart 207, 91-00-00) to the threads and all bearing surfaces.

c. Install the wheel and tire assembly with bushing, washer and nut.

NOTE

Make sure that the brake disc lugs engage the slots of the inner wheel half. Rotate the wheel while adjusting the axle nut to assure proper seating and check to see that there is no side motion.

d. While rotating the wheel, torque the axle nut to 250-300 inch-pounds to ensure that the bearings are properly seated, then back off the axle nut to zero torque.

e. While rotating the wheel, retighten the axle nut to 125-145 inch-pounds of torque in one continuous rotation before checking for alignment of the locking holes and installing the cotter pin.

f. If necessary, continue tightening the nut to the next available cotter pin position and install a new cotter pin.

MAIN WHEEL INSPECTION

On the fifth tire change after a new wheel has been installed, then on each third subsequent tire change up to a total of 20 tire changes and on each tire change thereafter, the wheel and tire should be removed from the airplane, disassembled, and inspected in accordance with the applicable vendor wheel and brake manual. Bolts should be magnafluxed for cracks or breaks, and the wheel halves should be dye-checked or fluorescent penetrant inspected.

The following inspections may be accomplished with the wheel on the airplane:

a. Inspect wheel half flanges for cracks and corrosion. Smooth minor abrasions, nicks and burrs with a fine file or emery cloth, removing as little metal as possible. Chemically treat and coat the surfaces as instructed in the applicable Wheel and Brake Manual.

b. Check for loose or missing bolts and nuts. Retighten or replace as necessary.

CAUTION

Wheels with loose or missing bolts should be removed from the airplane and fluorescent inspected for cracks.

c. Inspect for excessively worn or loose torque keys or keyway liners.

NOSE WHEEL AND TIRE

The wheel and tire assembly is shipped from the factory completely assembled. The bearings are packed with the proper grease and may be installed as received.

NOTE

Extended storage of the lubricated bearings may require relubrication. If this is necessary, refer to the applicable Wheel and Brake Manual (P/N 98-32751 or 98-34998) for instructions.

Install the nose wheel and tire assembly on the axle as follows:

a. Visually check the nut and axle for burrs and rough threads.

b. Apply Aeroshell 5 grease (38, Chart 207, 91-00-00) to the threads and bearing surfaces.

c. Install the wheel and tire assembly with bushing, washer and nut.

d. While rotating the wheel, tighten the axle nut to 250 to 300 inch-pounds torque to ensure that bearings are properly seated. Back off the nut to zero torque. While rotating the wheel, retighten the axle nut to 125 to 145 inch-pounds torque in one continuous rotation. Using a wrench, tighten the nut to the next cotter pin location and install a new cotter pin.

NOSE WHEEL INSPECTION

On the fifth tire change after a new wheel has been installed, then on each third subsequent tire change up to a total of 20 tire changes and on each tire change thereafter, the wheel and tire should be removed from the airplane, disassembled, and inspected in accordance with the applicable vendor Wheel and Brake Manual. Bolts should be magnafluxed for cracks or breaks, and the wheel halves should be dye-check or fluorescent penetrant inspected.

The following inspections may be accomplished with the wheel on the airplane:

a. Inspect wheel half flanges for cracks and corrosion. Smooth minor abrasions, nicks and burrs with a fine file or emery cloth, removing as little metal as possible. Chemically treat and coat the surface as instructed in the applicable Wheel and Brake Manual.

b. Check for loose or missing bolts and nuts. Retighten or replace as necessary.

CAUTION

Wheels with loose or missing bolts should be removed from the airplane and fluorescent penetrant inspected for cracks.

c. Inspect for excessively worn or loose torque keys or keyway liners.

TIRE VALVE - TUBELESS TIRES

To replace a valve core, use the standard extraction tool to unscrew the valve core. If the stem or its attaching parts require replacement, dismount and disassemble the wheel and tire in accordance with the instructions in the applicable Wheel and Brake Manual. Replace parts as necessary.

HYDRAULIC BRAKE SYSTEM

Brake system servicing is limited primarily to maintaining the correct fluid level in the reservoir. (Refer to Chapter 12-20-00.) The other requirement related to servicing involves inspecting the wheel brake assemblies for wear.

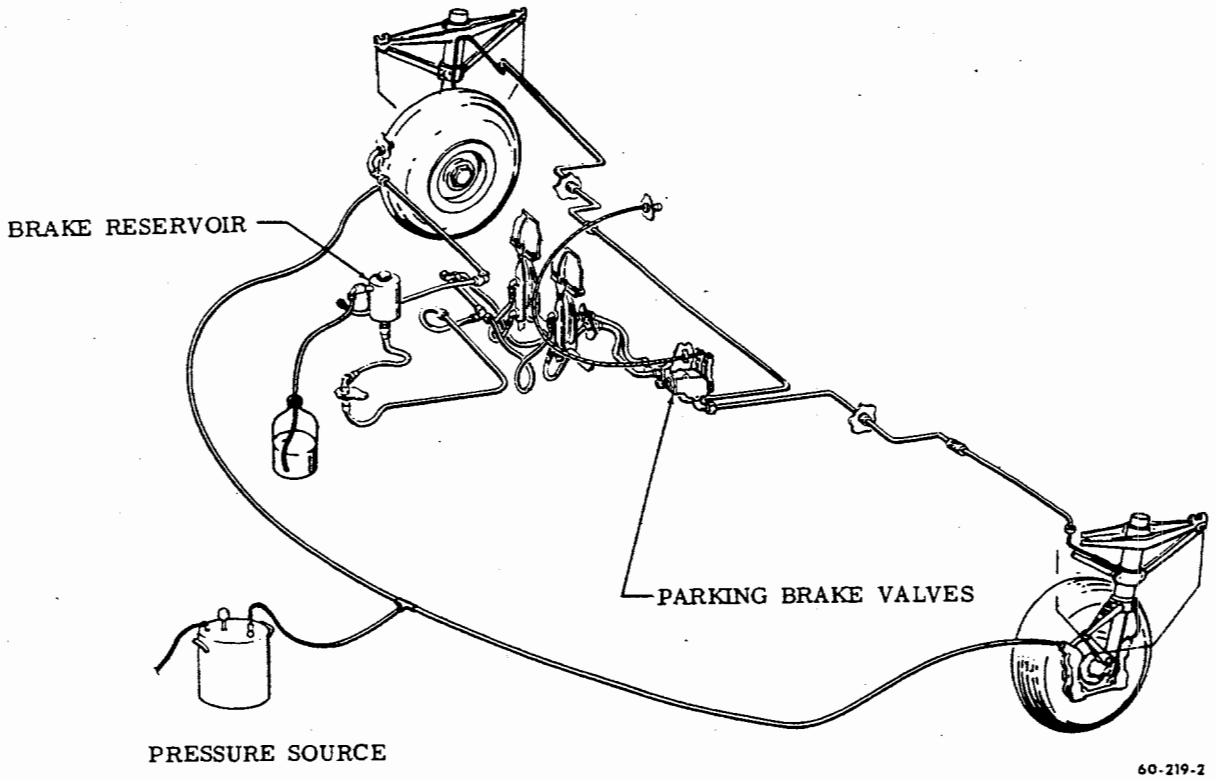
BRAKE SYSTEM BLEEDING

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 wherever the parking brakes will no longer hold. In the latter instance, the system should be further checked for leakage.

Use only MIL-H-5606 hydraulic fluid (13, Chart 207, 91-00-00) in the brake system, and ensure that no dirt or foreign matter is allowed to get into the brake system. Dirt can get under seals and cause leaks or clog the compensating ports in the master cylinders and cause the brakes to lock.

Beech Aircraft recommends the use of pressure pot brake bleeding. If the pressure pot bleeding method is not avaliable, electric bleeding is recommended. Use the gravity method only if the other two methods are not available. If the gravity system is used, pressure bleed the brakes at the earliest possible time. Using any method, the parking brake lever and toe brake pedals must both be fully released to open the compensating ports in the brake master cylinders.

If the brakes feel soft or "spongy" after the bleeding operation, air may be trapped in the cylinders. Remove the brake and lay



Pressure Bleeding Brake System
Figure 201

it on its side. Add brake fluid as needed through the bleed port and tap the brake lightly with a rubber hammer to dislodge any air bubbles. When air bubbles no longer appear at the port, install the brake and repeat the bleeding procedure.

BRAKE SYSTEM PRESSURE BLEEDING

Pressure bleeding is the most efficient method of bleeding the brake system and is also the recommended method. This procedure involves attaching a pressure pot to the brake assembly bleeder ports and back bleeding the system to the fluid reservoir. Procedures for utilizing the preferred pressure pot, the electric bleeder and the gravity bleed method are outlined below.

Brake System Bleeding Using the Pressure Pot Method (Figure 201)

NOTE

The line hook-up for pressure pot bleeding is the same as shown in Figure 201 except the electric bleeder is replaced with a pressure pot.

a. Disconnect the pressure equalization line from the reservoir and attach an extension line approximately three feet in length.

b. Place the end of the extension line in a clean receptacle to collect the brake fluid overflow.

c. Cut the safety wire and remove the screws from the bleeder ports of each brake assembly. Install a bleeder hose adapter into each brake bleeder port.

Fabricate a bleeder hose assembly for each set of brakes; connect the bleeder hose assemblies between the bleeder hose adapters and the pressure lines of the pressure pot.

d. Apply a constant pressure of approximately 15 pounds to the pressure pot. Open the pressure pot control valve.

e. Bleed the system until the draining fluid is free of air bubbles.

f. Close the pressure pot valve. Remove the bleeder hose adapters and hose assemblies from each landing gear. Install the screws into the bleeder ports of each brake assembly and safety wire.

g. Remove the extension line from the pressure equalization port on the reservoir.

h. Connect the pressure equalization line to the reservoir.

i. Remove the cap from the hydraulic fluid reservoir and add MIL-H-5606 hydraulic fluid (13, Chart 207, 91-00-00) as required to obtain a full reading.

j. Check the operation of the brakes. There should be no soft or spongy feeling at the brake pedals and the pedal pressure should be equal on both brakes.

Brake System Bleeding Using the Electric Bleeder Method

a. Disconnect the pressure equalization line from the reservoir and attach the electric bleeder fluid return line to the reservoir.

b. Cut the safety wire and remove the screws from the bleeder ports of each brake assembly. Install a bleeder hose adapter into each brake bleeder port. Fabricate a bleeder hose assembly for each set of brakes; connect the bleeder hose assemblies between the bleeder hose adapters and the pressure lines of the pressure pot.

c. Activate the bleeder and set the relief valve to approximately 15 pounds; this may be ascertained by observing the pressure gage prior to opening the elec tric bleeder control valve.

d. Open the electric bleeder contro valve and observe the returning flui through the inline sight glass. Pumpir the pilot's and copilot's pedals durin the bleeding process may help to dislodg any air bubbles trapped in the maste cylinders.

e. When the returning fluid shows n further evidence of air bubbles, close th electric bleeder control valve.

f. Disconnect the fluid infusio lines from the bleeder hose assemblies an remove the bleeder hose assemblies an adapters from the brake assemblies Install the screws into the bleeder port of each brake assembly and safety wire.

g. Disconnect the fluid return lin from the brake fluid reservoir and recon nect the pressure equalization line.

h. Check the brake reservoir flui level and add MIL-H-5606 hydraulic flui (13, Chart 207, 91-00-00) as required t obtain a full reading.

i. Check the operation of the brakes When the brake pedals are depressed ther should be no spongy feeling and the peda pressure should be equal on both brakes.

Brake System Bleeding Using th Gravity Bleeding Method

This method of bleeding is done from th master cylinder down to the brake assem bly. The brake fluid reservoir must b kept full during the bleeding operatior Since the pilot's and copilot's maste cylinders are plumbed in series, th entire system may be bled by operating th pilot's brake pedals in the followir manner:

a. Cut the safety wire and open th bleeder port screws of both brake assem blies on one landing gear.

b. Depress the pilot's correspondir brake pedal slowly and smoothly to elim nate air trapped in the system.

c. Hold the brake pedal in the depressed position and close the bleeder port screws at the brake assemblies.

d. Release the brake pedal.

e. Repeat steps "a, b, c, and d" until no more air bubbles appear in the drained fluid.

f. Open the bleeder port screws of both brake assemblies on the other landing gear and repeat steps "a, b, c, and d", depressing the other brake pedal until no more air bubbles appear in the drained fluid.

g. Tighten the bleeder port screws at all four brake assemblies and safety wire.

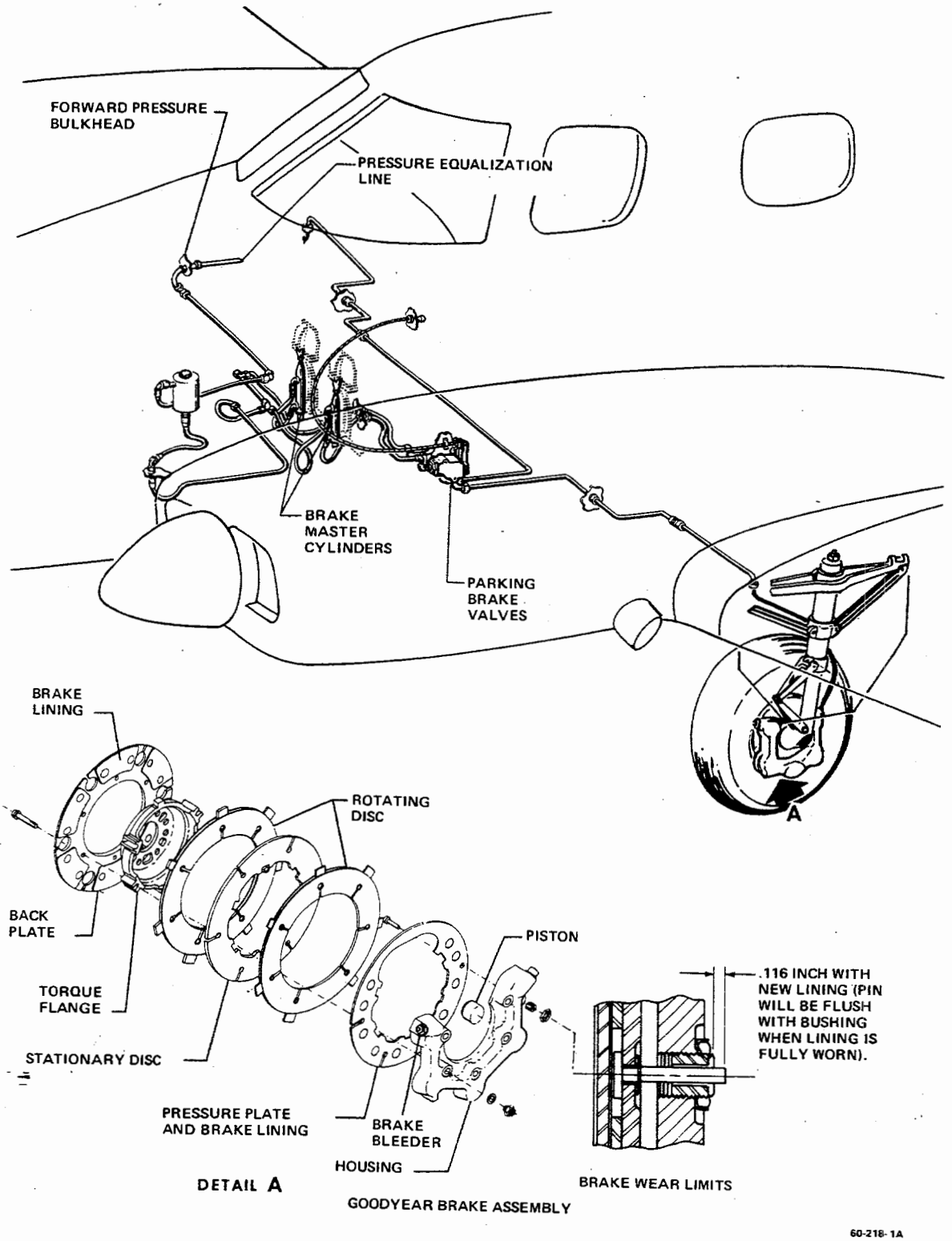
h. Check the brake reservoir fluid level and add MIL-H-5606 hydraulic fluid (13, Chart 207, 91-00-00) as required to obtain a full reading.

i. Check the brakes for proper operation. When the brake pedals are depressed there should be no spongy feeling and the pedal pressure should be equal for both brakes.

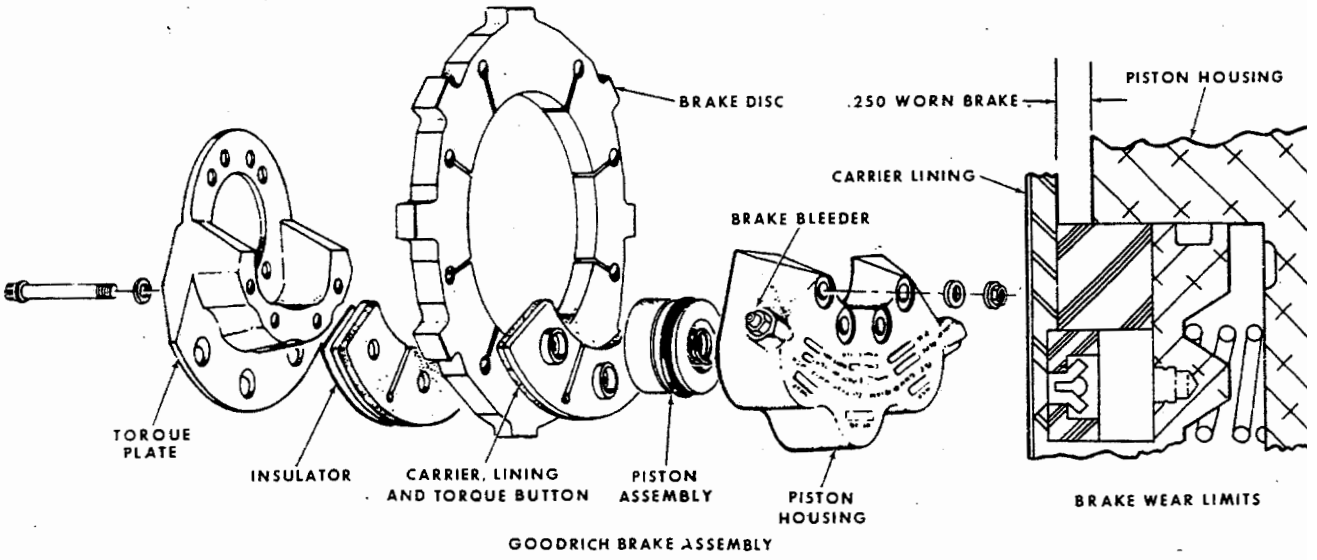
DUAL BRAKE SYSTEM BLEEDING

In airplanes having the optional dual brake system, the copilot's brake system is bled by closing the valve on the pressure pot and pumping the copilot's brake pedals to change the shuttle valve position. This causes hydraulic fluid to be routed through the copilot's system and this sytem should be bled as was the pilot's system.

After the pilot's and copilot's brakes have been bled, close the bleeder valve and repeat for the other wheel.



Brake System (Sheet 1 of 2)
Figure 202



Brake System (Sheet 2 of 2)
Figure 202

BRAKE ASSEMBLY REMOVAL
(Figure 202)

a. Place the airplane on a jack.

b. Remove the cotter pin, wheel retaining nut, washer and bushing. Slide the wheel off the axle.

c. Disconnect the brake hydraulic line and cap the open line and port. Remove the bolts securing the brake housing to the landing gear torque flange. Slide the brake off the axle.

BRAKE ASSEMBLY INSTALLATION

a. Slide the brake assembly onto the axle and install the bolts attaching the brake housing to the landing gear torque flange. Tighten the bolts to 340 to 360 inch-pounds of torque.

b. Connect the brake hydraulic line.

c. Clean and repack the wheel bearings with Aeroshell 5 grease (38, Chart 207, 91-00-00) and slide the wheel on the axle. Install the bushing, washer and wheel retaining nut.

d. While rotating the wheel, tight the axle nut to 250-300 inch-pounds torq to ensure that bearings are proper seated.

e. Back off the axle nut to ze torque.

f. While rotating the whee retighten the axle nut to 125 to 1 inch-pounds torque in one continuo rotation.

g. Using a wrench, tighten the n to the next available cotter pin positi and install a new cotter pin.

h. Bleed the brake system.

BRAKE WEAR LIMITS (GOODYEAR)

A brake wear indicator pin is attach to the pressure plate on each brake. T pin moves with the pressure plate as t brakes are applied. When the brakes a applied and the indicator pin is flu with its bushing, the lining has reach its wear limit. The indicator pin wi measure 0.116 inch above the top of t

its wear limit. The indicator pin will measure 0.116 inch above the top of the bushing when new brake linings are installed and the brakes are applied.

BRAKE WEAR LIMITS (GOODRICH)

If a measurement of .250 inch or more is noted between the piston housing and carrier lining, the brake assembly requires a lining inspection. If the carrier and lining thickness is under .180 inch, the brake assembly should be overhauled in accordance with the Wheel and Brake Manual (P/N 98-34998).

BRAKE MASTER CYLINDER REMOVAL

a. Close the parking brake valve by placing the control in the ON position.

b. Unsnap the floor mat and remove the floorboard section below the brake pedals.

c. Disconnect the two brake hydraulic lines at each master cylinder and mark the lines to assure correct reinstallation.

d. Remove the master cylinder attaching bolts and nuts and remove the master cylinder.

e. If new master cylinders are to installed, note the positions of the master cylinder 45-degree elbow fittings.

BRAKE MASTER CYLINDER INSTALLATION

a. Install the master cylinder with attaching bolts and nuts.

b. Connect the two brake hydraulic lines at each master cylinder as noted during removal.

c. Install the floorboard and the foor mat.

d. Open the parking brake valve by placing the control in the OFF position.

e. Service the brake reservoir and bleed the brake system.

BRAKE MASTER CYLINDER LINKAGE ADJUSTMENT

The proper linkage arrangement will adjust the brake pedals to a straight upright position. This is considered the best adjustment since it will prevent the pedals from hitting the bulkhead in their extreme forward position. Linkage adjustment is obtained by removing the clevis from the rudder pedal and turning the clevis on or off the piston rod as required. After both pistons are adjusted to the same length, tighten the jam nuts.

PARKING BRAKE VALVE REMOVAL

a. Bleed the brake system of all hydraulic fluid.

b. Remove the floorboards forward of the pilot's and copilot's seats.

c. Disconnect the parking brake cable from the parking brake valve by loosening the set screw and pulling the cable free of the cable attach fitting

d. Disconnect and cap the hydraulic lines from the parking brake valve.

e. Remove the attach bolts and remove the valve.

PARKING BRAKE VALVE INSTALLATION

a. Install the parking brake valve with the attaching bolts.

b. Connect the hydraulic lines to the valve.

c. Connect the parking brake cable to the valve by engaging the cable to the attach fitting ad tightening the set screw.

d. Install the floorboards forward of the pilot's and copilot's seats.

e. Service the hydraulic reservoir and bleed the brake system.

PARKING BRAKE ADJUSTMENT

a. Place the parking brake control in the off (valve open) position.

b. Remove the floorboards forward of the pilot's seats.

c. Loosen the set screw in the cable attach fitting and adjust the cable housing through the mounting block to obtain 1-1/2 inch travel between the cable housing and the cable attach fittings. The 1-1/2 inch clearance should be made with the parking brake valve lever in the open position.

d. Tighten the mounting block, insert the cable in the cable attach fitting, tighten and safety wire the set screw in the attach fitting.

e. Test the parking brake adjustment by pulling the parking brake handle out and operating the brake pedals.

f. If the brake pedals are not solid, place the parking brake control in the off position and recheck the rigging.

g. Inspect the parking brake valve for hydraulic fluid loss.

100-HOUR INSPECTION

WHEELS AND TIRES

a. Visually inspect wheels for cracks, nicks and general condition.

b. Check the wheels for loose or missing parts.

c. Inspect tires for breaks, blisters and excessive wear.

d. Check tires for proper inflation as instructed in Chapter 12-20-00.

NOTE

In service, the tire grows slightly due to shock loads during landing. Normally, this growth is balanced by tread wear so there is no increase in tire diameter.

NOTE

The use of recapped tires is not recommended by Beech Aircraft Corporation. The tires may pass the retraction test when first installed; however, recapped tires have a tendency to swell after use and may cause malfunction of the retract system or damage to the landing gear doors.

BRAKES

Check brake discs, linings and lines for wear, corrosion and security, brake housing, valves and lines for leakage.

"END"

STEERING - MAINTENANCE PRACTICES

Direct linkage to the rudder pedals allows the nose wheel to turn 15° each side of neutral position. Spring mechanisms in the linkage dampen excessive shock loads to the rudder pedals. When the landing gear is retracted, the nose wheel is automatically centered.

Friction of the nose wheel against the ground, while the airplane is static, inhibits the turning movement. Proper turning may be accomplished smoothly by allowing the airplane to roll and lightly depressing the appropriate rudder pedal.

ADJUSTMENT

a. With the airplane on jacks, turn the nose wheel as far to the left as the rudder pedal will permit. Check for a 15° travel from neutral position. If proper travel is not obtained, proceed as follows:

1. Disconnect the nose steering linkage at the nose steering bell crank.
2. Loosen the jam nut and thread the rod end in or out, as necessary, to obtain the proper travel.
3. Tighten the jam nut and connect the steering linkage to the bell crank.

b. Adjust the shimmy damper as follows:

1. Disconnect the shimmy damper at the steering yoke.
2. Turn the nose wheel as far to the left as the rudder will permit.
3. Adjust the shimmy damper piston to extend 1/16 to 1/4-inch beyond the attach point on the steering yoke.
4. Connect the shimmy damper to the steering yoke and tighten the jam nut on the piston rod.

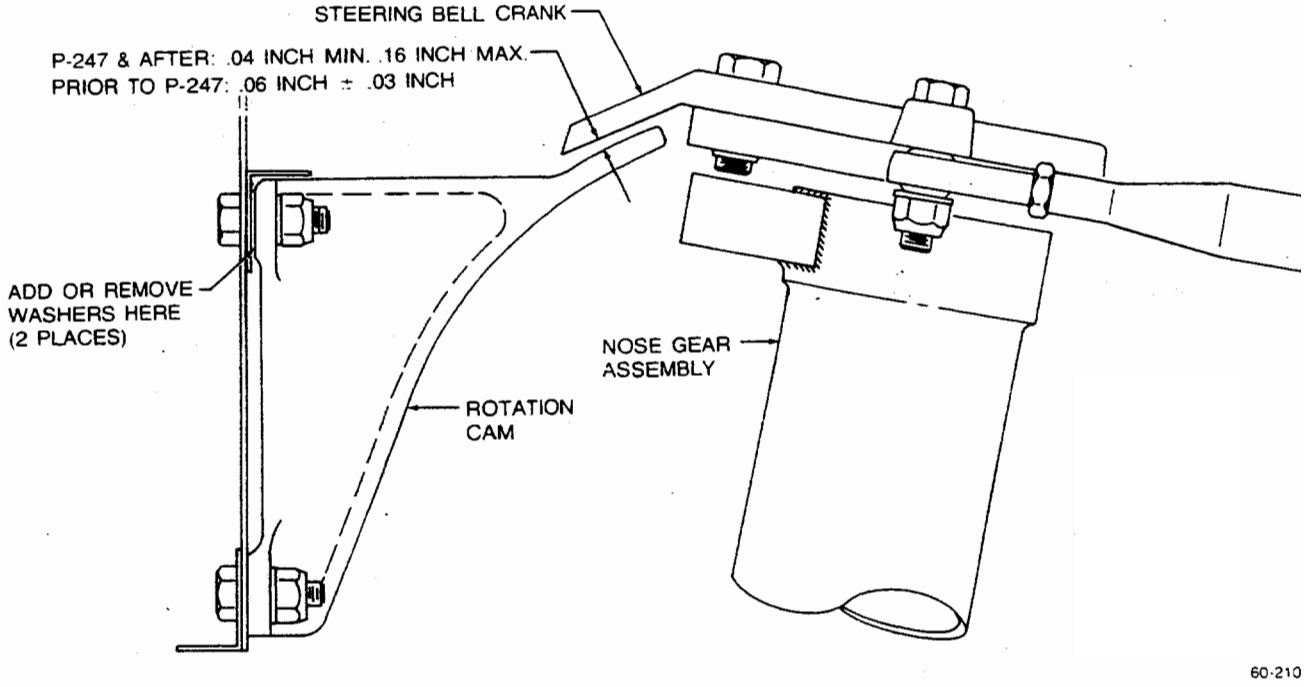
c. Adjust the steering bell crank clearance as follows:

1. Refer to Figure 201 and check the clearance between the bell crank and the rotation cam with the steering bell crank turned as far to the left and right (normal steering) as the rudder pedals will allow. Airplane serials prior to P-247 should have a clearance of .06 ± .03 inch. Airplane serials P-247 and after should have a clearance of .04 inch minimum and .16 inch maximum. Check the clearance with the bell crank in both the left and right positions.
2. If the clearance is other than the noted dimensions, AN960-416 or AN960-416L washers may be removed or added behind the rotation cam as required to maintain the correct clearance (refer to Figure 201). Up to a maximum of two AN960-416 and a maximum of four AN960-416L washers may be placed behind the rotation cam (2 places).

100-HOUR INSPECTION

STEERING LINKAGE - Check nose steering mechanism for condition, security and correct adjustment.

NOSE GEAR STEERING - Check the steering bell crank for cracks, condition, security and correct clearance.



Steering Bell Crank Clearance
Figure 201

"END"

POSITION AND WARNING - MAINTENANCE PRACTICES

SAFETY SYSTEM

No maintenance is required for the landing gear safety system, other than replacing defective units or checking the electrical wiring for condition, security of attachment, and tightness of electrical connections. The switches are preset and adjustment will not normally be required; however, should the system fail to function properly, the following checks and adjustments may be accomplished:

CHECK OF SYSTEM WITH SAFETY SWITCH IN TEST POSITION

a. Place the throttle in the closed or retarded position.

b. Place the battery master switch ON. The landing gear circuit breaker may be either in or out.

c. Place the landing gear safety system switch in the momentary full up (TEST) position. Noise or movement of the solenoid in the landing gear position switch indicates that the automatic landing gear extension part of the system is functioning properly. The on-off switch returns normally to the ON position unless the pilot intentionally places the switch in the OFF position.

MICROSWITCH ADJUSTMENT

The microswitch cannot be accurately adjusted on the ground. Before the microswitch is adjusted, it must be ascertained that the throttle warning horn switch is properly set. The microswitch may then be adjusted as follows.

a. With the airplane in flight, mark the throttle control at the control console when the manifold pressure gage registers approximately 17 ± 1 in. Hg.

b. With the airplane on the ground, move the throttle until the mark on the control is aligned with the control console as accomplished in step "a".

c. Adjust the microswitch until the cam clicks the switch closed with the throttle in the position indicated in the preceding step.

PRESSURE SWITCH ADJUSTMENT

The pressure switches are preset and will not normally require adjustment. Because of the built-in tolerance of these switches, they should not be tampered with unless the switch in question fails to actuate at an airspeed within 2 knots above or below the setting recommended for it. Even then the system plumbing and electrical wiring should be checked to ascertain that the source of trouble is not something other than improper adjustment of the pressure switches.

a. Place the airplane on jacks.

b. With the master switch ON, the landing gear circuit breaker in, and the landing gear warning circuit-breaker out, advance the throttle to its maximum position.

c. Place the landing gear safety position switch in the ON position.

d. Place the landing gear position switch in the UP position.

e. Clamp a section of soft rubber tubing over the pitot head inlet, making certain that the connection is airtight.

f. Crimp the end of the tubing and roll it up until the airspeed indicator registers 85 ± 2 knots. The landing gear will start retracting immediately if the pressure switch is properly adjusted.

CAUTION

To avoid rupturing the diaphragm of the airspeed indicator, the rubber tubing must be rolled SLOWLY.

g. If the landing gear failed to retract in the preceding step, turn the master switch OFF and adjust the pressure switch (outboard switch of the two installed in the left main wheel well) as follows:

1. Secure the rolled up tubing so that it will hold the airspeed indicator reading at 85 ± 2 knots.

2. Connect a continuity tester across the contacts of the pressure switch, then turn the adjustment screw until the switch closes at 85 ± 2 knots on the airspeed indicator.

h. Turn the master switch ON and roll up the rubber tubing until the airspeed indicator registers 130 kts. (to allow the switch to reset), then secure the tubing so that the airspeed indicator will hold that reading.

i. Retard the throttle.

j. Slowly bleed off pressure, until the airspeed indicator registers 120 ± 2 knots. The landing gear will extend immediately if the pressure switch is properly adjusted.

k. Should the landing gear fail to extend, turn the master switch OFF and adjust the pressure switch (inboard switch of the two installed in the left main wheel well) as follows:

1. Secure the rolled tubing so that it will hold the airspeed indicator reading at 120 ± 2 knots.

2. Connect a continuity tester across the contacts of the pressure switch, then turn the adjustment screw until the switch closes at the 120 ± 2 knots reading on the airspeed indicator.

i. Turn the master switch ON and check the landing gear safety system through the complete cycle of operation.

LANDING GEAR LIMIT SWITCHES - ADJUSTMENT

a. When adjusting the landing gear limit switches, observe the following precautions.

1. Do not allow the actuator to run full cycle into the internal stops as this may damage the sector gear. Adjust the limit switches for an early shut-down if this is considered a possibility.

2. Do not reverse the landing gear direction of travel without bringing the gear to a complete stop. If the direction of travel is reversed using the landing gear selector switch, damage to the sector gear may result.

3. Use the landing gear relay circuit breaker to start or stop the landing gear during rigging.

4. Battery voltage is not sufficient to properly cycle the landing gear during rigging. A 28.25 ± .25 volt auxiliary power unit capable of maintaining the initial setting within .25 volt during the extension and retraction cycles is recommended.

5. Do not operate the landing motor without proper cooling as this may damage the motor. Allow 5 minutes cooling time after each extension and retraction cycle.

6. After making an adjustment to the limit switches, operate the landing gear intermittently as the system nears the limits of the extension/retraction cycle to prevent damage due to overtravel.

7. Do not jack the airplane if an unbalanced condition exists. The fuel should be evenly distributed in the wings.

b. Adjust the landing gear limit switches as follows:

1. Place the airplane on jacks. Allow adequate floor clearance for the wheels during the retraction or extension cycle.

NOTE

The actuators (used on serials P-577 and after) may be installed on earlier airplanes. The new actuator is identified by part number 60-810030-5 or -7 and white paint on the upper actuator arm, and on the upper and lower actuator housing.

2. Retract the landing gear to its 2/3 position, then stop and inch the gear the remaining distar to the uplimit switch by intermittent operation of the land gear relay circuit breaker. Check the emergency hand cra for 1/8 to 1/4 turn (or 5/8 to 3/4 turn on airplanes using 810030-5 or -7 white actuators) free counterclockw movement before the sector gear contacts the geart internal stop. If proper internal clearance is not obtain adjust the landing gear uplimit switch. Locate the land gear uplimit switch adjustment screw on the actuator a and adjust the screw to stop the landing gear on its up cy at the point where proper internal clearance is maintain

CAUTION

The gearbox may be damaged if allowed to run full cycle into the internal stops. To preclude this possibility, the limit switches should initially be adjusted aft, for an early shut-down, if an out-of-adjustment condition is suspected.

NOTE

All adjustment and rigging of limit switches should be accomplished using an external power source adjusted to 28.25 ± .25 volt. Due to overtravel, the landing gear will not stop at the same position that the limit switches are actuated.

NOTE

On series P-139 and after, there are two switch assemblies, one on each side of the actuator.

3. Extend the landing gear to its 2/3 do position, then stop and inch the gear the remaining distar to the downlimit switch by intermittent operation of landing gear relay circuit breaker. Check the emerger hand crank for 1/8 to 1/4 turn counterclockwise (5/8 to turn with 60-810030-5 or -7 white actuators) before sector gear contacts the gearbox internal stop. If pro internal clearance is not obtained, locate the landing g downlimit switch adjustment screw on the upper arm of

landing gear actuator, and adjust the screw to stop the landing gear on its down cycle at the point where proper internal clearance is achieved.

WARNING

Do not operate the hand crank with the power on.

4. Extend and retract the gear two or three times to assure that the switches are correctly set. Check the hand crank each time to assure a free movement of 1/8 to 1/4 turn (5/8 to 3/4 turn on airplanes using 60-810030-5 or -7 white actuators) before the sector gear contacts the internal stop.

LANDING GEAR SAFETY SWITCH ADJUSTMENT

With the gear down, check the adjustment of the safety switch. Refer to the Landing Gear Illustration Figure 201 in 32-30-00, while adjusting the switch.

a. Remove the safety switch actuator rod (1) from the attaching bracket on the upper torque knee, then remove the retaining nut (2) and switch arm (3) from the switch shaft.

b. Jack the landing gear so the shock strut is compressed to .50 ± .12 inch from the extended position.

c. Connect the wire leads from a test light to pins inserted into the splices at wires number 1 and 3 (about 10 inches up from the safety switch).

d. Rotate the switch shaft clockwise until the test light comes on.

e. Remove the safety wire from the locking screw (4) on the switch arm (3) and back off the locking screw (4).

f. Install the switch arm (3) on the switch shaft in a position parallel to the upper torque knee and adjust the actuating rod end (1) to align with the attaching bracket on the torque knee. Install the actuating rod connecting bolt.

g. Position the shock strut to .87 ± .12 inch from the fully extended position and adjust the switch shaft counterclockwise at the adjusting screw (5) until the light goes out. When satisfactory adjustment is obtained, tighten the locking screw (4) and the retaining nut (2).

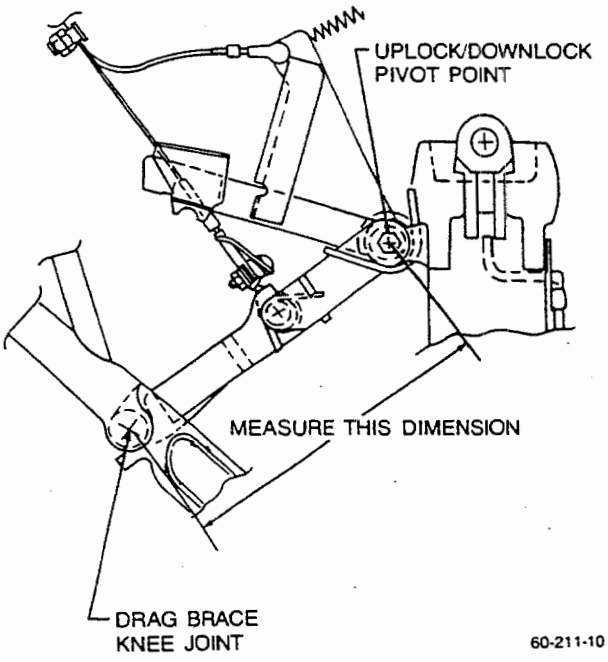
h. Recheck the landing gear travel to both dimensions described above before safety wiring the locking screw (4) to the switch arm.

LANDING GEAR POSITION LIGHTS ADJUSTMENT

The landing gear position lights on the instrument panel are operated by the uplock and downlock switches located on each landing gear.

a. Main Gear: With the main landing gear fully extended, measure the distance between the uplock/downlock pivot points and the lift leg knee joint on the main landing gear as shown in Figure 201. Adjust the downlock switch (outboard side of gear) to actuate the in-transit light when the lift leg knee joint overcenter dimension is reduced by .15 ± .10 inch. With the landing gear fully retracted adjust the uplock switches (inboard side of gear) to actuate the in-transit light when the actuator lacks five handcrank revolutions of being hard against the internal stop of the actuator. Adjust the switch as follows:

1. Handcrank the actuator until the sector gear contacts the internal stop.



Main Gear Downlock Switch Adjustment
Figure 201

2. Back the sector gear off the internal stop, by five revolutions of the handcrank.

3. Adjust the in-transit light switch to actuate in this position.

b. Nose Gear: With the nose gear fully extended adjust the downlock switch, located on the nose gear drag brace, so that the overtravel of the switch plunger is .050 to .125 inch after the actuation of the switch. Remove the wheel well access cover on the nose baggage compartment floor to gain access to the uplock switch. With the nose gear fully retracted, adjust the uplock switch (adjacent to the uplock hook) to actuate when the landing gear actuator lacks seven handcrank turns of being hard against the internal stop of the actuator. Adjust the switch as follows:

1. Handcrank the actuator until the sector gear is against the internal stop.

2. Back the sector gear off of the internal stop by seven revolutions of the handcrank.

3. Adjust the uplock switch to actuate at this position.

"END"

CHAPTER 33

LIST OF PAGE EFFECTIVITY

CHAPTER SECTION SUBJECT	PAGE	DATE
33-EFFECTIVITY/CONTENTS	1	Apr 18/80
33-00-00	1	Apr 18/80
33-40-00	101	Nov 2/73
	201	Nov 2/73
	202	Apr 18/80
	203	Apr 18/80
	204	Apr 18/80

CHAPTER 33 - LIGHTS

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"END"

GENERAL - DESCRIPTION AND OPERATION

HIGH INTENSITY LIGHTS - HOSKINS

The high intensity light system greatly increases the visibility of the airplane to other airplanes during night flight by means of pulsating strobe lights mounted adjacent to the wing tip and tail lights. The system is actuated by a switch in the exterior lighting group on the pilot's subpanel and is protected by a 10 amp circuit breaker on the copilot's circuit breaker panel. The strobe lights are powered by the master power assembly mounted in the aft fuselage directly behind the access door in the belly of the airplane. This assembly consists of three power supply modules (one for each strobe light) and a timing circuit module mounted on a common, negatively ground subchassis. The timing circuit module contains a small DC motor that rotates a notched transistorized power pack cam to actuate two switches mounted 180 degrees apart. Each time one of the switches is actuated, a relay in the affected power supply module closes to trigger its respective strobe light. A transistorized circuit in the power supply unit steps up the 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 flash tube of the strobe light by a specially shielded power cable. A charge of high voltage electricity is momentarily released to a coil in the flash tube assembly. The coil further steps up the charge to the point where it ionizes the xenon gas in the flash tube. The high voltage stored in the condenser then surges through the gas to produce the brilliant burst of energy that characterizes the strobe light.

HIGH INTENSITY LIGHTS-GRIMES AND SYMBOLIC DISPLAYS INC.

The function of the Grimes and SDI Strobe Light Systems are essentially the same as the Hoskins Strobe Light System. The Grimes and SDI systems do not incorporate a timing circuit, as the lights all flash at the same time. An electronic power supply module steps up the voltage of the airplane system to the level required to ionize the xenon gas in the flash-tubes. Each module contains a built-in flasher and the unit receiving input power acts as a master unit. An interconnecting unit located on the sides of each unit handles the input power and trigger pulsing of the system: these snap together when the units are placed side by side.

"END"

TROUBLESHOOTING
EXTERIOR LIGHTS

STROBE LIGHTS

TROUBLE	*PROBABLE CAUSE*	*REMARKS*
1. Lights inoperative.	a. Circuit breaker tripped.	a. Check for short circuit. Reset circuit breaker.
	b. Loose connection.	b. Check and tighten electrical connections.
	c. Battery defective.	c. Replace battery or use external power.
2. One bulb does not light.	a. Bulb burned out.	a. Replace bulb.
	b. Fixture not grounded.	b. Check for good bonding between fixture and structure. Tighten mounting screws.
	c. Loose connection.	c. Check all connections in circuit.
	d. Defective fixture or switch.	d. Replace fixture or switch.

LANDING LIGHTS

1. Lamp fails to light.	a. Circuit breaker tripped.	a. Check for short circuit. Reset circuit breaker.
	b. Lamp burned out.	b. Replace lamp.
	c. Loose connection or defective.	c. Tighten connections and check wire circuit continuity. Replace or repair wire if necessary.
	d. Landing-light switch defective.	d. Check continuity through switch. Replace if necessary.

"END"

EXTERIOR - MAINTENANCE PRACTICES

HIGH INTENSITY LIGHTS-HOSKINS

POWER CABLE CHECKS

a. Check that the individual conductors of the power cable are soldered to the connector pins as follows: red wire to the "A" pin, white wire (or striped wire of red and white) to the "B" pin, and black wire (or striped wire of red and black) to the "C" pin.

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.

b. Use 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.

c. Check the condition of the potting for the power supply cable at the terminals, connectors, and clamp cavities. The potting compound used to protect the system against moisture is RTV No. 102 silicon rubber.

d. Make sure the cable clamps are not tight enough to cause a short in the cable.

SYSTEM CHECK-OUT

The following check is recommended when the flash tube of a strobe light unit fails to fire:

a. Check that the flash tube is not broken and that the connectors are tight.

b. 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.

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.

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.

POWER SUPPLY UNIT 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 aircraft operation, perform the following check:

a. Open each power supply module and check the relay contacts for pitting. Replace those that are excessively pitted and clean the others.

b. 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 rnilliamperes 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.

TIMER MODULE 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 top. If necessary, loosen the top screw on the switch mounting plate and slide it in or out until the switches respond properly.

POWER SUPPLY UNIT REMOVAL

Remove the panel from the underside of the fuselage just aft of the rear pressure bulkhead to gain access to the power supply unit. This unit consists of three power supply modules (one for each light) and a timer module mounted on a common subchassis. The entire unit may be removed by disconnecting the wiring from the modules and removing the screws anchoring the subchassis to the support structure in the aft fuselage. Any one of the modules can be removed and replaced separately by simply disconnecting the wiring and removing the screws securing it to the subchassis.

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. Make sure the unit is hooked up as indicated in the wiring diagram, for it will be permanently damaged by reversed polarity.

POWER SUPPLY UNIT INSTALLATION

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, for it will be permanently damaged by reversed polarity.

NOTE

A harness wring-out and high potential test of 500 vac on the harness wiring should be conducted on the strobe light harness in the wing and fuselage prior to connection of either the strobe light or strobe power supply.

Place the power supply module on the support structure and secure in place with the attaching screws. Connect the power supply module to the airplane wiring in accordance with the Wiring Diagram Manual (P/N 60-590001-29).

HIGH INTENSITY LIGHTS-GRIMES AND SYMBOLIC DISPLAYS INC.

POWER SUPPLY UNIT REMOVAL

Remove the panel from the underside of the fuselage just aft of the rear pressure bulkhead to gain access to the power supply unit. The entire unit may be removed by disconnecting the wiring from the power supply and removing the screws anchoring the modules to the support structure.

CAUTION

Observe the precautions noted under STROBE LIGHT WIRING procedure when removing the power supply.

POWER SUPPLY UNIT INSTALLATION

NOTE

A harness wring-out and high potential test of 500 vac on the harness wiring should be conducted on the storbe light harness in the wing and fuselage prior to connection of either the strobe light or strobe power supply.

a. Place the power supply module on the support structure and secure it in place with the attaching screws.

b. Connect the power supply module to the airplane wiring in accordance with the Wiring Diagram Manual (P/N 60-590001-29).

STROBE LIGHT WIRING

NOTE

A harness wring-out and high potential test of 500 vac on the harness wiring should be conducted on the strobe light harness in the wing and fuselage prior to connection of either the strobe light or strobe power supply.

An incorrect hook-up of the wires at either the power input or between the strobe light assemblies and the power

supply unit will cause a reversal of polarity that results in serious component damage and failure. Care must be taken to ensure that the red wire is connected to positive power and the black wire to ground. On airplane serials prior to P-433 make sure that the red, white, and black wires are connected to pins "A", "B", and "C" of the connector respectively and that the connectors are properly assembled. On airplane serials P-433 and after make sure that the red, blue, and yellow wires are connected to pins "A", "B", and "C" of the connector respectively and that the connectors are properly assembled.

The shield for the wing cables should be grounded to the airplane structure at the wing break and the shield for the tail light cables should be grounded to the airplane structure at the power supply.

WARNING

Although a bleed-off resistor is incorporated in the power supply circuit, high voltage is involved in the circuit between the power supply and light assemblies. For this reason, turn the control switch for the strobe lights OFF and wait for at least 10 minutes to elapse before disconnecting the cables at the power supply or light assemblies and before handling or disassembling either of these units in any way. Failure to observe these precautions may result in physical injury from electrical shock.

STROBE LIGHT REPLACEMENT

CAUTION

To avoid damage to the strobe light system or possible physical injury from electrical shock, observe the precautions outlined under strobe light wiring before removing or installing the strobe light assembly.

CHART 201
LAMP BULB REPLACEMENT

LOCATION	*BULB PART NUMBER*
Annunciator Panel Lights	327
Edge Lights	D158-100-5T1
Post Lights	327
Compass Light	327
Instrument Flood Lights (Red)	1846R
Instrument Flood Lights (White)	1846
Map Light	1495
Flap Position Indicator Lights	FB-59
Landing Gear Position Lights	327
Reading Lights	1495
Threshold Light	313
Nose Baggage Compartment Light	303

CHART 201
LAMP BULB REPLACEMENT (Cont'd)

LOCATION	*BULB PART NUMBER*
Navigation Lights (Wing)	1524
Navigation Light (Tail)	1683
Rotating Beacon (Upper and Lower)	A7079B-24
Ice Light (P-4 thru P-402) (P-403 and after)	A7796A24 A7079B-24
Landing Lights	4596
Taxi Light (Nose Landing Gear)	4587
Strobe Light, Tail (Flashtube) (Hoskins)	31-0725-1
Strobe Light, Tail (Flashtube) (Grimes)	55-0221-1
Strobe Light, Wing (Flashtube) (SDI)	202331
Strobe Light, Wing (Flashtube) (Grimes)	55-0221-1
Recognition Lights (Wing) (P-386, P-401 and after)	1982

"END"

CHAPTER 34

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	202	Nov 2/73

CHAPTER 34 - NAVIGATION

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"END"

GENERAL - DESCRIPTION AND OPERATION

INSTRUMENT ACCESS

Engine instruments may be removed with the instrument panel in place. Pull the post-light shield straight off to provide clearance for the instruments. Loosen the retaining screw located below and to the left of the instrument, loosening the instrument retaining clamp. Sufficient wiring has been installed to enable you to pull the instrument out.

To gain access to the other instruments remove the four screws attaching the LH or RH panel door and pull it out far enough to disconnect plumbing and wiring, freeing the panel for removal.

NOTE

To avoid marring the finish, wrap padding around the control columns just below the instrument panel.

The glare-shield may be loosened and tilted back by releasing the four clips located on the underneath side.

INSTRUMENT OPERATION

The gyro instruments operate on filtered air provided by the pneumatic pressure system. Adjustment of the pneumatic pressure system is performed by adjusting the various regulators in a specified sequence. A PRESSURIZATION SYSTEM ADJUSTMENT CHART corresponding to applicable illustrations and a general adjustment procedure for each individual regulator is provided in Chapter 36-00-00. The air speed indicator, altimeter, and rate-of-climb indicator operate on air provided by the pitot and static pressure system. (Refer to Chapter 34-10-00.)

An Overhaul and Replacement Schedule for instruments utilized by the Duke series aircraft is found in Chapter 5-00-00.

"END"

FLIGHT ENVIRONMENT DATA DESCRIPTION AND OPERATION

The pitot and static pressure system provides a source of ram and static air for instrument operation. Ram air enters the pitot tube, located on the LH lower side (P-4 through P-152) or the RH lower side (P-153 and after) of the nose section forward of the nose landing gear doors, and is routed to the pilot's airspeed indicator. A heating element is installed in the pitot mast to prevent the mast from becoming obstructed by ice. Static air is taken from a static air port, located on each side of the aft fuselage. The static air is routed along the right side of the fuselage to the cabin differential pressure gage, rate-of-climb indicator, altimeter and airspeed indicator. Should the normal static system become inoperative, an emergency static air source control, located on the upholstery panel forward of the copilot's seat, may be opened to provide a source of static air. The emergency air port is located on the aft pressure bulkhead. The drain valves are located on the RH lower upholstery panel.

NOTE

When dual instruments are installed an optional dual pitot and static system is utilized. The optional pitot mast is plumbed to the copilot's airspeed indicator. The optional static air ports are located, one on each side of the aft fuselage and are plumbed to the copilot's rate-of-climb indicator, altimeter and airspeed indicator.

"END"

TROUBLESHOOTING
PITOT AND STATIC PRESSURE SYSTEM

	TROUBLE		*PROBABLE CAUSE*		*REMARKS*
1.	Heating element inoperative.	a.	Defective switch.	a.	Replace.
		b.	Grounded or open circuit.	b.	Check for continuity.
		c.	Defective heating element in pitot head.		
2.	Circuit breaker keeps tripping.	a.	Grounded wire.		
3.	Instruments inoperative or erratic in operation.	a.	Lines clogged.	a.	Drain lines at emergency static drain. Disconnect lines at instruments and blow out with low pressure air.
		b.	Line leaks.	b.	Check lines for looseness at all connection points.

"END"

FLIGHT ENVIRONMENT DATA - MAINTENANCE PRACTICES

(Figure 201)

PITOT SYSTEM LEAK TEST

A functional test of the pitot system can be made by using an observer in the cabin to watch the airspeed indicator while air pressure is built up by using a section of soft rubber tubing as follows:

a. Clamp the rubber tubing over the pitot mast inlet, making certain that the connection is air tight.

b. Crimp the end of the tubing and slowly roll it up until the airspeed indicator registers approximately 90% of its maximum reading.

CAUTION

To avoid rupturing the diaphragm of the airspeed indicator, roll up the rubber tubing slowly and do not build up excessive pressure in the line.

c. Secure the rolled up tubing so that it will hold the airspeed indicator reading.

d. If there is no decline in the reading after several minutes, there is no leak in the pitot system.

e. If a decline in the reading of the airspeed indicator is observed, check the pitot system plumbing for leaky hoses and loose connections.

CAUTION

Release the air pressure slowly by unrolling the rubber tubing; a sudden release of the air pressure may damage the airspeed indicator.

PITOT SYSTEM HOSE INSPECTION

After the pitot system is checked for leaks, inspect the hose sections for signs of deterioration. Check all polyethylene tubing for hardness or brittleness. Rubber hoses on which outer surfaces have checked or cracked, particularly at the bends or connecting points, or which have become hard, should be replaced. Replace the defective hose with MIL-H-5593 rubber hose (27, Chart 207, 91-00-00). When new hose is installed, recheck the system for leaks using the PITOT SYSTEM LEAK TEST procedure.

STATIC SYSTEM CLEANING

Blow low air pressure through the lines from the disconnected line at the airspeed indicator to the static ports. Cover each static port separately when blowing tc insure that each line is clear. Instrument error or possible damage may result if even one port is clogged with dirt or foreign matter.

CAUTION

Never blow air through the line toward the instrument panel; to do so will seriously damage the instruments. When blowing back through the line from the instrument panel, make sure that no air is blown into the instruments.

NOTE

Wax or polish applied to the static air buttons can cause wrong instrument readings. The static air buttons should be cleaned periodically with a cleaning solvent to insure that no film exists on them.

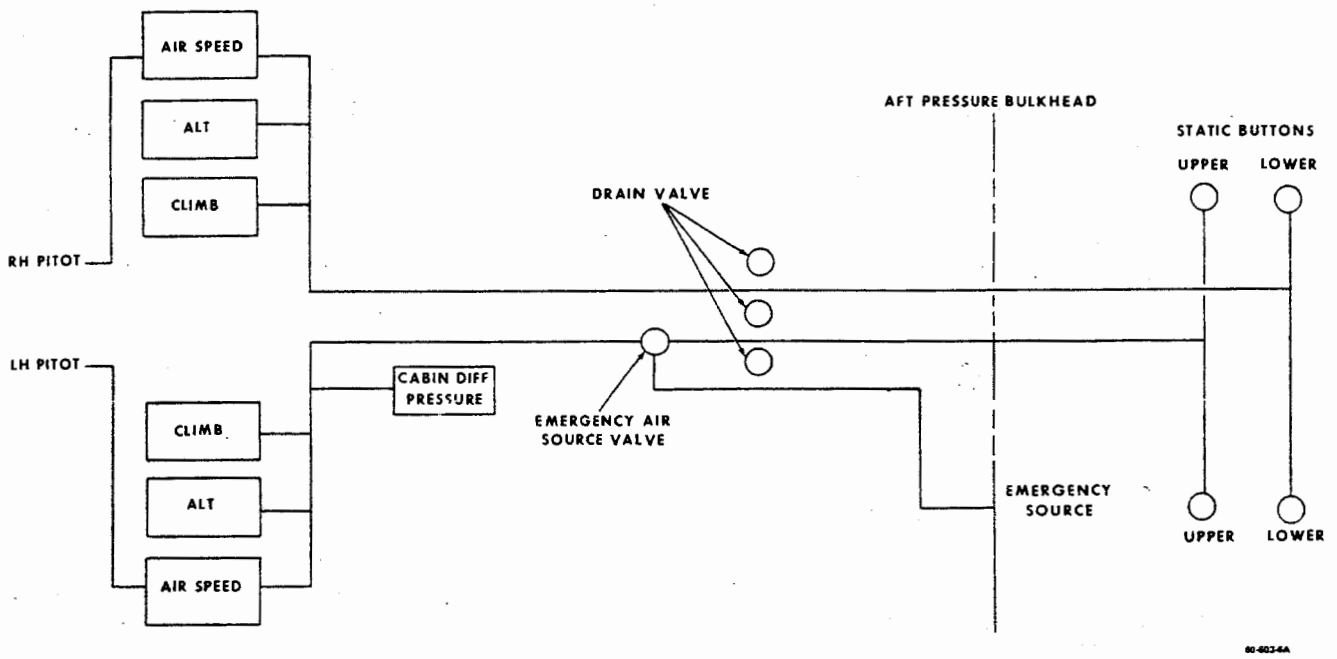
Drain the static air system by opening the static drain valves located on the upholstery panel forward of the copilot's seat.

STATIC SYSTEM LEAK TEST

The static system should be checked for leaks in accordance with the instructions in Federal Aviation Regulation 91.170.

CAUTION

To avoid damaging the airspeed indicators, the indicators should be removed from the system and the lines capped or an equal pressure should be applied to the pitot side of the indicators while leak testing the system.



Pitot and Static System Schematic
Figure 201

"END"

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CHAPTER 35 - OXYGEN

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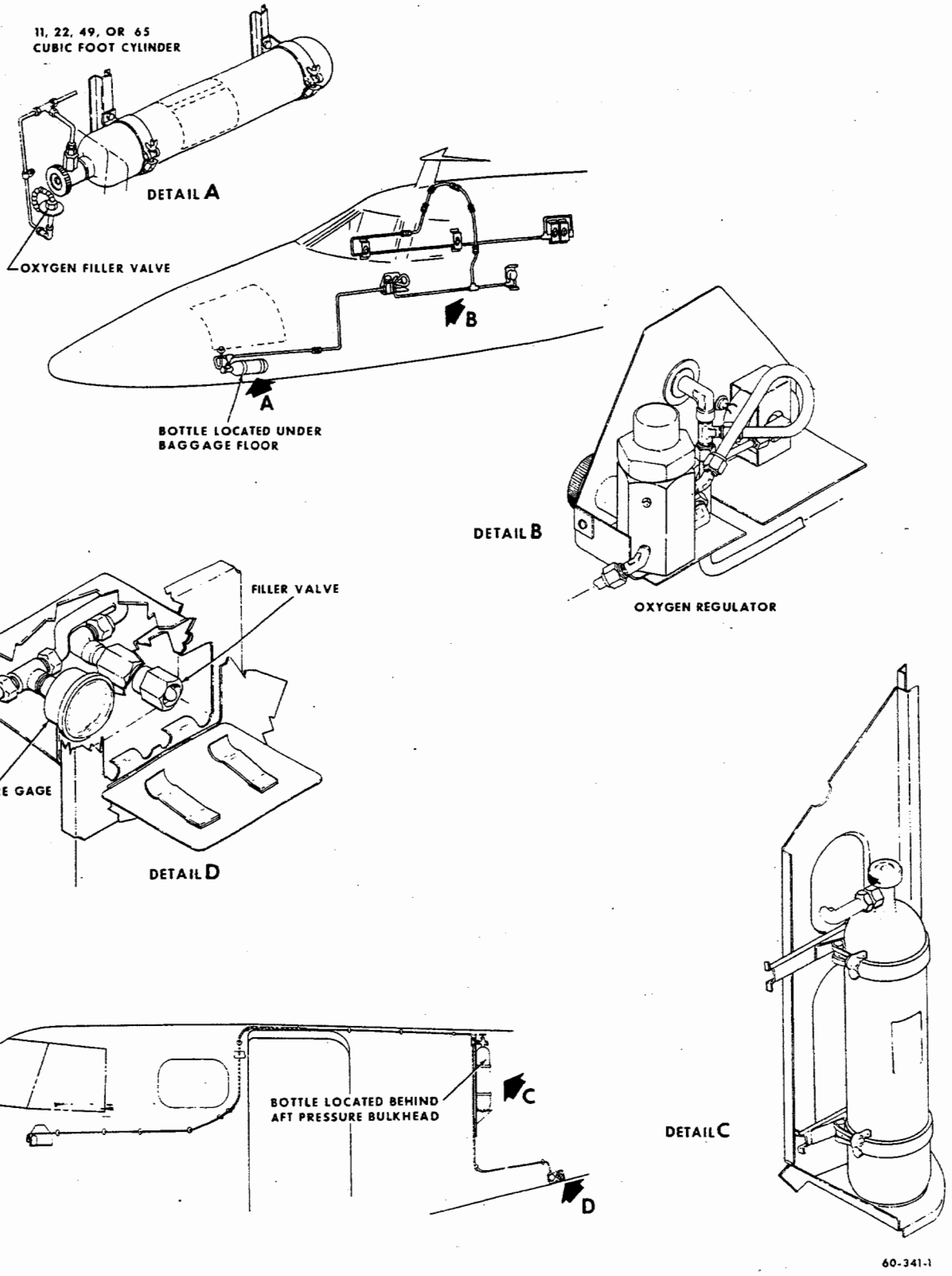
"END"

GENERAL - DESCRIPTION AND OPERATION

Oxygen for flights at high altitude is supplied by a cylinder mounted under the nose baggage compartment floorboards or aft of the rear pressure bulkhead. The system is serviced by a filler valve, accessible through a door on the lower LH side of the aft fuselage for the aft mounted bottle or the nose baggage compartment for the nose mounted bottle. 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 located on the pilot's sidewall. These components regulate the oxygen flow to the six cabin wall outlets. Oxygen masks are of the Scott 283 continuous-flow type. They are easily adjusted to fit the average person comfortably with a minimum leakage of oxygen and are considered adequate for continuous use up to 30,000 feet. When use of the oxygen is discontinued, it is necessary that the system be turned off by closing the control valve of the console. The pressure gage on the console indicates the supply of oxygen available but does not disclose whether the system is on or off. The regulator is a pressure-demand altitude compensating constant flow type which reduces system pressure to 50 to 55 psi at the passenger outlets.

WARNING

Proper safety measures must be employed while using oxygen, or a serious fire hazard will be created. NO SMOKING PERMITTED.



Oxygen System
Figure 1

"END"

GENERAL - MAINTENANCE PRACTICES

CAUTION

All persons handling and servicing oxygen systems should review proper precautions to be observed during servicing. FAA Advisory Circular 43.13-1A contains the necessary information.

OXYGEN SYSTEM TEST PROCEDURES

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 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 5 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 2 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 15 minutes, the drop in pressure shall not exceed 5 psi. In case leakage is excessive, apply MIL-L-25567 leak testing compound (14, Chart 207, 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.

OXYGEN SYSTEM 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 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.

a. After gaining access to the oxygen cylinder turn the oxygen cylinder valve toward the closed position until it is just cracked open.

b. Open the access panel for the filler valve, remove the protective cap, and attach the hose from an oxygen recharging cart to the filler valve.

c. Plug in an oxygen mask at each outlet in the cabin and pilot's compartment.

d. Open the cabin door, and turn the oxygen shutoff valve in the pilot's compartment to the open position.

e. Set the cart pressure regulator to deliver 50 psi of pressure to the system.

f. 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, return the cylinder valve to its normal operating position and service the system as described in Chapter 12-10-00.

OXYGEN CYLINDER REPLACEMENT

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.

a. Slowly turn the oxygen supply cylinder valve until fully closed.

b. Disconnect the line from the supply cylinder.

c. Cap the open line immediately with a clean fitting.

d. 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.

e. Remove the old cylinder from the clamps and install the new cylinder.

f. 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.

g. Connect the line fitting to the cylinder.
h. Open the cylinder shutoff valve.
i. Test the connections for leaks with MIL-L-25567 leak testing compound (14, Chart 207, 91-00-00).

OXYGEN CYLINDER RETESTING

Oxygen cylinders used in the Duke series aircraft are of two types. Light weight cylinders, stamped "3HT" on the side plate, must be hydrostatically tested every three years and the test data stamped on the cylinder. This bottle has a service life of 4,380 pressurizations or fifteen years, whichever comes first, and then must be discarded. Regular weight cylinders, stamped "3A" or "3AA" must be hydrostatically tested every five years and stamped with the retest date. Service life on these cylinders is not limited.

"END"

CHAPTER 36

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36-EFFECTIVITY/CONTENTS	1	May 30/75
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	208	May 30/75
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CHAPTER 36 - PNEUMATIC

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"END"

GENERAL - DESCRIPTION AND OPERATION

The pressure system on the Duke provides filtered air for deicer, autopilot and gyro instrument operation. Air pressure is supplied by two engine-driven dry air pumps. Pressure is controlled by a supply pressure regulator located in each nacelle. From the supply pressure regulator, air then flows to the pressure manifold, located below the pilot's and copilot's floorboards. Air from the manifold is distributed to the instruments and the deicer and autopilot systems (if installed).

Pressure from the pressure manifold to the gyro instruments is regulated. On airplane serials P-3 through P-182, a single regulator is located on the RH side of the pilot's compartment, forward and below the instrument panel. The gyro instrument pressure may be monitored on the gyro instrument pressure gage. On airplane serials P-183 and after, the pilot's and copilot's gyro instrument supplies are separated and each is supplied by separate pressure regulators. These regulators are located near the manifold, below the RH floorboards, forward of the main spar. A selector valve is provided to permit monitoring the pilot's or copilot's gyro instrument pressures on the gage.

"END"

GENERAL - MAINTENANCE PRACTICES

SERVICING

Impurities and foreign matter are removed from the air by two separate filters before entering the pressure manifold. On serials P-3 through P-246 which have not complied with Service Instructions 0595-194 the intake filter, located forward of the rear engine baffle, should be removed and cleaned every 100 hours. The paper filter on serials P-247 and after cannot be cleaned and should be replaced every 300 to 500 hours or annually.

CAUTION

In the event the paper filter is contaminated by solvent it must be replaced.

The sealed inline filter on the pressure side of the pressure pump, located in each nacelle aft of the firewall, should be replaced every 150 hours of operation for serials P-3 through P-159 except P-158 and every 300 hours of operation for serials P-158, P-160 and after.

ADJUSTMENT

Adjustment of the pneumatic pressure system is performed by adjusting the various regulators in a specified sequence. A PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART corresponding to applicable illustrations and a general adjustment procedure for each individual regulator are provided on the following pages.

On serials P-158 and after and those prior airplanes which have complied with Part II of Service Instructions 0433-190 that are equipped with an H-14 autopilot and a surface deice system incorporate a normally open by-pass valve for the H-14 Autopilot pneumatic pressure regulator. The valve causes air to bypass the regulator except in the deice mode thus eliminating air pressure drop across the regulator and permitting a lower system operating pressure for increased air pump life.

SINGLE STAGE SUPPLY PRESSURE REGULATOR

The single stage supply pressure regulator is used when the basic pressure system or the basic with H-14 autopilot pressure system is installed. The regulator is located in the RH side of each nacelle, aft of the firewall. The regulator may be adjusted as follows:

a. Remove the cap from the supply pressure test point located in the pressure line between the regulator and the inline air filter.

b. Install a test gage (0-20 psi) on the supply pressure test point.

c. Loosen the check nut and adjust the supply pressure regulator to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART. The engine should be running at 2500 rpm while the adjustment is being made. Rotate the adjusting screw clockwise to increase pressure and counterclockwise to decrease pressure.

d. Tighten the check nut, remove the test gage and reinstall the cap on the test point.

e. Repeat the above procedure on the opposite side of the airplane.

TWO STAGE SUPPLY PRESSURE REGULATOR

The two stage supply pressure regulator is used when the basic with deicer pressure system or the basic with deicer and H-14 autopilot pressure system is installed. The regulator is located in the RH side of each nacelle, aft of the firewall. The regulator may be adjusted as follows:

a. Loosen the check nut on the low pressure section of the regulator (section with the cutout solenoid attached) and adjust to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART. The engine should be running at 2500 rpm and the deicer system turned off while the adjustment is being made. Rotate the adjusting screw clockwise to increase pressure and counterclockwise to decrease pressure.

b. Tighten the check nut on the low pressure section of the regulator.

c. Loosen the check nut on the high pressure section of the regulator (section without the cutout solenoid attached) and adjust to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART. The engine should be running at 2500 rpm and the deicer system turned on while the adjustment is being made. Rotate the adjusting screw clockwise to increase pressure and counterclockwise to decrease pressure.

d. Tighten the check nut on the high pressure section of the regulator.

e. Repeat the above procedure on the opposite side of the airplane.

EJECTOR REGULATOR (P-308 and after; EQUIPPED WITH SURFACE DEICE)

A regulator is installed in the supply line for the ejector which develops the vacuum used in the cabin pressurization system on airplane serials P-308 and after, if equipped with surface deice. This regulator may be adjusted by the following suggested procedure:

a. Remove the plug from the port adjacent to the regulator outlet port.
b. Connect a pressure gage to this port.
c. Operate one engine at a speed of 2500 rpm.
d. Loosen the check nut and adjust the regulator to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART. Monitor the outlet pressure on the test gage and the inlet pressure of the regulator on the pneumatic pressure gage. Turn the adjustment clockwise to increase the pressure, counterclockwise to decrease the pressure. Tighten the check nut.
e. Remove the test gage and replace the plug.

GYRO INSTRUMENT PRESSURE REGULATOR

The gyro instrument pressure supply is regulated on all configurations of the pressure system. On airplane serials P-3 through P-182, a single gyro instrument pressure regulator is used. This regulator is located on the RH side of the pilot's compartment, forward and below the instrument panel.

On airplane serials P-183 and after, a pair of gyro instrument pressure regulators are located below the RH floorboards of the pilot's compartment, just forward of the main spar. Access may be gained to these regulators by removing the carpet and an access panel. A selector valve on the subpanel permits switching the gyro instrument pressure gage to monitor either the pilot's or copilot's gyro instrument pressure.

GYRO INSTRUMENT PRESSURE REGULATOR ADJUSTMENT (P-3 thru P-182)

a. Loosen the check nut and adjust the regulator to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART. Both engines should be running at 2500 rpm while the adjustment is being made. Rotate the adjusting screw clockwise to increase pressure and counterclockwise to decrease pressure.
b. Tighten the check nut on the regulator.

GYRO INSTRUMENT PRESSURE REGULATOR ADJUSTMENT (P-183 and after)

a. Remove the carpet and the instrument air regulator access panel from the RH floorboards just forward of the main spar.
b. Operate both engines at 2500 rpm while the adjustment is being made.
c. Select the PILOT position of the selector valve to permit monitoring the pilot's gyro instrument supply pressure.
d. Loosen the check nut and adjust the regulator supplying pressure to the pilots gyro instruments. Adjust the regulator to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART, CHART 201. Rotate the adjusting screw clockwise to increase, counterclockwise to decrease the pressure.
e. Tighten the check nut on the regulator.
f. Select the COPILOT position of the selector valve.
g. Loosen the check nut and adjust the regulator supplying pressure to the copilot's gyro instruments.
h. Tighten the check nut on the regulator.
i. Shut down the engines.
j. Replace the access panel and carpeting.

TURN AND SLIP PRESSURE REGULATOR

A turn and slip pressure regulator is installed for each pressure-driven turn and slip indicator. The regulator is located on the turn and slip indicator forward of the instrument panel. The regulator may be adjusted as follows:

a. Remove the glareshield and/or radio panel to provide access to the back side of the turn and slip indicator.
b. Remove the plug on the upper side of the regulator and install a test gage (0-5 in. Hg).
c. Loosen the check nut and adjust the turn and slip regulator to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART. Both engines should be running at 2500 rpm while the adjustment is being made. Rotate the adjusting screw counterclockwise to increase pressure and clockwise to decrease pressure.
d. Tighten the check nut, remove the test gage and reinstall the plug in the regulator.
e. Reinstall the glareshield and/or radio panel.

H-14 AUTOPILOT PRESSURE REGULATOR

The H-14 autopilot pressure regulator is used only with the basic with deicer and H-14 autopilot pressure system. The regulator is located inside the tail section just aft of the aft pressure bulkhead (see Figure 207). Adjust the regulator as follows:

a. Remove the cap from the autopilot pressure test point and install a test gage (0-20 psi).
b. Loosen the check nut and adjust the regulator to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART. Both engines should be running at 2500 rpm while the adjustment is being made. Rotate the adjusting screw clockwise to increase pressure and counterclockwise to decrease pressure.
c. Tighten the check nut, remove the test gage and reinstall the cap on the test point.

CHART 201
PNEUMATIC PRESSURE SYSTEM ADJUSTMENT

	Single Stage Supply Pressure Regulator	Two Stage Supply Pressure Regulator	Gyro Instrument Pressure Regulator	Turn and Slip Pressure Regulator	Autopilot Pressure Regulator	Aileron Servo Pressure Regulator
Basic System. (Figure 201 and Figure 202)	[1] 7.5 ± .5 psi * on test gage. Engine running at 2500 rpm. Adjust each side individually. **		[2] 5.25 ± .25 inches Hg * on gyro pressure indicator. Both engines running at 2500 rpm. †††	[3] 2.3 ± .2 inches Hg *** on test gage. Both engines running at 2500 rpm.		
Basic with H-14 Auto pilot (Figure 203)	[1] 11.8 to 12.3 psi on test gage. Both engines running at 2500 rpm with autopilot on. †		[2] 5.25 ± .25 inches Hg * on gyro pressure indicator. Both engines running at 2500 rpm. †††	[3] 2.3 ± .2 inches Hg *** on test gage. Both engines running at 2500 rpm.		[4] 5.3 to 5.8 psi on test gage. Both engines running at 2500 rpm with autopilot on.
Basic with Deice (Figure 204)		[1] 8.0 ± .5 psi * on pneumatic pressure gage. Both engines running at 2500 rpm with deice system off. † On airplane serials P-308 and after, adjust ejector regulator for 6-7 psi with 8.0 ± .5 psi on pneumatic pressure gage (Figure 206).	[3] 5.25 ± .25 inches Hg * on gyro pressure indicator. Both engines running at 2500 rpm. †††	[4] 2.3 ± .2 inches Hg *** on test gage. Both engines running at 2500 rpm.		
		[2] Adjust for peak pressure of 16 to 18 psi on pneumatic pressure gage. Both engines running at 2500 rpm with deice system on. †				
Basic With Deice, H-14 Autopilot and Pitch Trim (Figure 205)		[1] 11.3 to 11.8 psi on pneumatic pressure gage. †† Both engines running at 2500 rpm with autopilot on and deice system off. †	[3] 5.25 ± .25 inches Hg * on gyro pressure indicator. Both engines running at 2500 rpm. †††	[4] 2.3 ± .2 inches Hg *** on test gage. Both engines running at 2500 rpm.	[5] 11.0 to 11.5 psi on test gage. Both engines running at 2500 rpm with autopilot and deice system on.	[6] 5.3 to 5.8 psi on test gage. Both engines running at 2500 rpm with autopilot on and deice system off.
		[2] Adjust for peak pressure of 16 to 18 psi on pneumatic pressure gage. Both engines running at 2500 rpm with autopilot and deice system on. †				

Numbers designated ☐ outline sequence that regulators should be adjusted and refer to indexes on corresponding illustrations. Gyro Instrument Pressure Regulator and Turn and Slip Pressure Regulator shown on Basic Pressure System illustration (Figure 201 only).

*If airplane has more than two air-driven gyros, increase Supply Pressure Regulator setting minimum amount required to obtain 5.25 ± .25 inches Hg on gyro pressure indicator (maximum Supply Pressure Regulator setting to be 12.3 psi at test gage).

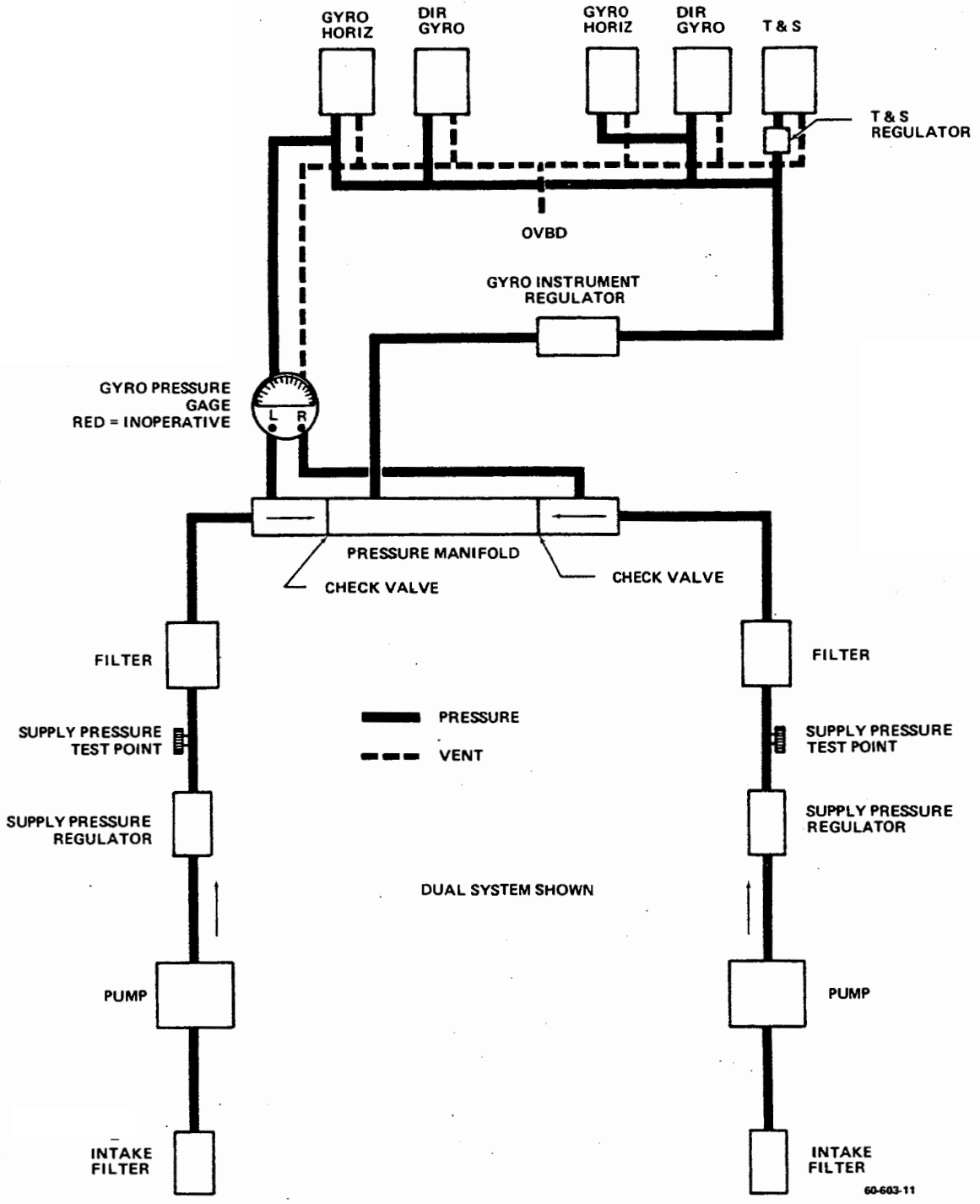
**Pressure will increase slightly with both engines running.

***Check in flight and if necessary, adjust to obtain standard rate turn.

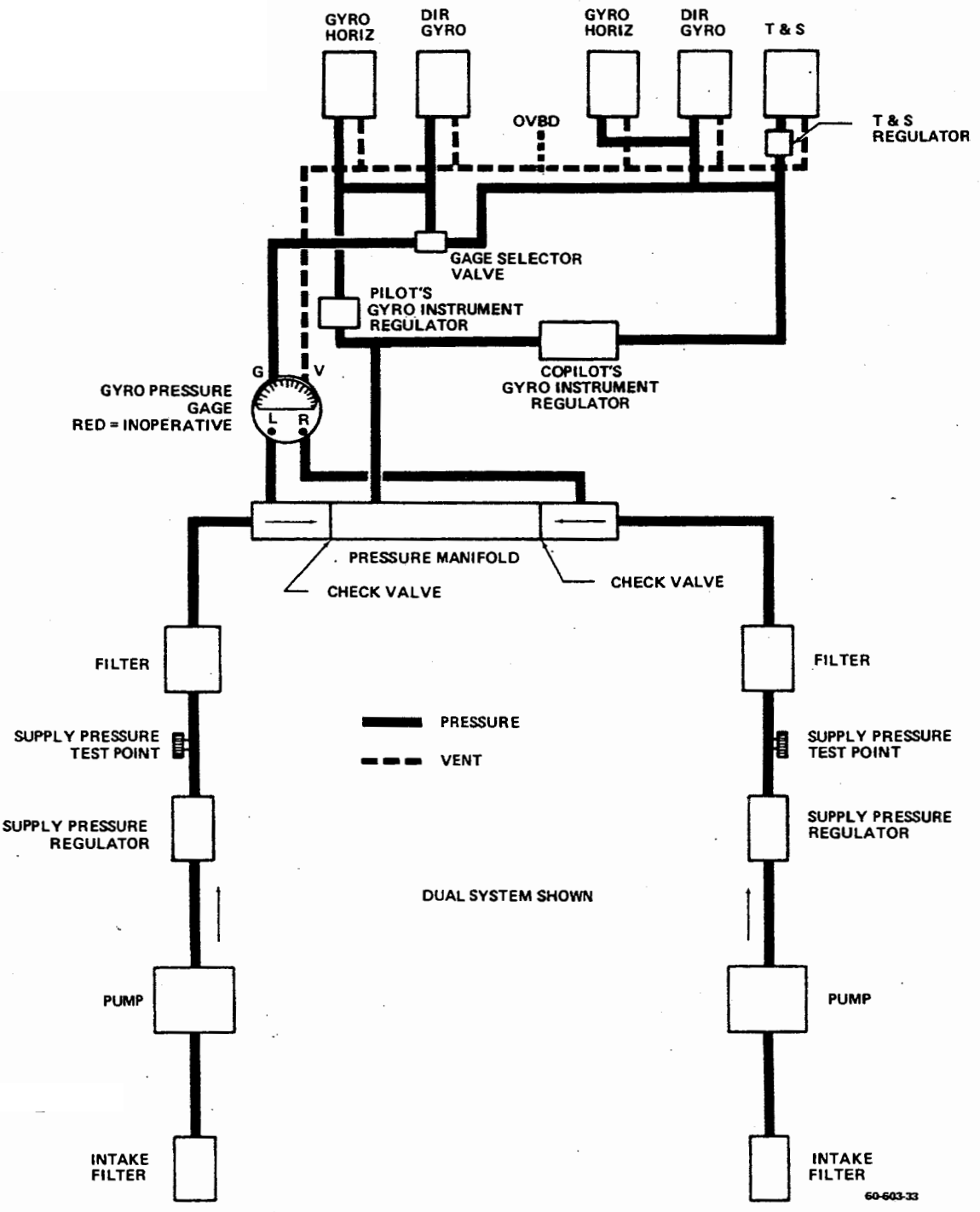
†Single-engine settings to be equalized at slightly lower pressure.

††13.5 to 14.5 psi on airplnaes prior to serials P-158 which do not have Kit No. 60-5015 S installed.

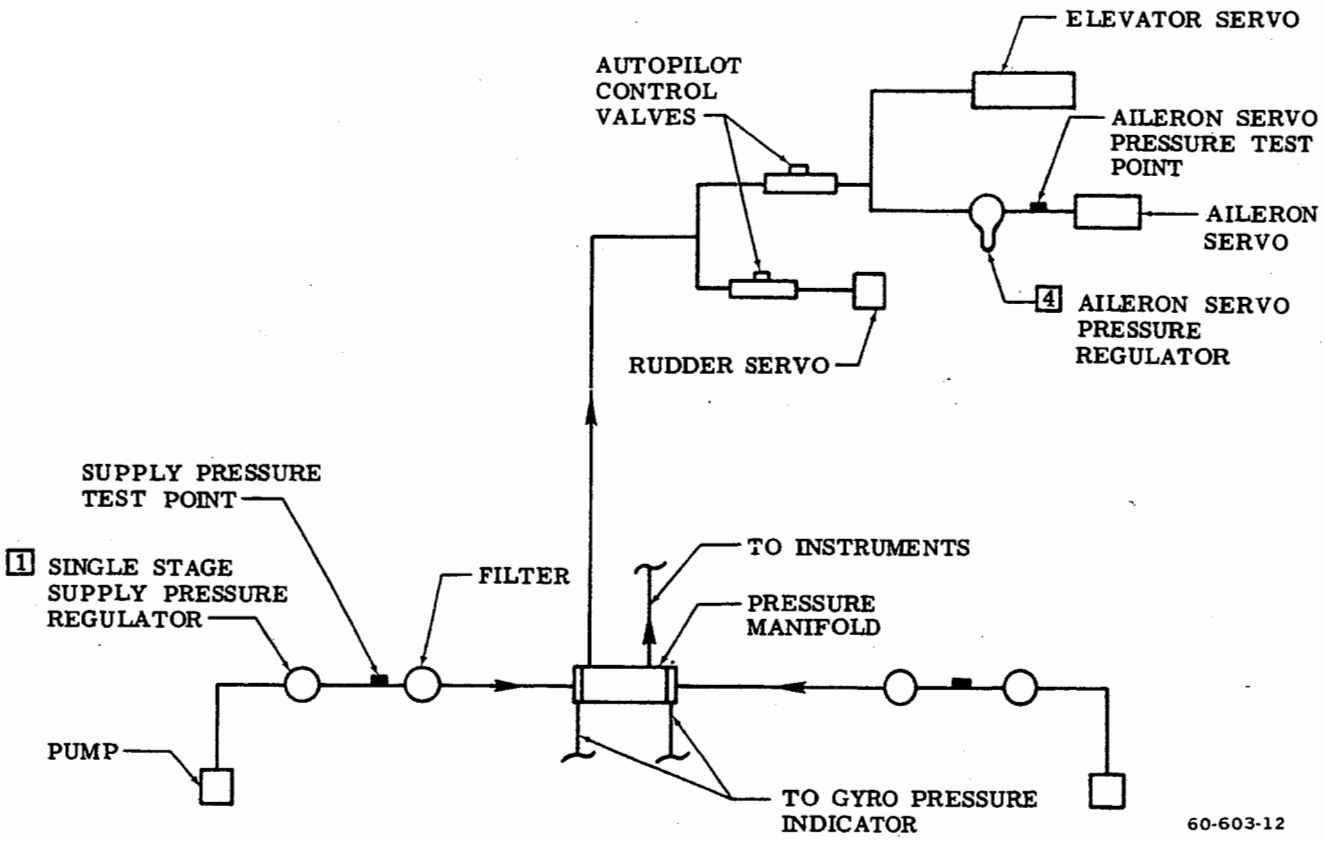
†††On airplane serials P-183 and after, select PILOT or COPILOT position of the gage selector to permit monitoring the gyro instrument pressure while adjusting the respective regulator.



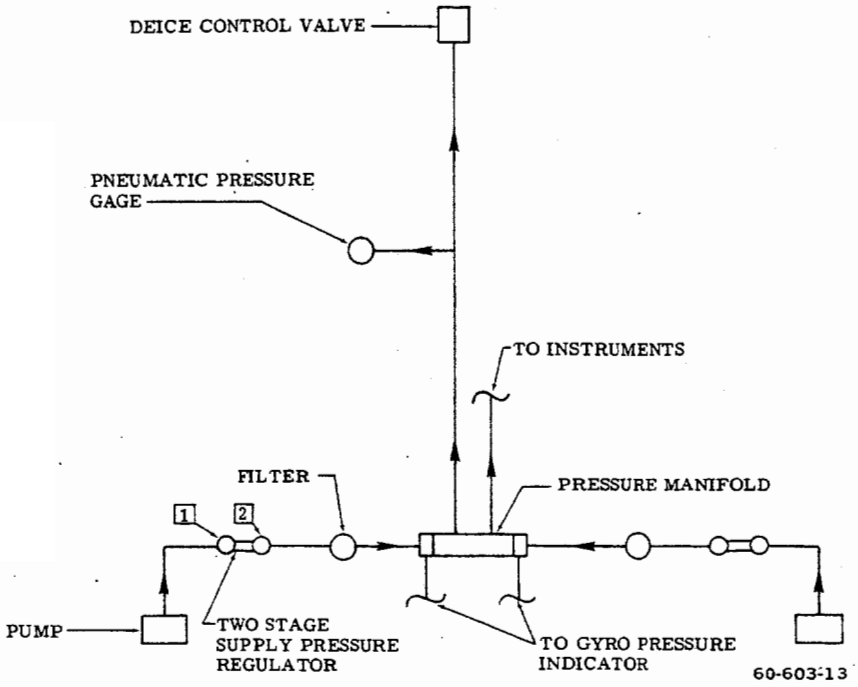
Pneumatic Pressure System (P-3 thru P-182)
Figure 201



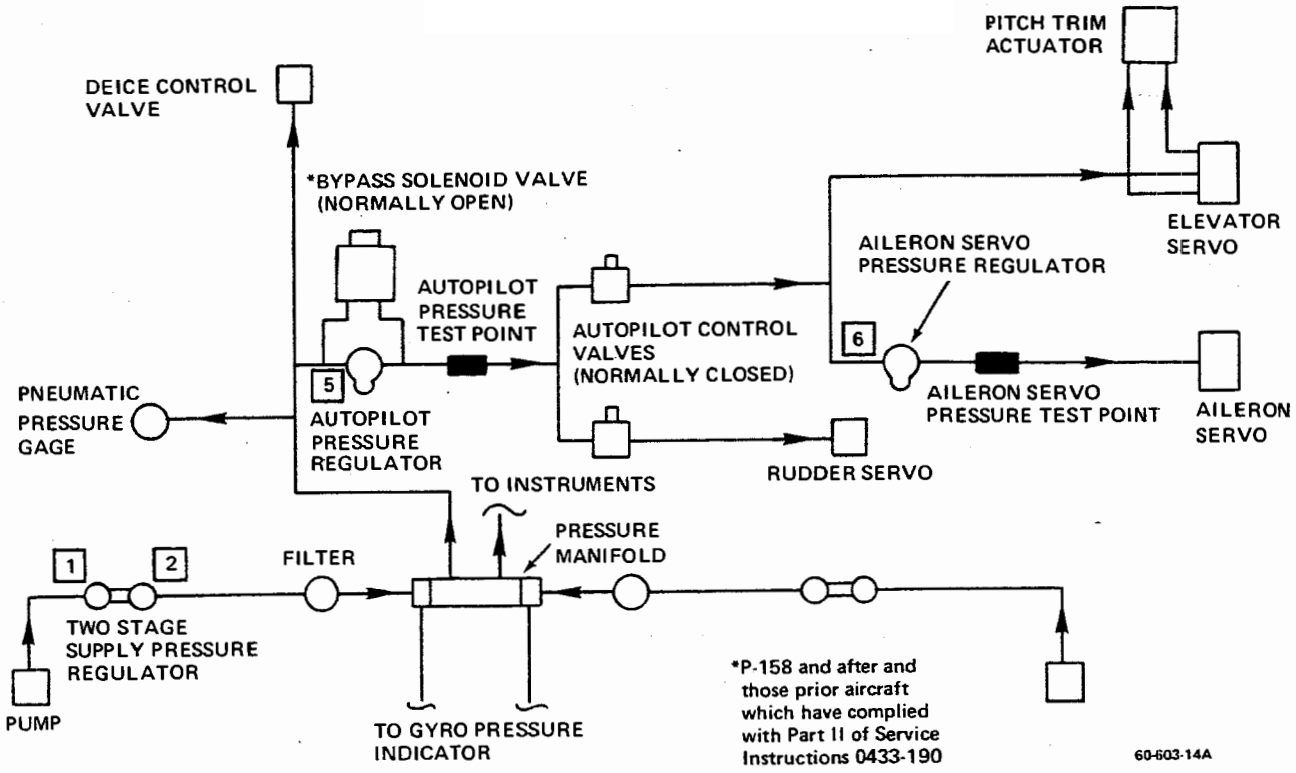
Pneumatic Pressure System (P-183 and after)
Figure 202



Pneumatic Pressure System (Basic with H-14 Autopilot)
Figure 203



Pneumatic Pressure System (Basic with Deice) (P-3 thru P-307)
Figure 204



Pneumatic Pressure System (Basic with H-14 Autopilot Deice and Pitch Trim)
Figure 205

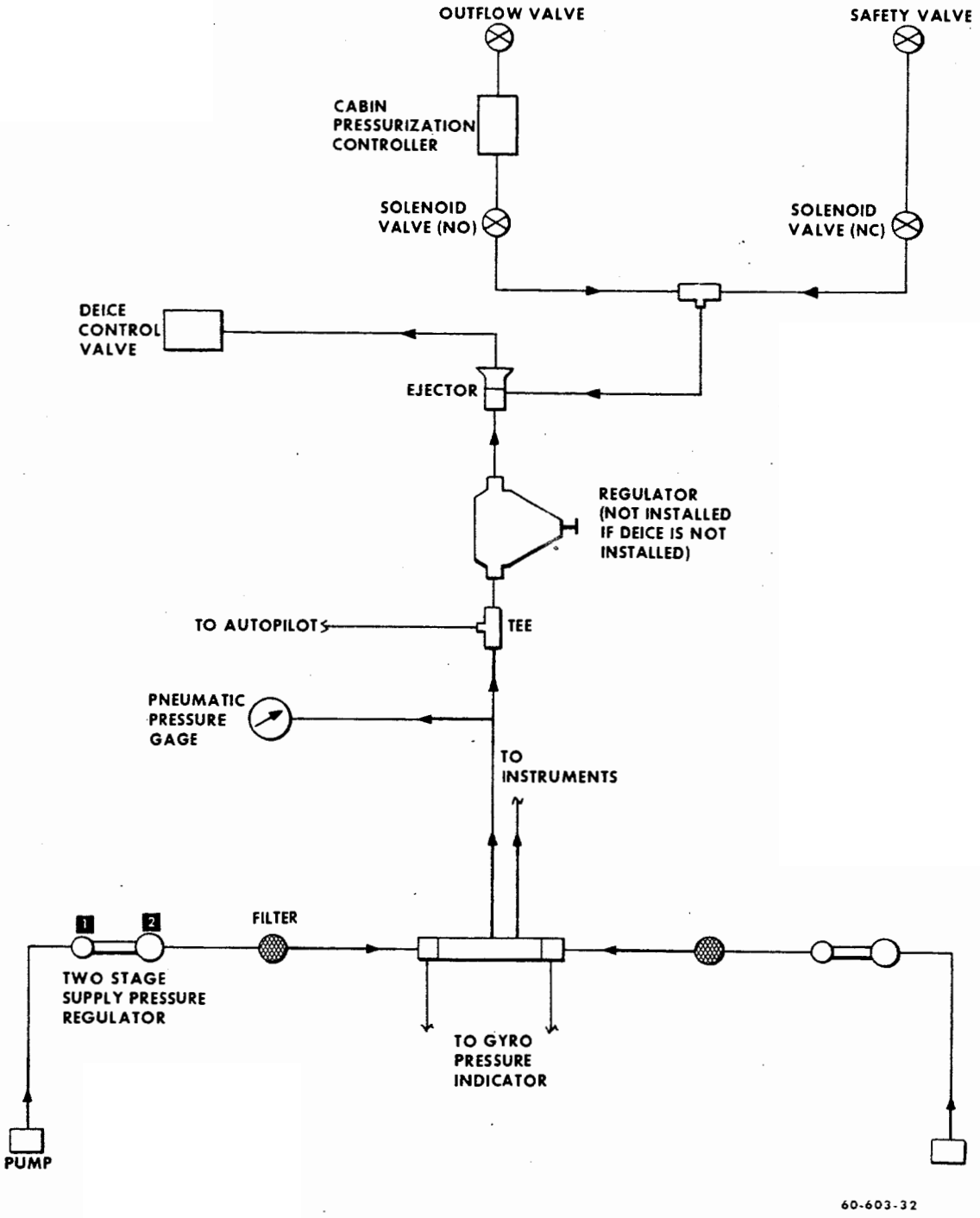
AILERON SERVO PRESSURE REGULATOR

The aileron servo pressure regulator is used when the H-14 autopilot is installed. The regulator is located inside the tail section just aft of the pressure bulkhead (see Figure 208). Adjust the regulator as follows:

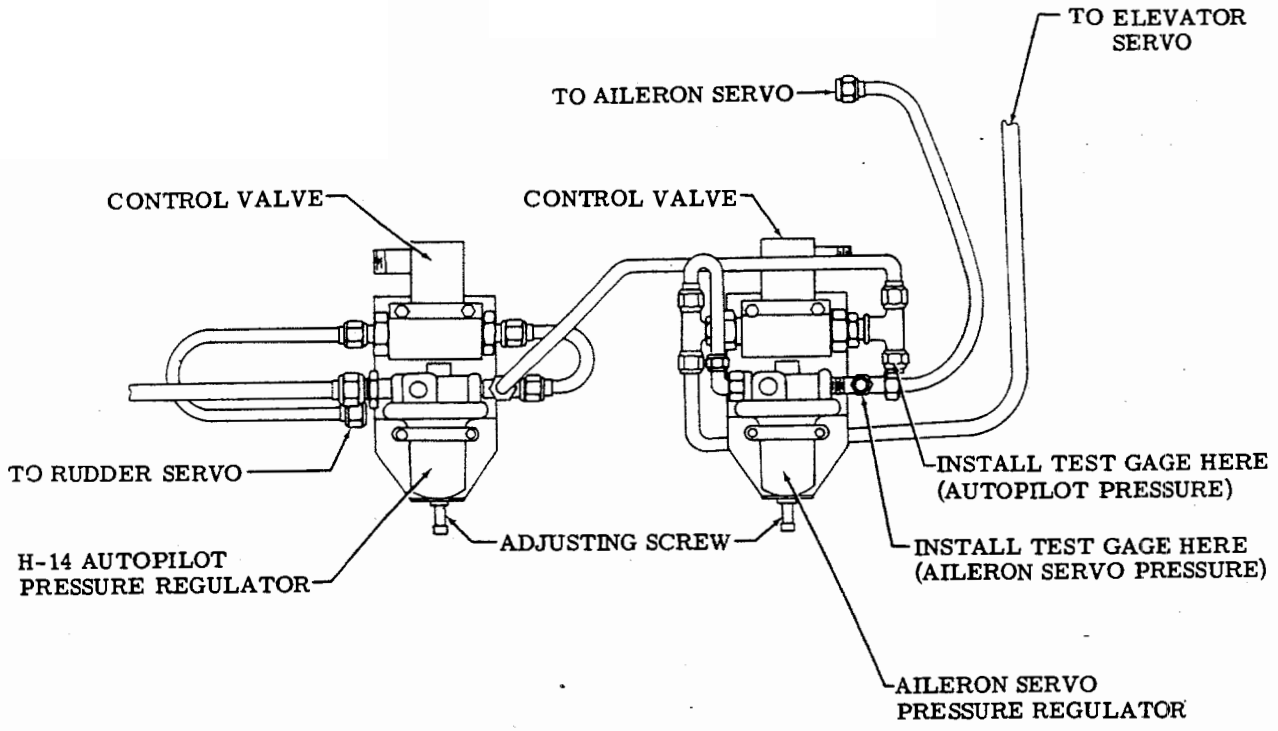
a. Remove the cap from the servo pressure test point and install a test gage (0-10 psi).

b. Loosen the check nut and adjust the regulator to the pressure indicated on the PNEUMATIC PRESSURE SYSTEM ADJUSTMENT CHART. Both engines should be running at 2500 rpm while the adjustment is being made. Rotate the adjusting screw clockwise to increase pressure and counterclockwise to decrease pressure.

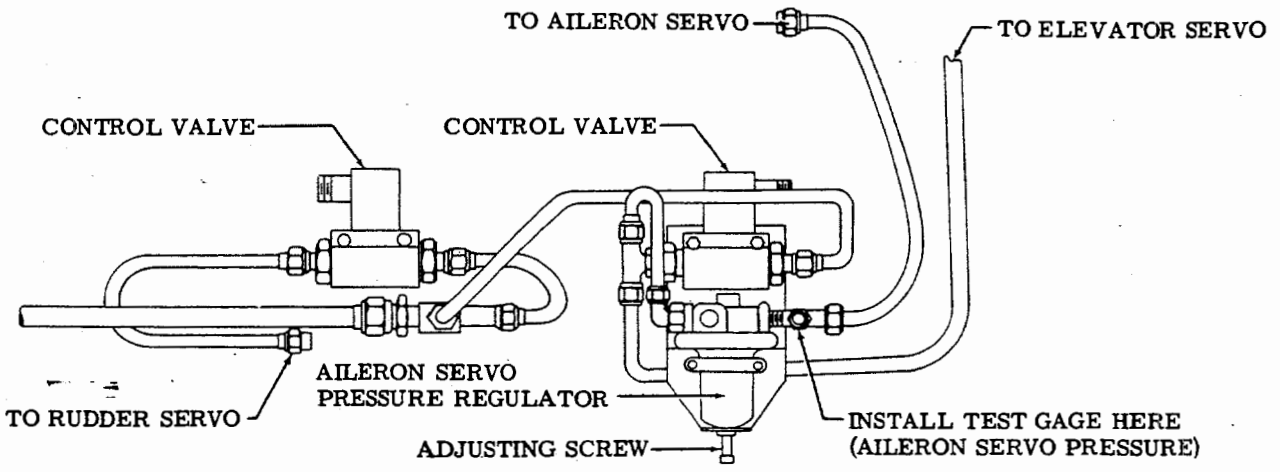
c. Tighten the check nut, remove the test gage and reinstall the cap on the test point.



Pneumatic Pressure System (Basic with Deice)
(P-308 and after)
Figure 206



Pressure Regulators (H-14 Autopilot and Aileron Servo)
Figure 207



Pressure Regulator (Aileron Servo)
Figure 208

"END"

CHAPTER 38

LIST OF PAGE EFFECTIVITY

CHAPTER SECTION SUBJECT	PAGE	DATE
38-EFFECTIVITY/CONTENTS	1	Aug 15/80
38-30-00	1	Aug 15/80
	201	Aug 15/80

CHAPTER 38 - WATER/WASTE

TABLE OF CONTENTS

"END"

WASTE DISPOSAL - DESCRIPTION AND OPERATION

The chemical toilet is located in the aft section of the airplane and is concealed by a bench-type seat in place of the optional fifth and sixth seat. The toilet is on the RH side of the seat and is accessible by lifting the RH seat cushion. The toilet is of the standard dry chemical type.

"END"

WASTE DISPOSAL - MAINTENANCE PRACTICES

CHEMICAL TOILET CLEANING

The toilet is of the standard dry chemical type. The toilet should be removed and emptied after each period of use. Any approved dry chemical may be used in accordance with the manufacturers instructions. A stiff (non-metallic) bristle brush and a water and detergent solution may be used to clean the toilet.

NOTE

For sanitary reasons, wear rubber gloves, elbow length, any time the toilet is to be cleaned or emptied.

a. Remove the RH seat cushion to gain access to the toilet.

b. Pull the toilet seat away from the seat deck. The toilet seat is attached with velcro fasteners.

c. Pull out the disposable waste bag and discard.

d. Place a new disposable waste bag in the bowl, making sure the bag overlaps the top of the bowl.

e. Place the toilet seat in position on the seat deck and press down in the area of the velcro fasteners.

f. Put the RH seat cushion back in position.

"END"

CHAPTER 51

LIST OF PAGE EFFECTIVITY

CHAPTER SECTION SUBJECT	PAGE	DATE
51-EFFECTIVITY/CONTENTS	1	Nov 2/73
51-00-00	1	Nov 2/73
	201	Nov 2/73

CHAPTER 51 - STRUCTURES

TABLE OF CONTENTS

"END"

GENERAL - DESCRIPTION AND OPERATION

Being of semimonocoque construction, the BEECHCRAFT Duke fuselage is pressurized to the skin between pressure bulkheads at stations 100.00 and 242.00 All skin, bulkheads and structure points, plumbing and wiring connections passing through a pressure wall, access doors, windows, control cables, and torque shafts are sealed to minimize air leakage. Although the carry through structure is an integral part of the fuselage, the wing panels may be removed at the attach points inboard of the nacelles. An emergency exit is installed on the right side of the fuselage at the forward cabin window. Individual passenger seats are provided, with the front seats installed facing aft.

"END"

GENERAL - MAINTENANCE PRACTICES

STRUCTURAL REPAIR

WARNING

Drilling, modification, or any type of work which creates a break in the pressure vessel is considered the responsibility of the owner or facility performing the work. Obtaining approval of the work is therefore, their responsibility.

In general, structural repair methods used on the BEECHRAFT Duke may be in accordance with AC 43. 13-2 AIRCRAFT INSPECTION AND REPAIR MANUAL. Never make a skin replacement or patch from a material thinner than the original skin. Patches should be of the next thicker material. The following considerations are recommended in addition to AC 43. 13-2 AIRCRAFT INSPECTION AND REPAIR MANUAL for repair of the pressure vessel of the Duke:

CAUTION

In the pressurized area, all skins, formers, stringers, etc., are structural members and should be treated as such.

a. All lap joints, including patches, must have at least two staggered rows of rivets.

b. All repair material must be free of any defects such as nicks, scratches, etc., which can cause stress rises.

c. Never dimple a structural member by driving the rivet head into the part.

d. Do not countersink deeper than 75% of the material thickness.

e. Scratches in the outer windows of acrylic plastic may be removed with 400 to 600 grit sandpaper, providing that not more than 0.30 inch of material is removed. Polish the repaired area smooth with buffing compound. The minimum thickness of the pilot's compartment side windows is 9/32 inch and for the cabin and baggage compartment windows, 7/32 inch. No crazing or cracks are permitted in the pressure windows.

REPAIR OF FIBERGLASS COMPONENTS

a. 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 portion of the part extending at least 2 inches beyond the cutout.

b. Prepare 3 patches of laminated glass cloth, such as Trevano, Uniglass, or their equivalent. Cut the first patch to the dimensions of the sanded area, the second patch 1/2-inch smaller than the first, and the third patch 1/2-inch smaller than the second.

c. Prepare the MIL-R-7575 resin (33, Chart 207, 91-00-00), for the patch in accordance with the manufacturer's instructions. Make sure that your hands are free of oil, grease, and dirt when handling the resin.

d. Apply an even coat of resin to the sanded area. Impregnate all three laminated glass cloth patches by laying the patches on clean waxed paper and working the resin through the fabric with a 2-inch brush.

e. Place the large patch over the cutout area, working out all air bubbles and wrinkles. If the patch starts to sag, place a support behind the repair area. Coat the support with automobile wax or waxed paper to prevent the resin from adhering to the support. Work out all air bubbles and wrinkles while installing the second patch over the first. Install the third patch over the second in the same manner.

f. Brush the repaired area with an even coat of resin. After the patches have cured for 24 hours at temperatures between 23°C (75°F) and 66°C (150°F), blend the patch into the contour of the part with fine sandpaper. Paint the repair to match the rest of the part.

"END"

CHAPTER 52

LIST OF PAGE EFFECTIVITY

CHAPTER SECTION SUBJECT	PAGE	DATE
52-EFFECTIVITY	1	Apr 18/80
52-CONTENTS	1	Apr 18/80
52-00-00	1	Nov 2/73
52-10-00	201	Apr 18/80
	202	Apr 18/80
	203	Apr 18/80
	204	Apr 18/80
52-60-00	201	Apr 18/80
	202	Apr 18/80
	203	Apr 18/80
52-70-00	201	Apr 18/80

"END"

CHAPTER 52 - DOORS

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"END"

GENERAL - DESCRIPTION AND OPERATION

The Duke is equipped with a fail safe cabin door latching mechanism. When the door latch bolts are in position, a spring-loaded secondary locking device maintains a safety locked condition. In addition, a pressure slide lock prevents inadvertent movement of either the secondary system or the door handle itself when pressurized. When the door is closed, the outside cabin door handle is spring loaded to fit into a recess in the door to create a flat, aerodynamically clean surface. The door may be locked with a key.

To open the door from the outside, press inward on the forward end of the handle to raise the aft end enough to grasp it. On serials P-123, P-127 and after, push the safety release button and lift the handle from its recess and turn it counterclockwise until the door opens. The door will swing out and forward over the left wing section. The door may be closed from the outside by rotating the handle clockwise. The three door latching bolts activate three switches mounted on the bulkhead behind the fuselage door frame. A fourth switch mounted on the door (serials P-4 through P-126, except P-123) is activated by the door handle latch mechanism. A cabin door warning light on the annunciator panel illuminates when the cabin door is not secure. All door switches must be activated to turn off the annunciator light.

To close the door from the inside, pull the door shut firmly with the handle in the forward position. Rotate the door handle aft in a counterclockwise manner until the safety lock bolt handle moves aft or the safety lock button pops outward. When the door handle has been rotated completely aft, (serials P-4 through P-126, except P-123) the safety lock bolt handle will snap forward to its original position.

At this point, the door is securely locked and cannot be opened except by moving the safety lock bolt full aft; or o the serials P-123, P-127 and after, pressing the safety loc button in. If there is residual pressure remaining in th cabin, the red "T" handle, located forward of the cabi door handle, must be pulled to override the pressur locking mechanism before the safety lock bolt or safet lock button will move. Once the safety lock bolt has bee pulled aft, or the safety lock button pressed in, the doo handle may be rotated forward to open the door.

CAUTION

If the cabin door handle is rotated in an attempt to open the door and the safety lock bolt (P-4 through P-126, except P-123) is not in the full aft position, damage may result to the safety lock bolt mechanism.

The Duke 60 Series aircraft are equipped with a retractabl assist step, attached to the fuselage under the cabin door, t aid in entering and leaving the aircraft. A cable, attached t the actuator arm on the right hand main landing gear strut extends the assist step when the landing gear is extended When the landing gear is retracted, the assist step i retracted and fits flush with the fuselage.

The CABIN DOOR light in the annunciator remain illuminated until the door is closed, latched and locked, since all three latch pin switches are wired in parallel with one another; and on serials P-4 through P-126, except P-123, with the door locked switch. When the cabin door i closed and latched, each latch pin compresses the actuator on its respective switch mounted on the bulkhead behind the aft frame of the doorway. When the cabin door handle is rotated to the locked position (serials P-4 through P-126, except P-123) a spacer at the bottom of the latch mechanism lock bolt compresses the arm on the door locked switch.

"END"

ENTRANCE STAIRS - MAINTENANCE PRACTICES

RETRACTABLE ASSIST STEP REMOVAL (Figure 201)

a. Remove the bolt (1) from the outboard side of the step (2).

b. Slide the step off far enough to clear the stops (6). Allow the step to rotate until the tension on the spring (5) is relieved.

c. Remove the step.

RETRACTABLE ASSIST STEP INSTALLATION

a. Place the spring in the step. Align the spring end in the corresponding hole in the step retainer (3).

b. Slide the step on the strut assembly shaft (4) and rotate counterclockwise until the remaining spring end is in place in the hole in the strut assembly shaft.

c. Continue rotating the step until the stops are aligned. Slide the step on until the stops engage.

d. Replace the bolt in the outboard side of the step.

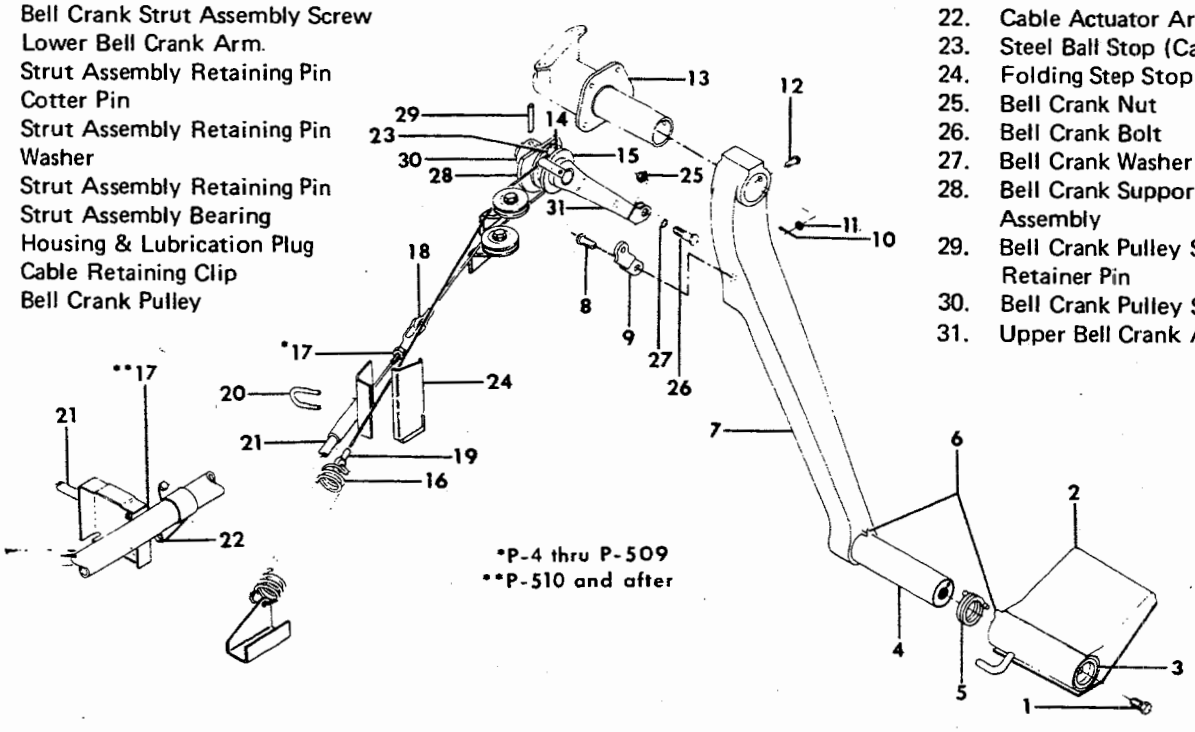
STEP ADJUSTMENT (FOLDED POSITION)

If the step is not flush with the skin when in the folded position it may be adjusted. This is done by loosening the two bolts in the stop (24), and sliding it up or down as needed.

STRUT ASSEMBLY REMOVAL (Figure 201)

a. Remove the small access plate below the cabin door in the area of the strut assembly (7).

1. Step Retaining Bolt
2. Cabin Step
3. Step Retainer
4. Step Assembly Shaft
5. Step Extension Spring
6. Step Extension Stops
7. Strut Assembly
8. Bell Crank Strut Assembly Screw
9. Lower Bell Crank Arm.
10. Strut Assembly Retaining Pin Cotter Pin
11. Strut Assembly Retaining Pin Washer
12. Strut Assembly Retaining Pin
13. Strut Assembly Bearing Housing & Lubrication Plug
14. Cable Retaining Clip
15. Bell Crank Pulley
16. Strut Assembly Extension Spring
17. Clevis Adjusting Rod End
18. Cable Clevis
19. Bell Crank Actuator Cable
20. U-bolt Cable Clamps
21. Covered Cable
22. Cable Actuator Arm
23. Steel Ball Stop (Cable)
24. Folding Step Stop
25. Bell Crank Nut
26. Bell Crank Bolt
27. Bell Crank Washer
28. Bell Crank Support Assembly
29. Bell Crank Pulley Shaft Retainer Pin
30. Bell Crank Pulley Shaft
31. Upper Bell Crank Arm



Retractable Assist Step
Figure 201

b. Remove the access plate from the bottom of the airplane in the area of the strut assembly.

c. Remove the assembly screw (8) which connects the lower bell crank arm (9) to the strut assembly.

d. Remove the cotter pin (10), washer (11) and pin (12) which secures the strut assembly to the bearing housing (13).

e. Remove the six strut assembly attaching bolts revealed by removing the bottom panel, and remove the bearing housing.

STRUT ASSEMBLY INSTALLATION

a. Position the bearing housing (13) through the bottom panel and install the six strut assembly attaching bolts. Install the bottom panel.

b. Position the strut assembly (7) to the bearing housing and secure with the pin (12), washer (11), and cotter pin (10).

c. Position the bell crank arm (9) to the strut assembly and secure with the attaching screw (8).

d. Install the access plate to the bottom of the airplane in the area of the strut assembly.

e. Install the small access plate below the cabin door in the area of the strut assembly.

CABLE REMOVAL (P-4 THRU P-509)
(Figure 201)

a. Remove the large access plate below the cabin door.

b. Remove the L-shaped cable retaining clip (14) from the bell crank pulley (15).

c. Lower the flaps.

d. Release tension from the extension spring (16) by backing off the clevis adjusting rod end (17) from the clevis.

e. Separate the bell crank actuator cable at the cable clevis (18).

f. Disconnect the bell crank actuator cable (19) from the extension spring.

NOTE

The cable/return spring connection is accessible through holes which are exposed when the flaps are in the down position.

g. Remove the bell crank actuator cable.

h. Remove the long access plate inboard and slightly aft of the RH main landing gear door.

i. Remove the U-bolt clamps (20) from each end of the covered cable (21).

j. Detach the covered cable from the actuator arm (22) on the landing gear strut.

k. Note the routing of the covered cable and remove.

CABLE INSTALLATION (P-4 THRU P-509)

a. When installing either of the cables, peel laminations may be removed as required to allow the cable actuator arm (22) to rotate with 3 ± 2 inch-pounds torque.

NOTE

When installing the cables, be certain the steel ball stop (23) on the cable is in place on the bell crank pulley and tighten the rod end into the clevis to a depth of one inch. This is all that is necessary to rig the step travel.

b. Position the covered cable and route as noted during removal.

c. Attach the covered cable to the actuator arm (22) on the landing gear strut.

d. Install the U-bolt clamps (20) to each end of the covered cable.

e. Install the long access plate inboard and slightly aft of the RH main landing gear door.

f. Position the bell crank actuator cable and connect to the extension spring (16) and the clevis adjusting rod end (17).

g. Raise the flaps.

h. Install the L-shaped cable retaining clip (14) to the bell crank pulley (15).

i. Install the large access plate below the cabin door.

CABLE REMOVAL (P-510 AND AFTER)
(Figure 201)

a. Remove the large access plate below the cabin door.

b. Remove the L-shaped cable retaining clip (14) from the bell crank pulley (15).

c. Remove the long access plate inboard and slightly aft of the RH main landing gear door.

d. Release the tension from the extension spring (16) by backing off the clevis adjusting rod end (17) from the cable actuator arm (22), located on the landing gear strut.

e. Lower the flaps.

f. Separate the bell crank actuator cable at the cable clevis (18).

g. Disconnect the bell crank actuator cable (19) from the extension spring.

NOTE

The cable/return spring connection is accessible through holes which are exposed when the flaps are in the down position.

h. Remove the bell crank actuator cable.
i. Remove the U-bolt clamps (20) from each end of the covered cable (21).
j. Detach the covered cable from the actuator arm (22) on the landing gear strut.
k. Note the routing of the covered cable and remove.

CABLE INSTALLATION (P-510 AND AFTER)

a. When installing either of the cables, peel laminations may be removed as required to allow the cable actuator arm (22) to rotate with 3 ± 2 inch-pounds torque.

NOTE

When installing the cables, be certain the steel ball stop (23) on the cable is in place on the bell crank pulley and tighten the rod end into the clevis to a depth of one inch. This is all that is necessary to rig the step travel.

b. Position the covered cable and route as noted during removal.
c. Position the bell crank actuator cable (19) and connect to the extension spring (16) and the non-adjustable clevis end of the covered cable.
d. Attach the clevis adjusting rod end (17) of the covered cable to the actuator arm (22) on the landing gear strut.
e. Install the U-bolt clamps (20) to each end of the covered cable.
f. Install the long access plate inboard and slightly aft of the RH main landing gear door.
g. Raise the flaps.
h. Install the L-shaped cable retaining clip (14) to the bell crank pulley (15).
i. Install the large access plate below the cabin door.

BELL CRANK ASSEMBLY REMOVAL (Figure 201)

a. Remove the screw that attaches the bell crank to the strut assembly.
b. Remove the bell crank nut (25), bolt (26) and washer (27) from the upper bell crank arm (31). Remove the lower bellcrank arm (9).
c. Remove the access plate on the under side of the airplane, near the bell crank.
d. Remove the access plate below the cabin door in the area of the bell crank.
e. Remove the bell crank actuator cable as described in CABLE REMOVAL.
f. Locate the bell crank support assembly (28) in the lower access opening and remove the bell crank pulley shaft retainer pin (29) from the bell crank pulley shaft (30).
g. Remove the upper bell crank arm (31), the bell crank pulley (15) and bell crank pulley shaft (30).

BELL CRANK ASSEMBLY INSTALLATION

a. Position the upper bell crank arm (31), pulley (15) and shaft (30) to the bell crank support assembly (28) and install the retainer pin (29) to the bell crank shaft (30).
b. Install the bell crank actuator cable as described in CABLE INSTALLATION.
c. Install the access plate below the cabin door in the area of the bell crank.
d. Install the access plate on the underneath side of the airplane, near the bell crank.
e. Install the lower bell crank arm (9) to the upper bell crank arm (31) and secure with the attaching bolt (26), washer (27) and nut (25).
f. Install the screw that attaches the bell crank to the strut assembly.

"END"

DOOR WARNING - MAINTENANCE PRACTICES

The door locked and door latched switches will not normally require adjustment except when a new switch is installed.

a. Adjust the latch pin switches as follows:

NOTE

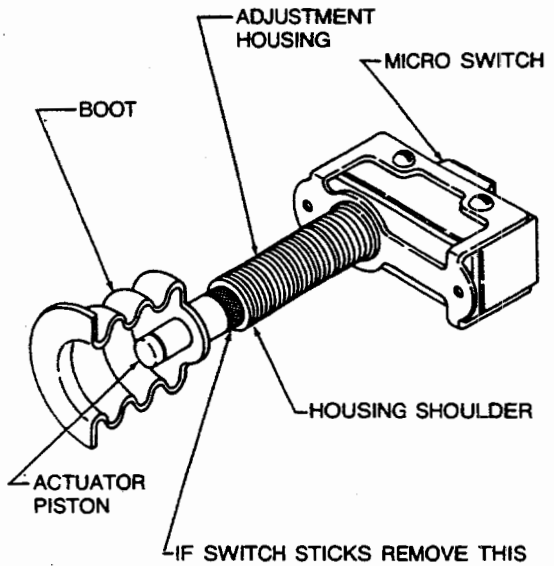
Before making adjustments to the latch pin switches be sure the latch pins are properly adjusted as outlined under DOOR LATCH ADJUSTMENT in this chapter.

1. Remove the cabin upholstery panels adjacent to the upper, lower and center latches to gain access to the switches.
2. Close and lock the cabin door.
3. Rotate the cabin door handle clockwise against the lock mechanism stop while in the locked position. This will eliminate the effect of play in the lock mechanism.
4. Back the switch adjustment off until the switch is not actuated.
5. Readjust the switch until it actuates.
6. Adjust upper and lower switch overtravel to .12 to .20 inch and middle switch overtravel to .07 to .11 inch.

NOTE

One complete revolution of the adjusting nut will provide .031 inch travel.

7. Reopen the cabin door and note if the switch remains in the actuated position due to the friction of the rubber boot on the actuating piston. If this occurs remove the safety wire on the boot, trim one-half of that portion of the boot which rides the actuating piston as indicated in Figure 201 and slide the boot along the piston until it rests against the shoulder of the adjustment housing.
8. Install new safety wire on the boot and close, lock and open the door to check for proper operation.
9. Install the cabin upholstery panels.



60-104-13

Switch Boot Modification
Figure 201

b. Adjust the door locked switch (P-4 thru P-126, except P-123) as follows:

1. Remove the upholstery panel under the window of the cabin door to gain access to the door locked switch.
2. Loosen the attaching screws and position the switch in its mounting slots so that the CABIN DOOR light goes out when the door is closed, latched, and locked.

"END"

CHAPTER 53

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CHAPTER 53 - FUSELAGE

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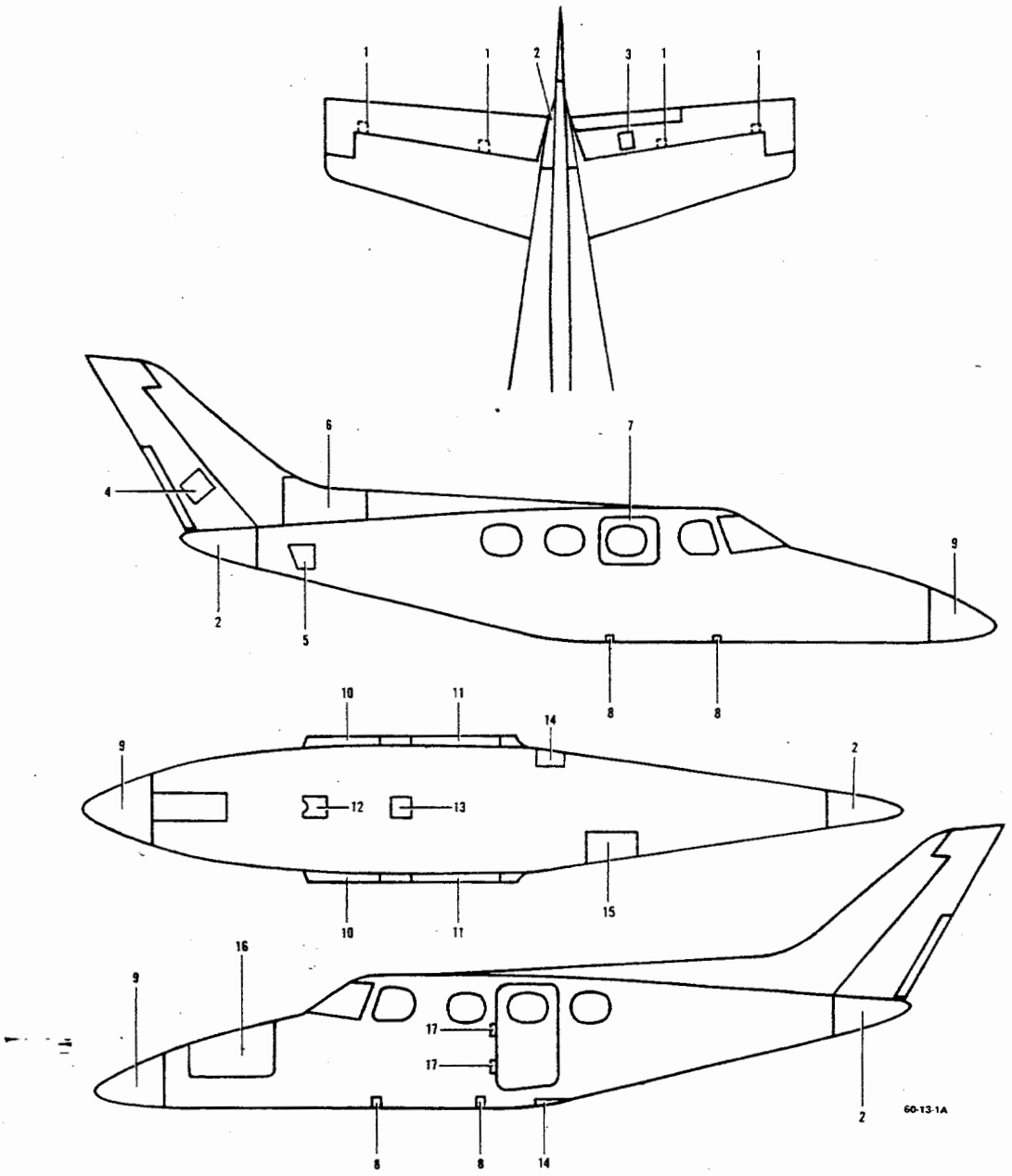
"END"

PLATES/SKIN - MAINTENANCE PRACTICES

FUSELAGE ACCESS OPENINGS

The panels, plates and doors as shown in Figure 201, provides maintenance access to the components, plumbing and cables enclosed within the fuselage. When installed, they continue the aerodynamic lines of the fuselage with little increase of drag.



Fuselage Access Openings
Figure 201 (Sheet 1 of 2)

1. Elevator Hinges
2. Tail Cone
3. Elevator Trim Tab Actuator
4. Rudder Trim Tab Actuator
5. Deicer Dump Valve, Elevator Bell Crank, Rudder Bell Crank and Control Cables
6. Vertical Stabilizer Deicer Hose
7. Emergency Exit
8. Wing Bolts
9. Nose Cone
10. Intercooler
11. Uplock Cable, Retract Rods, Retract Rod Pressure Seals and Wing Stub Fuel Fittings
12. Lower Pedestal
13. Landing Gear, Gearbox and Actuator
14. Assist Step Mechanism
15. Autopilot Aileron, Rudder and Elevator Servos, Magnetic Navigation Sensing Element (B-5P Autopilot) and Autopilot Computer (H-14 Autopilot)
16. Oxygen Filler and Bottle, Heater, Ram Air Plenum, Brake Fluid Reservoir, Avionics Equipment and Baggage Compartment
17. Cabin Door Hinges

Fuselage Access Openings
Figure 201 (Sheet 2 of 2)

"END"

CHAPTER 55

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	203	Feb 27/81

"END"

CHAPTER 55 - STABILIZERS

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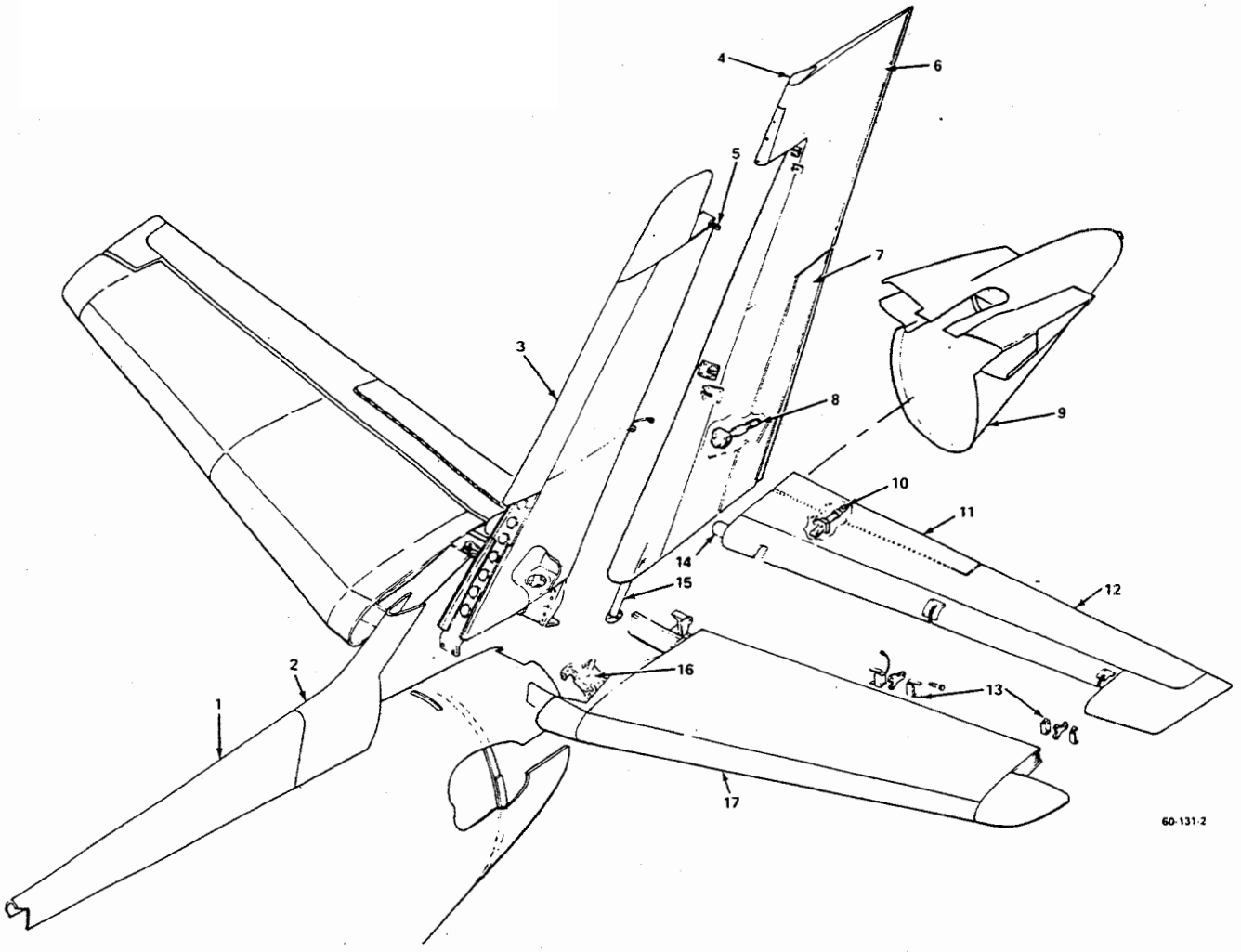
"END"

GENERAL - DESCRIPTION AND OPERATION



1. Dorsal Fairing
2. Dorsal Saddle Fairing
3. Vertical Stabilizer
4. Beacon
5. Rudder Hinge Bracket
6. Rudder
7. Rudder Trim Tab
8. Rudder Trim Tab Actuator
9. Tail Cone
10. Elevator Trim Tab Actuator
11. Elevator Trim Tab
12. Elevator
13. Elevator Hinge Brackets
14. Elevator Torque Tubes
15. Rudder Torque Tube
16. Rudder Bell Crank
17. Horizontal Stabilizer

Empennage
Figure 1

"END"

HORIZONTAL STABILIZER - MAINTENANCE PRACTICES

HORIZONTAL STABILIZER REMOVAL

a. Remove the tail cone and elevators. (Refer to Chapter 27-30-00.)

b. Remove the dorsal saddle fairing and lower vertical stabilizer fairings.

c. Remove the access cover on the lower aft right side of the fuselage.

d. Disconnect the surface deicer tubes at the horizontal stabilizers.

e. Remove the bolts attaching the horizontal stabilizers to the fuselage bulkheads.

f. Carefully pull the horizontal stabilizers outboard and clear of the fuselage.

HORIZONTAL STABILIZER INSTALLATION

a. Carefully move the horizontal stabilizers inboard into position at the fuselage bulkheads and install the attaching bolts. Torque all 5/16-24 bolts to 100 to 140 inch-pounds and 3/8-24 bolts to 160 to 190 inch-pounds.

NOTE

The attaching bolts shall have no threads bearing against structural members. Proper torque must be maintained without bolt rotation in the bolt holes.

b. Connect the surface deicer tubes at the horizontal stabilizers.

c. Install the access cover on the lower aft right side of the fuselage.

d. Install lower vertical stabilizer fairings and the dorsal fairing.

e. Install the elevators and tail cone. (Refer to Chapter 27-30-00.)

"END"

ELEVATOR - MAINTENANCE PRACTICES

ELEVATOR BALANCING
(Figure 201)

After repainting and/or repair, the finished elevator must be check balanced to ensure that its static moment about the hinge line is within the prescribed limits. The static moment for all completed elevator assemblies must fall within the range of 12.9 to 25.1 inch-pounds underbalance (tail heavy) at the measured moment about the hinge line. The static moment of the elevator is determined by multiplying the unbalanced weight of the elevator 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 percent balanced elevator assembly is 0.0 inch-pounds. Tail heaviness indicates static underbalance while nose heaviness indicates static overbalance.

NOTE

Control surfaces ordinarily need not be rebalanced unless they are repainted, repaired or have parts replaced. When repainting, hang the control surfaces by the trailing edge so excess paint will drain toward the leading edge.

The balance weight of the right elevator assembly is manufactured of lead or steel shot and epoxy resin. This weight is constructed in such a manner that adding weight is not recommended, but material may be removed to reduce weight. This weight should be sufficiently heavy to allow repainting and still check balance within the range of 12.9 to 25.1 inch-pounds underbalance (tail heavy). Even though the right elevator balance weight cannot be added to, the elevator MUST have the balance checked any time the elevator is repainted, repaired, or any work is done which might affect the balance.

CHECKING BALANCE

The balance must be checked in a draft free area with the elevator 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, the simplest is 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.

EQUIPMENT REQUIRED TO PERFORM CHECK BALANCING

a. A stand with knife edge supports as illustrated in Figure 201. The knife edges must be in the same horizontal plane.
b. A cup or similar light weight container.
c. Approximately 3 pounds of lead shot.
d. A certified beam balance weighing device calibrated in units of 0.01 pound or less.
e. A straight edge, ruler, and spirit level.

BALANCING PROCEDURE

COUNTERBALANCING METHOD

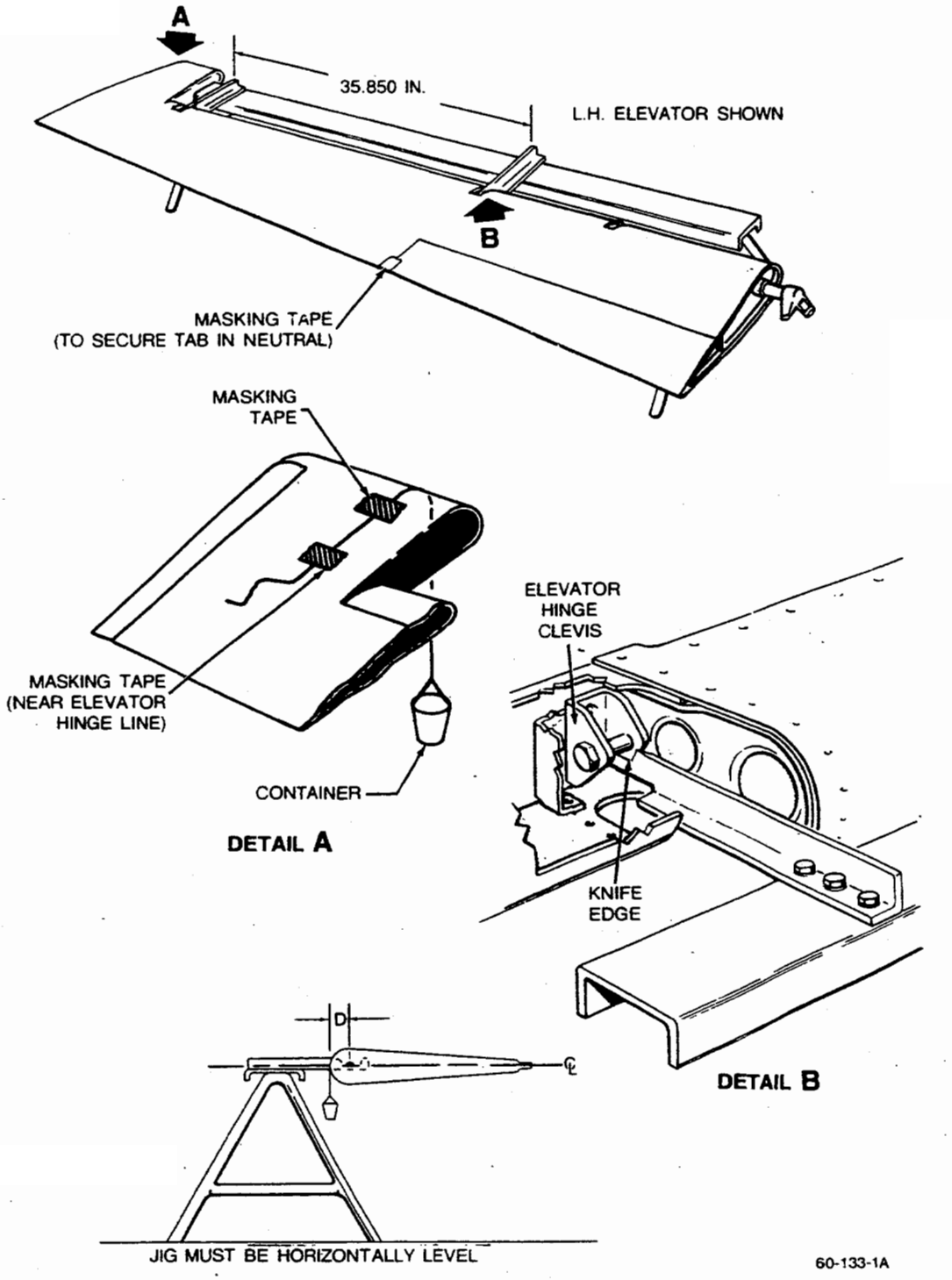
a. Locate the chord line by placing a stright edge at the inboard end of the elevator assembly so that one end is on the hinge center line (at the center of the torque tube) and the other end is centered on the trailing edge. Mark the chord line with a suitable marker, such as a grease pencil, then remove the straight edge.
b. Secure the trim tab in its neutral position with a small piece of masking tape.
c. 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.
d. To determine if weight should be added or removed, suspend a cup from a point near the inboard end of the balance weight assembly on the elevator leading 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.
e. 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.
f. Carefully measure the perpendicular distance "D" within 0.1 inch from the hinge line to the point of suspension of the cup.
g. Remove the cup, contents, and string, then weight 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.

h. Calculate the static balance as follows:

1. The weight of the cup and contents is designated by "W".



Balancing the Elevator
Figure 201

2. The underbalance moment is designated by "M".

3. M = W x D

4. The following is a typical example of a balancing calculation: Suspend a cup from the leading edge and add the required amount of lead shot. If the elevator balances with the chord line level at "W = 2.00 pound" and "D = 10.0 inches", then . . .

M = 2.00 x 10.0

M = 20.00 inch-pounds (The product of "W x D" must be accurate to within 0.10 inch-pounds.) In this instance, "M" is within the required static balance range and is therefore acceptable.

i. If the static balance does not fall within the range of 12.9 to 25.1 inch-pounds tail heavy (underbalance), weight must be added or removed (left elevator only) and the balance rechecked.

Remove the balance weight (left elevator only) and add or remove solder to bring the elevator balance within required limits. Coat the weight with a corrosion preventative material such as zinc chromate primer to insulate the dissimilar metals. Replace the balance weight and recheck the balance.

"END"

VERTICAL STABILIZER - MAINTENANCE PRACTICES

VERTICAL STABILIZER REMOVAL

a. Remove the tail cone and rudder. (Refer to Chapter 27-20-00.)

b. Remove the dorsal saddle fairing and the lower vertical stabilizer fairings.

c. Disconnect the antenna coaxial cable.

d. Disconnect the surface deicer tubes at the vertical stabilizer.

e. Remove the bolts attaching the vertical stabilizer to the fuselage bulkheads.

f. Carefully raise the vertical stabilizer clear of the fuselage.

VERTICAL STABILIZER INSTALLATION

a. Carefully lower the vertical stabilizer into position at the fuselage bulkheads and install the attaching bolts. Torque all 5/16-24 bolts to 100 to 140 inch-pounds and 3/8-24 bolts to 160 to 190 inch-pounds.

NOTE

The attaching bolts shall have no threads bearing against structural members. Proper torque must be maintained without bolt rotation in the bolt holes.

b. Connect the surface deicer tubes at the vertical stabilizer.

c. Connect the antenna coaxial cable.

d. Install the lower vertical stabilizer fairings and the dorsal saddle fairing.

e. Install the rudder and tail cone. (Refer to Chapter 27-20-00.)

"END"

RUDDER - MAINTENANCE PRACTICES

RUDDER BALANCING (Figure 201)

After repainting and/or repair, the finished rudder must be check balanced to ensure that its static moment about the hinge line is within the prescribed limits. The static moment for all completed rudder assemblies must fall within the range of 21.1 to 31.2 inch-pounds underbalance (tail heavy) at the measured moment about the hinge line. 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. The static moment of a 100 percent balanced rudder assembly is 0.0 inch-pounds. Tail heaviness indicates static underbalance while nose heaviness indicates static overbalance.

CHECKING BALANCE

The 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, anti-collision light, chain and cable assembly, 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, the simplest is counterbalancing: The application of a known force of weight at a measured distance from the hinge line to counter the unbalance moment of the rudder assembly.

EQUIPMENT REQUIRED TO PERFORM CHECK BALANCING

a. A stand with knife edge supports as illustrated in Figure 201. The knife edges must be in the same horizontal plane.

b. A can or similar light weight container.

c. Approximately 9 pounds of lead shot.

d. A certified beam balance weighing device calibrated in units of .01 pound or less.

e. A straight edge, ruler, and spirit level.

BALANCING PROCEDURE

COUNTERBALANCING METHOD

a. 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.

b. Secure the trim tab in its neutral position with a small piece of masking tape.

c. Fit the correct size bolts in the hinge brackets and mount the rudder on the knife edge supports. Ensure that the rudder is free to rotate about the hinge line.

d. Suspend a can from a point on the leading edge directly above the lower hinge skin cutout. 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 can must be free to hang vertically.

e. Add small quantities of lead shot to the can until the rudder balances with the chord line level. Check this by holding a spirit level aligned with the marked chord line.

f. Carefully measure the perpendicular distance "D" within 0.1 inch from the hinge line to the point of suspension of the can.

g. Remove the can, 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.

h. Calculate the static balance as follows:

1. The weight of the can and contents is designated by "W".

2. The underbalance moment is designated by "M".

3. M = W x D

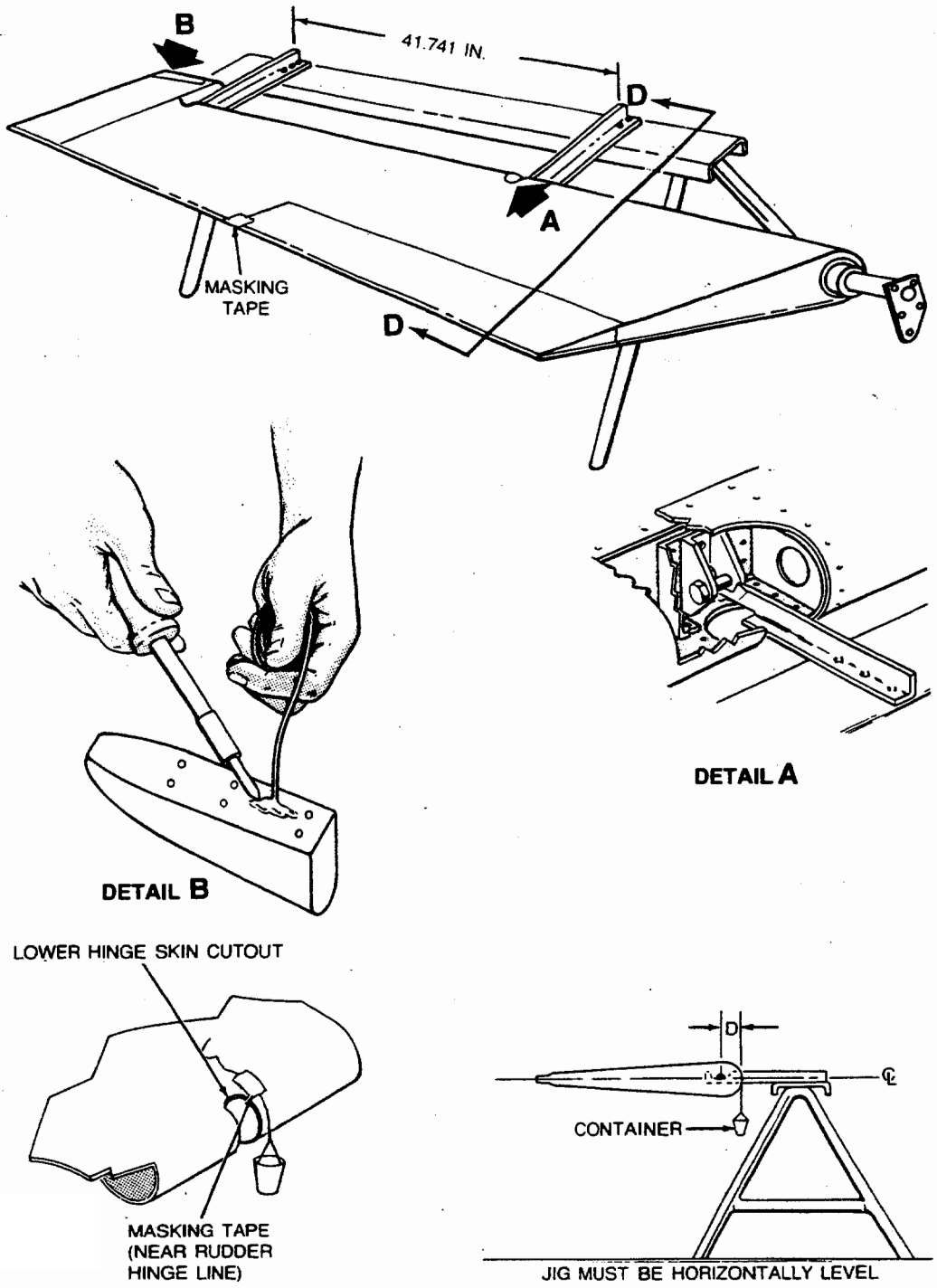
4. The following is a typical example of a balancing calculation: If the rudder balances with the chord line level at "W = 8.00 pound" and "D = 3.5 inches", then . . .

M = 8.00 x 3.5

M = 28.00 inch-pounds (The product of "W x D" must be accurate to within 0.10 inch-pounds.) In this instance, "M" is within the required static balance range and is therefore acceptable.

i. If the static balance does not fall within the range of 21.1 to 31.2 inch-pounds underbalance, remove the rudder horn weight and add or remove solder to bring the rudder balance within the required limits.

j. The weight of the solder to be added or removed is calculated as follows:



Balancing the Rudder
Figure 201

1. The weight of solder to be added or removed is designated "W_1".

2. The moment difference between the actual measurement and the required moment is designated "M_1".

3. The perpendicular distance from hinge center line to the point of solder removal or addition is designated "D_1".

4. $W_1 = M_1 \div D_1$

5. The following are typical examples of required solder changes:

a. If the rudder balances at "M_1 = 32.0 inch-pounds" then "M_1 = 32.0 - 31.2 or 0.8 inch-pounds" and "D_1 = 8.5 inches" then W_1 = 0.8 ÷ 8.5 or 0.09 pounds of solder to be added.

b. If the rudder balances at "M_1 = 20.0 inch-pounds", then "M_1 = 20.0 - 21.1 or -1.1 inch-pounds" and "D_1 = 8.5 inches", then W_1 = -1.1 ÷ 8.5 or -0.13 pounds of solder to be removed.

"END"

CHAPTER 56

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CHAPTER 56 - WINDOWS

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"END"

GENERAL - MAINTENANCE PRACTICES

CLEANING PLASTIC WINDOWS

A commercial cleaning compound made specifically for acrylic plastic windows may be used. When using a commercial cleaner follow the instructions on the container.

If a commercial cleaner is not available, these instructions should be followed:

Cleaning of the acrylic plastic windows should never be attempted when dry. The window should first be flushed with water or a mild soap solution, then rubbed slightly with a grit-free soft cloth, chamois or sponge. Stubborn grease or oil deposits are readily removed with aliphatic naphtha or hexane. Rinse with clear water.

CAUTION

Do not use thinner or aromatic abrasive cleaners to clean the windows as they will damage the surface of the plastic. Aliphatic naphtha and similar solvents are highly inflammable, and extreme care must be exercised when used.

"END"

FLIGHT COMPARTMENT - MAINTENANCE PRACTICES

WINDSHIELD REMOVAL
(Figure 201)

a. Remove the windshield heater wiring from the lower corners of the windshield (heated windshield only).
b. Remove the screws that hold the retainer strips on the front of the windshield.
c. Remove the retainer strips from around the windshield.
d. Remove the windshield.
e. Remove the old sealer from the retainer strips and window frame with toluol.

NOTE

Do not allow the toluol to come in contact with the plastic window as it will craze the surface.

WINDSHIELD INSTALLATION
(Figure 201)

a. It is suggested that Y9136 teflon tape (8, Chart 205, 91-00-00) should be applied to all areas of the windshield that will come in contact with the windshield frame or the windshield retainer strips. The tape will keep the sealer from bonding permanently to the windshield and will facilitate future windshield removal procedures.
b. Apply PR1221 sealer (9, Chart 205, 91-00-00) to the windshield frame.
c. Place the windshield in position and secure by starting an attaching screw in each corner. Fill the gap between the windshield and the windshield frame with PR1221 sealer (9, Chart 205, 91-00-00).
d. Place the retainer strips in position and secure in place with the attaching screws. Tighten the screws (in alternate rows progressively) to a torque of 16 to 20 inch-pounds.

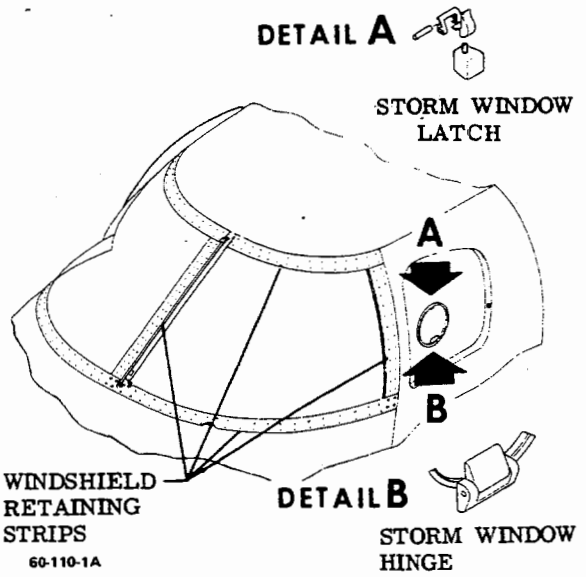
CAUTION

Overtightening the screws will cause damage to the windshield.

e. Connect the windshield heater wiring to the lower corners of the windshield as they were on the old windshield (heated windshield only).

STORM WINDOW LATCH REMOVAL
(Figure 201)

a. Using an Allen wrench in the end of the latch attaching bolts to prevent the bolts from turning, remove the nuts from the bolts.
b. Remove the latch.



Windshield Replacement
Figure 201

NOTE

Care should be taken not to damage the attaching bolts or the plexiglass block to which the hinge attaches. If either of these are damaged beyond use, the entire window assembly must be replaced.

STORM WINDOW LATCH INSTALLATION
(Figure 201)

a. Position the latch on the storm window.
b. Using an Allen wrench in the end of the latch attaching bolts to prevent the bolts from turning, install the nuts on the bolts and secure.

STORM WINDOW HINGE REMOVAL
(Figure 201)

a. Loosen the two set screws in the plexiglass hinge block and punch the hinge pin out from one end.
b. Remove the hinge block.

NOTE

The hinge pin and the set screws are the only replaceable parts in the hinge. If either hinge block is damaged, the window assembly to which it is attached must be replaced.

STORM WINDOW HINGE INSTALLATION (Figure 201)

a. Position the hinge block and install the hinge pin.
b. Secure the hinge pin with the set screws.

FLIGHT COMPARTMENT SIDE WINDOW REMOVAL (Figure 202)

a. Remove the screws from the inner window frame and remove the window frame.
b. Remove the L-shaped retaining clips from around the window.
c. Remove the window. If the window is not bonded tightly to the airplane skin, remove the window intact. If the window is tightly bonded, the bulk of the window may be cut away, leaving a narrow strip of window bonded to the airplane skin. This strip may then be peeled or broken away.

CAUTION

Do not use a sharpened metal tool or knife to cut the sealant from around the window or damage to the area may result.

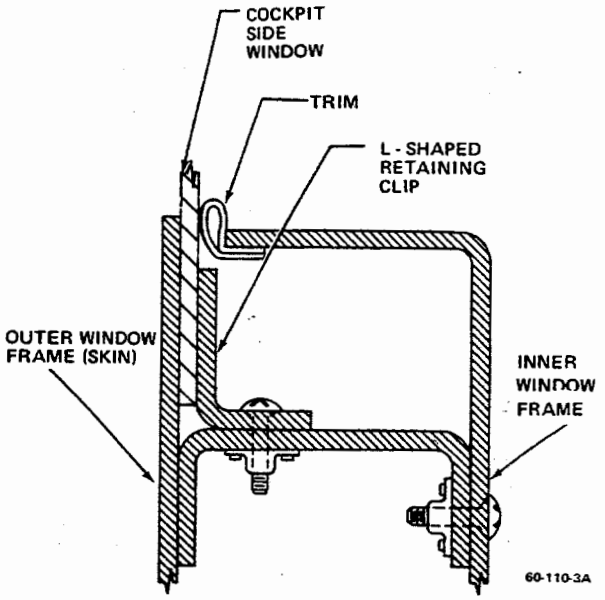
d. Clean the old sealant from the airplane skin by scraping with a sharpened wood or plastic tool. Sand lightly to remove all traces of the old sealant.

FLIGHT COMPARTMENT SIDE WINDOW INSTALLATION (Figure 202)

a. Clean all bonding surfaces with naphtha (20, Chart 207, 91-00-00). Clean both the airplane skin and the plastic window bonding surfaces.
b. Rough sand the bonding surface, of both the airplane skin and the window, using 60 grit emery cloth. Exercise caution not to sand the surface of the window which will be exposed.
c. Thoroughly clean each sanded surface, using a clean white rag dampened with naphtha (20, Chart 207, 91-00-00). Continue cleaning until no discoloration appears on the rag.

NOTE

Do not use red shop rags for cleaning, as they contain oil.



Flight Compartment Side Window
Figure 202

d. Immediately prior to applying the sealant, clean the metal part (bond area only), using EC3911 degreasing primer (45, Chart 207, 91-00-00). Remove all traces of powder by brushing or wiping with a clean, lint free cloth.

CAUTION

Do not apply EC3911 degreasing primer to fiberglass or plastic parts.

e. Prepare the sealant, EP711 (5, Chart 205, 91-00-00), per the manufacturers instructions.
f. Apply a uniform layer of the prepared sealant, 1/16 inch to 1/8 inch thick over the contact bond area of the airplane skin.
g. Carefully position the window in place. Apply sufficient pressure to force a small amount of sealant out at the periphery of the bond area. Maintain a light pressure while securing the window, using the L-shaped retaining clips and attaching screws.
h. Permit the sealant to cure.
i. After the sealant has cured, use a sharpened wood or plastic tool to remove the excess fillet of sealant on the exposed surface of the window.
j. Clean the window using naphtha (20, Chart 207, 91-00-00).
k. Install the inner window frame. Secure it, using the attaching screws.

"END"

CABIN - MAINTENANCE PRACTICES

CABIN SIDE WINDOW REMOVAL (Figure 201)

a. If the curtains are installed, remove the spring clips from each end of the window curtain tracks and slide the curtains to the center of the window.

b. Remove the attaching screws (exposed by moving the curtains) from the inner window frame and remove the inner window frame.

c. Remove the attaching screws from around the inner window. Remove the inner window and the inner window molding. See Figure 201, Cabin Side Window.

d. Remove the U-shaped retaining clips.

e. Remove the window. If the window is not bonded tightly to the airplane skin, remove the window intact. If the window is tightly bonded, the bulk of the window may be cut away, leaving a narrow strip of window bonded to the airplane skin. This strip may then be peeled or broken away.

CAUTION

Do not use a sharpened metal tool or knife to cut the sealant from around the window or damage to the area may result.

f. Clean the old sealant from the airplane skin by scraping with a sharpened wood or plastic tool. Sand lightly to remove all traces of the old sealant.

CABIN SIDE WINDOW INSTALLATION (Figure 201)

a. Clean all bonding surfaces with naphtha (20, Chart 207, 91-00-00). Clean both the airplane skin and the plastic window bonding surfaces.

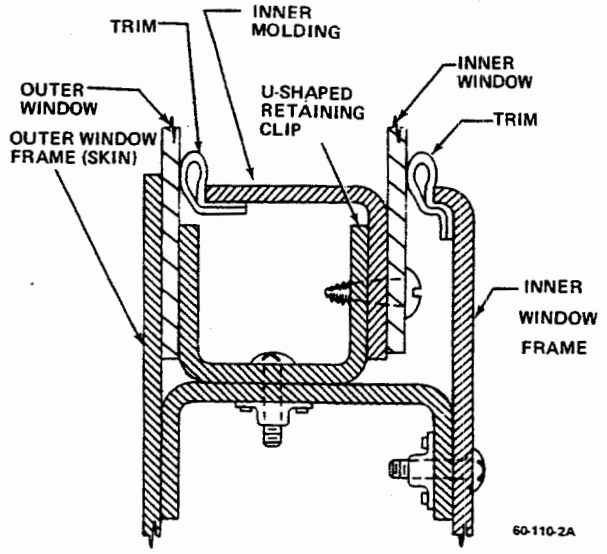
b. Rough sand the bonding surface, of both the airplane skin and the window, using 60 grit emery cloth. Exercise caution not to sand the surface of the window which will be exposed.

c. Thoroughly clean each sanded surface, using a clean, white rag dampened with naphtha (20, Chart 207, 91-00-00). Continue cleaning until no discoloration appears on the rag.

NOTE

Do not use red shop rags for cleaning, as they contain oil.

d. Immediately prior to applying the sealant, clean the metal part (bond area only), using EC3911 degreasing primer (45, Chart 207, 91-00-00). Remove all traces of powder by brushing or wiping with a clean, lint free cloth.



Cabin Side Window
Figure 201

CAUTION

Do not apply EC3911 degreasing primer to fiberglass or plastic parts.

e. Prepare the sealant, EP711 (5, Chart 205, 91-00-00), per the manufacturers instructions.

f. Apply a uniform layer of the prepared sealant, 1/16 inch to 1/8 inch thick over the contact bond area of the airplane skin.

g. Carefully position the window in place. Apply sufficient pressure to force a small amount of sealant out at the periphery of the bond area. Maintain a light pressure while securing the window, using the U-shaped retaining clips and attaching screws.

h. Permit the sealant to cure.

i. After the sealant has cured, use a sharpened wood or plastic tool to remove the fillet of excess sealant on the exposed surface of the window.

j. Clean the window using naphtha (20, Chart 207, 91-00-00).

k. Install the inner window molding and position the inner window, secure to the U-shaped outer window retaining clips with the attaching screws.

l. Install the inner window frame in position and secure with the attaching screws.

m. If curtains are installed, slide the curtains into position and install the spring clips.

"END"

CHAPTER 57

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CHAPTER SECTION SUBJECT	PAGE	DATE
57-EFFECTIVITY	1	Jun 13/84
57-CONTENTS	1	Nov 30/83
57-00-00	1	Nov 2/73
	201	Nov 30/83
	202	Nov 30/83
	203	Nov 30/83
	204	Nov 30/83
	205	Jun 13/84
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	209	Nov 30/83
	210	Jun 13/84
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	215	Nov 30/83
	216	Nov 30/83
	217	Nov 30/83
57-30-00	201	Oct 27/75
	202	Oct 27/75
57-40-00	201	Nov 2/73
57-50-00	201	Feb 27/81
	202	Feb 27/81
	203	Feb 27/81

"END"

CHAPTER 57 - WINGS

TABLE OF CONTENTS

"END"

GENERAL - DESCRIPTION AND OPERATION

The all-metal wing group consists of the fuselage carry-through structure, outboard wing panels, leading edge, wing tips, flaps, aileron and aileron trim tabs, and the integral fuel cells. The wing tips, flaps, ailerons, and fuel cells are readily removable. To remove the wing assembly, engine removal is required.

CARRY-THROUGH STRUCTURE

The carry-through structure, to which the wing assemblies are attached, is riveted to the fuselage and forms an integral part thereof. The upper forward carry-through extruded spar cap is of clad 2024-T3 aluminum alloy; while the lower spar cap is of 2014-T6 aluminum alloy. A web of clad 2024-T3 aluminum alloy sheet encloses the area between both spar caps. The aft (one piece) extruded spar cap is of clad 2416-T6 aluminum alloy.

OUTBOARD WING

Two spars, their attaching ribs and skin, constitute the box beam construction used throughout the wing. The outer wing spars are of the same construction as the carry-through structure, except that a combination of clad 2014-T4 aluminum alloy extrusions and formed clad 2024-T4 aluminum alloy U-channel members comprise the main spar caps while those of the rear spar are composed of formed clad 2024-T3 and 2024-T4 aluminum alloy angles and clad 2024-T3 cap strips. The stamped ribs and formed stringers used throughout the wing are of clad 2024-T3 aluminum alloy. Clad 2024-T3 aluminum skin covers the entire wing. The wing tips are formed of clad 6061-T4 aluminum alloy sheets and are attached to the wing with screws. Two fuel cells are located in the leading edge, and a nacelle and box section cell is located between the main and rear spar in each wing assembly. Each fuel cell cavity is lined with clad 2024-T3 aluminum alloy sheet.

"END"

GENERAL - MAINTENANCE PRACTICES

WING TIP REMOVAL

NOTE

On aircraft equipped with pneumatic surface deicers, the deicer boot must be removed from the wing tip before the wing tip can be removed. Refer to Chapter 30-10-00 for deicer boot removal and installation procedures.

a. Remove the two access plates, located on the lower side of the wing tip.

b. Loosen the clamps and disconnect the three vent lines from the fuel vent float valve.

c. Remove the screws attaching the wing tip to the wing.

d. Disconnect the electrical leads to the navigation and landing lights.

WING TIP INSTALLATION

a. Connect the electrical leads to the navigation and landing lights.

b. Position the wing tip to the wing and install the attaching screws.

c. Install the three vent lines on the fuel vent float valve. Torque the hose clamps to 25 ± 5 inch-pounds.

d. Reinstall the deicer boots (If applicable).

e. Reinstall the access plates on the lower side of the wing tip.

WING REMOVAL

a. Drain and purge all fuel cells.

b. Remove the wing mounting bolt access plates from the top and bottom of the wing.

c. Place the aircraft on a three point jack to prevent an unbalanced condition of the airplane after the wing is removed.

d. Place a wing stand under the wing that is not being removed and place a stand under the tail. Place two adjustable screw jacks under the wing being removed, one jack just inboard of the nacelle and one near the wing tip.

e. Remove the engine as instructed in Chapter 71-00-00.

f. Open the brake cylinder bleed ports and pump all fluid from the system.

g. Retract the landing gear until the inboard landing gear doors are fully open.

h. Disconnect the inboard door actuating rod from the control horn.

i. Disconnect the landing gear actuator rod from the V-brace in the wheel well.

j. Disconnect the aileron cables at the turnbuckles in the wheel well and remove the roll pins from the inboard aileron cable pulley brackets. Disconnect the aileron tab cables and aileron tab stops in the left wheel well.

k. Disconnect the hydraulic brake line at the inboard connection in the wheel well.

l. Disconnect the fuel lines between the wing root rib and the fuselage.

m. Remove the leading edge cover of the wing located between the fuselage and nacelle.

n. Disconnect the pressurization ducting in the leading edge of the wing stub, and disconnect the firewall shutoff control cable.

o. Disconnect the flap drive shaft at the flap actuator and remove the clamps attaching the shaft housing to the wing.

p. Remove the lower aft nacelle fairing assembly.

q. Remove the inboard nacelle fairing.

r. Remove the clamps securing the wire bundles to the wing inboard leading edge. Disconnect the wire bundles at the terminals located on the aft side of the firewall.

s. Disconnect the wiring to the electrical components located in each side of the upper nacelle.

t. Disconnect and cap all plumbing between the wing root rib and the fuselage.

WARNING

The two air conditioner lines between the right wing root rib and the fuselage are high pressure lines. Before disconnecting the two lines, loosen the fitting just enough to bleed off the pressure slowly.

u. Disconnect the flap wire bundle and safety switch wiring in the left wheel well. Disconnect the plumbing and electrical wiring (boost pump and fuel quantity transmitter) in the wheel well.

v. Remove the clamps securing the engine controls to the leading edge.

w. Position two support jacks under the wing.

NOTE

Outline the position of the wing on the fuselage, using a grease pencil. This will aid realignment when the wing is reinstalled.

CAUTION

If bolt binding occurs, adjust the position of the wing until the bolt disengages freely. Do not screw or drive a bolt into, or out of the fittings.

x. Remove the wing attach bolts from the fittings.
y. Remove the wing by pulling it straight away from the fuselage.

NOTE

The soft aluminum washers between the upper wing attach fittings and the preload indicating washer under the nut at the lower forward wing attach point must be discarded and new components installed when the wing is reinstalled.

WING INSTALLATION

a. Using a nonmetalic brush and naphtha or methyl ethyl ketone (20 or 21, Chart 207, 91-00-00), clean the wing attach fittings and hardware (bolts, washers, and nuts). Inspect the wing attach fittings and attaching hardware as instructed under WING BOLT, NUT, AND FITTING INSPECTION.

WARNING

Wing bolts and nuts that have reached their life limit (10 years after the initial inspection) must not be reused (see Chart 202).

b. Coat the fitting bolt bores and bearing faces, bolts, washers and nuts with MIL-C-16173 Grade II corrosion preventive compound (43, Chart 207, 91-00-00).
c. Place the slide in the fuselage fitting at the leading edge attach point as shown in Figure 205.
d. Guide the flap shaft and landing gear retract rod into their respective positions.
e. Align the wing and fuselage fittings, install the new soft aluminum washers between the upper wing attach fittings, and insert the bolts into the fittings.

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 through the fittings. Do not screw or drive a bolt into, or out of the fittings. Bolts, nuts and washers must be oriented as shown in the applicable illustration for each location (Figure 201, 202, 203, 204 and 205).

f. Start the nuts on the bolts and rotate the wing trailing edge until the wing aligns with the outline drawn on the fuselage. After alignment is established, verify that the lower forward bolt is not binding on the bolt bore. If bolt binding is encountered, adjust the position of the wing until the bolt moves freely in the fittings.

CAUTION

When torquing the wing nuts, assure that the wrenches do not come into contact with the wing attach fittings. Such an occurrence could result in damage to the fittings and false torque readings.

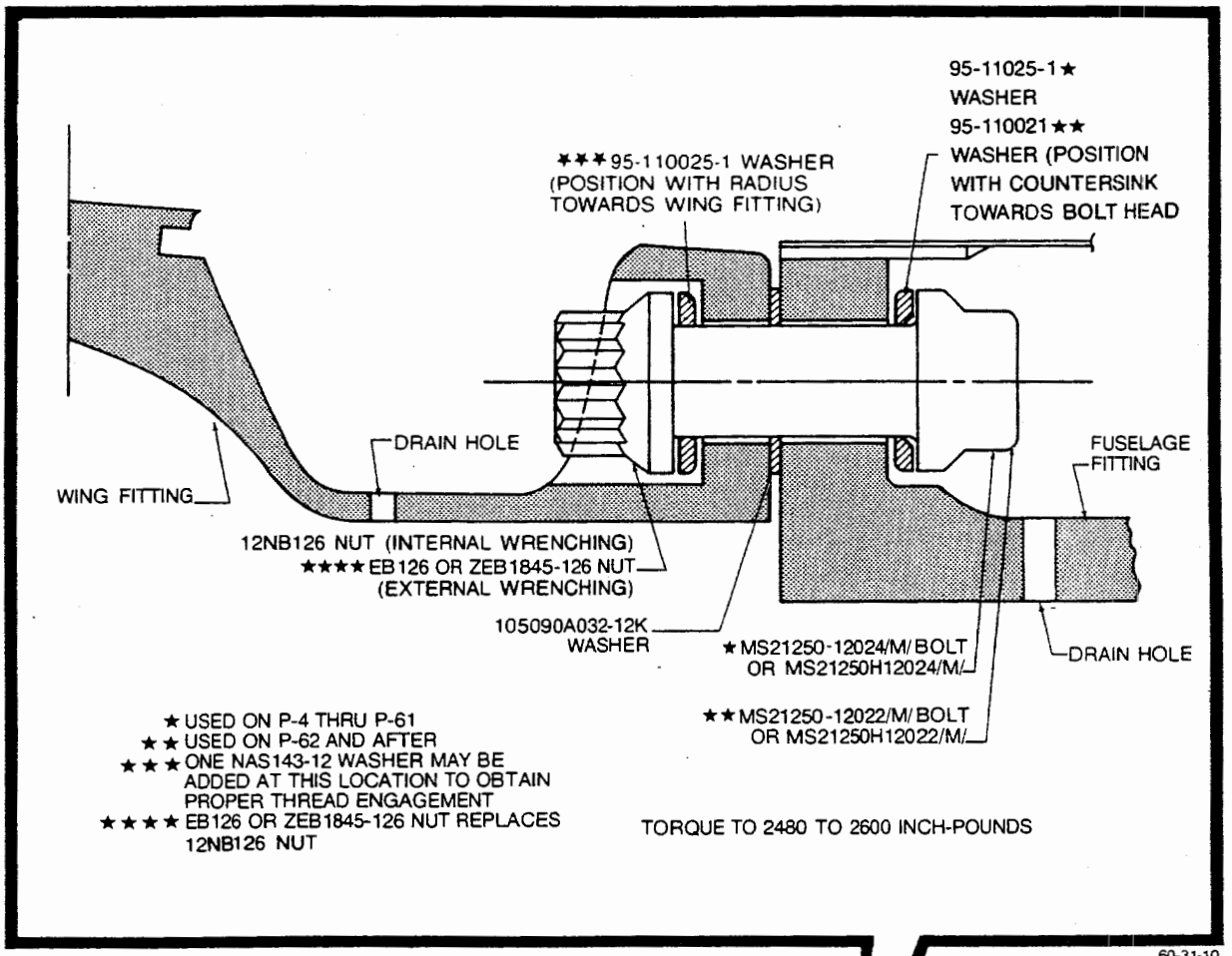
g. Tighten the upper forward nut and remove the holding force from the wing cradle (if used). Torque the remaining three nuts in the following order: upper aft, lower forward, and lower aft. When a torque wrench adapter is used, the length of the adapter must be added to the length of the torque wrench and the proper torque value computed as detailed in Chapter 20-00-00.

NOTE

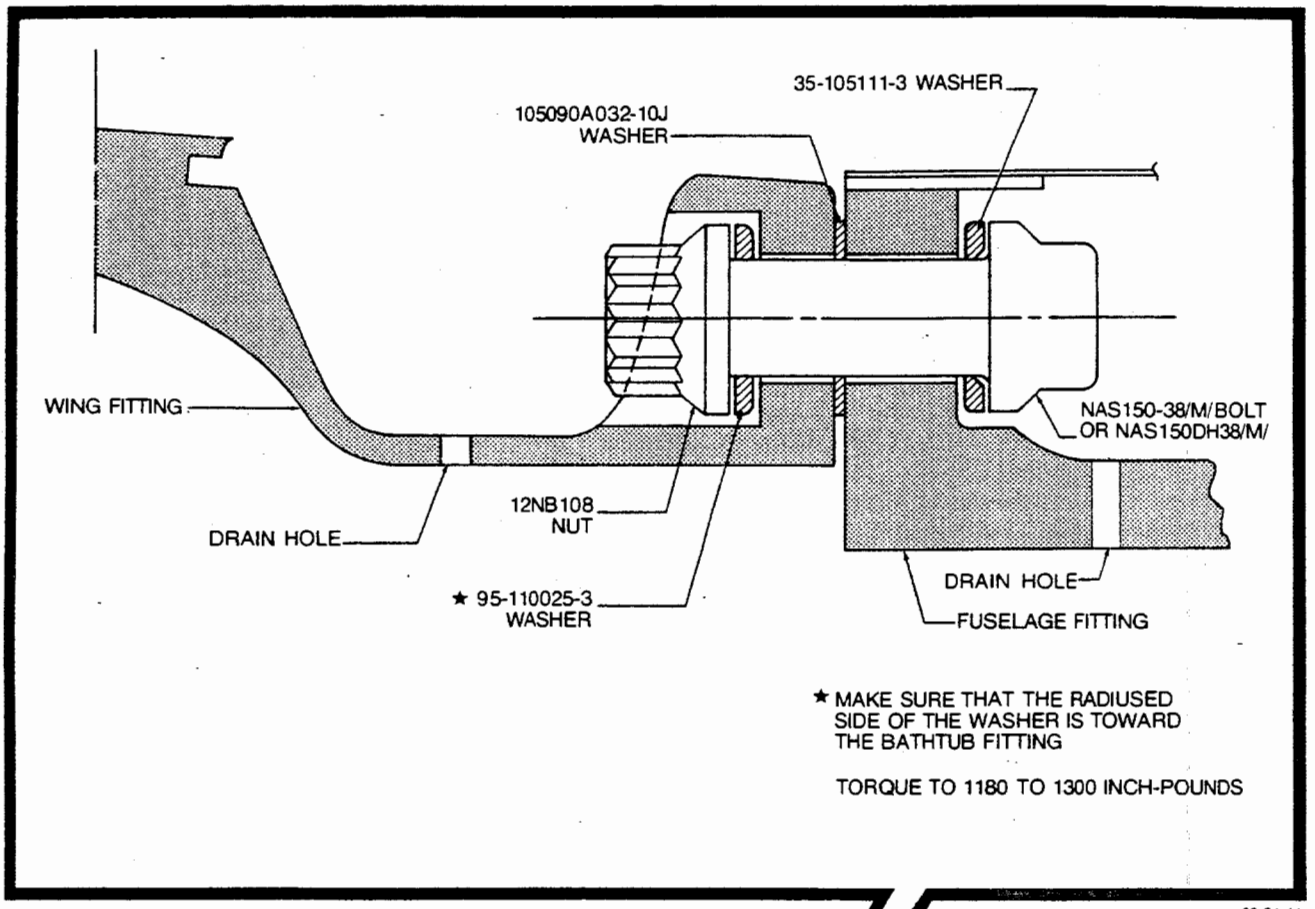
Each nut must be torqued to the value shown in the appropriate illustration for each location (Figure 201, 202, or 204). However, the lower forward attach point is not torqued to a specific torque value and must be tightened as instructed in Figure 206.

CAUTION

Before the lower aft nut is torqued, a slight gap may exist between the fittings. This gap must not exceed a width of .030 inch. No gap should remain after the nut is torqued. Torque the wing attach bolts at the nut end. Do not rotate the bolt in the bolt bore.

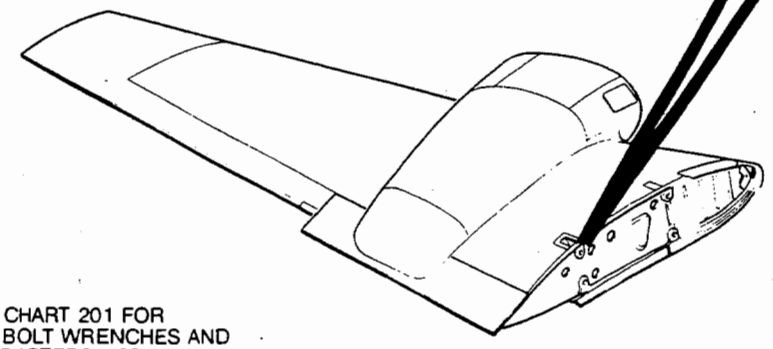


Upper Forward Wing Bolt Installation
Figure 201



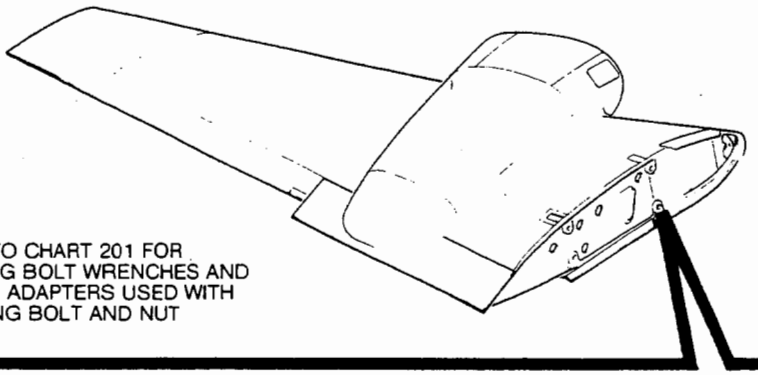
CAUTION

INSPECT THE WING ATTACH FITTINGS TO ENSURE THAT THE DRAIN HOLES ARE CLEAR.



REFER TO CHART 201 FOR THE WING BOLT WRENCHES AND TORQUE ADAPTERS USED WITH THIS WING BOLT AND NUT.

Upper Aft Wing Bolt Installation
Figure 202

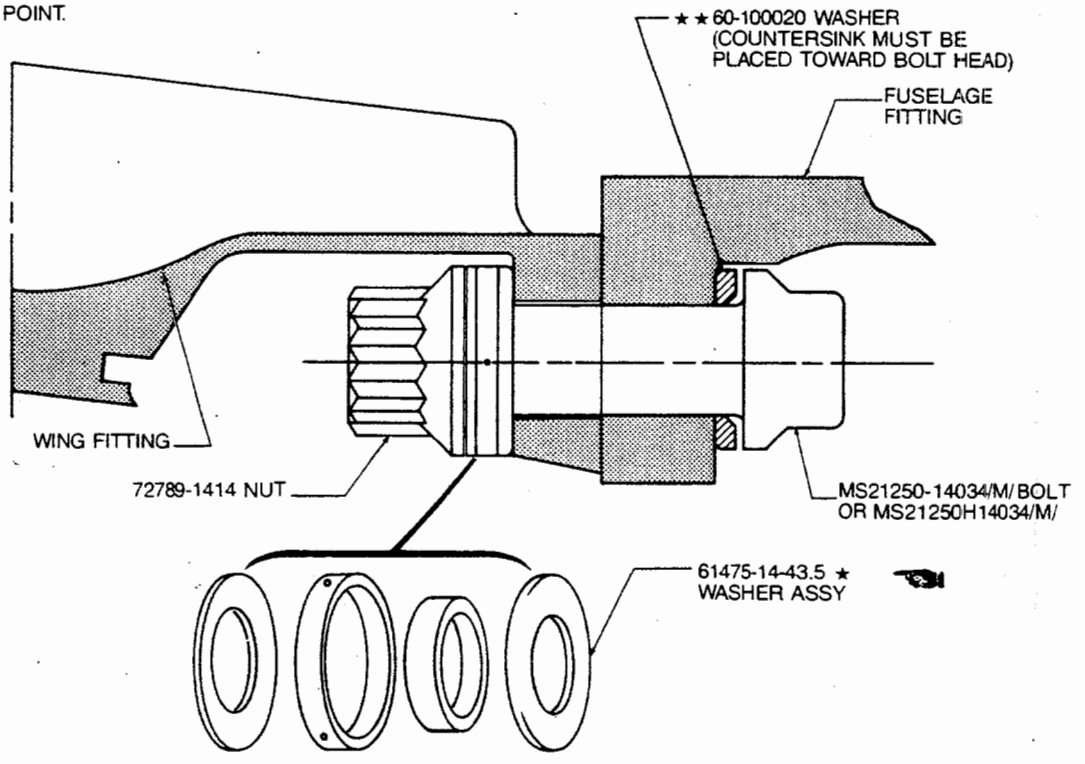


REFER TO CHART 201 FOR THE WING BOLT WRENCHES AND TORQUE ADAPTERS USED WITH THIS WING BOLT AND NUT

★★ MAKE SURE THAT THE RADIUSED SIDE OF THE WASHER IS TOWARD THE FITTING.

WHEN 30 ± 5 POUNDS OF TANGENTIAL FORCE WILL NO LONGER MOVE THE INDICATOR RING ON THE PRELOAD INDICATING WASHER THE BOLT IS TIGHT. DO NOT TIGHTEN THE BOLT BEYOND THIS POINT.

★ DO NOT MIX WITH PARTS FROM OTHER PACKAGES. THE WASHER ASSEMBLY CONSISTS OF MATCHED (PACKAGED) PARTS AND IS ASSEMBLED AS FOLLOWS:
(1) FLAT WASHER WITH SQUARE SHOULDERS: UNDER NUT.
(2) RINGS, DIAMETERS ALLOW NESTING: BETWEEN WASHERS.
(1) FLAT WASHER WITH (1) RADIUSED EDGE: RADIUSED EDGE NEXT TO WING FITTING.



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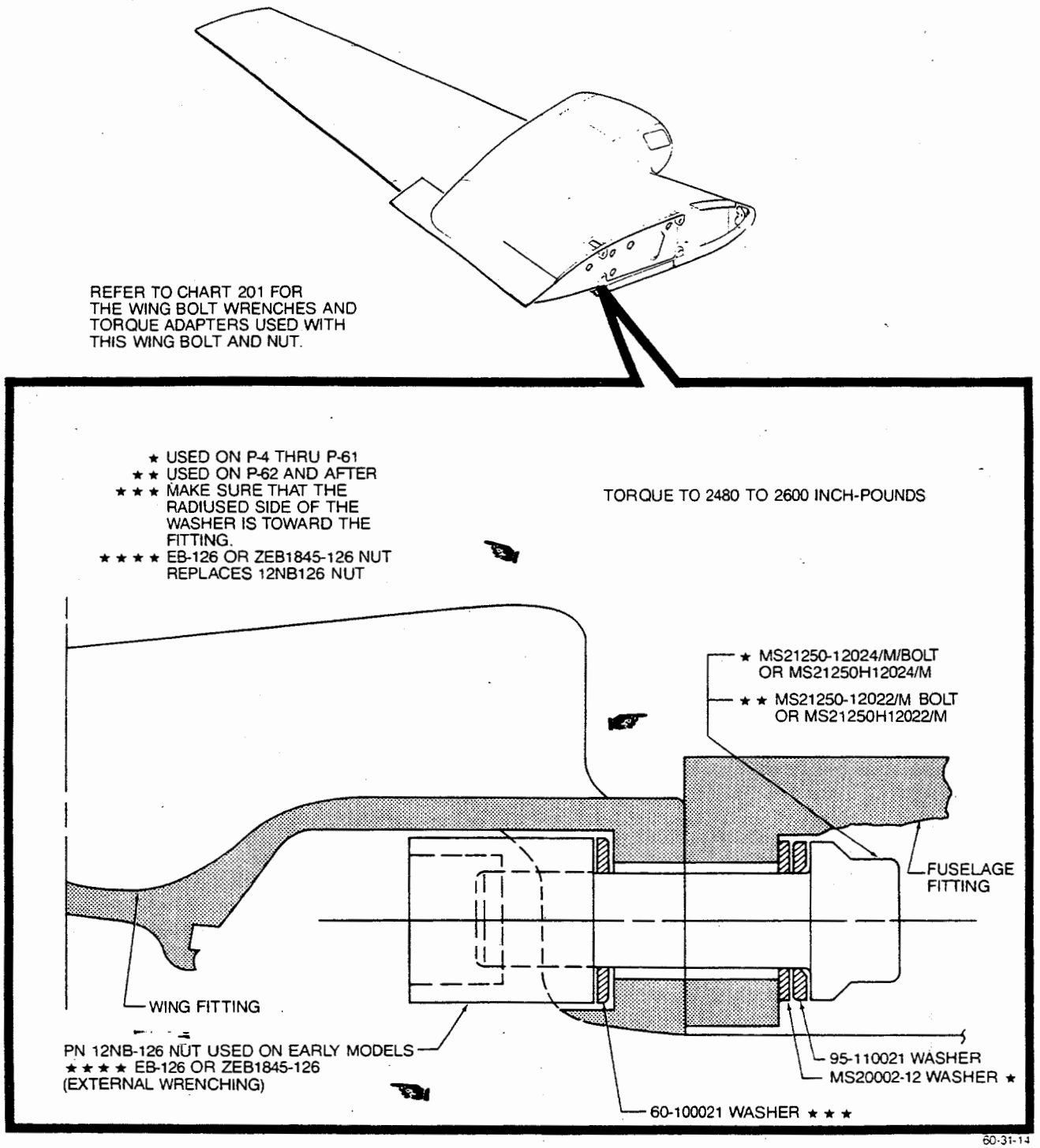
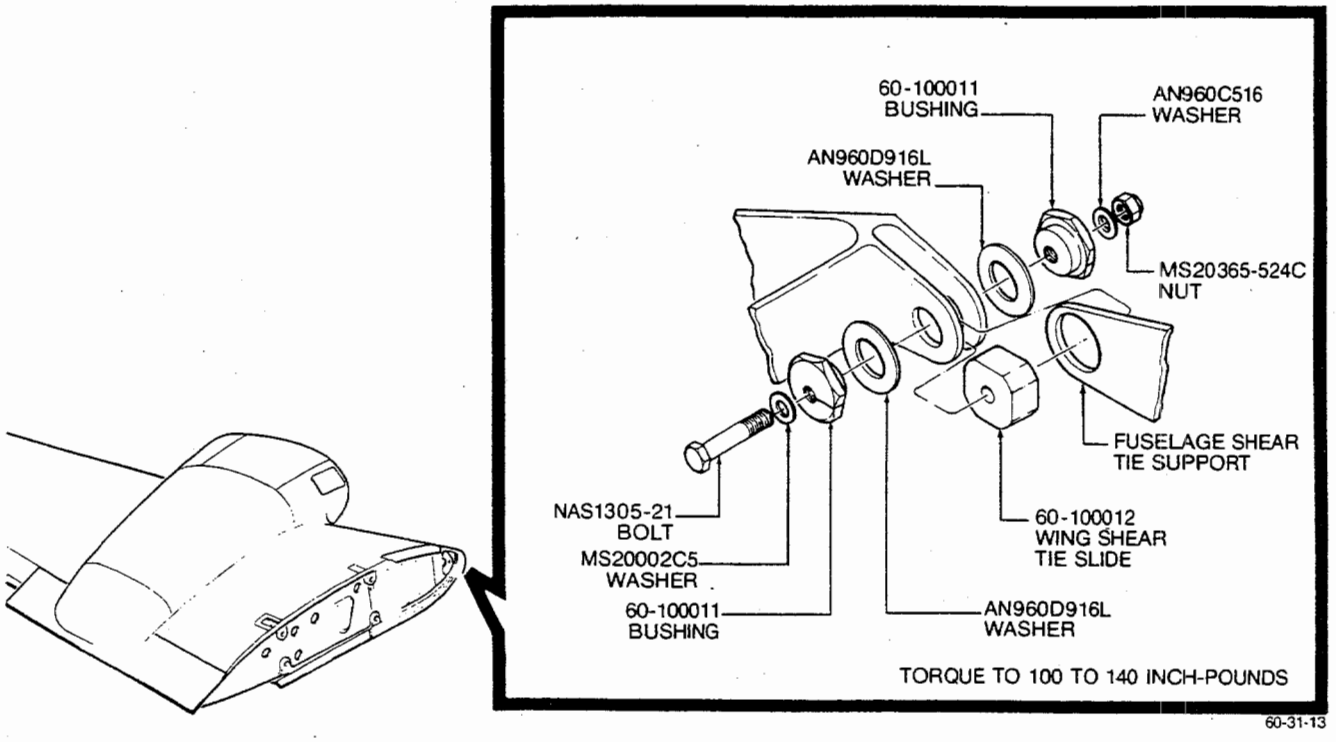
Lower Forward Wing Bolt Installation
Figure 203



Lower Rear Wing Bolt Installation
Figure 204



The Forward And Aft Travel Of The 60-100011 Bushings Must Not Exceed .025 Inch, After The Bolt Is Torqued.

Leading Edge Attach Point
Figure 205

h. Install the attaching components in the leading edge fitting. See Figure 205 for proper arrangement of the components. When the components are properly arranged, torque the bolt to the value shown in Figure 205. Check the bushings to assure that the scribe marks align.

CAUTION

Do not lubricate the fittings or attach hardware at the leading edge attach point (Figure 205). The torque value shown in this figure is for dry hardware only.

i. Coat the threads that protrude through the nuts at the forward and aft spar attach points with MIL-C-16173 Grade II corrosion preventive compound (43, Chart 207, 91-00-00).

j. Route the engine control cables along the leading edge, through the engine firewall, and secure in place with clamps.

k. Route the electrical wire bundles along the leading edge and secure in place with clamps, connect wire ends to the terminals on the aft side of the firewall.

l. Connect all fuel, air condition, and deicing plumbing between the wing root and fuselage.

m. Connect the pressurization ducting and fuel selector cable in the leading edge of the wing stub.

n. Install the inboard leading edge cover.

o. Connect the electrical wiring to the fuel boost pump and transmitter.

p. Connect the flap and safety switch wiring in the left wheel well.

q. Connect the flap drive shaft at the flap actuator and clamp the shaft housing to the wing.

r. Install the roll pins in the aileron and aileron trim tab pulley brackets.

s. Install the aileron tab cable stops and connect the aileron cables and the tab cables to the turnbuckles. Rig the aileron control system as instructed in Chapter 27-10-00.

t. Connect the landing gear and the inboard main gear door actuating rods. Check the landing gear rigging as instructed in Chapter 32-30-00.

u. Connect the brake hydraulic line and bleed the brake system as instructed in Chapter 32-40-00.

v. Install the engine as instructed in Chapter 71-00-00.

w. Charge the air condition system with refrigerant as instructed in Chapter 21-50-00.

x. Install all removed access plates and covers.

y. Install the nacelle fairings.

z. Remove the airplane jack and service the fuel cells as instructed in Chapter 12-10-00. Check for fuel leaks.

aa. Perform an engine run-up; check and adjust, a necessary, as instructed in Chapter 71-00-00.

ab. Test fly the airplane, and adjust the wing an engine, as necessary.

ADJUSTING THE WING

After a wing is installed or repaired, flight tests may sho one wing to be chronically heavy. This condition may b corrected by altering the angle of incidence, using th following procedure:

CAUTION

When adjusting the wing, always replace the soft aluminum washers and the 61475-14-43.5 washer assembly at the forward lower wing attach point. Check the torque at the first 100-hour inspection by making certain the center outer ring of the 61475-14-43.5 washer assembly does not turn by finger pressure. Check the remaining wing bolts for proper torque at the first 100-hour inspection after a wing has been installed.

a. Raise the trailing edge of the light wing t decrease its lift as follows:

1. Mark the position of the wing on the rea wing bolt fittings.

2. Loosen the mounting nut at the lower rea wing fitting. Remove the mounting bolt and nut at the lowe forward wing fitting. Remove and replace the 61475-14-43. washer assembly at the lower forward attachment a outlined under CAUTION in this procedure ADJUSTIN THE WING. Loosen the mounting bolt on the forward fitting

3. Remove the upper mounting bolt and nut from the two upper wing fittings. Install new aluminur washers between the upper wing and fuselage fittings Raise the wing trailing edge, install the upper wing mountin bolts and nuts and torque all the wing nuts to the specifie torque.

4. Flight test the airplane. If the same wing i still heavy, accomplish step "b".

b. Lower the trailing edge of the heavy wing t increase its lift as follows:

1. Mark the position of the wing on the rea wing bolt fittings.

2. Loosen the mounting bolt nut at the lowe rear wing fitting. Remove the mounting bolt nut at the lowe

forward wing fitting. Remove and replace, the 61475-14-43.5 washer assembly at the lower forward attachment as outlined under CAUTION in this procedure. Loosen the mounting bolt on the forward fitting.

3. Remove the upper mounting bolt and nuts from the two upper wing fittings. Install new aluminum washers between the upper wing and fuselage fittings. Lower the wing trailing edge, install the upper wing mounting bolts and nuts, and torque all the wing nuts to the specified torque.

c. If the combination of steps "a" and "b" does not correct the wing heavy condition, rig the flap down on the heavy wing by screwing the actuator out. Do this only as a last resort, since it will create a drag on the airplane.

1. RH Wing Heavy

(a) Disconnect the flap actuator from the flap and screw the actuator arm out to eliminate the wing heavy condition.

(b) Connect the flap actuator to the flap.

2. LH Wing Heavy

(a) Lower the flaps to provide access to flap up limit switch.

(b) Loosen the attaching screws on the flap up limit switch.

(c) Adjust the position of the switch in the elongated holes to rig the flap down enough to eliminate the wing heavy condition.

(d) Tighten the attaching screws on the flap up limit switch.

(e) Disconnect the flap actuator from the RH flaps and screw the actuator arm in to bring the RH flap up to the same position it was before the limit switch was adjusted.

(f) Connect the flap actuator to the RH flap.

Adjust the flaps only as a last resort, since it will create a drag on the airplane.

WING DISASSEMBLY

a. Support the wing on a suitable cradle.

b. Remove the wing tip, aileron, wing flap, fuel cells and other equipment as required by the work to be accomplished.

c. Remove the screws around the spar caps and the root ribs.

d. Vise-grip pliers may be used to remove the steel hinge pins. Remove the pins from the box section first, then the leading edge.

CAUTION

Do not attempt to spin the hinge pins out with a drill motor; the heating and expansion of the pin will cause the pin to seize in the hinge and break.

WING ASSEMBLY
(Figure 207)

a. Before assembling the spar to the wing sections it is advisable to drive the hinge pins through the hinge sections to remove any burrs and foreign material.

b. Use a new hinge pin, liberally coated with graphite (29, Chart 207, 91-00-00).

c. Position the spar on the leading edge and align the hinge sections.

d. Using an E-2, or equivalent size, rivet gun and the telescoping tube kit (P/N 35-588S), drive the hinge pin in until the tip is completely through the hinge, but not against the wing attach fitting. The pin must be supported with the telescoping tubes during the driving operation. Start the pointed end of the pin in the hinge and support the pin with the longest tubes against the hinge, then drive the pin. Remove the tubes as necessary, until the pins are completely inserted. Trim the hinge pins as necessary to extend 4.88 ± .12 inches beyond the end of the spar. Bend the end of the pins at a 90° angle and install retaining plates.

NOTE

It is imperative that the larger tube be held firmly against the hinge throughout the driving procedure in order to prevent the pin from kinking in the intervening space.

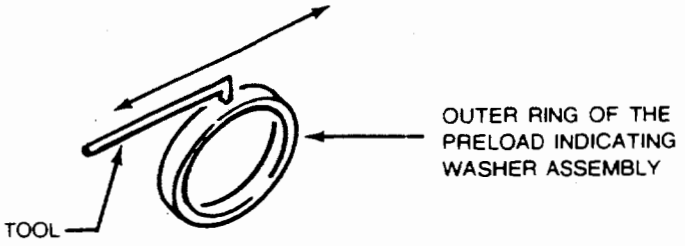
e. Install the box section in the same manner as the leading edge.

NOTE

If necessary, place a phenolic block against the spar and vibrate the spar with another rivet gun.

f. Install the screws around the spar caps and root ribs.

g. Install all components which had been removed.



To tighten the lower forward wing attach bolts, insert the tool into one of the holes in the outer ring of the preload indicating washer assembly. Rotate the ring back and forth while tightening the nut. The bolt is tight when the outer ring can no longer be rotated using 30 - 5 pounds tangential force applied as shown by the symbol above the tool. Do not tighten the bolt beyond this point.

Lower Forward Wing Bolt Tightening Procedure
Figure 206

60-31 15

CHART 201
WING BOLT WRENCHES AND TORQUE ADAPTERS

POSITION	*BOLT PART NO.*	*WRENCH PART NO.*	*NUT PART NO.*	*NUT TORQUE ADAPTER*
UPPER FORWARD	MS21250-12024/M/ or MS21250H12024/M/ or MS21250-12022/M/ or M/S21250H12022/M/	TK1817 922-4	12NB126 (internal wrenching) or EB126 (external wrenching) or ZEB1845-126 (external wrenching)	TS1171-2 or TS1176-2 TS1176-10 or TS1171-10
UPPER AFT	NAS150-38/M/ or NAS150DH38/M/	TS1222-4 or TS1222-8	12NB108	TS1171-1 or TS1176-1 or 50-590013
LOWER FORWARD	MS21250-14034/M/ or MS21250H14034/M/	TK1817 922-5	72789-1414	50-590014
LOWER AFT	MS21250-12024/M/ or MS21250H12024/M/ or MS21250-12022/M/ or M/S21250H12022/M/	TK1817 922-4	12NB126 (internal wrenching) or EB126 (external wrenching) or ZEB1845-126 (external wrenching)	TS1171-2 or TS1176-2 TS1171-10 or TS1176-10

METAL STALL STRIPS

The stall strips installed on airplanes without wing deicer boots are manufactured from clad 6063 aluminum alloy extrusion and riveted to the leading edge. A 10.50-inch strip is located outboard of wing station 123.06 on the left wing, and a 7.62-inch strip is located outboard of wing station 122.93 on the right wing.

WING BOLT NUT AND FITTING INSPECTION

NOTE

Read this entire section before removing any wing bolt for inspection.

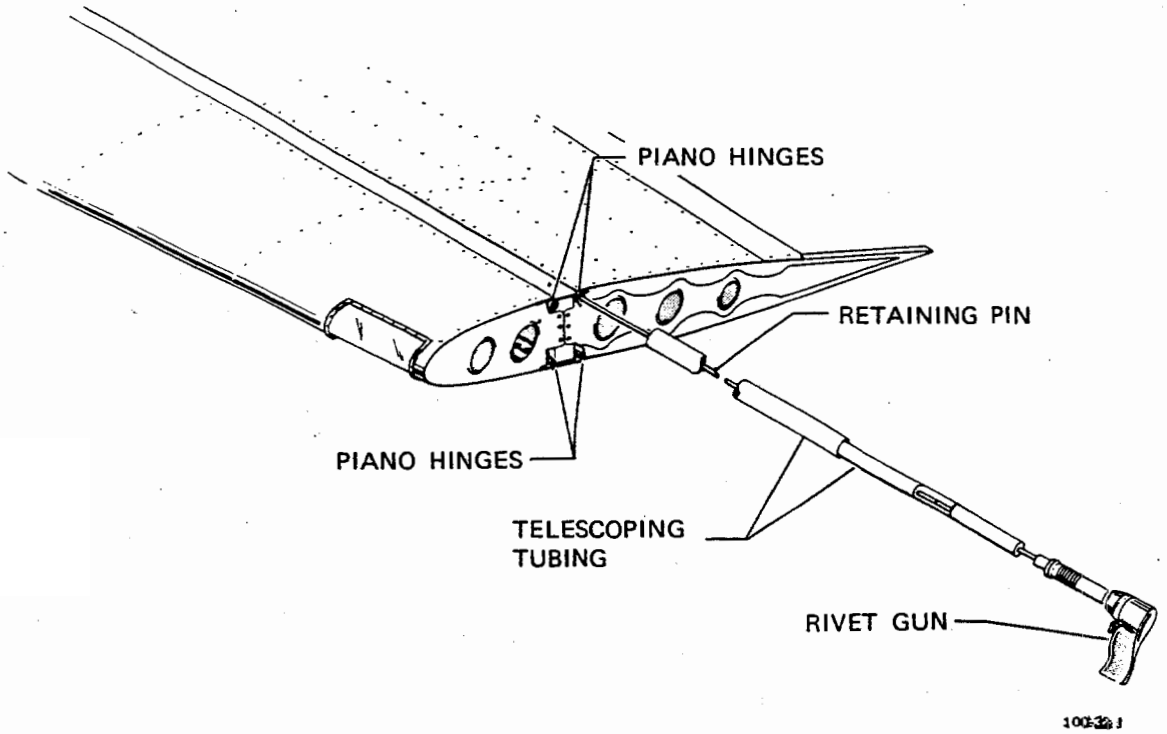
WARNING

The wing bolts and nuts installed in all Model 60 (Duke series) airplanes that are five years old or older must be removed and inspected. If the bolts and nuts prove to be free of all damage, they may be reinstalled for an additional five year period. At the end of this period the bolts and nuts must again be removed and inspected. Ten years after the initial inspection, all wing bolts and nuts must be replaced with new hardware. Render unserviceable all components removed in compliance with Chart 202.

a. Before removing any wing bolt, draw an outline of the wing position on the fuselage with a grease pencil. If wing bolt binding is encountered and the wing must be shifted, the outline will be helpful in returning the wing to its original position.

CAUTION

There should be no wing bolt binding during removal or installation of the bolts. Do not screw or drive a bolt in or out of the fittings. If wing bolt binding is encountered, place the airplane on a three point jack and raise until the wheels are clear (see Chapter 7-00-00 for jacking



Wing Hinge Pin Installation
Figure 207

instructions). Place a wing stand under each wing and a tail stand under the aft fuselage. Defuel the wing, loosen the remaining three bolts and rotate the wing until the binding bolt moves freely through the fittings. Replace the soft aluminum washers between the upper wing attach fittings and the preload indicating washer under the nut at the lower forward wing attach point. Retorque the nuts in the order outlined in this chapter under WING INSTALLATION.

NOTE

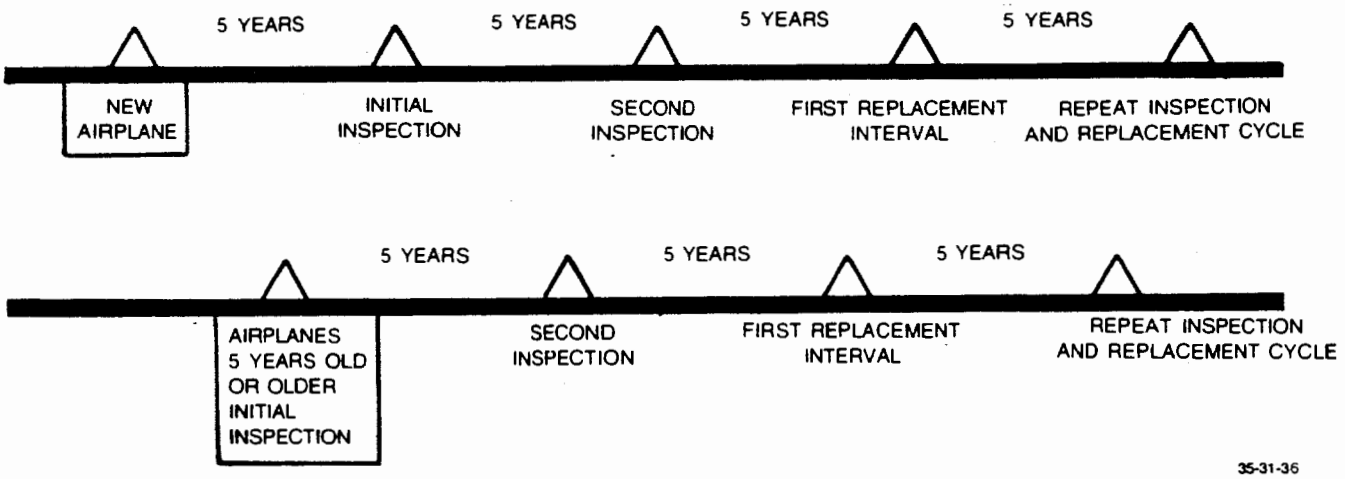
Beech Aircraft Corporation supplies hardware that has been given an additional magnetic particle inspection since manufacture. These components may be identified by the green dye on the head of the bolt and on some portion of the nut.

WARNING

Use only the components specified in the applicable illustrations. DO NOT INSTALL THE BLACK P/N H20 NUTS; these nuts have been dry film lubricated with molybdenum disulfide. When MIL-C-16173 Grade II corrosion preventive compound is added to these nuts, the additional lubrication may cause improper preload in the bolt when it is torqued.

b. Starting at the lower forward wing attach point on each side, remove, inspect, and retorque one bolt and nut set at a time until the complete set of eight bolts and nuts have been inspected. The leading edge attach fittings and hardware (Figure 205) ARE NOT a part of this inspection requirement.

c. Using a nonmetallic brush, thoroughly clean the bolt, washers, and nut with naphtha or methyl ethyl ketone (20 or 21, Chart 207, 91-00-00).



NOTE

The first inspection for airplanes five years old or older must be performed at the first scheduled inspection following the issue date of revision A14.

NOTE

At each replacement interval, all wing attach bolts, washers, and nuts must be replaced with new hardware.

Wing Bolt And Nut Inspection And Replacement Cycle
Chart 202

CAUTION

Assure that the radiused washers shown in Figures 201, 202, 203 and 204 have a full radius with no sharp edges that could damage the wing fittings.

d. If the bolts and nuts do not exceed the life limit shown in Chart 202, visually inspect each bolt and nut with a 10-power or stronger magnifying glass; inspect for corrosion, cracks and mechanical damage. The cadmium plating may have areas that appear rubbed, discolored or polished. These areas are usually the result of previous installation procedures and are of no consequence. A bolt should not be rejected because of cadmium plating deterioration; however, any component that is cracked, corroded or shows signs of mechanical damage must be replaced.

e. Using the magnetic particle inspection process described in this chapter, check each bolt for circumferential crack indications and each nut for longitudinal crack indications. If the bolts and nuts prove to be free of all damage (corrosion, cracks, and mechanical damage), they may be reused after demagnetization and cleaning.

f. Clean the fitting bolt bores with naptha or methyl ethyl ketone (20 or 21, Chart 207, 91-00-00). Do not strip the epoxy paint from this area. Inspect the surface condition of each fitting; focus special attention on the washer seat and bolt bore area. If scoring, corrosion pitting or washer impressions are discovered in this area, contact the Commercial Service Department of Beech Aircraft Corporation. If the fittings are satisfactory, coat the bolt bores and bearing faces with Alodine 1200, 1200S or 1201 (48, Chart 207, 91-00-00). Allow the coating to remain on the surface for approximately five minutes. When the approximate time has elapsed, wash the treated areas with water and blow dry (do not wipe dry). Paint the treated areas with zinc chromate primer (26, Chart 207, 91-00-00) and allow to dry.

g. Coat the bearing faces and bolt bores of the fittings, the complete bolt, washers, and nut with MIL-C-16173 Grade II corrosion preventive compound (43, Chart 207, 91-00-00).

h. Install the bolts, washers and nut into the fittings.

CAUTION

Ensure that the wing bolt wrenches do not bottom out on the wing fittings when torquing the nut. This could result in damage to the wing fittings and erroneous torque readings.

i. Torque the nut to the wet torque value shown in the appropriate illustration (Figure 201, 202, 203 or 204). When a torque wrench adapter is used, the length of the adapter must be added to the length of the torque wrench and the proper torque value computed as detailed in Chapter 20-00-00.

j. Coat the exposed threads that protrude through the nut with MIL-C-16173 Grade II corrosion preventive compound (43, Chart 201, 91-00-00).

k. Check that the decal shown in Figure 208 is affixed to the appropriate locations on the airplane.

l. At the first scheduled inspection after the wing bolts have been inspected or replaced, check each bolt for proper torque and inspect the drain holes in the upper wing fittings to assure that they are unobstructed.

MAGNETIC-PARTICLE INSPECTION

Magnetic-Particle Inspection is a method for locating surface and subsurface discontinuities in ferromagnetic materials (i.e. materials capable of being magnetized); consequently, nonferromagnetic materials (such as aluminum alloys, magnesium alloys, copper alloys, lead, titanium alloys, nickel base alloys and many stainless steel alloys) cannot be inspected by this method. Magnetic-Particle Inspection is based upon the principle that any discontinuities lying in a direction generally transverse to the direction of the magnetic field of the part magnetized for the test will cause a leakage field to be formed at and above the surface of the part. The presence of the leakage field denoting the discontinuity is detected by the use of finely divided ferromagnetic particles over the surface of the part. Some of the particles are magnetically gathered and held by the leakage field to form an outline indicating the location, size, shape and extent of the discontinuity. In general, magnetic particle inspection utilizes a variety of types of equipment for magnetization, as well as several methods for application of ferromagnetic particles to the test part. Additionally, the ferromagnetic particles are available in a selection of colors (including fluorescent) and particle shapes. Magnetic particle inspections required by this manual can best be accomplished by utilizing the "wet continuous method" on the standard wet horizontal type equipment, with either visible or fluorescent magnetic particles suspended in a petroleum base vehicle (normally kerosene). Since magnetic particle indications are best obtained when the discontinuity lies in a direction transverse to the magnetic field, the following procedures are recommended for optimum detection of discontinuities in both bolts and nuts.

NOTICE

**WING BOLTS ARE LUBRICATED
SEE MAINTENANCE MANUAL
FOR CORRECT TORQUE VALUES**

WHEN THE CORROSION PREVENTIVE COMPOUND HAS BEEN APPLIED TO THE WING BOLTS, AFFIX THE ABOVE DECAL TO THE FOLLOWING LOCATIONS:

1. On the side of the fuselage immediately above the RH forward and aft wing bolt covers.
2. On the wing immediately forward of the LH forward and aft wing bolt covers.
3. On the wing immediately forward of the lower forward wing bolt covers on both sides.
4. On the wing immediately aft of the lower aft wing bolt covers on both sides.

Lubricated Bolt Identification Placard Location
Figure 208

WARNING

Improper operation of the magnetic particle inspection because of faulty equipment or untrained operators can jeopardize the airworthiness of parts being inspected. Minute electrical arc burns caused during inspection by improper operation of the test equipment can result in eventual failure of the part.

Bolts: Inspection of a bolt is accomplished by longitudinal magnetization in a multiturn low-fill factor coil (i.e. the inner diameter of the coil greatly exceeds the bolt diameter). For proper magnetization the bolt is positioned close to the coil inside wall with the bolt length perpendicular to the winding direction. The magnetic particle suspension is flowed on the bolt and the appropriate current is applied to achieve adequate field strength. Using the described procedure, laboratory testing has indicated that the ampere turn values listed in Chart 203 provide for optimum detection of discontinuities perpendicular to the bolt axis.

CHART 203
MAGNETIC-PARTICLE INSPECTION (BOLTS)

BOLT DIAMETER	*TOTAL BOLT LENGTH INCLUDING HEAD TO NEAREST 1/4 INCH*	*AMPERE TURNS*
5/8 INCH	2 1/2 INCH	7900
5/8 INCH	2 3/4 INCH	7100
5/8 INCH	3 INCH	6600
3/4 INCH	3 INCH	7900
3/4 INCH	3 1/4 INCH	7400
3/4 INCH	3 1/2 INCH	6700
3/4 INCH	3 3/4 INCH	6300
7/8 INCH	3 1/2 INCH	7900
7/8 INCH	3 3/4 INCH	7400
7/8 INCH	4 INCH	6900
7/8 INCH	5 INCH	5500
1 INCH	5 INCH	6300

*Amperage requirement is the ampere turns value divid by the number of turns on the coil. For example: A 1-in diameter x 5-inch long bolt tested on a 5-turn coil wou require 6300 ÷ 5, or 1260 amps.

CHART 204
MAGNETIC-PARTICLE INSPECTION
(NUTS)

NUT SIZE	*CENTRAL CONDUCTOR SIZE*	*AMPERAGE*
5/8 INCH	1/2 INCH	500 AMPS
3/4 INCH	5/8 INCH	600 AMPS
7/8 INCH	3/4 INCH	700 AMPS
1 INCH	7/8 INCH	800 AMPS

Nuts: Inspection of a nut is accomplished by circular magnetization on a central conductor (usually a copper rod) the approximate size of the nut inside diameter. For proper magnetization, the central conductor bar is inserted through the nut and the bar is positioned between the heads of the wet horizontal equipment. The magnetic particle suspension is flowed on the nut and the appropriate current is applied through the central conductor to achieve adequate field strength. Using the described procedure, laboratory testing has indicated that the amperage values listed in Chart 204 provide for optimum detection of discontinuities parallel to the nut axis.

After magnetic particle inspection, the parts must be carefully demagnetized and cleaned of the ferromagnetic particles. Examine parts for any possible evidence of electric arc burn that may have occurred during the inspection.

WING MAIN SPAR CAP INSPECTION

The outboard wing main spar caps must be inspected for corrosion annually.

WARNING

All areas of the upper and lower spar caps must be inspected from the attach fitting to the outboard end.

BEECHCRAFT KIT NO. 58-4002-1S provides the parts and information necessary to install a new 000-110011-7 LH spar and a new 000-110011-8 RH spar on the 60 and A60 series airplanes. The kit does not contain the spars which must be ordered separately. Parts for installing new spars on the B60 series airplanes may be ordered from the Model 60 series parts catalog.

NOTE

Special emphasis should be placed on airplanes that have been operated or stored for extended periods (5 years or longer) in geographical locations where atmospheric conditions are highly conducive to corrosion.

Inspection of the upper and lower spar caps should be accomplished in the following manner:

a. Examine the forward and aft sides of the spar cap where it meets the skin. If a whitish, salt-like, nonmetallic substance is noted in these areas, a thorough inspection must be performed to determine if corrosion has occurred. Wax or paint trapped between the edge of the skin and the exposed section of the spar cap should not be misinterpreted as corrosion.

NOTE

To gain access to the upper spar caps in the nacelle area, remove the solid black panels shown in Figure 210.

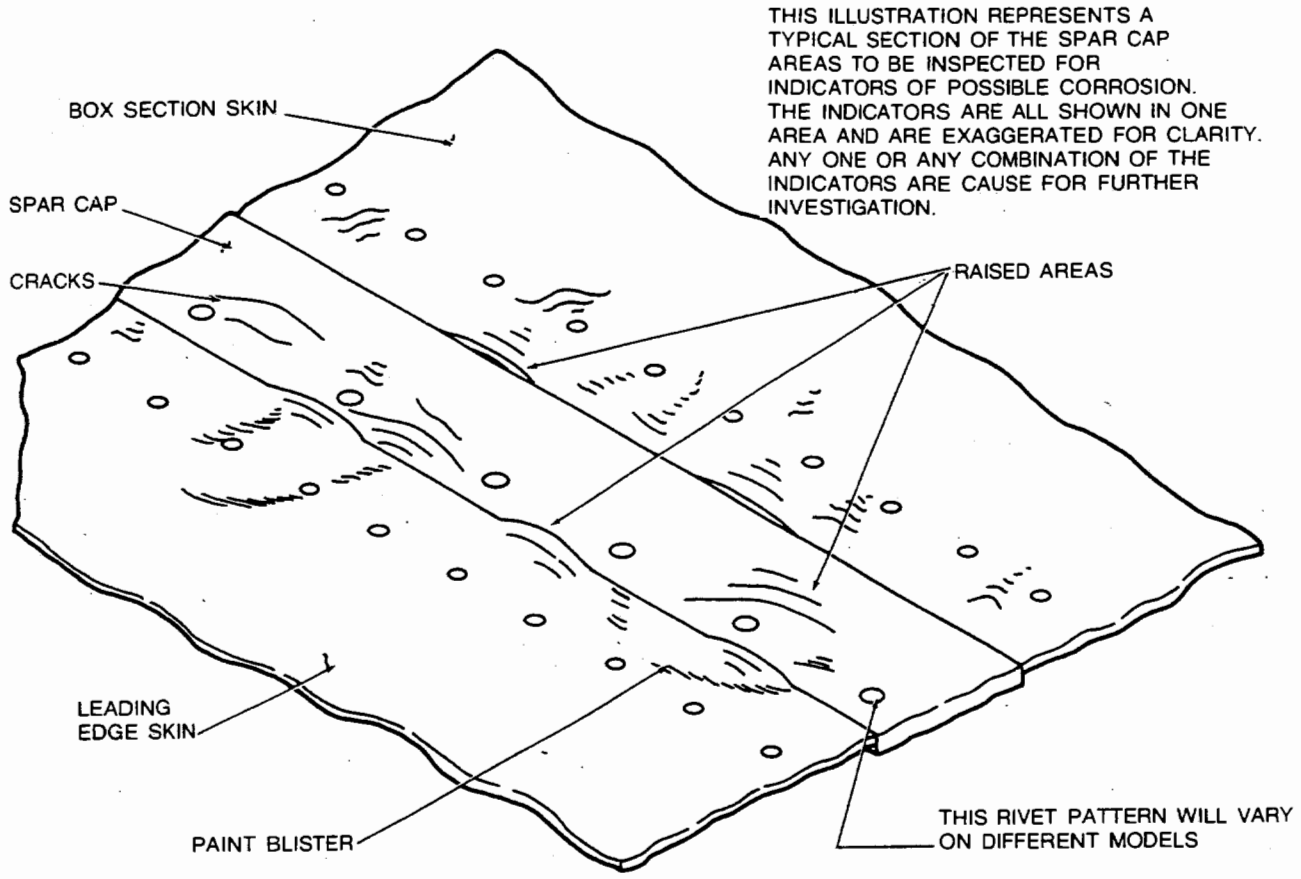
b. Wash all exposed areas of the upper and lower spar caps.

c. Visually inspect all exposed areas of the upper and lower spar caps for irregularities, such as paint blisters, raised or uneven areas, and cracks. The exposed areas of the spar caps are extruded flat and irregularities could be an indication of corrosion. Thoroughly investigate all irregular areas to determine if any damage has occurred.

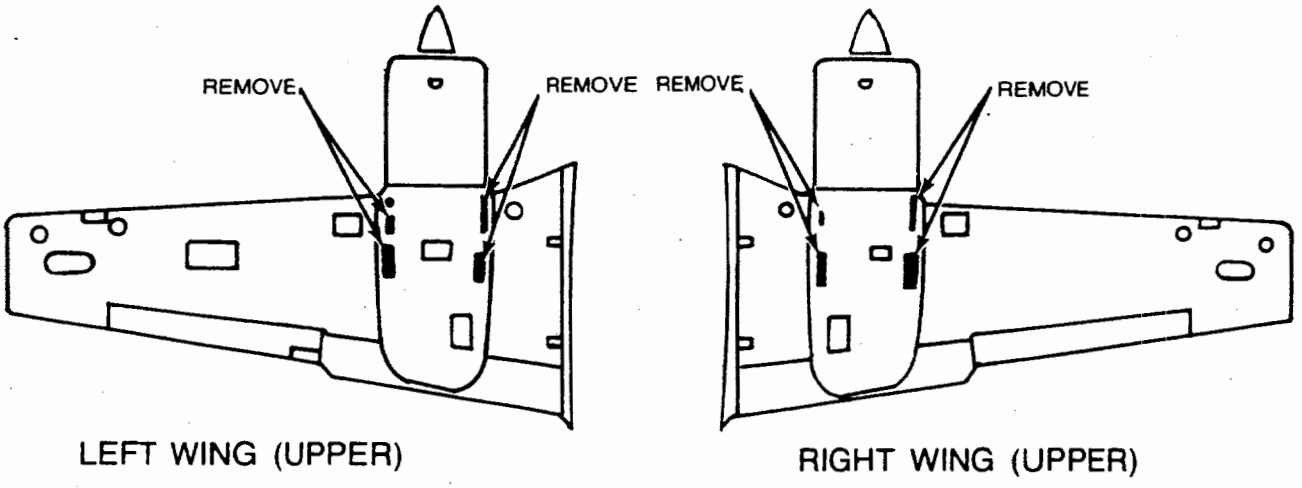
NOTE

Uneven or raised areas on the spar caps may be detected by sliding the fingers over the surface, by moving a straight edge over the surface or by sighting down the length of the spar cap surface.

d. If unusual conditions are encountered that cannot be resolved locally, contact the Commercial Service Department of Beech Aircraft Corporation for evaluation and determination of any corrective action that may be required.



Spar Cap Inspection
Figure 209



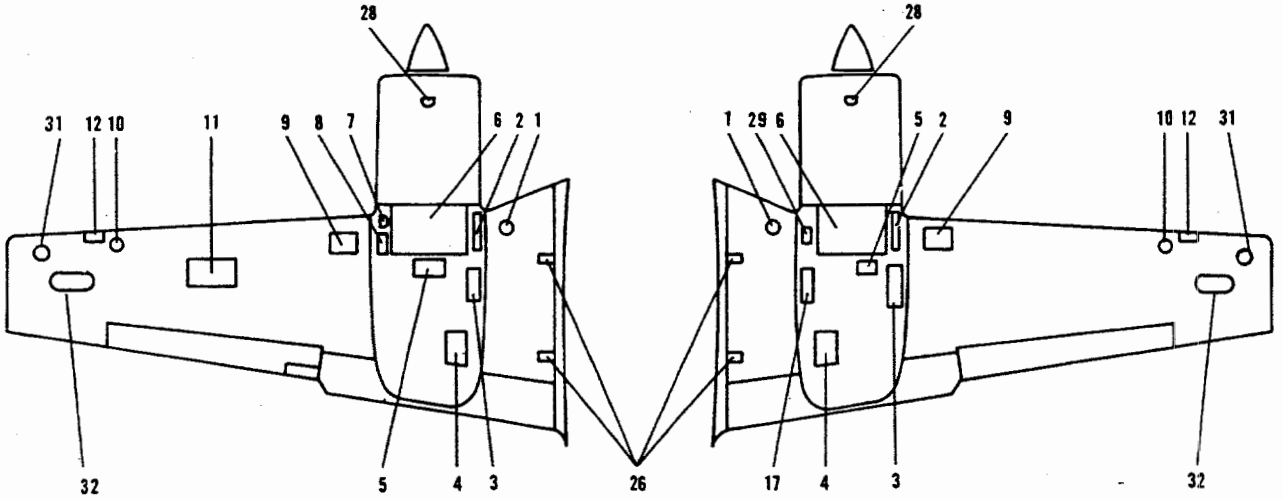
Upper Spar Cap Access Panels
Figure 210

"END"

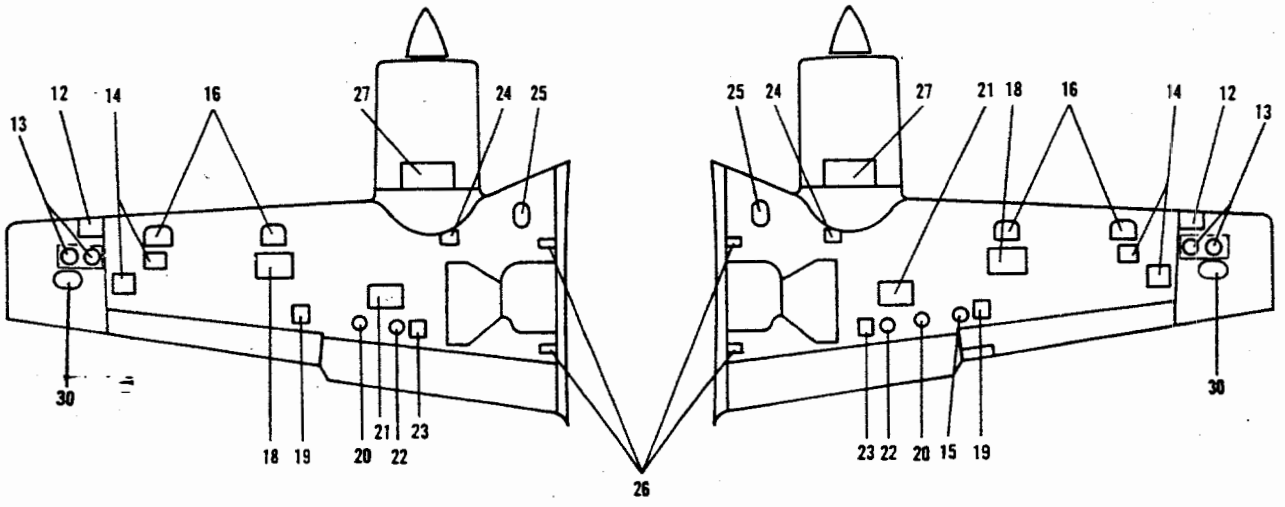
PLATES/SKIN-MAINTENANCE PRACTICES

WING ACCESS OPENINGS

The panels, plates and doors as shown in Figure 201, provides maintenance access to the components, plumbing and cables enclosed within the wing. When installed, they continue the aerodynamic lines of the wing with little increase of drag.



LEFT WING (UPPER) RIGHT WING (UPPER)



RIGHT WING (LOWER) LEFT WING (LOWER)

60-12-1A

Wing Access Openings
Figure 201 (Sheet 1 of 2)

1. Leading Edge Fuel Cell Transmitter
2. Alternate Air and Fuel Pressure Solenoid
3. Nacelle Fuel Cell Transmitter and Plumbing
4. Nacelle Fuel Cell and Vent Line Plumbing
5. Fuel Vent Check Valve and Plumbing
6. Battery, Battery Relays, Voltage Regulators, Overvoltage Relays, Starter Relays, Paralleling Rheostat, Fuel Flow Inverter, Load Meter Shunt, Fuse Block, Radio Inverter, Radio Inverter Circuit Breaker and Relay, External Power Diode and Current Limiter for Battery
7. External Power Plug
8. Reverse Current Diode, External Power and LH Control Relay
9. Leading Edge Fuel Cell Transmitter and Fuel Cell Installation
10. Fuel Filler
11. Remote Compass
- **12. Landing Light
- * 13. Wing Tip Tiring and Fuel Vent Float Valve
14. Fuel Siphone Valve
15. Aileron Tab Actuator
16. Leading Edge Fuel Cell
17. Nacelle Fuel Cell Plumbing
18. Box Section Fuel Cell
19. Aileron Actuator and Pulleys
20. Aileron Cable, Fuel Vent and Battery Vent
21. Box Section Fuel Cell
22. Fuel Vent Line and Aileron Tab Cable
23. Fuel Vent Line
24. Landing Gear Attach Bolt
25. Fuel Boost Pump
26. Wing Attach Bolt
27. Cowl Flap
28. Oil Level Indicator
29. Reverse Current Diode
- **30. Remote Compass Detector
- †31. Fuel Filler
- †32. Wing Tip Access Opening

* One (1) retangular opening on P-223 and after without wet wing tip
** P-223 through P-347, P-349 through P-364
† Optional P-348, P-365 and after

Wing Access Openings
Figure 201 (Sheet 2 of 2)

"END"

ATTACH FITTINGS - MAINTENANCE PRACTICES

The major fittings in each wing are the supporting structures adjacent to the attachment points for the flap actuator, flap tracks and flap, the aileron hinge brackets and hinges, the main landing gear, support brace and landing gear doors, and the engine mount. Minor fittings include brackets to support cable pulleys, bell cranks, and similar components. The main gear is bolted to heavy aluminum alloy fittings attached to the main and rear spar. The support brace is attached in the same manner. If the landing gear hinge bolt fittings are cracked, or if the spars are warped or buckled, replacement is necessary.

WING FRONT SPAR CAP INSPECTION

Perform this inspection on all Dukes, which are 5 years or older, in the areas and by the methods defined in Service Instructions No. 0514-035, Rev. 1.

"END"

FLIGHT SURFACES - MAINTENANCE PRACTICES

BALANCING THE AILERON (Figure 201)

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 0.2 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 percent balanced control surface is 0.0 inch-pounds. A tail-heavy surface exhibits static underbalance. A nose-heavy surface exhibits static overbalance.

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, the simplest is counterbalancing: The application of a known force or weight at a measured distance from the hinge line to counter the unbalance moment of the aileron assembly.

EQUIPMENT REQUIRED TO PERFORM CHECK BALANCING BY COUNTERBALANCING METHOD

a. A stand with knife edge supports as illustrated in Figure 201. The knife edges must be in the same horizontal plane.

b. A cup or similar light weight container.

c. Approximately 1 pound of lead shot.

d. A certified beam balance weighting device calibrated in units of .01 pound or less.

e. A straight edge, ruler, and spirit level.

BALANCING PROCEDURE

COUNTERBALANCING METHOD

a. 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.

b. Secure the trim tab (LH only) in its neutral position with a small piece of masking tape.

c. Fit 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.

d. To determine if weight should be added or removed, suspend a cup from a point near the center of the aileron trailing edge. Use a short length of small diamter string secured to the surface with a small piece of masking tape. (See Figure 201.) The cup must be free to hang vertically.

e. 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.

f. The distance "D" must be perpendicular to the hinge line. Measure "D" from the hinge line to the suspension point of the cup.

g. Remove the cup, contents, and string, then weight 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.

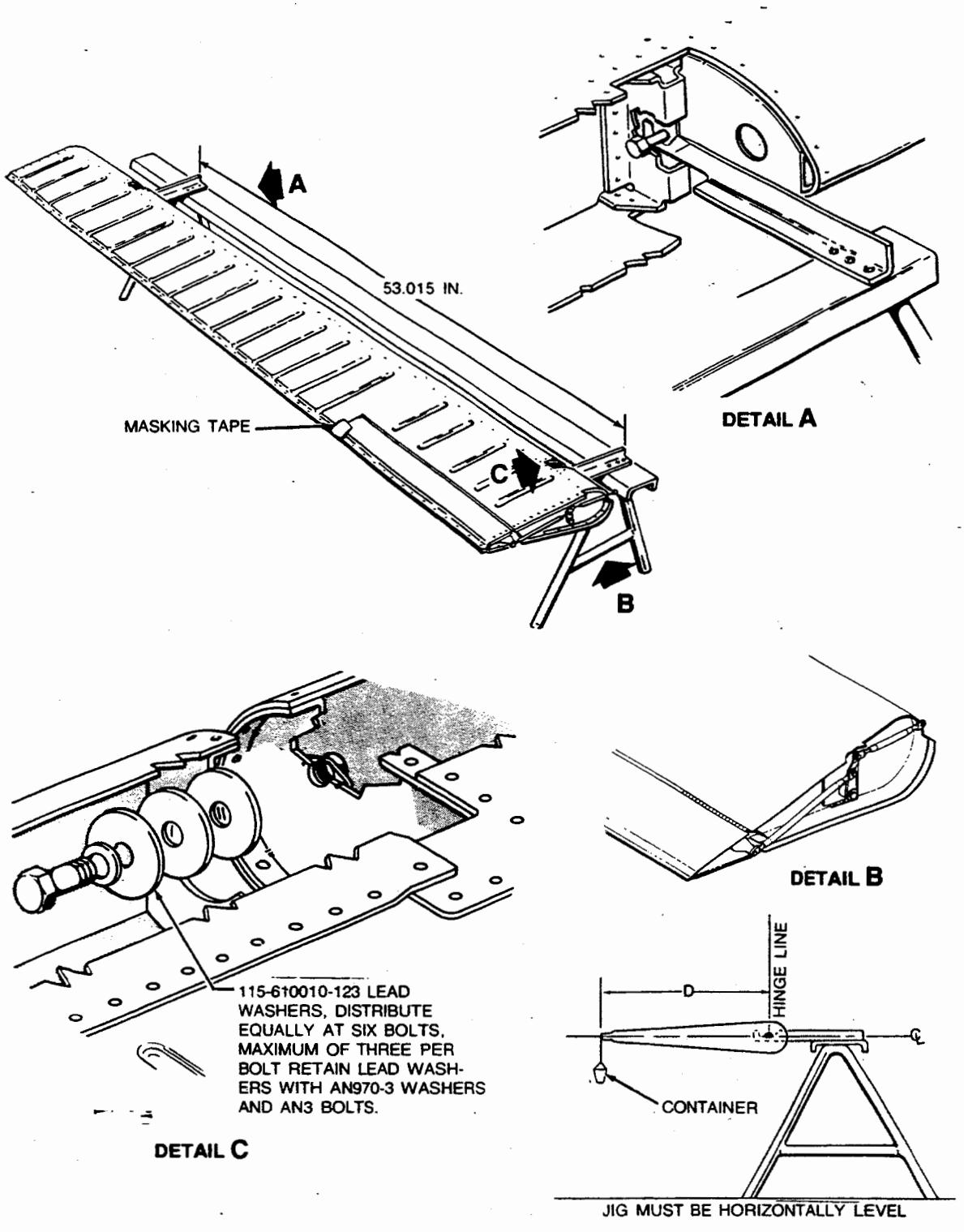
h. Calculate the static balance as follows:

1. The weight of the cup and contents is designated by "W".

2. The overbalance moment is designated by "M".

3. M = W x D.

4. The following is a typical example of a balancing calculation: If the aileron balances with the chord line level at "W = .15 pound" and "D = 10.0 inches", then . . .



Balancing the Aileron
Figure 201

M = .15 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.

i. If the static balance is not as noted in BALANCING THE AILERON, in this chapter, add or remove the lead washers as needed to attain the desired balance.

NOTE

A maximum of three lead washers (115-610000-123) may be added on each of six AN3 bolts near the leading edge, to bring the aileron balance within limits. Equally distribute and attach the washers with AN3 bolts of suitable length, and use one AN970-3 washer between the head of bolt and lead washer (see Figure 201).

"END"

CHAPTER 61

LIST OF PAGE EFFECTIVITY

CHAPTER SECTION SUBJECT	PAGE	DATE
61-EFFECTIVITY	1	Oct 19/77
61-CONTENTS	1	Oct 19/77
61-00-00	1	Oct 19/77
	101	Nov 2/73
	102	Nov 2/73
	201	Oct 19/77
	202	Oct 19/77
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	205	Nov 2/73
	206	Oct 19/77
	207	Nov 2/73

"END"

CHAPTER 61 - PROPELLERS

TABLE OF CONTENTS

"END"

GENERAL - DESCRIPTION AND OPERATION

PROPELLERS

On airplanes P-3 thru P-384, P-387, P-402 and P-403, the engines are equipped with 74 inch Hartzell HC-F3YR/C7479-2R or HC-F3YR-2F/FC7479-B2R three bladed, full feathering, constant speed, air dome propellers. Centrifugal force from the propeller counterweights, assisted by air pressure in the propeller dome, moves the blades to high pitch. Engine oil under governor-boosted pressure moves the blades to low pitch. On airplanes P-385, P-386, P-388 and after except P-402 and P-403, the engines are equipped with 74 inch Hartzell HC-F3YR-2UF/FC7479B-2R three bladed, full feathering, constant speed, air dome propellers. Centrifugal force from the propeller counterweights, the feather assist spring, and air pressure from the propeller dome moves the blades to high pitch. Engine oil under governor-boosted pressure moves the blades to low pitch. The propeller hub area and the air dome are enclosed by a spinner and bulkhead assembly.

PROPELLER SYNCHRONIZER

The propeller synchronizer automatically matches the left "slave" propeller rpm to that of the right "master" propeller. To prevent the left propeller from losing excessive rpm if the right propeller is feathered while the synchronizer is on, the synchronizer 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 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 left engine compressor mounting bracket. The motor then trims the left propeller governor through a flexible shaft and trimmer assembly to exactly match the right 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 pedestal 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 left propeller rpm will automatically be adjusted to correspond with the right. To change rpm, adjust both propeller controls at the same time. This will keep the left governor setting within the limiting range of the right propeller. If the synchronizer is ON but is unable to adjust the left propeller rpm to match the right, 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 left propeller to be governed by the propeller lever), synchronize the propellers manually, and turn the synchronizer switch ON.

PROPELLER SYNCHROSCOPE

A propeller synchroscope, located in the tachometer case, operates to give an indication of synchronization of propellers. If the right propeller is operating at a higher rpm than the left, the face of the synchroscope, a black and white cross pattern, spins in a clockwise rotation. Left or counterclockwise, rotation indicates a higher rpm of the left propeller. This instrument aids the pilot in obtaining complete manual synchronization of the propellers.

"END"

TROUBLESHOOTING
PROPELLER SYNCHRONIZER

TROUBLE	PROBABLE 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 (left) governor pickup gives open or short circuit reading on pins 6 & 8	c. Broken or grounded wire in magnetic pickup or slaved governor.	c. Repair or replace magnetic pickup.
d. Master (right) governor pickup gives open or short circuit reading on pins 7 & 8.	d. Broken or grounded wiring 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 .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 (Cont'd)

TROUBLE	*PROBABLE 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 O-ring 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.

"END"

GENERAL - MAINTENANCE PRACTICES

PROPELLER BLADE BEARING LUBRICATION
(Figure 201)

a. Remove the propeller spinner dome.
b. Remove the safety wire and covers from the six zerks.
c. Remove one zerk from each blade.
d. Lubricate the blade bearings with Hartzell DG Grease by placing the grease gun fitting on the remaining zerk of each blade. Fill until the grease is visible in the hole where the opposite zerk was removed.
e. Reinstall the zerk on each blade.
f. Clean excess grease from the propeller, reinstall the grease zerk covers and safety.
g. Reinstall the spinner dome.

PROPELLER REMOVAL

a. Remove the attaching screws and remove the spinner dome and cap.
b. Remove the retaining screws and the nose cowling fairing channels from the base of the propeller hub.
c. Remove the safety wire and remove the nuts around the propeller hub base with the special propeller torque wrench adapter (P/N 922 60-960000).

NOTE

When propeller deicer equipment is installed, it is necessary to disconnect the terminal wires on the starter ring gear.

d. Pull the propeller carefully from the mounting studs.

CAUTION

Do not damage the threads on mounting studs and be careful not to damage the spinner.

PROPELLER INSTALLATION

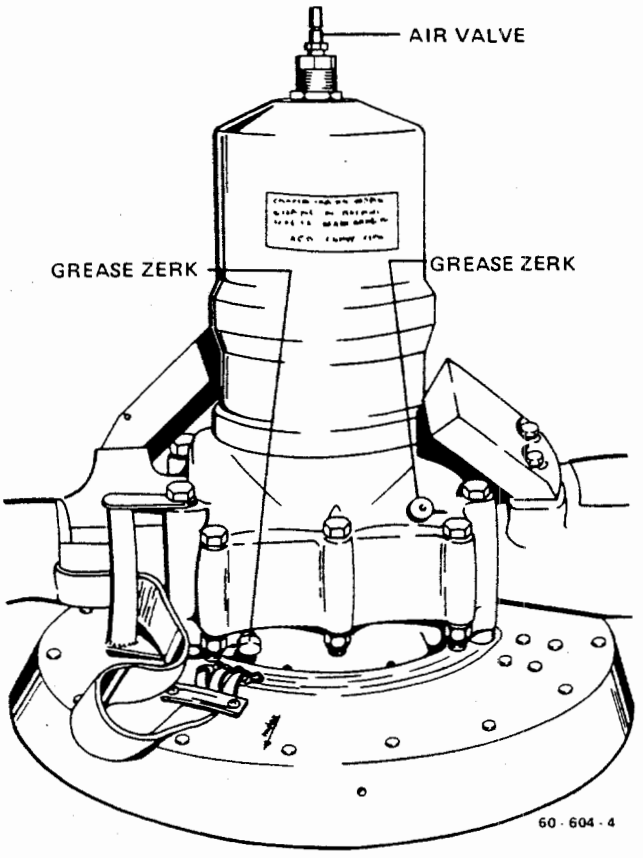
a. Install a new O-ring in the propeller flange extension and carefully position the propeller on the mounting studs. Install nuts and washers and snug down in a diagonal pattern. Torque nuts to 90 to 100 foot-pounds and safety wire.

CAUTION

Do not damage the threads on mounting studs and be careful not to damage the spinner.

NOTE

When using the special propeller torque wrench adapter, the torque valve must be recomputed. (Refer to Chapter 20-00-00.)

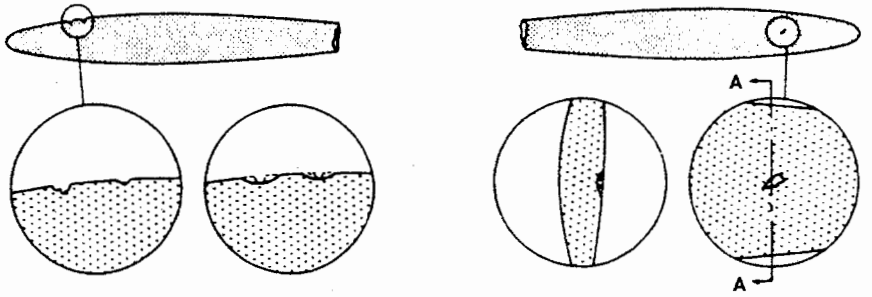


Propeller Servicing Point
Figure 201

b. Connect propeller deicer terminal wires, if installed, on the starter ring gear.
c. Install the nose cowling fairing channels with attaching screws.
d. Position the spinner dome and cap, and install the attaching screws.
e. Connect a dry air or nitrogen supply line to the air valve and fill to 80 psi for HC-F3YR-2/C7479B-2R, HC-F3YR-2F/FC7479-2R, HC-F3YR-2/C7479-2R, or HC-F3YR-2F/FC7479B-2R propellers. This should be done at 70 degrees F. Increase 2 psi for every 10 degrees of temperature increase. Decrease 2 psi for every 10 degrees of temperature decrease. Fill to 41 psi for HC-F3YR-2UF/7479-2R or HC-F3YR-2UF/FC7479B-2R propellers. This should be done at 70 degrees F.

NOTE

70 to 100 degrees F. Pressurize dome to 41 psi
40 to 70 degrees F. Pressurize dome to 38 psi
0 to 40 degrees F. Pressurize dome to 36 psi



Minor Propeller Blade Repair
Figure 202

PROPELLER BLADE REPAIR.
(Figure 202)

Minor nicks, dents and gouges may be dressed out by approved line personnel. Blend any nicks or gouges into the leading edge with smooth curves, and generous radii as shown in Figure 202. Reanodize the reworked area by the chromic acid process only.

PROPELLER GOVERNOR REMOVAL

a. Remove cotter pin, nut, washer and bolt attaching the adjusting rod end to the governor.

b. Disconnect the oil line, from the outboard side of the governor.

c. Remove the nuts and washers around the governor base. Remove the governor and cover the engine boss.

PROPELLER GOVERNOR INSTALLATION
(Figure 203)

a. Remove the cover from the engine boss; install a new gasket with the raised portion of the screen facing up.

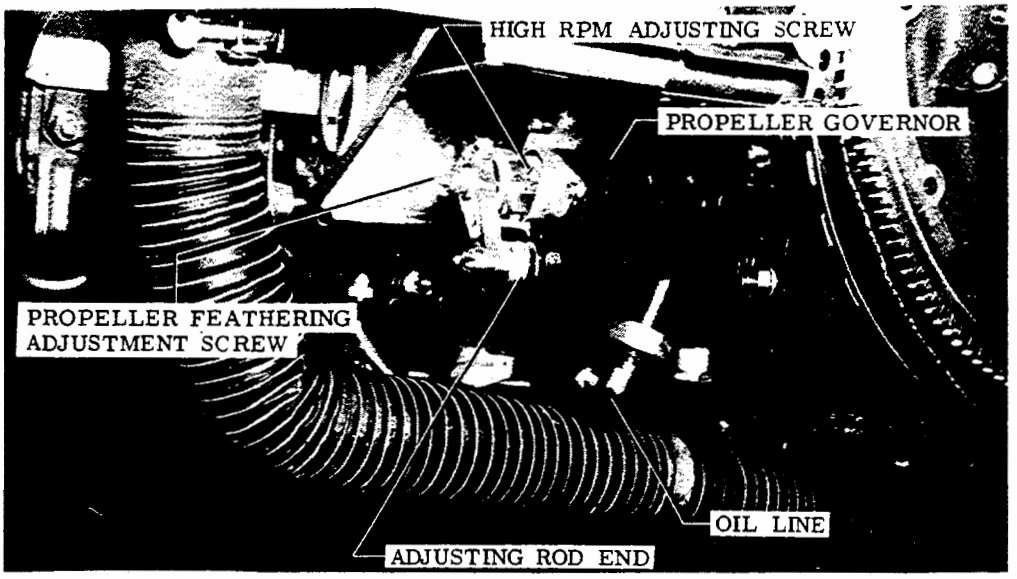
b. Align the governor spline with the engine drive spline, and install washers and nuts. Diagonally torque the nuts to 150 inch-pounds.

c. Connect the oil line to the outboard side of the governor.

d. Install the bolt, washer, nut and cotter pin attaching the adjusting rod end to the governor.

NOTE

To insure proper adjustment, do not turn the adjusting rod end when installing the governor.



Propeller Governor Installation
Figure 203

PROPELLER GOVERNOR ADJUSTMENT
(Figure 203)

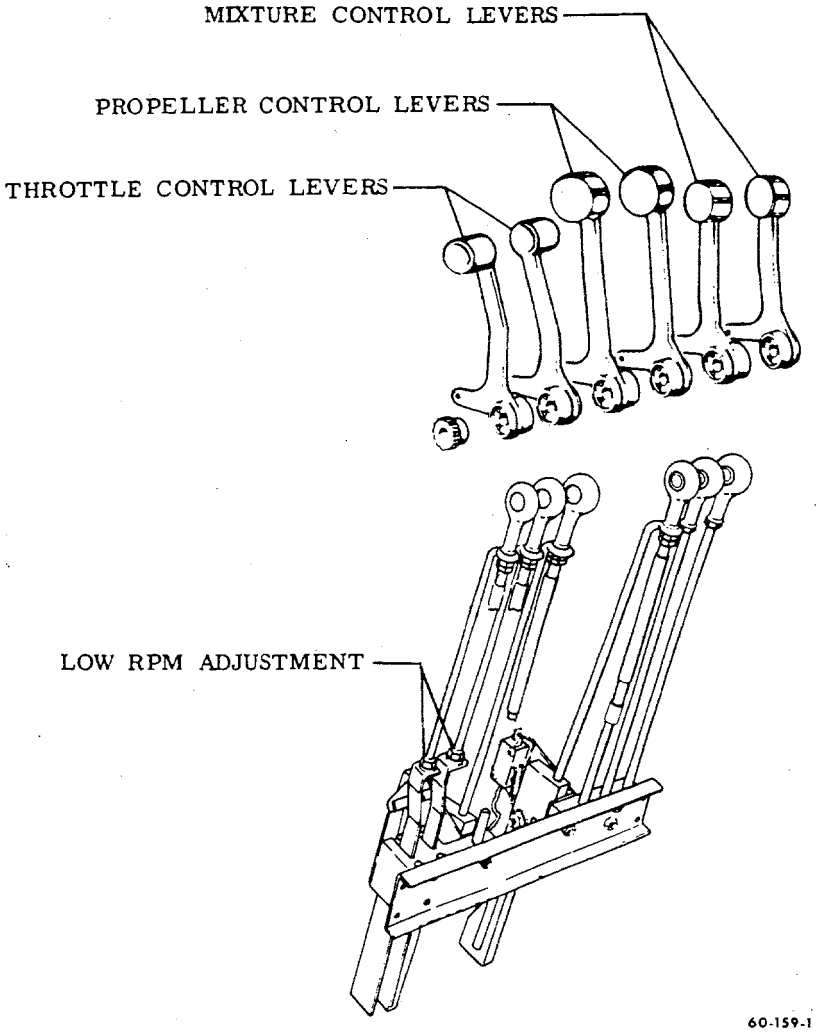
The propeller governor can be adjusted for high and low rpm setting and a feathering adjustment. The high rpm adjustment must be checked while the aircraft is in flight. Observe the take-off rpm to see if it exceeds the redline figure. If excessive rpm is observed, land the plane and adjust the high rpm screw inward to reduce the rpm to the redline figure. The high rpm adjustment screw is located at the rear of the governor just forward of the speed adjusting control lever. One complete revolution of the screw reduces the propeller rpm by approximately 25 to 30 revolutions.

FEATHERING ADJUSTMENT

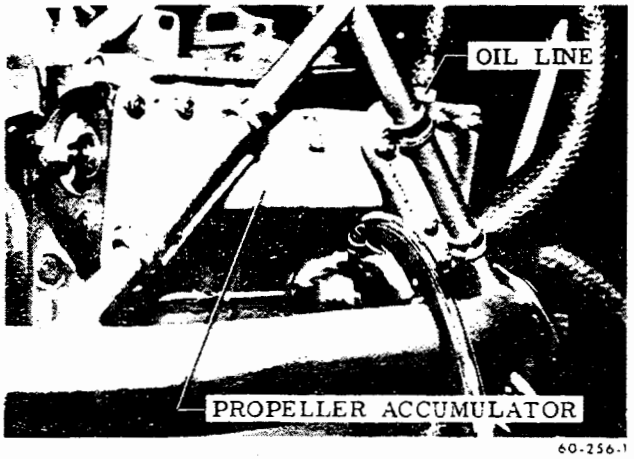
To adjust the feathering action, pull the control back through the detent and observe the point at which the rpm setting begins to fall off sharply, then bring the propeller back to low rpm. The point at which propeller feathering starts should be at 2,100 rpm. If adjustment is required turn the square-head screw on the end of the governor control shaft inward or outward to correct the setting. One half revolution of the screw inward will lower the feathering rpm approximately 100 revolutions. (See Figure 203.)

LOW RPM ADJUSTMENT
(Figure 204)

The low rpm adjustment is made while the airplane is on the ground. To make this adjustment, pull the propeller lever back against the detent. Slowly move the throttle control lever forward until the rpm stabilizes. Observe the rpm setting. If the rpm varies from the specified low rpm setting of 2350 rpm the low rpm setting must be adjusted. The low rpm adjustment is made on the detent rod, which is located behind the instrument panel on the governor control



Propeller Low RPM Adjustment
Figure 204



Unfeathering Accumulator
Figure 205

linkage. To increase the setting, lengthen the rod; to decrease the setting, shorten the rod.

PROPELLER ADJUSTMENT

For high and low pitch adjustments, service, overhaul and maintenance procedures refer to the manufacturers applicable FAA Approved Propeller Manuals.

PROPELLER ACCUMULATOR

The propeller accumulators are located on the lower rear section of each engine. The accumulators should be inspected every 100 hours and charged with dry air or nitrogen to 125 psi.

PROPELLER ACCUMULATOR REMOVAL

(Figure 205)

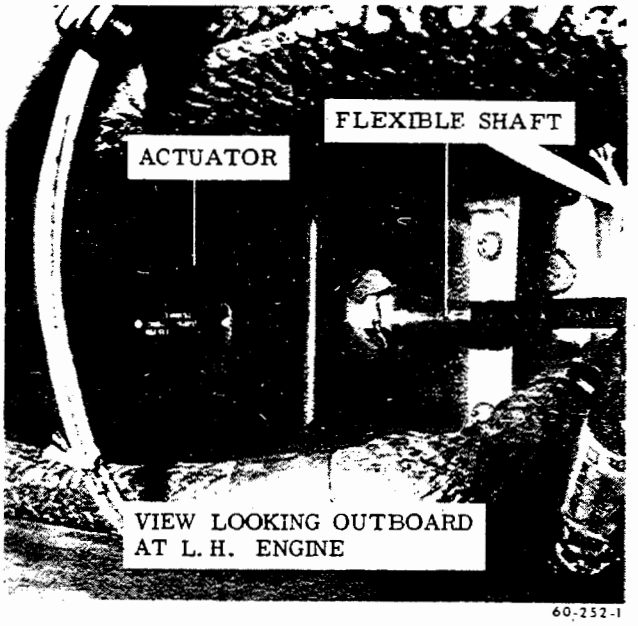
a. Check the propeller control lever for unfeathering (low pitch) position, to release accumulator pressure.

CAUTION

This system has approximately 300 psi of pressure with the propeller in full-feather position.

b. Remove the oil line from the end of the accumulator.

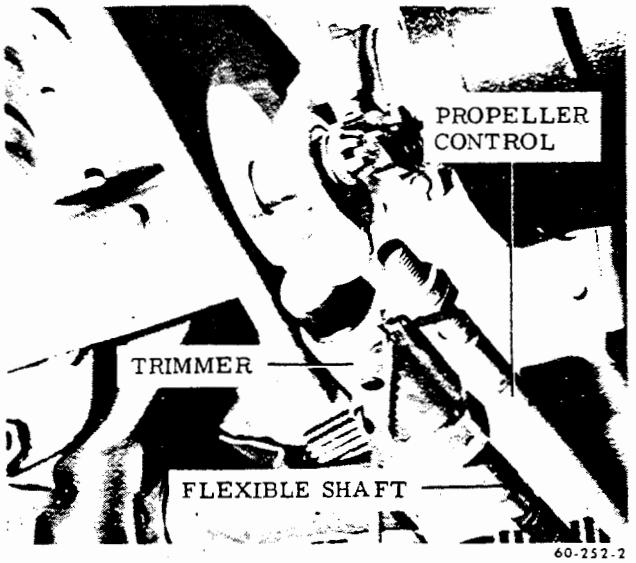
c. Remove the four clamps that hold the accumulator mounting brackets to the engine mount and remove the accumulator.



Propeller Synchronizer Actuator
Figure 206

PROPELLER ACCUMULATOR INSTALLATION

a. Position the accumulator against the engine mount



Propeller Synchronizer Trimmer
Figure 207

and install the four attaching clamps.

b. Connect the oil line to the end of the accumulator.

c. Charge the accumulator with dry compressed air or nitrogen to 125.

SYNCHRONIZER FUNCTIONAL TEST
(Figure 206)

Proper operation of the propeller synchronizer can be determined by the following method. Turn the synchronizer on in the normal manner, then decrease the right propeller rpm in small increments. Both propellers should decrease together until the actuator reaches the end of its travel, then the left propeller should stabilize its rpm while the right continues to decrease. Increase the right propeller rpm until both propellers commence to increase rpm together. Turn the system OFF. An unsynchronized condition will develop as the actuator runs to its midrange position. When the synchronizer is turned ON, the left propeller should again synchronize with the right.

SYNCHRONIZER RIGGING
(Figure 207)

a. Disconnect the flexible trimmer shaft from the actuator.

b. 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.

c. 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.

d. Rotate the shaft to the center of its range.

e. 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.

f. With both the trimmer and the actuator motor centered, connect the flexible shaft.

g. Rig the propeller controls in the normal manner.

SYNCHRONIZER 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 tests outlined below.

SYNCHRONIZER WIRING CHECK
(Chart 201)

a. 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.

b. Unplug the control box, turn the aircraft master switch OFF, and pull the synchronizer circuit breaker before proceeding further with these checks.

CHART 201
SYNCHRONIZER RESISTANCE CHART

TEST BETWEEN RECEPTACLE NUMBERS:	OBTAIN		
		With actuator uncentered 180°	
	With actuator centered	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 1 ohm	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

c. Complete resistance checks. (See CHART 201.)

CAUTION

Zero the ohmmeter and read on the X1 to X10 scale during the following checks. Do not use a probe greater than .045 inch in thickness. Insert and remove the probe carefully to avoid damaging the pin connectors.

NOTE

Later propeller governors are equipped with an improved type of magnetic pickup. The new type has a solid steel pickup end; while the old type has a visible ceramic core. Any combination of new or old magnetic pickups is acceptable between engines.

d. Complete the following checks.

1. Using an ohmmeter, test that the resistance between pin receptacles 7 and 8 of the Jones cinch socket (see Figure 208) is 52 to 68 ohms (new pickup), or 90 to 110 ohms (old pickup) with pin receptacle 6 disconnected at the pickup.

2. Check that the resistance between pin receptacle 6 and 8 is 52 to 68 ohms (new pickup), 90 to 110 ohms (old pickup) with pin receptacle 7 disconnected at the pickup.

e. Make the following checks with an ohmmeter connected to the pin receptacles of the Jones cinch socket (P/N 233032).

1. Check that an open circuit (very high-resistance) exists between pin receptacle 8 and airplane ground and between pin receptacle 2 and ground.

2. Check that a closed circuit (zero ohms) exists between pin receptacle 1 and ground.

f. Turn the airplane master switch ON and reset the synchronizer circuit breaker, but leave the control box unplugged.

g. 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.

h. 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 left (slave) engines and pins 7 and 8 for pickup voltage on the right (master) engine. These values should be between 1/2 volt minimum and 3 volts maximum.

i. 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.

j. Check the speed pickups mounted on the governor for oil leaks or looseness.

k. 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. The normal output torque is 15 ounce-inches. Make sure the actuator is returned to the center of its range.

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.

If the response of the system to the preceding checks has been satisfactory, the airplane is ready for flight.

FLIGHT CHECKS

a. 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.

b. 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 airplane 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.

100-HOUR INSPECTION

PROPELLER - Inspect the propeller for nicks, dents, cracks, evidence of leakage, condition and security.

AIR DOME - Check the propeller air dome for correct pressure.

DEICER BOOTS - Check the boots for hot spots, exposed heating element wires, tears and security to the blades.

SPINNER AND BULKHEAD - Check the spinner and bulkhead for nicks, dents, cracks, condition and security. Check deicer boot wires for security at the starter ring gear.

PROPELLER GOVERNORS - Inspect the governor for oil leakage, condition and security.

CONTROL LEVERS - Check levers for smooth and free movement and cushion. Check controls at the governor for security and full travel against the stops.

ACCUMULATOR - Inspect for oil leakage, condition and security. Check air pressure.

SYNCHRONIZER - Check all components of the system for condition and security. Any discrepancies noted during ground or flight should be isolated using the troubleshooting chart and the wiring check. Consult Woodward Governor Bulletins 33032A and 33049C for further detailed information.

"END"

CHAPTER 71

LIST OF PAGE EFFECTIVITY

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	103	Nov 2/73
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	202	Nov 2/73
	203	Nov 2/73
71-10-00	201	Nov 2/73

CHAPTER 71 - POWER PLANT

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"END"

GENERAL - DESCRIPTION AND OPERATION

The Duke is equipped with Lycoming TIO-541-E1A4 and/or TIO-541-E1C4 engines. They are rated at 380 horsepower at 2900 rpm and 41.5 in. Hg, and are turbochargered for high performance at altitudes to 30,000 feet. The engines power three-bladed, 74 inch diameter, constant speed, full feathering, hydraulically controlled propellers.

"END"

TROUBLESHOOTING
ENGINE

TROUBLE	PROBABLE CAUSE	REMARKS
1. Failure of engine to start.	a. Lack of fuel.	a. Check fuel system for leaks. Fill fuel cell. Clean dirty lines, strainers or fuel valves.
	b. Overpriming.	b. Unload engine by standard clearing procedure.
	c. Incorrect throttle setting.	c. Open throttle to 1/4 of its range.
	d. Defective spark plugs.	d. Clean and adjust or replace spark plugs.
	e. Defective ignition wire.	e. Check with tester and replace any defective wires.
	f. Improper operation of magneto.	f. Clean points. Check timing.
	g. Internal failure.	g. Check oil screens for metal particles. If found, complete overhaul of engine is indicated.
2. Failure of engine to idle properly.	a. Incorrect idle mixture.	a. Adjust mixture control.
	b. Incorrect idle speed.	b. Adjust idle speed.
	c. Leak in induction system.	c. Tighten all connections, replace any defective parts.
	d. Uneven cylinder compression.	d. Check condition of piston rings and valve seats.
	e. Faulty ignition system.	e. Check ignition system.
3. Low power and uneven running.	a. Mixture too rich, indicated by sluggish engine, red exhaust flame. Extreme cases indicated by black smoke at exhaust.	a. Readjust fuel injector.
	b. Mixture too lean; indicated by overheating and back-firing.	b. Check fuel lines for restrictions. Readjust mixture.
	c. Leak in induction system.	c. Tighten all connections, replace any defective parts.
	d. Defective spark plugs.	d. Clean and gap or replace spark plugs.

TROUBLESHOOTING
ENGINE (Cont'd)

TROUBLE	PROBABLE CAUSE	REMARKS
3. Low power and uneven running.(Cont'd).	e. Improper fuel.	e. Fill cell with fuel of recommended grade.
	f. Magneto breaker points not working properly.	f. Clean points, check timing.
	g. Defective ignition wire.	g. Check wires with tester, replace any defective wires.
	h. Defective spark plug terminal.	h. Check and replace connectors if necessary.
4. Failure of engine to develop full power.	a. Leak in the induction system.	a. Tighten all connections replace any defective parts.
	b. Throttle lever out of adjustment.	b. Check travel of throttle linkage.
	c. Improper fuel flow.	c. Check strainers and flow at fuel injector.
	d. Restriction in air scoop.	d. Examine air scoop and remove any obstruction.
	e. Improper fuel.	e. Drain and refill cell with fuel of recommended grade.
	f. Faulty ignition.	f. Check ignition system.
5. Rough engine.	a. Cracked engine mount.	a. Replace or repair mount.
	b. Defective mounting bushing.	b. Replace bushing.
	c. Uneven compression.	c. Check compression.
6. Low oil pressure.	a. Insufficient oil.	a. Fill sump with oil.
	b. Air lock or dirt in relief valve.	b. Remove and clean oil pressure relief valve.
	c. Dirty oil strainers.	c. Remove and clean oil strainers.
	d. High oil temperatures.	d. See step 7.
	e. Defective pressure gage.	e. Replace gage.
	f. Stoppage in oil pump inlet passage.	f. Check line for obstruction.
7. High oil temperature.	a. Insufficient oil supply.	a. Fill sump with oil of recommended grade.

TROUBLESHOOTING
ENGINE (Cont'd)

TROUBLE	PROBABLE CAUSE	REMARKS
7. High oil temperature.(Cont'd).	b. Low grade of oil.	b. Drain and fill sump with oil conforming to specifications.
	c. Clogged oil lines or strainers.	c. Clean oil lines and strainers.
	d. Excessive blow-by.	d. Usually caused by worn or stuck rings.
	e. Failed or failing bearings.	e. Examine oil strainers for metal particles. If found, overhaul of engine is indicated.
	f. Defective temperature gage.	f. Replace gage.
8. Excessive oil consumption.	a. Low grade of oil.	a. Fill sump with oil conforming to specifications.
	b. Failing or failed bearings.	b. Check sump for metal particles.
	c. Worn piston rings.	c. Install new rings.
	d. Incorrect installation of piston rings.	d. Install new rings.
	e. Failure of rings to seat (nitrided barrels).	e. Use mineral base oil. Climb to cruise altitude at full power and operate at 75% cruise power setting until consumption stabilizes.

"END"

GENERAL - MAINTENANCE PRACTICES

ENGINE REMOVAL

CAUTION

The engine induction air is supplied through a fiberglass duct located in the right hand section of the engine compartment. Care should be taken when removing or installing the engine that no dirt or foreign objects, be allowed to enter the induction system. Be careful not to damage the fiberglass ductwork attached to the firewall.

a. Check the magneto switches for "OFF" position.

WARNING

To be safe, treat all magnetos as hot whenever the ground lead is disconnected. To ground the magneto, disconnect the magneto switch lead wire at the capacitor and ground the capacitor pole. If this is impractical, remove the ignition harness distributor cap, or disconnect the spark plug leads.

b. Remove the upper and lower engine cowling. (Refer to Chapter 71-10-00.)

c. Remove the propeller. (Refer to Chapter 61-10-00.)

d. Disconnect all plumbing at the firewall. Be sure to cap all open lines and fittings.

CAUTION

Place the fuel selector valve handle in the "ON" position to relieve approximately 60 psi of pressure in the fuel line from the firewall to the fuel pump.

e. Disconnect and identify all electrical wiring at the firewall.

f. Disconnect all engine controls.

g. Place a wing stand under the opposite wing and a support under the tail.

h. Position the engine hoist and attach the hoisting sling to the three lifting eyes on the engine.

i. Remove the slack from the hoisting cable and remove the bolts that attach the engine mounts to the firewall.

j. Remove the engine and place in a suitable work stand.

ENGINE BUILD-UP

Engine build-up consists of the removal of accessories and equipment from the old engine and installing them on the new engine. Refer to the Lycoming Engine Overhaul Manual, P/N 60294-6, for proper torque values.

NOTE

Tag or identify all hoses, bolts, washers, nuts, electrical connectors, and note harness clamp locations for reinstallation on the new engine. Cap all open hoses and engine ports to prevent contamination.

NOTE

Torque engine mount (isolators) bolts to 250-300 inch-pounds.

ENGINE INSTALLATION

a. Observe the WARNING and CAUTION notes in ENGINE REMOVAL.

b. Position the engine hoist and attach the hoisting sling to the three lifting eyes on the engine.

c. Move the hoist into position in front of the firewall, align the bolt holes of the engine mount and those of the firewall. Install the engine mount bolts and torque to 350 - 390 inch-pounds.

NOTE

If the engine mount bolt nuts are replaced, use a magnet to ensure they are steel.

f. Disconnect the hoisting sling and move the hoist clear of the aircraft.

g. Connect all electrical wiring at the firewall.

h. Connect all plumbing at the firewall, and all ducting.

i. Connect and adjust all engine controls. (Refer to Chapter 61-00-00.)

j. Install the propeller. (Refer to Chapter 61-00-00.)

k. Install the upper and lower engine cowling. (Refer to Chapter 71-10-00.)

l. Perform an engine run-up and complete final adjustments.

NOTE

If a new or newly overhauled engine has been installed, the engine fuel and oil system must be depreserved and serviced. (Refer to Chapter 12-10-00 for Servicing.)

GROUND RUNNING AND WARM-UP

Because the turbocharged engines depend on forward air speed for cooling, caution should be used to prevent overheating on the ground. The following precautions should be followed when performing power checks or engine run up:

a. Head the aircraft directly into the wind.
b. Operate the engines on the ground with the propeller in the low pitch position (unfeathered position).
c. Maintain the cylinder head temperature between 150°C and 246°C. Never allow the cylinder head temperature to exceed 246°C.
d. The turbine inlet temperature (TIT) gage, must not exceed 900°C.
e. Extended periods of idling at low rpm may result in fouled spark plugs.
f. The mixture control should remain in the "FULL RICH" position unless leaning is required during the checkout.
g. Warm up the engine at 1000 to 1500 rpm.

NOTE

The oil pressure should be within the red radial in 30 seconds; normal oil pressure should be approximately 90 psi at maximum rpm.

The oil seals incorporated in the turbocharger derive their sealing capabilities from oil pressure. At low engine idle speeds, the seals will allow some oil seepage onto the turbocharger shaft, which will cause coking (oil and carbon deposit buildup). Therefore, during taxiing, engine runup or ground test, it is advisable to maintain idling speeds of approximately 1200 rpm until the turbocharger temperature has stabilized. This engine speed will exert enough oil pressure against the shaft seals to prevent oil seepage. When the turbocharger temperature has stabilized, the engine may be shut down with a minimum of coking.

IDLE SPEED AND MIXTURE ADJUSTMENT
(Figure 201)

a. Start the engine and run at 1000 to 1500 rpm until the oil and cylinder head temperature gages read normal.
b. Check magneto drop-off. (See MAGNETO DROP-OFF CHECK, Chapter 74-10-00.) Maximum drop-off should not exceed 175 rpm. If the magneto drop-off is within limitations, proceed with the idle adjustment.
c. Slowly retard the throttle lever to the idle position. The engine tachometer should indicate 700 rpm (normal idle setting). To adjust, turn the idle speed adjusting screw at the throttle lever stop until the desired rpm is reached.
d. When the idle speed has stabilized, move the mixture control lever with a smooth, steady pull into the "IDLE CUT-OFF" position. Observe the tachometer for any change during the leaning out process.

CAUTION

Return the mixture control to the FULL RICH position before the rpm can drop to a point where the engine cuts out.

Idle and Mixture Adjustment
Figure 201

An increase in rpm while leaning out indicates the idle mixture is on the rich side of best power. An immediate decrease in rpm (if not preceded by a momentary increase) indicates that the idle mixture is on the lean side of best power. The desired idle setting is a compromise between one that is rich enough to provide a satisfactory acceleration under all conditions and lean enough to prevent spark plug fouling or rough operation. A rise of 25-50 rpm, during the leaning process, will usually satisfy both of these conditions.
e. Adjustment of the mixture is accomplished by turning the "STAR" adjustment screw, one or two notches, in the direction required, as noted on the linkage blocks with an R for Rich, and an arrow for direction of rotation.

NOTE

For major adjustments refer to Bendix Manual Form 15-468.

f. After each idle mixture adjustment change, clear

the engine by running it up to 2000 rpm before making a mixture check.

g. Recheck the idle speed as stated in step "c". Make final idle speed adjustment, if necessary.

NOTE

If the idle setting does not remain stable, check the idle linkage; any looseness on this linkage will cause erratic idling. In all cases, allowances should be made for the effect of weather conditions upon idling adjustments.

THROTTLE—LANDING GEAR WARNING HORN SWITCH ADJUSTMENT

a. In flight, place the propeller lever in low pitch. Slowly pull both throttle levers back until 12 to 14 in. Hg manifold pressure is indicated. Mark this position on the quadrant.

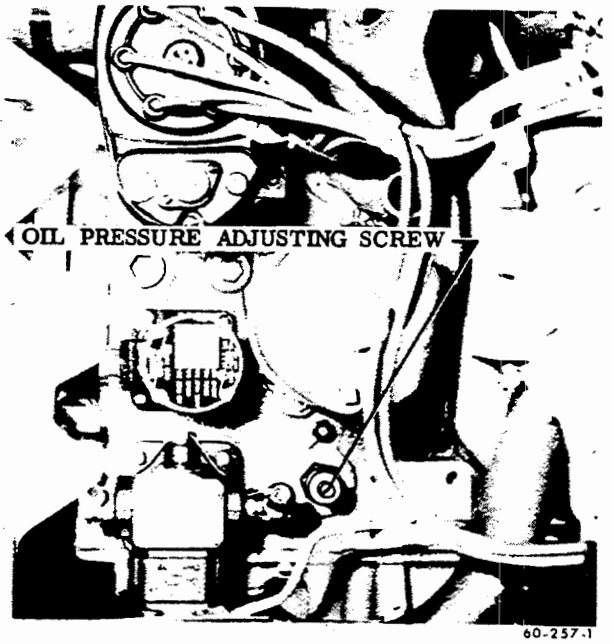
b. Land the airplane and shut the engine down.

NOTE

The landing gear warning horn micro-switches are located on a bracket, at the lower end of the throttle linkage, in the console.

c. Position the throttle levers on the mark previously made. Raise or lower the micro-switches until the cams "click" the switches closed. Secure the switches in this position.

d. Fly the airplane to check adjustment.



Oil Pressure Adjustment
Figure 202

OIL PRESSURE ADJUSTMENT
(Figure 202)

The oil pressure adjustment screw is located approximately 3 inches directly below the oil filter housing. To adjust, turn the adjusting screw clockwise to increase or counterclockwise to decrease oil pressure. Run the engine at 2600 rpm with normal operating oil temperature and set the oil pressure at 80 psi.

"END"

COWLING - MAINTENANCE PRACTICES

COWLING REMOVAL

a. Check the magneto switches for OFF position.

b. Disconnect the cowl flap actuator from the cowl flap.

c. Open both cowl doors, remove the two screws on each side of the nose bug.

d. Support the lower cowl and remove either the upper or lower two screws in each cowl fairing channels aft of the propeller spinner.

e. Remove the screws across the aft end of the upper and lower cowls and remove cowls from the engine.

COWLING INSTALLATION

a. Check the magneto switches for OFF position.

b. Position the upper cowl over the engine and install screws across the aft end.

c. Support the lower cowl against the engine and install the screws across the aft end.

d. Install the four screws on each side of the nose bug and in each cowl fairing aft of the propeller spinner.

e. Connect the cowl flap actuator to the cowl flap. Check cowl flap actuator rigging.

COWL FLAP AND ACTUATOR REMOVAL

a. With the cowl flap open, disconnect the cowl flap actuator rod end from the cowl flap clip.

b. Remove the bolts from the cowl flap hinges and remove the cowl flap.

c. Remove the AMP terminals from the cowl flap actuator leads.

d. Remove the bolt holding the cowl flap actuator to the actuator support arm and remove the actuator.

COWL FLAP AND ACTUATOR INSTALLATION

a. Install the bolt attaching the upper end of the actuator to the nacelle structure.

b. Connect the AMP terminals to the actuator leads.

c. Align the cowl flap hinges and install the attaching bolts.

d. Connect the actuator rod end to the cowl flap clip. Check actuator rigging.

RIGGING THE COWL FLAP ACTUATOR

a. Set the actuator to 11.43 inches open (extended).

b. Install the actuator on the actuator support arm and attach the rod end to the cowl flap clip.

c. Adjust the actuator so that the cowl flap sides will extend into the cowl by .25 inches minimum when open. If the cowl flap does not meet the minimum required extension into the cowl, adjust the actuator to obtain this position. A minimum of .06 inch clearance is required between the actuator lug and actuator support, and between the actuator rod end lug and the cowl flap clip through the entire range of movement. Install a new cotter pin.

"END"

CHAPTER 72

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CHAPTER 72 - ENGINE

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NOTE

Refer to the Lycoming Engine Overhaul Manual, P/N 60294-6, for detailed information on the above subjects.

"END"

CHAPTER 73

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73-30-00	201	Oct 19/77

CHAPTER 73 - ENGINE FUEL AND CONTROL

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"END"

FUEL FLOW INDICATING - MAINTENANCE PRACTICES

FUEL FLOW INDICATOR

The fuel flow indicator is an aid to the pilot when leaning the engines during flight. The major components of the system are the transducer or transmitter and the indicator. The transducer or transmitter is located in the fuel line of each engine forward of the firewall. This device generates a signal that is directed to the fuel flow indicator. The fuel flow indicator gives a readout in pounds of fuel per hour. The indicator has dual pointers, one for each engine and is located in the top center of the instrument panel. The circuit breaker is located in the upper copilots side panel.

Early airplanes (prior to P-465 except P-427, P-428, P-429, P-431, P-432, P-433) have a system which operates on 26 volts ac, 400 cycles. On these airplanes there is an inverter located near fuselage station 119.28 and to the rear of the firewall of the left engine. On these early airplanes there is a fuel flow fuse in the lower pilots side panel.

On airplane serials P-427 through P-429, P-431 throu P-433 and P-466 and after the system operates on 28 vo dc, airplane bus voltage.

TRANSDUCER OR TRANSMITTER REMOVAL

a. Open the right engine cowl.
b. Disconnect the eiectrical connection.
c. Disconnect the two fuel lines, and cover the open ing to prevent contamination.
d. Remove the bolts holding the transducer or tran mitter in place, and remove the unit.

TRANSDUCER OR TRANSMITTER INSTALLATION

a. Install the transducer or transmitter with the attacl ing bolts. Safety wire the bolts.
b. Connect the two fuel lines.
c. Connect the electrical connection.
d. Close the engine cowl.

"END"

CHAPTER 74

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74-20-00	201	Nov 2/73

CHAPTER 74 - IGNITION

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"END"

GENERAL - DESCRIPTION AND OPERATION

Each engine is equipped with two Bendix S-1200 series magnetos. The left magneto incorporates a retard breaker point assembly which provides a fixed retard and long duration spark for easier starting. The right magneto has only the conventional breaker points which are grounded out when the engine is being started.

"END"

ELECTRICAL POWER SUPPLY - MAINTENANCE PRACTICES

MAGNETO DROP-OFF CHECK

T he magneto drop-off may be checked as follows:

a. Throughly warm up engine and set the propeller control in low pitch. Place the mixture control in "FULL RICH".

b. Set the throttle to produce 2000 rpm.

c. Note the amount of rpm drop-off as the magneto switch is turned from "BOTH" to "LEFT" and back to "BOTH", and then to "RIGHT" position.

CAUTION

Operation on one magneto should not exceed 5 seconds to avoid fouling the spark plugs.

d. Normal magneto drop-off is approximately 100 rpm on either magneto and should be within 50 rpm of each other. If the magneto drop-off persistently exceeds 175 rpm, an inspection to determine the cause should be accomplished. Common causes are incorrect grade of fuel, fouled or incorrectly gapped spark plugs, incorrectly timed magnetos, incorrect air/fuel ratio.

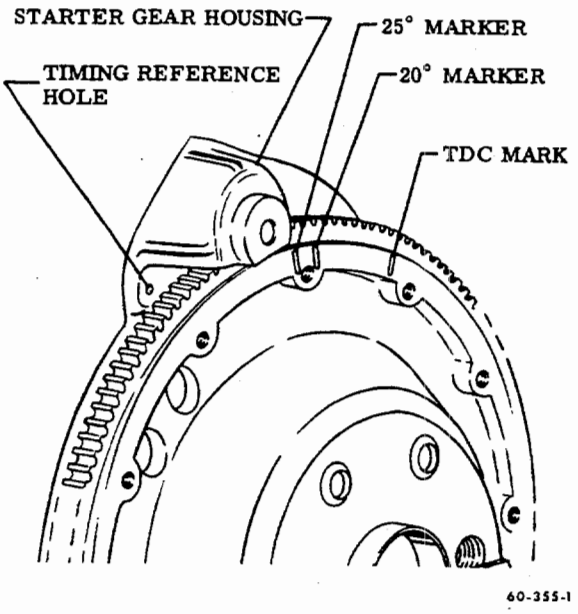
MAGNETO BREAKER POINT ADJUSTMENT

Every 100 hours check the breaker points for condition, clearance and timing. Breaker point clearances for the magnetos are .016 ± .006 for the retard points and .016 ± .003 for the conventional points. If the points are burned or worn excessively, do not try to redress the contact surfaces. Install a complete new breaker assembly if the points are found to be in an unsatisfactory condition. Wipe the breaker compartment free of any oil or dirt with a clean cloth.

MAGNETO TIMING
(Figure 201)

a. Remove a spark plug from No. 1 cylinder and turn the crankshaft in the direction of normal rotation until the compression stroke is reached.

b. Continue turning the crankshaft until the 20° BTC advance timing mark, on the forward face of the starter ring gear, is in alignment with the small hole located on the face of the starter housing.



Magneto Timing Reference Points
Figure 201

c. Remove the inspection plug on the left magneto and turn the drive coupling in the direction of normal rotation until the first marked tooth is aligned in the center of the inspection hole. Without allowing the gear to turn from this position, assemble the gasket and magneto to the engine.

d. Using an electric timing light, fasten the ground wire to any unpainted portion of the engine and one of the positive wires to a suitable terminal connected to the ground terminal of the magneto. Then turn the crankshaft several degrees from the advance timing mark in the direction opposite to that of normal rotation.

e. With the timing light on, turn the crankshaft slowly in the direction of normal rotation until the mark on the starter ring gear aligns with the hole in the starter housing. If the timing is correct the timing light should go out.

NOTE

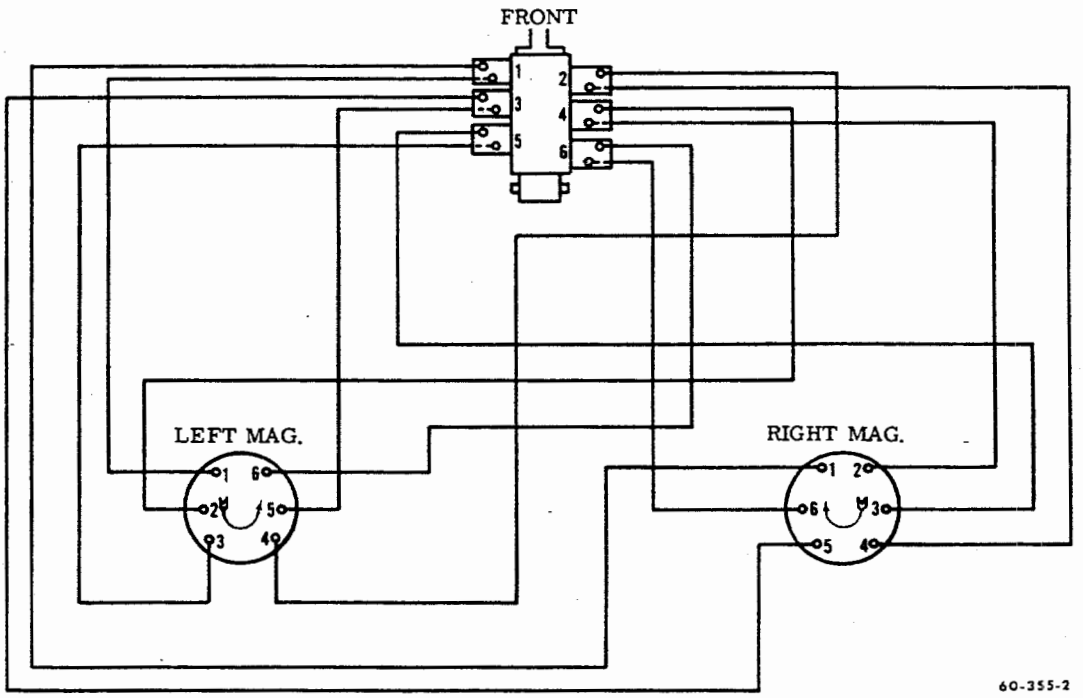
When a battery powered timing light is used, the light will go on when the marks align.

"END"

DISTRIBUTION - MAINTENENCE PRACTICES

(Figure 201)

In the event that an ignition harness or an individual lead is to be replaced, consult the Magneto Wire Routing Diagram, Figure 201, to be sure that the harness is correctly installed. Mark locations of clamps and clips to be certain that the replacement is clamped at the correct locations.



FIRING ORDER: 1-4-5-2-3-6

Magneto Wire Routing Diagram
Figure 201

"END"

CHAPTER 77

LIST OF PAGE EFFECTIVITY

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77-EFFECTIVITY/CONTENTS	1	Nov 2/73
77-00-00	201	Nov 2/73

CHAPTER 77 - ENGINE INDICATING

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"END"

GENERAL - MAINTENANCE PRACTICES

TIT INDICATOR CALIBRATION
(Figure 201)

CAUTION

Damage to the turbocharger turbine blades, excessive turbine coking and excessive oil consumption may be caused by turbine inlet temperatures above 900°C (1650°F).

To prevent a turbine inlet over-temperature condition due to an inaccurate TIT indicator reading, the indicator should be checked every 100 hours and calibrated if required.

The following procedure may be used to check and calibrate the TIT indicator.

a. Remove the TIT probe from the turbocharger intake manifold on the RH engine. (Do not disconnect the wires from the probe.)

b. Using the AICal test equipment, heat the probe to 900°C.

c. If the TIT indicator reads 900°C, the indicator is properly calibrated. If the reading is not 900°C, the calibration screw on the face of the instrument should be adjusted to obtain this reading.

d. If the seal was broken on the calibration screw, reseal by applying a small amount of torque seal as shown in Figure 201.

e. Reinstall the probe in the turbocharger intake manifold.

f. Repeat the above procedure on the LH engine.

ALCAL CALIBRATION UNIT

The AICal Calibration unit, available locally through the Beechcraft Parts and Service Outlets, provides a simple and accurate method for checking and, if necessary, recalibrating aircraft piston engine EGT systems. If the red line temperature is exceeded by the TIT indicators, the calibration unit will quickly determine if the fault lies with the indication system or the engine. The following method will accomplish the TIT calibration test:

a. Light the AICal unit and support it from the engine cowling.

b. Place the TIT thermocouple into the comparator port of the AICal unit until it is touching the reference thermocouple.

c. Raise the heat of the AICal unit until the temperature of 1650°F (900°C) is indicated on the unit's reference meter.

d. Because both thermocouples are measuring the same temperature, the aircraft-installed TIT indicator should indicate the same red line temperature. If the indicator corresponding to the engine being tested does not register 1650°F (900°C), refer to the adjustment procedure outlined under TIT INDICATOR CALIBRATION in this chapter.

e. Replace the TIT thermocouple in the turbocharger intake manifold. Lubricate the threads on the probe with MIL-A-907D anti-seize compound (Chart 208, 91-00-00).

TIT Indicator Calibration
Figure 201

"END"

CHAPTER 79

LIST OF PAGE EFFECTIVITY

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79-EFFECTIVITY/CONTENTS	1	Oct 19/77
79-00-00	1	Oct 19/77
	201	Nov 2/73
	202	Nov 2/73

CHAPTER 79 - OIL

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"END"

GENERAL - DESCRIPTION AND OPERATION

The engines are equipped with a wet sump, pressure type oil system with a capacity of 13 quarts. The sump fastens to the bottom of the crankcase and incorporates two oil drain plugs and the oil suction screen housing. A pressure relief valve installed on P-227and after protects the oil radiator, on these serials, from damage caused by cold weather pressure surges. The oil system may be checked through the access doors in the engine cowling. A calibrated dip stick attached to the filler cap indicates the oil level. Due to the canted position of the engines, the dip sticks are calibrated for either the right or left engine and are not interchangeable.

"END"

GENERAL - MAINTENANCE PRACTICES

The oil should be changes every 75 to 100 hours under normal operating conditions and the oil filter changed every 50 hours.

All TIO-541 series engines are limited to using only ashless dispersant multi-grade oil conforming to MIL-L-22851 (see Chart 201) or a Lycoming approved synthetic oil. Oil equivalent to SAE 50 or SAE 60 above 60°F; below 30°F SAE 40 is recommmended. However, newly overhauled engines of this series may be run-in on the test stand with single viscosity, grade SAE 50 oil conforming with MIL-L-6082, (6, Chart 207, 91-00-00).

The determining factor for choosing the correct grade of oil is the oil inlet temperature observed during flight; however, inlet temperatures consistently near the maximum allowable indicate a heavier oil is needed. Lycoming recommends use of the lightest weight oil that will give adequate cooling.

CHANGING THE OIL FILTER

The oil filter should be replaced every 50 hours. Replace the filter as follows:

a. Disconnect the fuel line to the inlet side of the engine-driven fuel pump. Cap the fuel line (P-4 through P-20) and push the line down to allow clearance for oil filter removal.

b. Cut the safety wire and remove the filter housing.

c. After making sure all traces of gasket material and cement are removed from the oil filter adapter recess, install the new aluminum ring and O-ring seal which replace the existing rubber gasket. (Refer to Lycoming Service Bulletin No. 337.)

d. Torque the retainer bolt to 25 to 30 foot-pounds and safety.

e. Remove cap (P-4 through P-20) and reinstall the fuel line.

CHART 201
APPROVED ENGINE OILS (ASHLESS DISPERSANT)

SPECIFICATION	*PRODUCT*	**VENDOR*
MIL-L-22851	Global Concentrate A	Delta Petroleum Company Inc. P. O. Box 10397 New Orleans, La. 70121
	Paranox 160 and 165	Enjay Chemical Company 60 West 49th Street New York, N. Y. 10020
	RT-451, RM-173E, RM-180E	Mobil Oil Corporation 150 East 42nd Street New York, N. Y. 10017
	Shell Concentrate A Code 60068 Aeroshell W120 Aeroshell W80	Shell Oil Company One Shell Plaza Houston, Texas 77002
	TX-6309 Aircraft Engine Oil Premium AD120 Aircraft Engine Oil Premium AD80	Texaco Inc. 135 East 42nd Street New York, N. Y. 10017
	PQ Aviation Lubricant 753	American Oil and Supply Co. 238 Wilson Avenue Newark, N. J. 07105

CHAPTER 80

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80-10-00	101	Nov 2/73
	201	Nov 2/73

CHAPTER 80 - STARTING

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"END"

GENERAL - DESCRIPTION AND OPERATION

Duke series aircraft are equipped with a 24-volt starter which engages with the accessory drive gear. The starter is located on the top forward end of each engine.

When the ignition switch is placed in the START position current is supplied by the battery bus which energizes the applicable starter relay providing current to the starter.

"END"

TROUBLESHOOTING
STARTER SYSTEM

	TROUBLE		*PROBABLE CAUSE*		*REMARKS*
1.	Both starters inoperative.	a.	Circuit breaker tripped in starter switch circuit.	a.	Check for short circuit; reset.
		b.	Battery relay inoperative.	b.	Check continuity of battery system.
		c.	Low battery.	c.	Test battery. If low, replace or start with external power.
		d.	Loose connections or open circuit between battery relay and left starter relay.	d.	Check connections and continuity.
2.	One starter inoperative.	a.	Starter relay inoperative.	a.	Check relay terminal connections and continuity of solenoid energizing circuit. If energizing circuit is closed and relay does not operate, replace relay.
		b.	Poor ground at starter.	b.	Test continuity from armature lead to ground. Repair if necessary.
		c.	Open circuit.	c.	Check continuity to starter.
		d.	Defective starting motor.	d.	Check brushes, springs, condition of commutator; replace if necessary.

"END"

CRANKING - MAINTENANCE PRACTICES

STARTER LUBRICATION

No lubrication is required on the starting motor except at overhaul.

STARTER REMOVAL

a. Open engine cowling.
b. Disconnect and tape starter lead.
c. Remove starter mounting nuts.
d. Lift starter up and aft to remove.

STARTER INSTALLATION

a. Place starter into position and install mounting nuts and cap screws. Tighten securely.
b. Connect starter lead.
c. Fasten engine cowling.

STARTER CIRCUIT CHECKS

a. The starter circuit wiring should be inspected at regular intervals to determine that all connections are clean and tight and that the insulation is sound.

b. The starter circuit should be checked to determine if there is excessive resistance in the circuit. This test is made with a low reading voltmeter while cranking the engine.

1. The voltage loss from the battery positive terminal to the starter terminal should not exceed 0.3 volts.
2. The voltage loss from the battery ground terminal to the starter frame should not exceed 0.1 volt.

If there are greater voltage losses than indicated above, additional checks should be made to locate the high resistance connections.

STARTER BRUSHES

The starter brushes should slide freely in their holders and make full contact on the commutator. The brushes should be replaced when they have worn to one half their original length (approximately 1/4 inch). Proper brush spring tension with new brushes installed is 32 to 40 ounces. This tension is measured with a scale hooked under the brush spring near the brush and the reading taken just as the spring leaves the brush.

"END"

CHAPTER 81

LIST OF PAGE EFFECTIVITY

CHAPTER 81 - TURBINES

TABLE OF CONTENTS

"END"

GENERAL - DESCRIPTION AND OPERATION

TURBOCHARGER SYSTEM

(Figure 1)

The turbocharger is standard equipment on the Duke. It increases the power output and efficiency of the engine by supplying compressed air to the intake manifold. In operation, engine exhaust gas passing over the turbine wheel causes the turbocharger compressor, mounted on the same shaft, to rotate. Ambient, ram air, supplied through the RH cowl door, is filtered and routed to the compressor where it is compressed and delivered to the engine and through a sonic nozzle to the cabin. The sonic nozzle, located on the intake housing, between the turbocharger compressor and throttle valve, supplies air flow for cabin pressurization. As the engine power increases, the flow of exhaust also increases, resulting in a proportionate increase in the speed of the rotating assembly and turbocharger output.

CONTROL SYSTEM

The turbocharger control system is automatic and functions continuously as engine power, speed and altitude are varied. The variable pressure controller, wastegate, wastegate actuator and engine oil are the principal components of the control system. The pressure controller senses compressor outlet pressure and regulates the oil pressure controlling the wastegate actuator position. The wastegate actuator is a hydraulic cylinder with spring tension holding the wastegate butterfly valve open. When oil pressure increases in the actuator, the spring tension is overcome and the butterfly valve closes, routing all exhaust through the turbocharger turbine. The variable pressure controller regulates the oil pressure in the actuator by means of an aneroid bellows which is sensitive to pressure changes at the induction manifold. The metering valve, which is connected to the bellows within the controller, is held closed by spring tension and vacuum. As the induction manifold pressure increases, the force of the aneroid bellows causes the metering valve to open. The controller is regulated by a cam which is connected to the throttle valve. Through this linkage, the pressure setting of the controller is varied proportionally to the amount of power the pilot selects with the throttle. The control system prevents the engine from exceeding 41.5 in. Hg manifold pressure; however, rapid movement of the throttle with low oil temperature or operation at low rpm with high manifold pressure may result in an overboost condition. An overboost condition may cause turbocharger surge, detonation or detuning of the engine counterweight system; any of which may cause serious engine damage.

OVERBOOST CONTROL

On serials P-247 and after, the engine incorporates a relief valve in the induction system which is set to relieve at approximately 44 in. Hg. (See Figure 1.) This valve will open only in the event of a malfunction in the variable absolute pressure controller system.

CAUTION

To avoid exceeding the normal limits, particularly in cold weather, the last 1-1/2 inches of throttle travel should be applied slowly while observing the manifold pressure. Momentary overboost to the limits of the relief valve (44 in. Hg) will have no detrimental effect on the engine, but is indicative of a malfunctioning variable absolute pressure controller. If overboost is more than momentary, or occurs when engine oil temperatures are normal, the controller should be checked by a authorized facility.

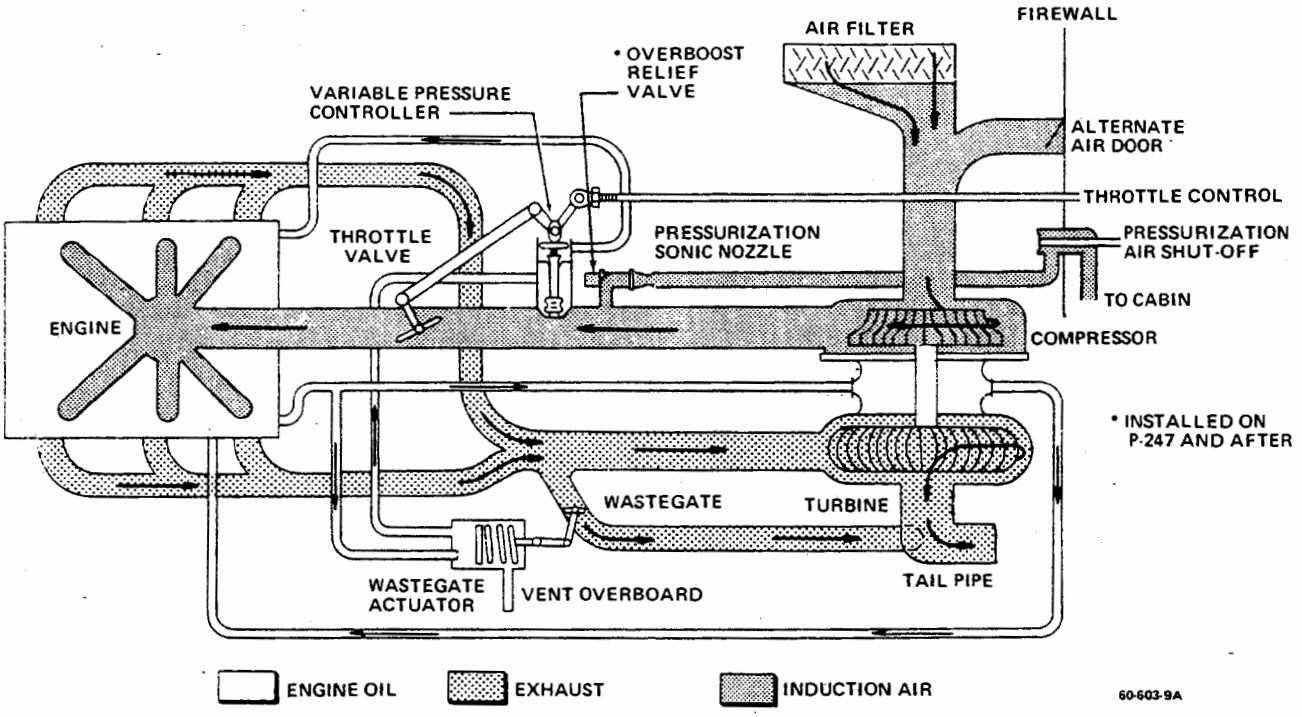
ENGINE AIR INDUCTION SYSTEM

Engine induction air is available as two sources, primary and alternate air. The primary air source is supplied through an intake duct, located on the engine RH cowl door, passes through an air filter, and then into the turbocharger.

NOTE

The air filter, located in the air box assembly, has a service life of 500 hours with periodic cleaning.

When the primary source of air is obstructed, the turbocharger forms a suction that opens the "Alternate Air Source" door and permits the required volume of air flow for normal engine performance. The alternate air door is located on the firewall behind the induction air box assembly.



Turbocharger System
Figure 1

"END"

TROUBLESHOOTING
TURBOCHARGER

	TROUBLE		*PROBABLE CAUSE*		*REMARKS*
1.	Excessive noise or vibration.	a.	Improper bearing lubrication.	a.	Supply required oil pressure. Clean or replace oil line. If trouble continues, remove turbocharger and return to approved overhaul station for overhaul or repair.
		b.	Leak in engine intake or exhaust manifold.	b.	Tighten loose connections, or replace manifold gaskets as necessary.
2.	Engine will not deliver rated power.	a.	Clogged manifold system.	a.	Clean all ducting.
		b.	Foreign material lodged in compressor impeller or turbine.	b.	Remove turbocharger and return to approved overhaul station for overhaul or repair.
		c.	Excessive dirt buildup in compressor.	c.	Service engine induction air filter and check for leakage. Remove turbocharger and return to approved overhaul station for overhaul or repair.
		d.	Leak in engine intake or exhaust manifold.	d.	Tighten loose connections, or replace manifold gaskets as necessary.
		e.	Rotating assembly bearing seizure.	e.	Remove turbocharger and return to approved overhaul station for overhaul or repair.
		f.	Restriction in return lines from actuator to wastegate controller.	f.	Remove and clean lines.
		g.	Wastegate controller out of adjustment.	g.	Have wastegate controller adjusted.
		h.	Oil pressure too low.	h.	Tighten fittings, replace lines or hoses. Increase oil pressure.
		i.	Inlet orifice to actuator clogged.	i.	Remove inlet line at actuator and clean orifice.
		j.	Wastegate controller malfunction.	j.	Replace unit.
		k.	Wastegate butterfly not closing.	k.	Low pressure, butterfly shaft binding.

TROUBLESHOOTING
TURBOCHARGER (Cont'd)

	TROUBLE		*PROBABLE CAUSE*		*REMARKS*
2.	Engine will not deliver rated power (Cont'd).	l.	Impeller binding, frozen or fouling housing.	l.	Remove turbocharger and return to approved overhaul station for overhaul or repair.
		m.	Piston seal in actuator leaking.	m.	Replace actuator or disassemble and replace packing.
3.	Critical altitude lower than specified.	a.	Controller not getting enough oil pressure to close by-pass valve.	a.	Check pump outlet pressure, oil filters and lines for leaks or obstructions.
		b.	Chips under metering valve in controller holding it open.	b.	Replace controller.
		c.	Metering jet in actuator plugged.	c.	Remove actuator and clean jet.
		d.	Actuator piston seal leaking excessively.	d.	Clean cylinder and replace piston seal.
		e.	Wastegate valve sticking.	e.	Clean and free action.
4.	Engine surges or smokes.	a.	Air in oil lines or actuator.	a.	Bleed system.
		b.	Control metering valve stem seal leaking oil into manifold.	b.	Replace controller.
		c.	Actuator to by-pass valve linkage binding.	c.	Correct cause of binding.
		d.	Clogged breather.	d.	Check breather for restriction to air flow.

NOTE

Smoke would be normal if engine has idled for a prolonged period.

5.	High deck pressure (Compressor discharge pressure).	a.	Controller metering valve not opening.	a.	Replace controller.
		b.	Exhaust by-pass valve sticking closed.	b.	Shut-off valve in return line inoperative.
		c.	Controller return line restricted.	c.	Clean or replace line.
		d.	Oil pressure too high.	d.	Reduce oil pressure.

TROUBLESHOOTING
TURBOCHARGER (Cont'd)

TROUBLE		*PROBABLE CAUSE*		*REMARKS*
5. High deck pressure (Compressor discharge pressure) (Cont'd).	e.	Wastegate actuator piston locked in closed position.	e.	Disassemble actuator, check condition of piston and packing.
	f.	Wastegate controller malfunction.	f.	Replace controller.

"END"

GENERAL - MAINTENANCE PRACTICES

INSTALLATION AND INITIAL RUN-IN OF TURBOCHARGER

Immediately prior to mounting the unit, prime the turbocharger lubrication system by inverting the turbocharger and filling the center housing with new, clean oil through the oil drain. Rotate the assembly by hand to coat the bearings and the thrust washer with oil.

Coat the threads of the attaching bolts or studs with high temperature thread lubricant. Connect the ducts and make sure all connections are air tight.

Flush oil through the oil supply line to assure the line is clean and unobstructed. Connect the oil supply line at the engine. To be sure that oil is being supplied to the turbocharger, hold the compressor impeller by hand and start the engine.

WARNING

Do not attempt to stop impeller after unit is rotating.

As soon as oil appears at the end of the oil inlet line, attach the line to the turbocharger and allow the rotating assembly to spin.

Operate the engine at a load and listen for sounds of metallic contact from the turbocharger. If any such noise is apparent, shut down immediately and replace the unit.

For a list of approved turbocharger overhaul and repair facilities, refer to the Component Maintenance Manual P/N 60-590001-27.

RIGGING THE THROTTLE AND TURBOCHARGER PRESSURE CONTROLLER LINKAGE
(Figure 201)

a. With the injector connecting rod installed, determine that the throttle lever moves freely from idle to full open throttle.

b. Adjust the pressure controller rod so that, with the pressure controller cam arm against the full boost stop (full forward position), the throttle lever is approximately .020 - .030 inch from the full throttle position.

VARIABLE PRESSURE CONTROLLER ADJUSTMENT
(Figure 202)

The variable pressure controller is mounted directly to the turbocharger discharge ducting between the oil filler neck and the engine throttle valve.

Adjustment of the controller is made as follows:

a. Head the aircraft into the wind. Set the brakes and chock the wheels.

b. Warm up the engine until the oil temperature reaches a minimum of 185° F.

c. Set the propeller control lever in the high RPM position. Slowly and smoothly apply the throttle until 41 in. Hg manifold pressure or the full throttle position is reached.

CAUTION

Do not exceed 41.5 in. Hg manifold pressure.

d. If at the full throttle position the manifold pressure has not reached 41 in. Hg:

1. Slowly and smoothly shut down the engine.
2. Loosen the locknut on the adjusting screw.
3. Turn the adjusting screw counterclockwise to increase the manifold pressure. (One full turn equals approximately 1 in. Hg manifold pressure).
4. Retighten the locknut on the adjusting screw.

e. If the manifold pressure reaches 41 in. Hg before obtaining full throttle:

1. Slowly and smoothly shut down the engine.
2. Loosen the locknut on the adjusting screw.
3. Turn the adjusting screw clockwise to decrease the manifold pressure. (One full turn equals approximately 1 in. Hg manifold pressure).
4. Retighten the locknut on the adjusting screw.

CAUTION

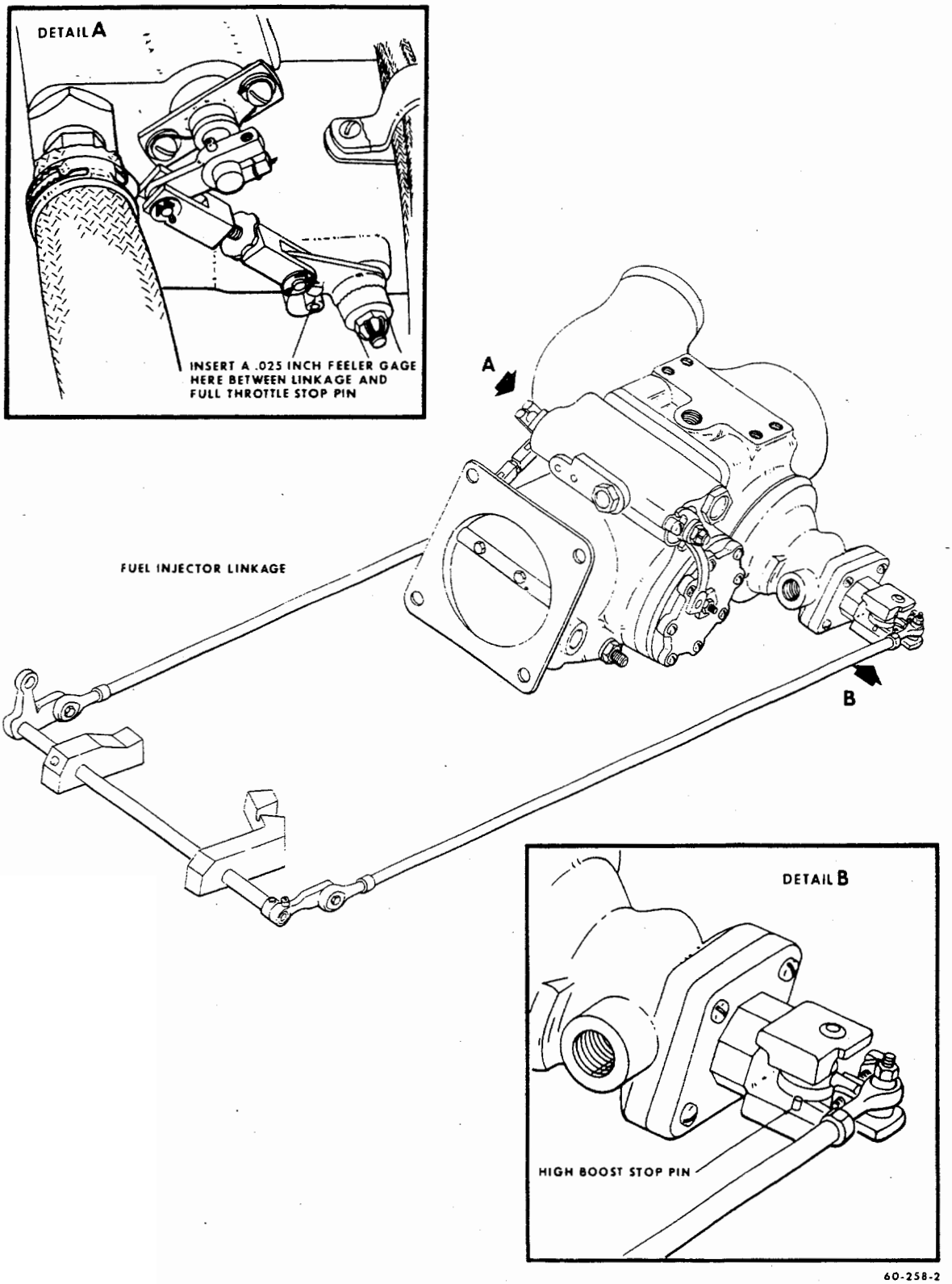
Do not exceed 41.5 in. Hg manifold pressure.

f. Repeat steps "b." through "e." until the manifold pressure at full throttle is 41 in. Hg.

LUBRICATION OF WASTEGATE BUTTERFLY SHAFTS

Rust deposits may form in the area of the wastegate butterfly shaft bosses as a result of water vapor accumulation if the aircraft is subjected to short intervals of engine operation.

This condition occurs only when the unit is new and combustion deposits have not formed a protective barrier



Throttle and Turbocharger Controller Adjustment
Figure 201

Variable Pressure Controller
Figure 202

on the shaft surface. Units which are binding after long service time are coked internally and must be removed for cleaning or replacement.

When this condition is noted, remove the exhaust discharge stack and apply Mouse Milk or Kano Kroil (37, Chart 207, 91-00-00) liberally to the shaft and boss. After a few minutes attempt to turn the shaft. A light tap on the shaft end will assist in freeing the shaft. Once the shaft is free the engine may be started and a power check made to confirm the turbocharger output, either on the ground or in flight.

ADJUSTMENT OF TURBOCHARGER WASTEGATE VALVE
(Figure 203 or 204)

The linkage connecting the butterfly valve to the actuator is adjustable. It is adjusted at the time of valve installation so that the piston in the actuator cylinder bottoms at the same time, or just before, the butterfly valve seats in its bore. Misadjustment of the linkage may cause the butterfly valve to seat before the actuator piston bottoms and will result in damage to the linkage, as the hydraulic closing forces are high at engine idle or during cold engine operation. The linkage adjustment is used to establish critical altitude, which is the altitude at which the wastegate butterfly just fully closes. The adjustment must be made with full hydraulic or air pressure applied to the inlet port of the wastegate actuator. The full open butterfly stop adjustment, located on the center of the actuator cover, may be adjusted to stop the butterfly at the position required for safe engine operation. In the event the butterfly valve fails to close or fully open, adjustment of the valve is made as follows:

a. Disconnect both the inlet and outlet oil lines at the wastegate actuator. Plug the actuator outlet port and connect an air pressure supply line to the inlet port. This line must have a pressure gage installed to maintain 50-60 psi into the wastegate actuator.

b. Loosen the locknut on the adjusting turnbuckle and remove the cotter pin, washers and pin.

c. Apply 50 - 60 psi to the wastegate actuator and adjust the closed position of the wastegate butterfly valve by rotating the turnbuckle counterclockwise to fully close the wastegate butterfly. After the butterfly has made contact with the bore, back off the turnbuckle clockwise until the hole and slot align.

NOTE

Maintain a clearance of .005 to .025 (P-4 through P-143), .005 to .015 (P-144 and after) between butterfly edge and bore.

d. Reinstall the pin, washer and cotter pin. Tighten the locknut against the clevis with 80 to 100 in-lbs. torque.

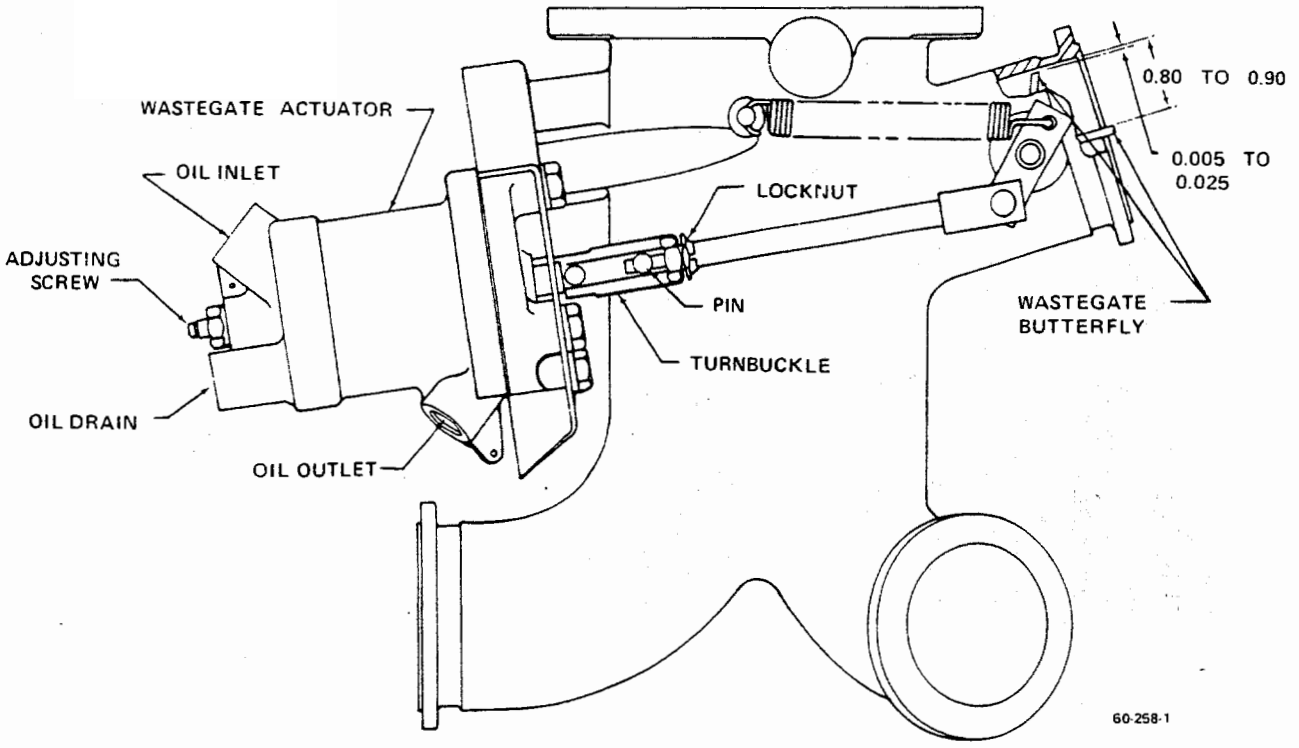
e. With zero air pressure in the wastegate actuator, adjust the full open stop position of the butterfly valve with the adjusting screw located on the end of the actuator.

NOTE

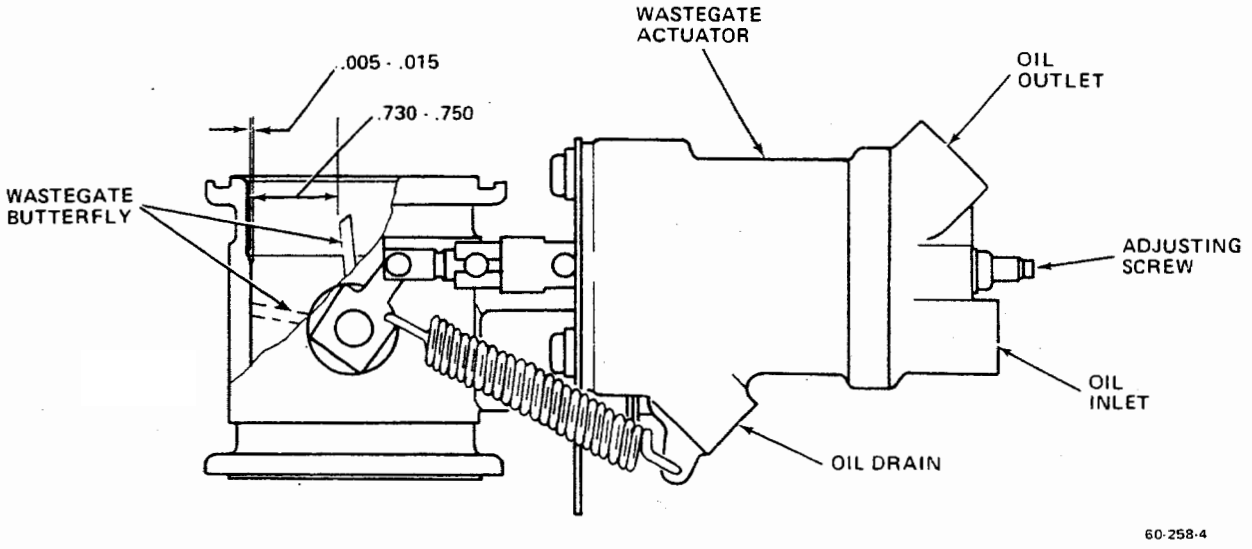
Maintain a minimum clearance of .80 to .90 (P-4 through P-143), .730 to .750 (P-144 and after) between butterfly edge and bore.

TURBOCHARGER CRITICAL ALTITUDE TEST
(Figure 205)

The following procedure provides a means of checking turbocharger performance. Refer to the Turbocharger Performance Graph. This graph indicates the minimum acceptable critical altitude the aircraft can achieve while maintaining 41.0 in. Hg manifold pressure. To check the turbocharger performance against the graph it will be necessary to flight test the aircraft. Place the aircraft in a climb configuration and note the altitude at which the manifold pressure begins to drop off; then observe the outside air temperature gage. Locate these points on the axes of the graph and project lines from these points toward the center of the graph. The point at which the lines intersect is the aircraft's critical pressure altitude. If this point is located below the minimum acceptable pressure altitude line on the graph, a thorough check of the turbocharging system, including variable controller, induction system leaks and wastegate adjustment, should be accomplished.

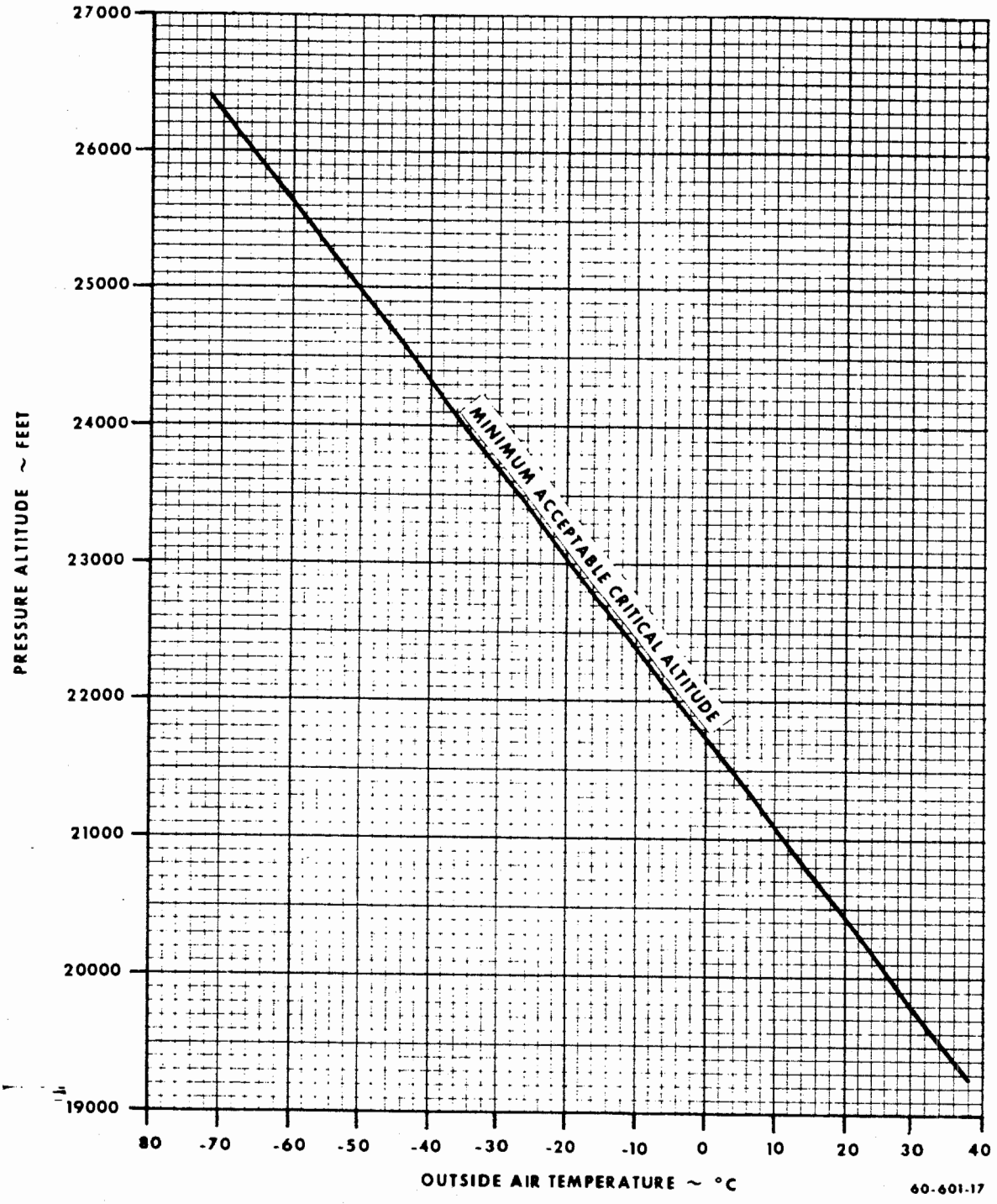


Wastegate Adjustment (P-4 thru P-143)
Figure 203



Wastegate Adjustment (P-144 and after)
Figure 204

CRITICAL PRESSURE ALTITUDE VS OAT



Turbocharger Performance Graph
Figure 205

"END"

CHAPTER 91

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	208A	Oct 27/75
	209	Oct 27/75
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	211	Nov 30/83
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"END"

CHART 91 - CHARTS

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"END"

CHART 201

TABLE OF TORQUES

ENGINE MOUNTING

Engine Mount Bracket Bolts And Nuts (At Firewall)	350 to 390 in. lbs.
Engine Shock Mount Bolts And Nuts	250 to 300 in. lbs
Engine Mounting Bracket Bolts	270 in. lbs.

ENGINE COMPONENTS

Air Conditioner Compressor Mounting Nuts	160 to 190 in. lbs.
Generator Mounting Bracket Bolts (Side Bracket)	150 in. lbs.
(Bottom)	75 in. lbs.
Engine Spark Plugs	360 to 420 in. lbs.
Engine Spark Plug Lead Connections	25 in. lbs.
Engine Oil Filter Retainer Bolt	25 to 30 ft. lbs.
Propeller Starter Gear/Slip Ring to Crankshaft	60 to 70 in. lbs.
Turbocharger Exhaust Clamp P/N MVT-68637-450-M	50 to 60 in. lbs.

HORIZONTAL AND VERTICAL STABILIZER

All 5/16 - 24 bolts	100 to 140 in. lbs.
All 3/8 - 24 bolts	160 to 190 in. lbs.

WING MOUNTING

NOTE

Wing mounting bolt torque should be checked at the first 100 hour inspection and 100 hours after wing reinstallation, replacing a wing bolt(s) or adjusting the wing. If it is necessary to retorque the wing bolts at this time, the bolts should then be checked at the next 100 hour inspection. This check should continue at each 100 hour inspection until it is no longer necessary to retorque the bolts. Check bolt torque at the nut end only.

Leading Edge Wing Mounting Bolt (Dry Torque Only)	100 to 140 in.-lbs
Upper Front Wing Mounting Bolt ((Wet Torque Only)	2480/2600 in.-lbs
Upper Rear Wing Mounting Bolt (Wet Torque Only)	1180/1300 in.-lbs
Lower Rear Wing Mounting Bolt (Wet Torque Only)	2480/2600 in.-lbs
Lower Forward Wing Mounting Bolt (See Chapter 57-00-00 for Torquing Instructions)	

LANDING GEAR

Strut Connecting Arm Bolts	290 to 410 in. lbs.
Horizontal Brace Bolts	25 to 40 in. lbs.
Main Gear Axle Nut	15 to 20 ft. lbs.
Main Gear Hinge Bolts	250 to 800 in. lbs.

CHART 201 (Cont'd)

TABLE OF TORQUES

FUEL SYSTEM (Attaching Bolts For)

Filler Necks	45 to 55 in. lbs.
Nacelle Transmitter	45 to 55 in. lbs.
Leading Edge Fuel Cell Outlet Plate	20 to 30 in. lbs.
Transmitter	25 ± 5 in. lbs.
Fuel Boost Pump	45 to 55 in. lbs.
Access Plates	45 to 55 in. lbs.
Fuel Cell Interconnect Clamps	25 ± 5 in. lbs.
Rubber Fuel Nipple Clamps	25 ± 5 in. lbs.

HEATER SYSTEM

Spark Plug	28 ft. lbs.
Spark Plug High Voltage Lead	20 ft. lbs.
Fuel Spray Nozzle	20 ft. lbs.

CHART 202

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 manual.

SIZE	*TORQUE LIMITS RECOMMENDED (INCH-POUNDS)*		*MAXIMUM ALLOWABLE TORQUE (INCH-POUNDS)*	
	AN365 AND AN310 NUTS COLUMN 1	AN364 AND AN320 NUTS COLUMN 2	AN365 AND AN310 NUTS COLUMN 3	AN364 AND AN320 NUTS COLUMN 4
8-36	12-15	7-9	20	12
10-32	10-25	12-15	40	25
1/4 - 28	50-70	30-40	100	60
5/16 - 24	100-140	60-85	225	140
3/8 - 24	160-190	95-113	390	240
7/16 - 20	450-500	270-300	840	500
1/2 - 20	480-690	290-410	1100	660
9/16 - 18	800-1000	480-600	1600	960
5/8 - 18	1100-1300	660-780	2400	1400
3/4 - 16	2300-2500	1300-1500	5000	3000
7/8 - 14	2500-3000	1500-1800	7000	4200
1 - 14	3700-5500	2200-3300	10000	6000
1 - 1/8 - 12	5000-7000	3000-4200	15000	9000
1 - 1/4 - 12	9000-11000	5400-6600	25000	15000

NOTE

The above values apply to Class 3 threads, cadmium plated and non-lubricated.

CHART 203

TORQUING COARSE THREAD SERIES BOLTS LOADED IN SHEAR

SIZE	*TORQUE LIMITS RECOMMENDED (INCH-POUNDS)*		*MAXIMUM ALLOWABLE TORQUE (INCH-POUNDS)*	
	AN365 AND AN310 NUTS COLUMN 1	AN364 AND AN320 NUTS COLUMN 2	AN365 AND AN310 NUTS COLUMN 3	AN364 AND AN320 NUTS COLUMN 4
8-32	12-15	7-9	20	12
10-24	20-25	12-15	35	21
1/4 - 20	40-50	25-30	75	45
5/16 - 18	80-90	48-55	160	100
3/8 - 16	160-185	95-110	275	170
7/16 - 14	235-255	140-155	475	280
1/2 - 13	400-480	240-290	880	520
9/16 - 12	500-700	300-420	1100	650
5/8 - 11	700-900	420-540	1500	900
3/4 - 10	1150-1600	700-950	2500	1500
7/8 - 9	2200-3000	1300-1800	4600	2700
1 - 8	3700-5000	2200-3000	7600	4500
1 - 1/8 - 8	5500-6500	3300-4000	12000	7200
1 - 1/4 - 8	6500-8000	4000-5000	16000	10000

NOTE

The above values apply to Class 3 threads, cadmium plated and non-lubricated.

CHART 204

FLARE FITTING TORQUE CHART

TORQUE – INCH-POUND

TUBING OD INCHES	*ALUMINUM - ALLOY AND TUBING FLARE 10061 OR AND 10078*		*STEEL TUBING FLARE AND 10061*		*HOSE END FITTING AND HOSE ASSEMBLIES*	
	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
3/4	300	500	900	1000	500	850
1	500	700	1200	1400	700	1150
1-1/4	600	900	---	---	---	---
1-1/2	600	900	---	---	---	---
1-3/4	---	---	---	---	---	---
2	---	---	---	---	---	---

INSTALLATION OF FLARED FITTINGS

When installing flare fittings, make sure the male threads are properly lubricated in accordance with Chart 208. Torque the fittings in accordance with Chart 204 above. Do not overtorque.

CHART 205

SEALING MATERIALS

SEALING

Because the BEECHCRAFT Duke is a pressurized aircraft, sealing the skin and bulkhead seams, the windows, doors, etc., is of prime importance. Control cables and torque shafts have removable rubber seals. When making a structural repair or modification which creates a break in the pressure vessel, the mating surfaces must be sealed with the proper sealer. All other components piercing the pressure vessel or attached to it must be sealed with the sealers described in Chart 205. To assure effective bonding of the sealers, be sure to clean all mating surfaces, mating parts and rubber seals thoroughly.

ITEM	*PRODUCT*	*VENDOR*
1.	A56B Cement	B. F. Goodrich Co., Akron, Ohio
2.	EC-2141	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
3.	Permatex Sealer No. 2	Permatex Co., Inc., Kansas City, Kansas
4.	Presstite Sealer No. 155	Presstite Engineering Co., St. Louis, Missouri
5.	EP-711 Sealer	Coast Pro Seal and Mfg. Co., Los Angeles, California
6.	EC-1792	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
7.	EC-1239 Sealer	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
8.	Y-9136 Teflon Tape	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
9.	PR-1221 Sealer	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
10.	EC-750	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
11.	EC-801	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
12.	EC-1300L	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
13.	EC-612	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
14.	EC-776	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
15.	EC-801A-1/2	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
16.	EC-1368	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota

Chart 205 (Cont'd)

SEALING MATERIALS

ITEM	*PRODUCT*	*VENDOR*
17.	EC-1403	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
18.	Bostic 1008	United Shoe Machinery Corp., B. B. Division, 748 Memorial Drive, Cambridge, Massachusetts
19.	Pro Seal 890 B-1/2	Coast Pro-Seal and Manufacturing Co., Los Angeles, California
20.	EC-1675 B-1/2	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota

NOTE

Proper procedure for applying the sealers may be found in the directions on the sealer container and/or in the chapter of this manual applying to that area being worked.

CHART 206

APPLICATION OF SEALING MATERIALS

APPLICATION	*MATERIAL*	*REMARKS*
Metal Surfaces and Wet Wing Tip	O-A-51, Acetone; MIL-M-13999, Methyl Ethyl Ketone; *TT-N-95, Naphtha	Clean all surfaces prior to application of adhesive or sealer. Adhesives or sealers may be applied to zinc chromate primed surfaces.
Synthetic or Natural Rubber	MIL-M-13999, Methyl Ethyl Ketone; TT-T-548, Toluol	Clean all surfaces to be bonded. Where possible, scuff surface by wire brushing or sanding.
Silicone Rubber	TT-X-916, Zylene	Clean all surfaces with zylene only when a surface lubricant, i.e. soapstone has been applied by the manufacturer.
Rubber Door Seals	EC-847	Rubber seals are installed to prevent air leakage. The seals must be cut with mitered corners to fit the periphery of the door jam. Apply a thin coating of adhesive to both the areas where the seal will be placed and the side of the seal to be bonded. Allow to dry completely, then lightly wipe the surface to be bonded with a cloth dampened with MIL-M-13999 methyl ethyl ketone to reactivate the adhesive. Position the seal carefully as it has immediate adhesion when laid in place.
Unused Drill Holes	EC-1239, *PR-890 B-1/2 or *EC-1675 B-1/2	Plug holes which go through pressure boundaries with rivets, or seal with EC-1239.
Plumbing Fittings	EC-1239 B-1/2 or PR-1221 B8	Apply to all plumbing fittings which pass through a pressure boundary.
Electrical Fittings	EC-1239, EC-776	Seal all electrical plugs which pass through a pressure boundary with EC-1239. Apply EC-776 on the screw head used for attachments upon completion of the installation.

CHART 206 (Cont'd)

APPLICATION OF SEALING MATERIALS

APPLICATION	*MATERIAL*	*REMARKS*
Faying Surface Seal	PR-1221 B8, EC-1239 B-1/2 EC-1792 *PR-890 B-1/2 *EC-1675 B-1/2	Apply with a bead approximately 1/8 to 3/16 inch in height, depending on the width of the lapped skin or angles. Apply sufficient sealant so there will be a slight squeeze out of sealant along the edge on the pressure side of the assembly when the assembly is completed.
Fillet Seal	PR-1221 B8, EC-1239 B-1/2 *PR-890 B-1/2 *EC-1675 B-1/2	Apply with a gun capable of extruding beads at least three times the thickness of the skin being sealed. Work the tip of the gun so it packs the sealant into the cracks, crevices or skin lap. The fillet must make complete contact with the immediate area.
Gap Seal	EC-1239 *PR-890 B-1/2 *EC-1675 B-1/2	Apply with a gun or spatula over the edges of the flanges forming at least 1/4 inch lap with a center thickness of 1/4 to 1/2 inch, depending on the size of the hole. Gaps and voids larger than 3/8 inch in two dimensions should be sealed with a mechanical closure.
Coating Seal	EC-776 *PR-890 B-1/2 *EC-1675 B-1/2	Apply as necessary to prevent pressure leaks on an area double the size of the rivet butt, working the sealer with the brush to insure that the rivet is completely covered.
Window Seal	EC-1202	Apply around periphery of all window panels prior to installation in frames. Tape width is to extend equally on both sides of panels.
Sealant Ribbon	Prestite 155	Apply around periphery of all removable inspection doors and panels.

*Wet Wing Tip

CHART 206 (Cont'd)

APPLICATION OF SEALING MATERIALS

TERMS

Faying Surface Seal refers to a seal where the sealing material is applied between the joint surfaces before assembly.

Fillet Seal refers to a seal over a joint or seam that fills the gap. It is not to be applied in the nature of a bridge.

Gap Seal refers to a seal where the sealing material is packed into a gap or void and lapped over the edges. These seals should not be used on gaps or voids larger than 3/8 inch in two dimensions.

Pot Life refers to the workable life of a two-part sealant mixture.

Sealing Ribbon refers to an extruded section ribbon sealer used on removable inspection doors.

Sealing Tape refers to a sealer made in the form of a tape and supported by a loose woven tobacco cloth. This tape is used in areas where a wide flat layer of sealer is required, such as a window installation.

CHART 207

CONSUMABLE MATERIALS

Only the basic number of each Military Specification is included in the Consumable Materials Chart. No attempt has bee made to update the basic number with the letter suffix that designates the current issues of the various specifications.

Vendors that are listed as meeting the Federal and Military Specifications are provided as reference only and are nc specifically recommended by Beech Aircraft Corporation; consequently, any product conforming to the specification listed ma be used. The products listed below have been tested and approved for aviation usage by Beech Aircraft Corporation, th vendor, or by compliance with the applicable specifications. Other products that are locally procurable which conform to th requirements of the applicable Military Specification may be used even though not specifically included herein.

It is the responsibility of the operator/user to determine the current revision of the applicable Military Specification prior t usage of that item. This determination may be made by contacting the vendor of a specific item.

ITEM	*MATERIAL*	*SPECIFICATIONS*	*VENDOR PRODUCTS*
1.	Fuel, Engine	100/130 octane (If not available, use 115/145) Never use 91/96 octane fuel.	
2.	Oil, Engine (Ashelss Disperant)	MIL-L-22851	Global Concentrate A Delta Petroleum Company Inc., P. O. Box 10397 New Orleans, La. 70121 Paranox 160 and 165 Enjay Chemical Company 60 West 49th Street New York, N. Y. 10020 RT-451, RM-173E, RM-180E Mobil Oil Corporation 150 East 42nd Street New York, N. Y. 10017 Shell Concentrate A Code 60068 Aeroshell W120, Aeroshell W80 Shell Oil Company One Shell Plaza Houston, Texas 77002 TX-6309 Aircraft Engine Oil, Premium AD120, Premium AD80 Texaco Inc. 135 East 42nd Street New York, N. Y. 10017 PQ Aviation Lubricant 753 American Oil and Supply Co. 238 Wilson Avenue Newark, N. J. 07105

CHART 207 (Cont'd)
CONSUMABLE MATERIALS

ITEM	MATERIAL	SPECIFICATIONS	VENDOR PRODUCTS
2.	Oil, Engine (Ashless Dispersant) (Cont'd)	MIL-L-22851	Chevron Aero Oil Grade 120 Chevron Oil Co. 1200 State Street Perth Amboy, N. J. 08861 Esso Aviation Oil E-120, Enco Aviation Oil E-120, Esso Aviation Oil A-100. Enco Aviation Oil A-100, Esso Aviation Oil E-80, Enco Aviation Oil E-80 Enco Aviation Oil E-80 Exxon Company, U.S.A. P. O. Box 2180 Houston, Texas 77001 Chevron Aero Oil Grade 120 Standard Oil Co. of Calif. 225 Bush Street San Francisco, Calif. 94120
3.	Corrosion Preventive Compound	MIL-C-6529	Anti-Corrode No. 205 Cities Service Oil Co. 60 Wall Tower, New York 5, N. Y. Rust Foil No. 652-2, Franklin Oil and Gas Co., Bedord, Ohio Kendex No. 7038, Kendall Refining Co., Bradford, Pa.
4.	Lubricating Oil	MIL-L-6081	Gulflite Turbojet Oil No. 1010, Gulf Oil Corp. Pittsburgh, Pa. Aeroshell No. 3-1286, Shell Oil Co., 50 West 50th Street, New York, N. Y. 10020 Jet Engine Oil No. 1010, Texaco Inc., 135 East 42nd Street, New York N. Y. 10017

CHART 207 (Cont'd)
CONSUMABLE MATERIALS

ITEM	MATERIAL	SPECIFICATIONS	VENDOR PRODUCTS
5.	Lubricating Oil	SAE 20 or SAE 10W30	
6.	Lubricating Oil, Aircraft Reciprocating (Piston) Engine	MIL-L-6082	Conoco Aero Oil No. 1065, Continental Oil Co. Ponca City, Oklahoma Phillips 66 Aviation Engine Oil, Grade 1065 Phillips Petroleum Co. Bartlesville, Oklahoma Skelflite No. 100, Skelly Oil Co., El Dorado, Kansas Avrey 10V1100, Socony Mobil Oil Co. Inc., 150 East 42nd Street, New York, N. Y. 10017 Or any approved airplane engine oil (graded at 1065 or 1100).
7.	Lubricating Oil (Gear)	MIL-L-6086 Grade M	Trojan Gear Oil No. 6086 M. Cities Service Oil Co., 60 Wall Tower, New York 5, N. Y. Aeroshell Fluid 5 M, Shell Oil Co., 50 West 50th Street, New York, N. Y. L-1195, Sinclair Refining Co., 600 Fifth Avenue, New York, N. Y.
8.	Lubricating Grease (General Purpose)	MIL-G-7711 (Superseded by MIL-G-81322, See Item 9)	

CHART 207 (Cont'd)
CONSUMABLE MATERIALS

ITEM	MATERIAL	SPECIFICATIONS	VENDOR PRODUCTS
9.	Lubricating Grease NOTE MIL-G-81322 is not compatible with Aeroshell No. 5 and contains chemicals harmful to painted surfaces.	MIL-G-81322	Mobilgrease 28 Mobil Oil Corporation Shoreham Building Washington D.C. 20005 Aeroshell 22, Shell Oil Co., 50 West 50th Street, New York, N. Y.
10.	Lubricating Grease (Gear)	Mobile Compound G. G.	
11.	Lubricating Grease (Aircraft and Instruments, Low and High Temperature) NOTE Precautions should be taken when using MIL-G-23827 and MIL-G-81322, since these greases contain chemicals harmful to painted surfaces.	MIL-G-23827	Supermil Grease No. A72832, American Oil Co., 910 South Michigan Avenue Chicago, Ill. 60680 Royco 27A, Royal Lubricants Co., River Road, P. O. Box 95, Hanover, N. J. 07936 Shell 6249 Grease, Shell Oil Co., 50 West 50th St., New York 20, N. Y.
12.	Lubricant, Molybedenum Disulfide Powder	MIL-M-7866	Molykote Z Wilco Co., Wichita, Kansas Molykote Z Standard Oil of Kentucky Molykote Z, Haskell Engineering and Supply Company, 100 East Graham Place Burbank, California 91502 Moly-Paul No. 4, K. S. Paul Products Ltd. Nobel Road, London, England

CHART 202 (Cont'd)
CONSUMABLE MATERIALS

ITEM	MATERIAL	SPECIFICATIONS	VENDOR PRODUCTS
13.	Hydraulic Fluid	MIL-H-5606	Brayco 756D, Bray Oil Co., 3344 Medford Street, Los Angeles 63, California PED 3656 Standard Oil Co. of California 225 Bush Street, San Francisco 20, California
14.	Oxygen-System Leak Testing Compound	MIL-L-25567	
15.	Solvent, Dry Cleaning or White Spirit	PD680 or British Specification 245	
16.	Lubricating Oil	SAE-10	
17.	Air conditioner Refrigerant	R-12	
18.	Oil (Air Conditioner Compressor) 500 Viscosity		Suniso No. 5 Virginia Chemical and Smelting Co., West Norfolk, Virginia Texaco Capella E, Texaco Inc., 135 East 42nd Street, New York, N. Y. 10017
19.	Aviator's Breathing Oxygen	MIL-O-27210	
20.	Naphtha	TT-N-95	
21.	Methyl Ethyl Ketone	MIL-M-13999	
22.	Toluol (Toluene)	TT-T-548	
23.	Paint Remover		Turco No. 4260 Turco Products Inc., Los Angeles, California
24.	Epoxy Primer		Ameron Industrial Coatings Division P.O. Box 2153, Wichita, Kansas

CHART 207 (Cont'd)
CONSUMABLE MATERIALS

ITEM	MATERIAL	SPECIFICATIONS	VENDOR PRODUCTS
25.	Wash Primer		Ameron Industrial Coatings Division P.O. Box 2153, Wichita, Kansas
26.	Zinc Chromate Primer	MIL-P-8585	
27.	Rubber Hose	MIL-H-5593	
28.	Oil, Engine Preservative	MIL-L-21260	
29.	Graphite, Lubricating NOTE Lubricating Graphite Superseded by Item 12	SS-G-659 (Supersedes MIL-G-6711)	Anchor Packing Co., 401 Broad St. Philadelphia, PA. 19108 Bel-Ray Co., Inc., P.O. Box 526, Farmingdale, N.J. 07727 Crane Packing Co., 6400 Oakton St., Morton Grove, Ill. 60053 Ducommum Inc., Super-Temp Div., 11120 S. Norwalk Blvd., Santa Fe Springs Ca. 90670
30.	Lubricating Grease	Enco Andok-B	Humble Oil Co. Houston, Texas
31.	Solvent		CRC-2-26, Corrosion Reaction Consultants Limeklim Pike, Dresher, Pa.
32.	Laminated Glass Cloth	MIL-F-9084	Trevano, Coast Manufacturing and Supply Inc., Box 71, Livermore, California Uniglass, United Merchants and Manufacturing Inc. 1407 Broadway, New York, New York 10018
33.	Resin	MIL-R-7575	Laminac 4116, American Cyanamid Co., Wallingford, Connecticut Glidpol 1001, The Glidden Company, 925 Euclid Ave. Cleveland, Ohio

CONSUMABLE MATERIALS
CHART 207 (Cont'd)

ITEM	MATERIAL	SPECIFICATIONS	VENDOR PRODUCTS
34.	Lubricating Grease	Aeroshell 7A	Shell Oil Co., 50 West 50th Street, New York, N. Y.
35.	Urethane Primer		U. S. Paint Lacquer and Chemical Co., 1501 N. Belmont, P. O. Box 8151, Wichita, Kansas 67208
			Ameron Industrial Coatings Division P.O. Box 2153, Wichita, Kansas
36.	Thread Locking Compound	Loctite Sealant, Grade A	Loctite Corp. 705 N. Mountain Road Newington, Conn. 06111
37.	Penetrating Oil	Mouse Milk	Worldwide Aircraft Filters Corp., 1685 Abram Ct. San Leandro Calif. 94577
		Kano Kroil	Kano Laboratories Inc. Nashville, Tennessee
38.	Lubricating Grease	Aeroshell Grease 5,	Aeroshell Grease 5, Shell Oil Co., 50 West 50th Street, New York, N. Y.
39.	Cement	EC2262	Minnesota Mining and Manufacturing Company St. Paul, Minneosta
40.	Primer	Locquic "N"	Loctite Corp. 705 N. Mountain Road Newington, Conn. 06111
41.	Cleaner	Turco Metal-glo No. 3	Turco Products, Inc., Los Angeles, Calif.
42.	Paint Stripper	Turco 4260	Turco Products Inc. Los Angeles, California

CHART 207 (Cont'd)
CONSUMABLE MATERIALS

ITEM	MATERIAL	SPECIFICATIONS	VENDOR PRODUCTS
43.	Corrosion Preventive Compound	MIL-C-16173 Grade 2	Braycote 137, Bray Oil Co. 1925 N. Marianna Avenue Los Angeles, California 90032
			Petrotech 1-4 Penreco, P.O. Box 671, Butler, Pa. 16001
44.	Lubricating Grease	MIL-G-7118	
45.	Primer, Degreasing	EC3911	Minnesota Mining and Manufacturing Co., St. Paul, Minnesota
46.	Lubricating Silicone	G-322L	General Electric Waterford, New York 12188
47.	Anti-Seize Compound	Loctite 76764 Paste Form 1 lb Bursh Top Can	Loctite Corp. 705 N. Mountain Road Newington, Conn. 06111
48.	Coating	Alodine 1200, 1200S or 1201	Amchem Products Inc. Spring Garden Street Ambler, Pennsylvania 19002

CHART 208

THREAD LUBRICANTS

The vendor products appearing in this chart have been selected at random to help field personnel determine products conforming to the specifications listed in this publication. The brand names are listed for ready reference and are not specifically recommended by Beech Aircraft Corporation. Any product which conforms to the referenced specification may be used.

SYSTEM	*MATERIAL*	*SPECIFICATION*	*VENDOR PRODUCTS*
Fuel	Petrolatum	VV-P-236	
Oil, Manifold Pressure, Air Pressure	Lubricating Grease (Gasoline and Oil Resistant)	MIL-G-6032	L-237, Lehigh Chemical Co. Chestertown, Maryland Rockwell 950, Rockwell Mfg. Pittsburgh 8, Pa. Royco 32, Royal Lubricants Co. Hanover, New Jersey
Deicer, Static, Pitot	Anti-Seize, White Lead Base	TT-A-580	Armite Product Armite Laboratories, Los Angeles, California
Autopilot (Pipe Threads Only)	Anti-Seize, White Lead Base	JAN-A-669	
Brakes	Hydraulic Fluid or Anti-Seize, White Lead Base	MIL-H-5606 or TT-A-580	
Air Conditioner	Anti-Seize, Graphite Petrolatum or Anti-Seize, White Lead Base	MIL-T-5544 or TT-A-580	
Oxygen, High Pressure Side	Tape, Tetrafluroethylene	MIL-T-27730	Permacel Tape Corp., New Brunswick, New Jersey
Turbocharger Inlet Probe	Anti-Seize Compound	MIL-A-907D	Anti-Seize Compound C5A, Fel-Pro Inc. 7450 McCormick Skokie, Illinois

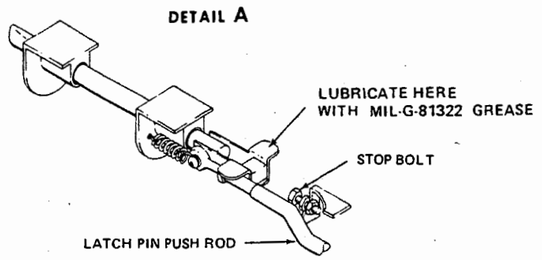
CHART 209

APPROVED ENGINE OILS (ASHLESS DISPERSANT)

SPECIFICATION	*PRODUCT*	**VENDOR*
MIL-L-22851	Global Concentrate A	Delta Petroleum Company, Inc. P.O. Box 10397 New Orleans, La. 70121
	Paranox 160 and 165	Enjay Chemical Company 60 West 49th Street New York, N.Y. 10020
	RT-451, RM-173E, RM-180E	Mobil Oil Corporation 150 East 42nd Street New York, N.Y. 10017
	Shell Concentrate A Code 60068 Aeroshell W120 Aeroshell W80	Shell Oil Company One Shell Plaza Houston, Texas 77002
	TX-6309 Aircraft Engine Oil Premium AD120 Aircraft Engine Oil Premium AD80	Texaco Inc. 135 East 42nd Street New York, N.Y. 10017
	PO Aviation Lubricant 753	American Oil and Supply Co. 238 Wilson Avenue Newark, N.J. 07105
	Chevron Aero Oil Grade 120	Chevron Oil Co. 1200 State Street Perth Amboy, N.J. 08861
	Esso Aviation Oil E-120 Enco Aviation Oil E-120 Esso Aviation Oil A-100 Enco Aviation Oil A-100 Esso Aviation Oil E-80 Enco Aviation Oil E-80	Humble Oil and Refining Co. P.O. Box 2180 Houston, Texas
	Chevron Aero Oil Grade 120	Standard Oil Co. of Calif. 225 Bush Street San Francisco, Calif. 94120

*The vendor products appearing in this chart have been selected at random to help field personnel determine products conforming to to the specifications in this publication. The brand names are listed for ready reference and are not specifically recommended by Beech Aircraft Corporation. Any product which conforms to the referenced specification may be used.

"END"



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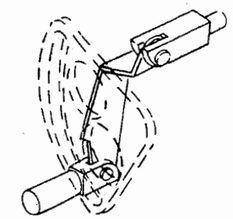

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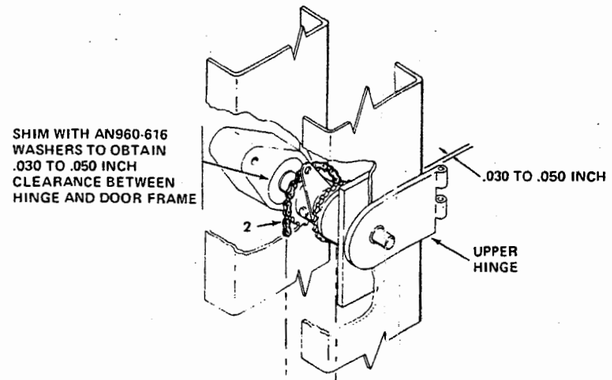

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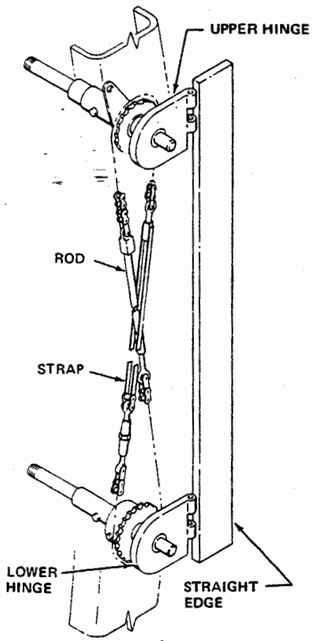

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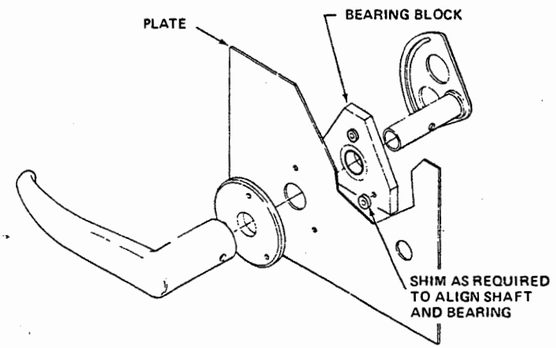
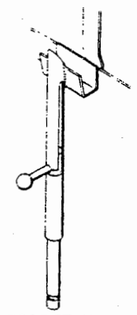
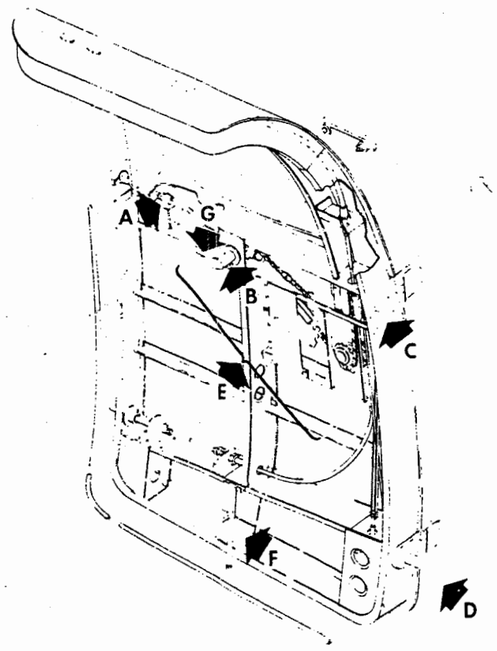


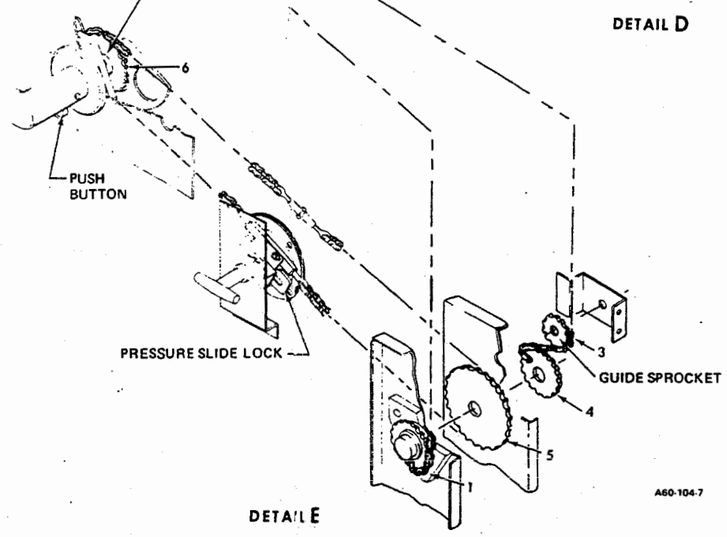
DETAIL G



DETAIL F

DOOR STAY





DETAIL E

Cabin Door Mechanism (P-123, P-127 and after)
Figure 202

Check that the chains are positioned on sprockets (5 and 6, Detail E) so that the interior handle can be rotated for its full travel without interference between the chain turnbuckle and the sprockets.

NOTE

If the handle binds while being rotated, check the chain for proper deflection under firm thumb pressure as indicated in Detail E. Adjust the chain turnbuckle as necessary to obtain this deflection. If the chain is at the proper tension but the handle still binds, check for alignment of the plate and bearing (see Detail G) with the handle shaft. Shim between the bearing block and plate with AN960-10L washers as necessary to eliminate the misalignment.

e. Check that the lock bolt (P-4 thru P-126, except P-123) pulls aft to clear the lock (see Detail B, Figure 201) when the exterior handle is in the fully open position. If necessary, reposition the lock arm support at its slotted mounting holes to obtain this clearance.

f. Adjust the stop bolt (see Detail A) untif the complete travel of the center latch bolt from the locked position is 1-1/4 inches.

CAUTION

To prevent damage to the ears of the spring in the exterior handle, (P-4 thru P-126, except P-123) the stop bolt must be adjusted to provide .010 to .030 inch clearance (see Detail F, Figure 201) between the ears of the spring and the lug in the casting of the handle when the handle reaches the limit of its travel.

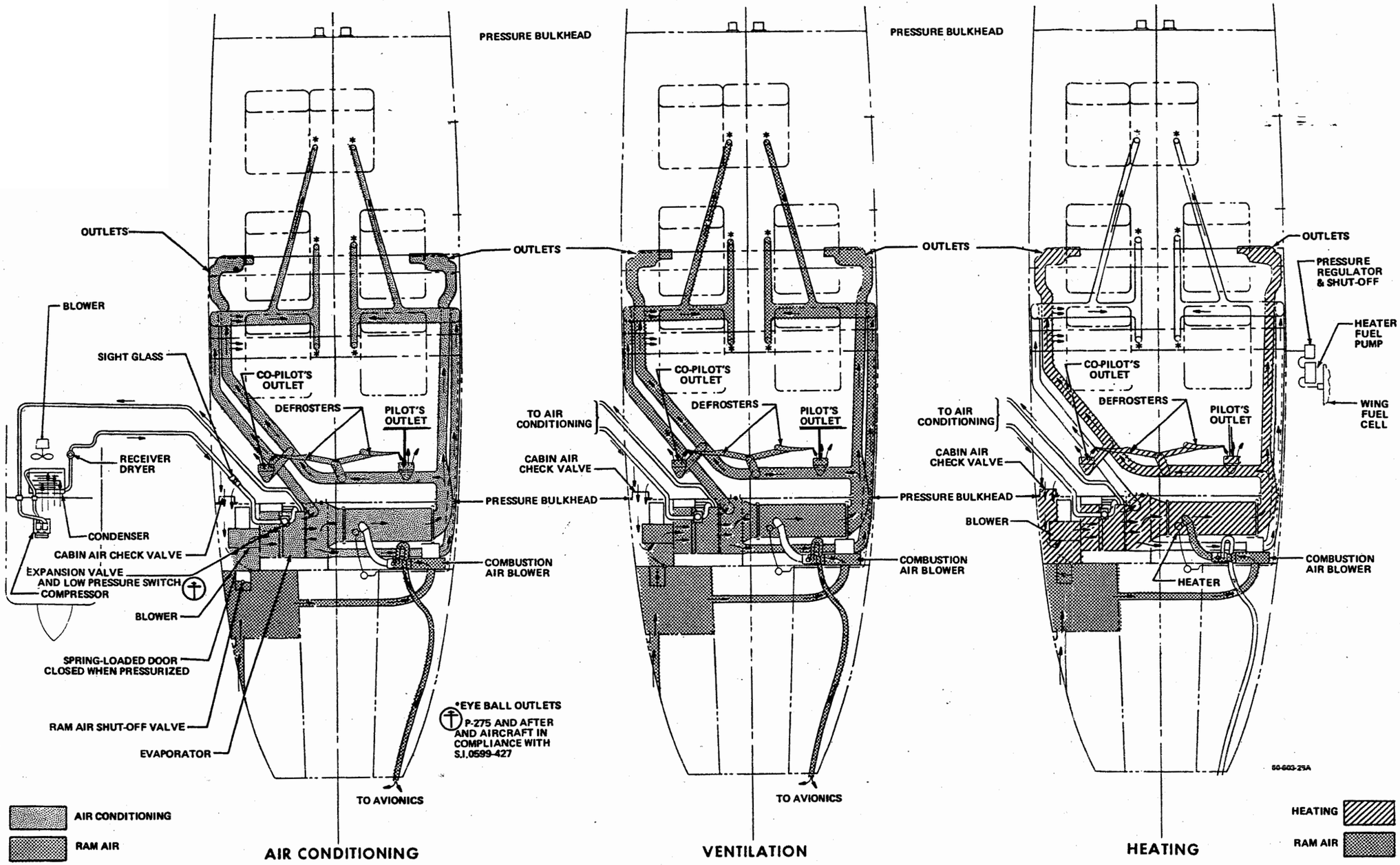
Check that the center latch pin extends 1-1/4 inches beyond the door frame when the hinges are in the closed position and 1/4 inch when in the open position. If necessary, adjust the length of the latch pin push rod (see Detail A) to obtain these settings.

g. Unsafety and adjust the length of the upper and lower latch pins until they extend one inch beyond the door frame with the hinges in the closed position.

LUBRICATION OF CABIN DOOR LATCHING MECHANISM

The latch mechanism is lubricated upon assembly and will not normally require further lubrication except when parts are replaced, then the chains and all points of friction except oilite bearings should be lubricated with MIL-G-81322 grease (9, Chart 207, 91-00-00).

"END"



Air Conditioning Distribution System (P-247 and after)
Figure 203

"END"

PASSENGER/CREW - MAINTENANCE PRACTICES

CABIN DOOR REMOVAL

a. Remove the upholstery panels adjacent to the hinges at the forward side of the door frame to gain access to the hinge attaching points.

b. Disconnect the electrical leads for the cigarette lighter and door locked indicator light at the forward side of the door frame.

c. Remove the nut securing the shaft of each hinge assembly to the door frame.

d. Remove the door by pulling the hinge shafts out of the mounting holes in the door frame.

CABIN DOOR INSTALLATION

a. Align the door with the door frame and insert the upper and lower hinge shafts into the mounting holes in the door frame.

b. Reinstall the washers that go between the hinge bearing mount and the shoulder of each hinge shaft.

c. Secure the hinges to the door frame structure with the attaching washer and nut. Check for a clearance of .03 to .05 inch between the hinge half and the door frame throughout the arc of the hinge rotation as the door is latched and locked. Add AN960-616 or AN960-616L washers as necessary between the hinge bearing mount and the shoulder of each hinge shaft to obtain this clearance.

d. Make any adjustments necessary for proper operation of the latch mechanism.

DOOR LATCH ADJUSTMENT
(Figures 201 and 202)

The following adjustments are possible only if the latch mechanism is properly rigged:

a. Unsafety and adjust the length of the upper and lower latch pins until they extend one inch beyond the door frame with the hinges in the closed position.

b. Adjust the stop bolt (see Detail A) until the complete travel of the center latch bolt from the locked position is 1-1/4 inches.

CAUTION

To prevent damage to the ears of the spring in the exterior handle (P-4 thru P-126, except P-123) the stop bolt must be adjusted to provide .010 to .030 inch clearance (see Detail F, Figure 201) between the ears of the spring and the lug in the casting of the handle when the handle reaches the limit of its travel.

Check that the center latch pin extends 1-1/4 inches beyond the door frame when the hinges are in the closed position and 1/4 inch when in the open position. If necessary, adjust the length of the latch pin push rod to obtain the foregoing settings.

c. If the proper settings cannot be obtained by the preceding adjustments, the latching mechanism must be rerigged.

LATCH MECHANISM RIGGING
(Figures 201 and 202)

The door latch mechanism is prerigged at the factory and should not normally require further adjustment except when damaged parts, such as chains and sprockets, require replacement. After removing all upholstery panels from the door to gain access to the latching mechanism, rig the door as follows:

a. Check for a clearance of .030 to .050 inch between each hinge half and the door frame throughout the arc of hinge rotation as the door is latched and locked. If necessary, add AN960-616 or AN960-616L washers between the shoulder of the hinge shaft and the hinge bearing (see Detail C) until the proper clearance is obtained.

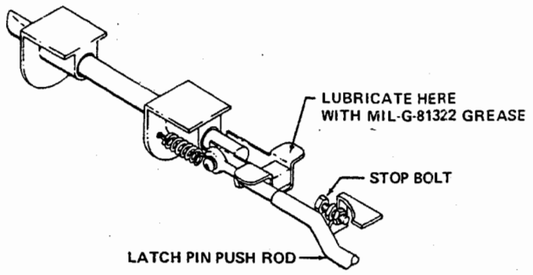
b. Use a straight edge to ascertain that the upper and lower hinge pins (see Detail D) are aligned with one another when the hinges are in the fully open position. Adjust the position of the chains on the upper and lower sprockets until the hinges can be rotated to the fully closed position without interference between the strap or rod terminals and the chain sprockets.

c. Check that the chains are positioned on sprockets (1, 2, 3, and 4, Details C and E) so that the hinges can be rotated to the fully open position without interference between the turnbuckle terminals and sprockets.

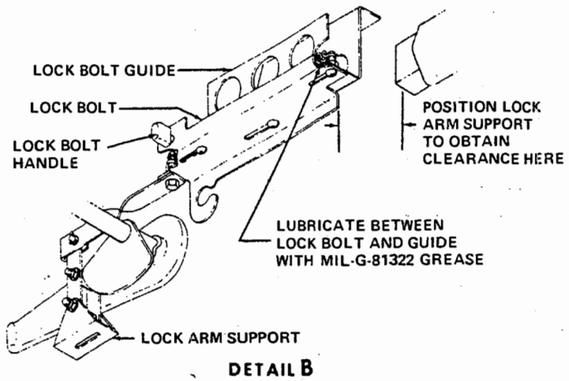
d. Align the hinges with the straightedge and place the interior handle in the fully open position.

NOTE

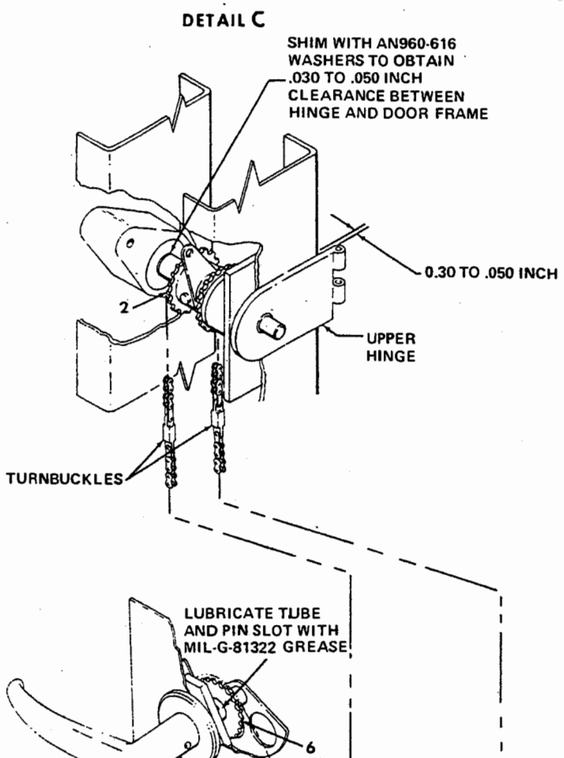
If the chain between sprockets (5 and 6, Detail E) has been removed, it must be reinstalled at this point. The chain must be installed with the exterior handle in the neutral position and the interior handle in the fully open position to avoid overloading the latching mechanism requiring excessive force to turn the door handles.

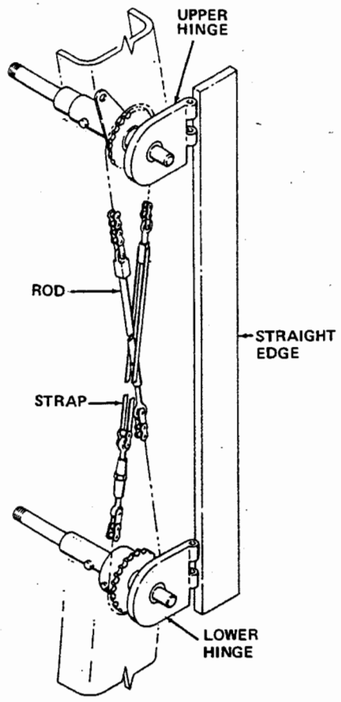


DETAIL A

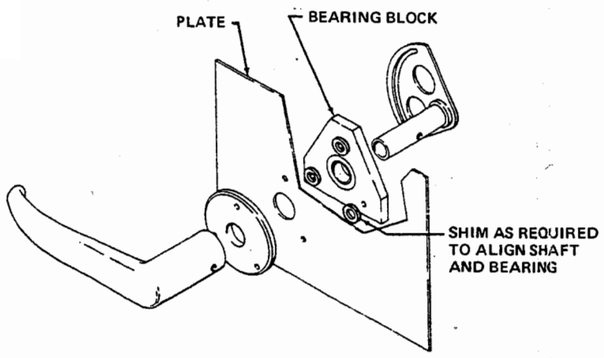


DETAIL B

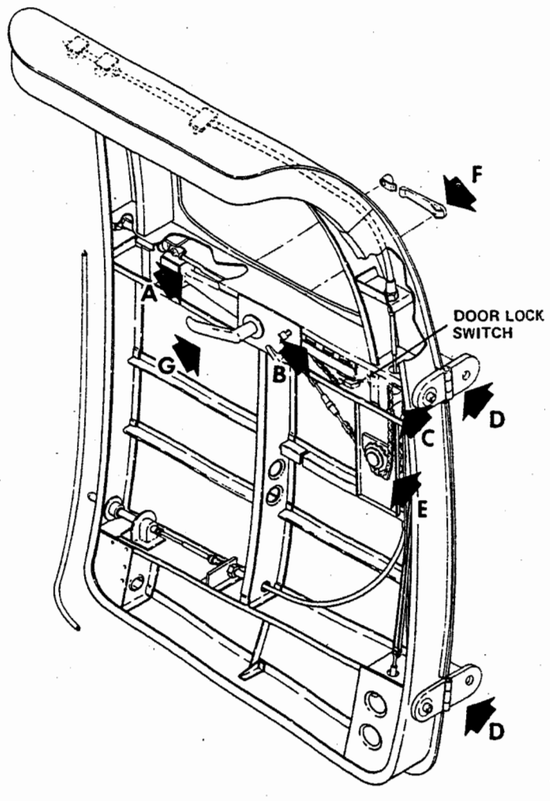


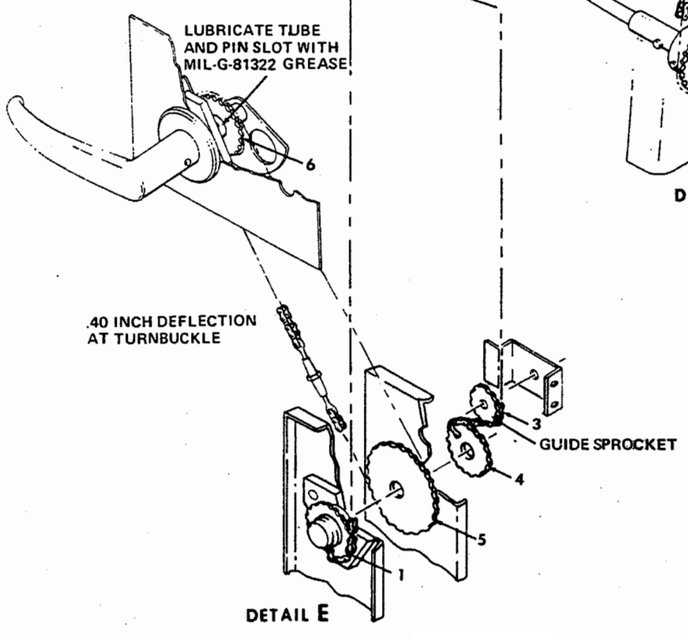


DETAIL D



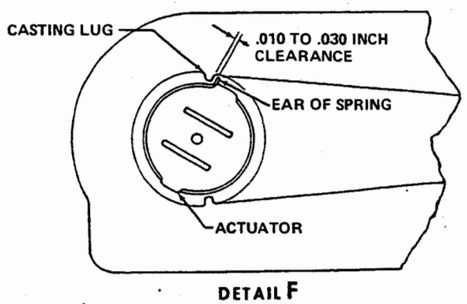
DETAIL G





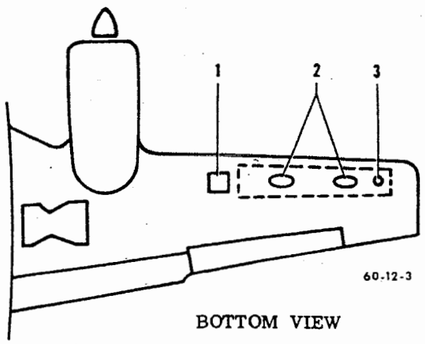
DETAIL E

60-104-2A



DETAIL F

Cabin Door Mechanism (P-4 thru P-126, except P-123)
Figure 201



BOTTOM VIEW

1. Fuel Plumbing Access
2. Fuel Cell Access
3. Filler Neck

Outboard Leading Edge Fuel Cell Access Openings
Figure 205

f. Unsnap the fuel cell and remove it from the wing cavity through one of the access openings (2).

NOTE

Tape the edge of the access hole to protect the fuel cell during removal and installation. If the fuel cell is to be stored for a period of 10 days or longer, coat the inside of the cell with light engine oil to prevent cracking or deterioration.

OUTBOARD LEADING EDGE FUEL CELL INSTALLATION

a. Carefully insert the fuel cell into the wing cavity through access openings (2) and snap in place.

b. Connect all fuel and vent plumbing. Torque the rubber fuel fitting nipples to 25 ± 5 inch-pounds.

c. Install the internal fuel cell interconnect clamps. Torque clamps to 25 ± 5 inch-pounds.

d. Install the access plates (2) and plumbing access plate (1) on the under side of the wing. Torque the access plates to 45 to 55 inch-pounds.

e. Install the filler neck with a new gasket. Torque the bolts to 45 to 55 inch-pounds and safety wire.

NOTE

Use sealer (3, Chart 205, 91-00-00) between the skin and the adapter flange when installing the filler neck.

WET WING TIP REMOVAL

a. Disconnect external power from the airplane. Place battery and generator switches in the off position.

b. Defuel the airplane, to the point where fuel cannot be seen from the inboard filler position.

c. Remove access plates from the outboard lower wing.

d. Remove deice boot from wing tip leading edge (if installed). (Refer to Chapter 30).

e. Working through the access opening in the under side of the wing, loosen the clamps on the 3 inch fuel interconnect and vent lines.

f. Disconnect the electrical connections.

g. Support wing tip. Using a 1/8 inch drill bit, drill out rivets along the connecting strap and remove the strap.

h. Remove the support angle screws at the juncture of the wing tip and wing front and rear spar. Remove the wing tip.

WET WING TIP INSTALLATION

a. Disconnect external power from the airplane. Place battery and generator switches in the off position.

CAUTION

Support the wing tip on a platform which will give firm support but will allow some flexibility of movement of the wing tip to facilitate proper alignment. Protect the wing tip surface from scratches, dents and other damage during installation.

b. Support the wing tip in the proper position to attach to the wing. Connect the hoses from the wing to the 3 inch fuel interconnect and vent lines. Secure with clamps.

c. Connect electrical connections.

d. Move wing tip into position and secure support angles to the front and rear spars with screws.

e. Using MS20426AD3 rivets, rivet the connecting strap to both the wing and wing tip.

f. Install the deicer boots (if required). (Refer to Chapter 30.)

NOTE

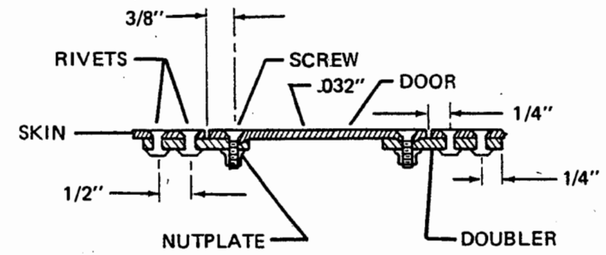
Repair of the wet wing tip is permissible providing the damaged area is far enough from the rib to allow a doubler or plate to be installed. Holes cut to remove damaged area must be round or at least have generous radii. Should a stringer be damaged or fall within the repair area it must be bridged across and be attached to the repair.

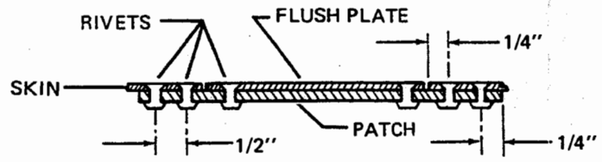
CAUTION

It must be realized the wet wing tip is a highly stressed area; consequently, the repair structure must be of equal capability.

GENERAL INSTRUCTIONS

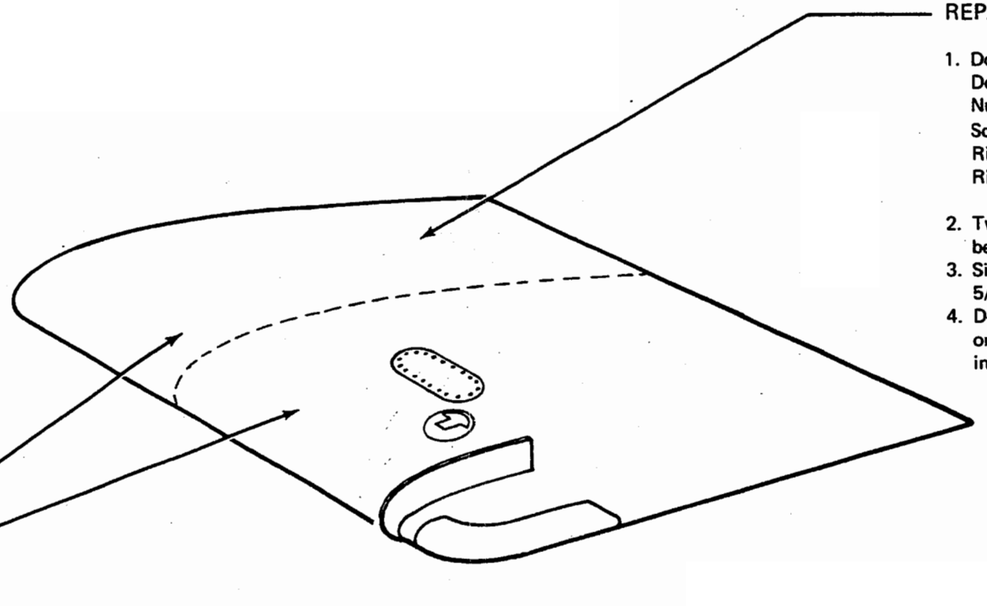
1. Seal edges of doubler or plate, rivet butts and nut plates with PR890B-1/2 (19, Chart 205, 91-00-00) or EC 1675B-1/2 (20, Chart 205, 91-00-00).
2. Fill the rivet heads, screw heads and the patch/skin joints with PR890B-1/2 (19, Chart 205, 91-00-00) or EC 1675B-1/2 (20, Chart 205, 91-00-00).
3. Pressure test (0.50 + 0.25 - 0.00 psig) before applying filler to outer surface.





REPAIR OF ACCESSIBLE AREAS

1. Rivets - AN426AD4-5
   Patch material - .032 2024 T3 Aluminum ALCLAD
   Plate material - .032 2024 T3 Aluminum ALCLAD
2. Two rows rivets through patch and skin (patch plate to be on inner surface of cell. --If the plate is too large for entry through access openings, use method for inaccessible area.)
3. Rivet E.D. 1/4"–spacing 1/2" between rows and rivets.
4. Fit flush plate and secure with only enough rivets to prevent filler from cracking.

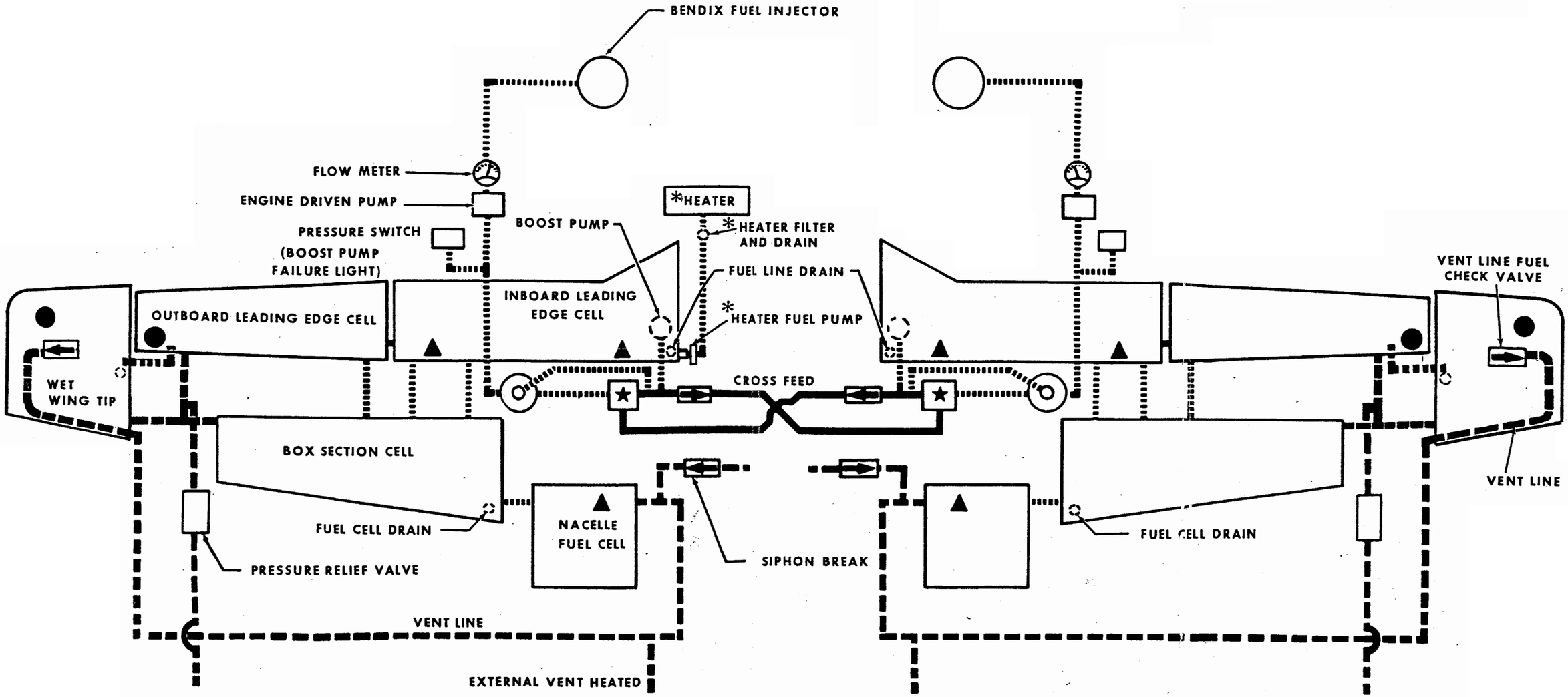


REPAIR OF INACCESSIBLE AREAS

1. Door material - .032 2024 T3 Aluminum ALCLAD.
   Doubler material - .051 2024 T3 Aluminum ALCLAD.
   Nut Plates - K1000-832 or equivalent.
   Screws - AN507-8R–length to be determined.
   Rivets - AN426AD3–for attaching nut plates.
   Rivets - AN426AD4–for attaching doubler.
2. Two rows rivets in doubler and skin--1/4" E.D., 1/2" spacing between rows and rivets.
3. Single row of screws through doubler and door–3/8" E.D., 5/8" spacing. Dimple door and countersink doubler.
4. Doubler may be cut on one side only in order to place it on inside of cell. Cut side of doubler to be placed on inboard or outboard side of repair.

60-608-1

Minor Wet Wing Tip Repair
Figure 206



60-603-34

Optional Fuel System Schematic
(P-348, P-365 and after)
Figure 2

"END"

GENERAL - DESCRIPTION AND OPERATION (Figure 1)

FUEL CELLS

The fuel system installation consists of an inboard main fuel cell and an outboard cell in the leading edge, a nacelle tank, a wing panel fuel cell in each wing and a wet wing tip tank (optional on serials P-348, P-365 and after). All of the fuel cells in each wing and wing tip are interconnected in order to make all of the usable fuel in each wing available to its engine when the fuel selector valve is turned ON. The interconnecting fuel cells are serviced either through the single filler on each wing or the filler in each of the optional wet wing tips, providing single point filling for each side. The combined capacity of the standard and optional systems is shown below:

SERIALS	*CAPACITY IN GALLONS*	*USABLE IN GALLONS*
P-3 thru P-195 inboard leading edge fuel cells unbaffled	207	192
P-3 and after with inboard leading edge baffled fuel cells	207	202
Optional fuel system P-348, P-365 and after	237	232

FUEL CROSSFEED

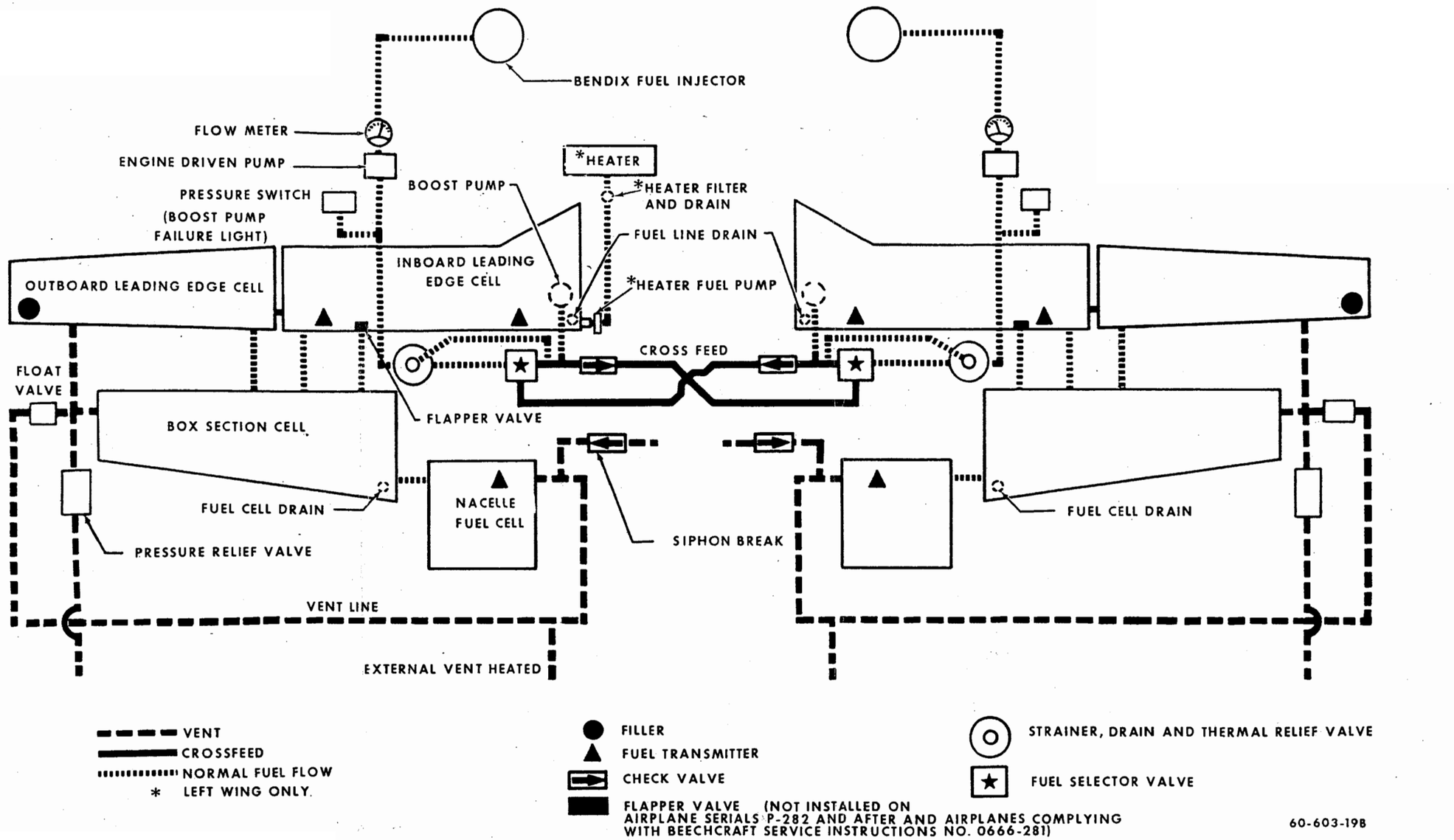
The separate identical fuel supplies for each engine are interconnected by crossfeed lines. During normal operation, each engine uses its own fuel pumps to draw fuel from its respective fuel tank arrangement. However, on crossfeed operations, the entire usable fuel supply of both wings can be consumed by either engine.

FUEL BOOST PUMPS

Submerged, tank-mounted fuel boost pumps are provided for each engine and are located in the inboard leading edge tanks. They are controlled by separate ON-OFF toggle switches located on the pilot's subpanel. The fuel boost pumps provide for near maximum engine performance should the engine-driven pump fail. Fuel boost pump failure is indicated by illumination of a FUEL PRESS light on the panel.

FUEL CELL DRAINS

The fuel system is drained by six snap-type drains under the wings. A drain is located in each inboard leading edge fuel cell, box section fuel cell and fuel strainer. An additional fuel strainer drain for the heater fuel line is located in the nose wheel well.



Fuel System Schematic
Figure 1

EXTENSION AND RETRACTION - MAINTENANCE PRACTICES

LUBRICATION

Lubricate the landing gear retract system as detailed in the Lubrication Chart in Chapter 12-20-00.

LANDING GEAR MOTOR AND ACTUATOR REMOVAL

When it is necessary to remove only the landing gear motor, accomplish steps "a", "b", "g" and "m".

a. Remove the cabin front seat.
b. Remove the carpet and access covers on top and directly behind the front carry through structure.

CAUTION

When removing the actuator from the airplane, do not remove the actuator drive shaft from the actuator. The drive shaft is indexed to the sector gear and the actuator will have to be disassembled to reinstall the drive shaft.

c. Disconnect the fuel selector valve controls and move them as far to one side as the cable slack will allow.
d. Disconnect the main landing gear retract rods at the actuator.
e. Disconnect the landing gear door retract rods at the actuator.
f. Remove the screws securing the landing gear limit switch assembly(ies) to the actuator and move the switch assembly(ies) aside to permit removal of the actuator.

NOTE

On serials P-139 and after, there are two switch assemblies, one on each side of the actuator.

g. Disconnect electrical wiring from the landing gear motor and the dynamic brake relay.
h. Remove the landing gear actuator access door on the bottom of the fuselage, and remove the nose gear actuator retract arm and linkage from the actuator.
i. Through the actuator access opening, remove the four nuts attaching the actuator to the fuselage structure.
j. Remove the landing gear actuator upper support retaining nut.
k. Remove the screws securing the upper support plate to its cross member.
l. Remove the landing gear actuator. The cross member may have to be moved upward to gain clearance as the actuator is lifted to clear its attaching studs.
m. Remove the three bolts attaching the landing gear motor to the actuator.

LANDING GEAR MOTOR AND ACTUATOR INSTALLATION

If only the landing gear motor has been removed, accomplish steps "a", "g", "j" and "k".

a. Install the three bolts attaching the landing gear motor to the actuator and secure with safety wire.
b. Position the actuator against the fuselage structure and install the four attaching bolts.

NOTE

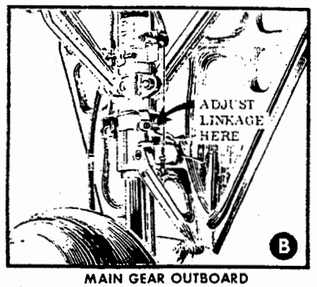
Prior to installing the actuator nuts, apply thread locking compound (36, Chart 207, 91-00-00) to the actuator stud threads.

c. Install the upper support plate to the cross member.
d. Install the actuator upper support retaining nut.
e. Secure the landing gear limit switch assembly(ies) to the actuator.
f. Connect the main landing gear retract rods and door retract rods to the actuator. An AN960-616 washer is installed between the main gear retract rods and the actuator arm.
g. Connect the electrical wiring to the landing gear motor and the dynamic brake relay.
h. Connect the fuel selector valve control and check rigging. (Refer to Chapter 28-20-00).
i. Connect the nose landing gear rod and linkage to the actuator.

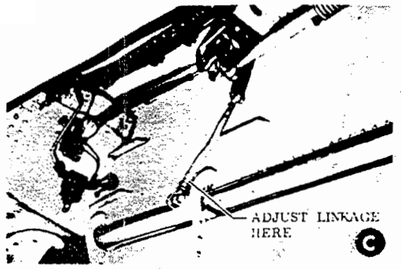
NOTE

When connecting the nose gear retract rod to the actuator, make certain the index mark on the arm and the actuator shaft coincide.

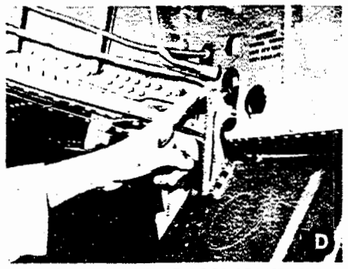
Check rigging of the landing gear system. When



MAIN GEAR OUTBOARD DOOR ADJUSTMENT

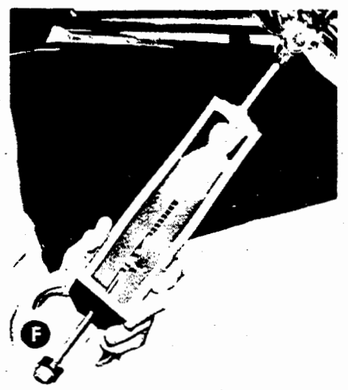


NOSE GEAR DOOR ADJUSTMENT

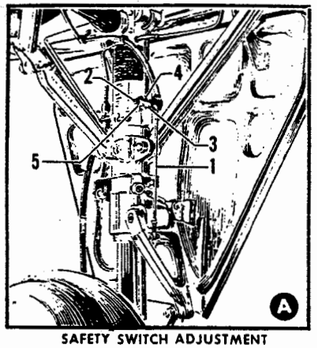


MAIN GEAR INBOARD DOOR ADJUSTMENT

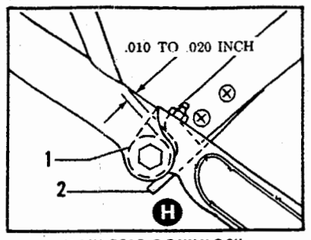
MAIN GEAR UPLOCK
1. Uplock Roller
2. Uplock Block



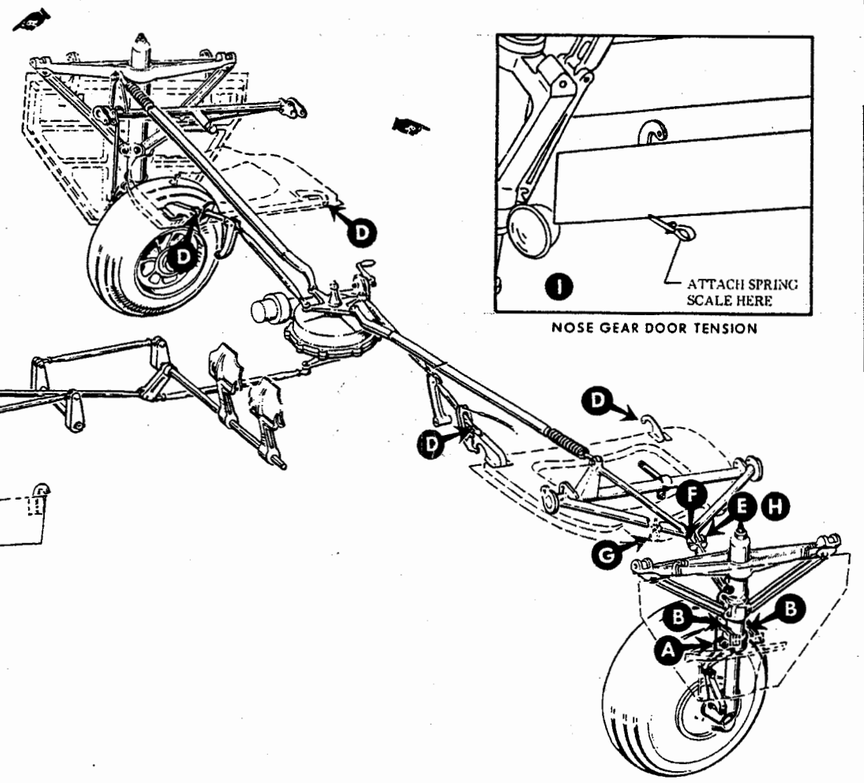
CHECKING MAIN GEAR DOWN LOCK TENSION



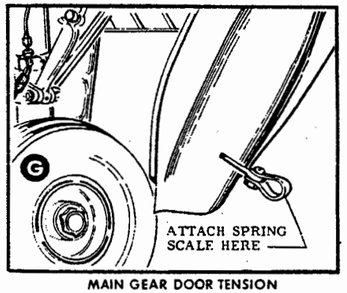
SAFETY SWITCH ADJUSTMENT
1. Actuator Rod
2. Retaining Nut
3. Switch Arm
4. Locking Screw
5. Adjusting Screw



MAIN GEAR DOWNLOCK
1. Roller
2. Downlock Block



NOSE GEAR DOOR TENSION



MAIN GEAR DOOR TENSION

60-223-1

Landing Gear System
Figure 201

ELEVATOR AND TAB - MAINTENANCE PRACTICES

ELEVATOR REMOVAL

a. Remove the tail cone and the access plate on the side of the fuselage beneath the RH horizontal stabilizer.

b. Loosen the elevator trim tab cables.

c. Disconnect the trim tab cables from the trim tab actuator chain by removing the master link at each end of the chain.

d. Disconnect the elevator push rods from the elevator torque tube fittings.

e. Remove the bolt from the inboard end of the elevator torque tube.

f. Remove the hinge bolts. Disconnect the elevator bonding cables and remove the elevator.

ELEVATOR INSTALLATION

a. Connect the elevator bonding cables. Position the elevator on the stabilizer and install the hinge bolts, washers and nuts. Tighten and safety.

b. Install the bolt in the inboard end of the elevator torque tubes. Torque the attaching bolts to 50-70 inch pounds.

c. Attach the elevator push rods to the elevator torque tube fittings. Torque the attaching bolts to 20-25 inch-pounds.

d. Position the elevator trim tab cables on the ends of the trim tab actuator chain and install the master links.

e. Adjust the tension on the elevator trim tab cables as noted on the Elevator Rigging Illustration, Figure 201.

NOTE

Check the trim tab system for correct movement of the control surface with respect to the movement of the trim tab control. When the elevator tab control is moved toward the NOSE DOWN position, the tab should move UP.

f. Connect the tail light wires and install the tail cone. Install the access plate on the side of the fuselage beneath the RH horizontal stabilizer.

ELEVATOR CABLE REMOVAL
(Figure 201)

a. Remove the tail cone and the access plate on the side of the fuselage beneath the RH horizontal stabilizer.

b. Remove the pilot's and copilot's seat and the floorboards in the pilot's compartment.

c. Remove the forward passenger seats and the floorboards between main and rear spar.

d. Remove the access panel in the floorboard aft of the rear spar.

e. Remove the cable retaining pins from the pulley brackets and the pressure seals from the rear pressure bulkhead.

f. Disconnect the elevator cables, in the aft fuselage, at the turnbuckles and connect lead lines to the forward cables.

g. Disconnect the cables at the forward bell crank. Identify both forward cables in relation to their attaching point on the bell crank. Remove the cables.

h. Disconnect the cables at the aft bell crank. Identify both aft cables in relation to their attaching point on the bell crank arms. Remove the cables.

ELEVATOR CABLE INSTALLATION
(Figure 201)

a. Route the aft elevator cables forward and connect to the applicable bell crank arms as noted during cable removal.

b. Route the forward elevator cables aft and connect to the bell crank as noted during cable removal.

c. Install all cable retaining pins in all pulley brackets.

d. Using PD680 solvent (15, Chart 207, 91-00-00), clean the cables for the length of travel through the pressure seals. Lubricate to one inch beyond the cleaned area with MIL-G-23827 grease (11, Chart 207, 91-00-00).

e. Fill the pressure seals with MIL-G-23827 grease (11, Chart 207, 91-00-00). Install the seals.

f. Connect the cables to the turnbuckles in the aft fuselage and rig the cable system.

g. Install the aft floorboard access panel, the floorboards between the main and aft spar, and the forward seats.

h. Install the floorboards in the pilot's compartment, and the pilot's and copilot's seats.

i. Install the tail cone and the access plate beneath the RH horizontal stabilizer.

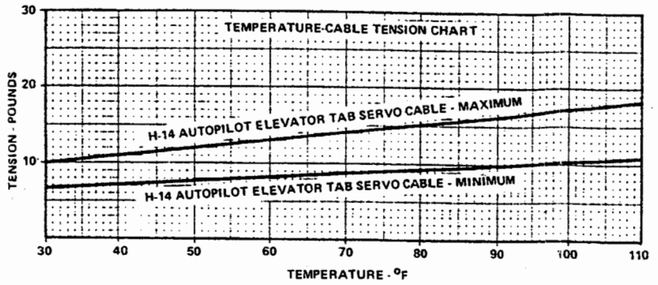
ELEVATOR CONTROL SYSTEM RIGGING
(Figure 201)

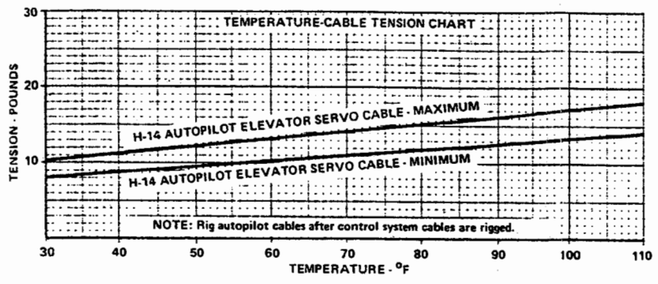
WARNING

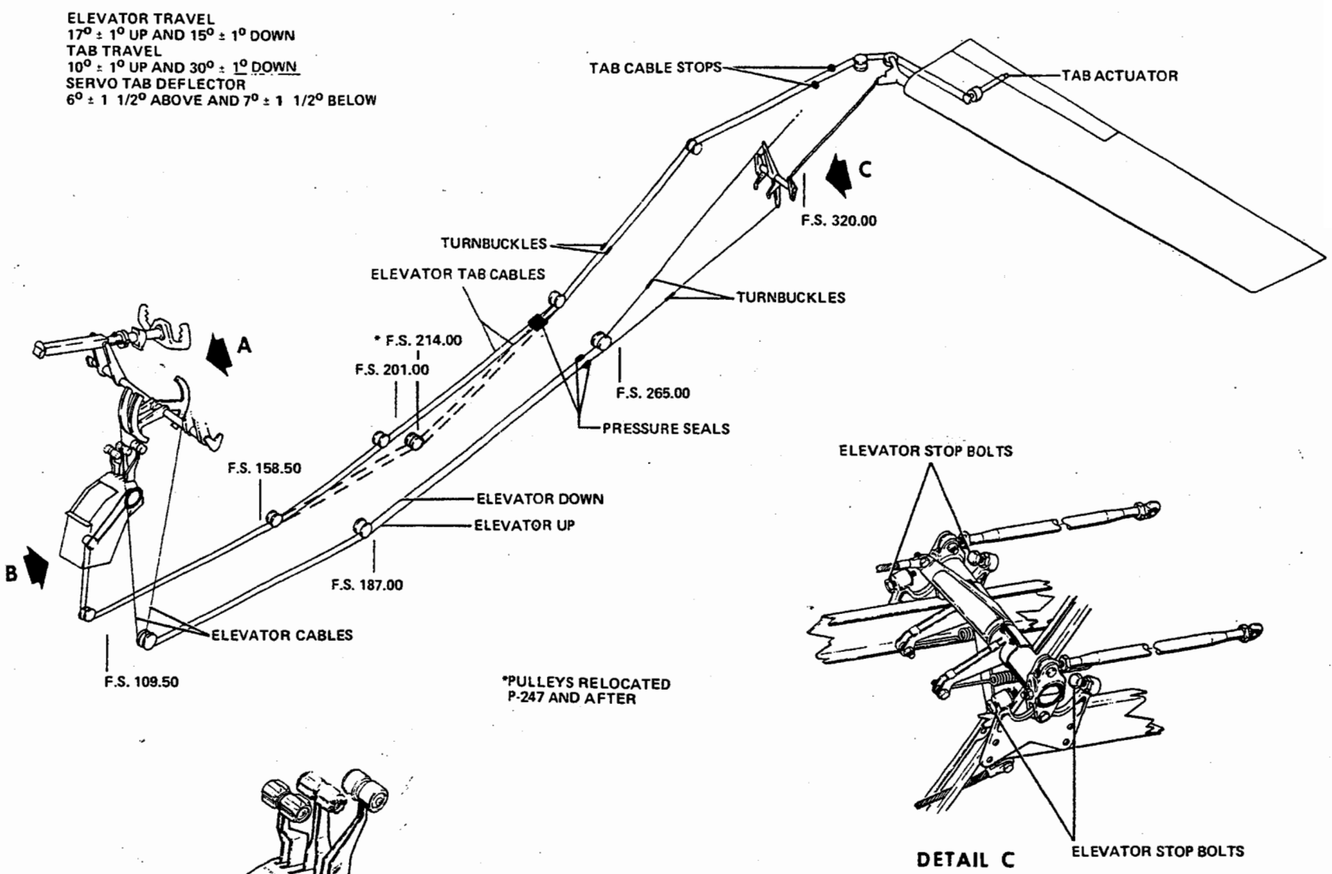
When replacing or installing control cables, bell cranks and other control system components, observe the color coding on all parts. DO NOT connect coded parts of one color to coded parts of a different color.

NOTE

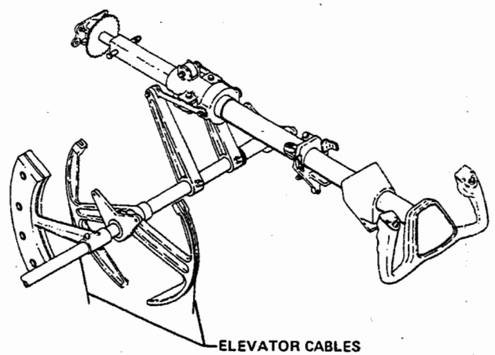
BEECHCRAFT recommends the use of the elevator travel gage shown in SPECIAL TOOLS in Chapter 12-20-00.



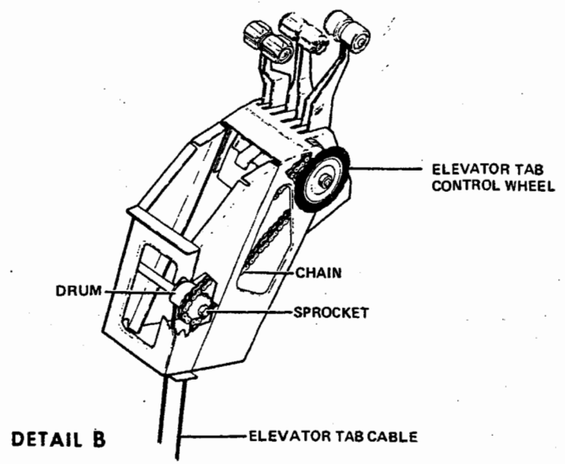




DETAIL C



DETAIL A



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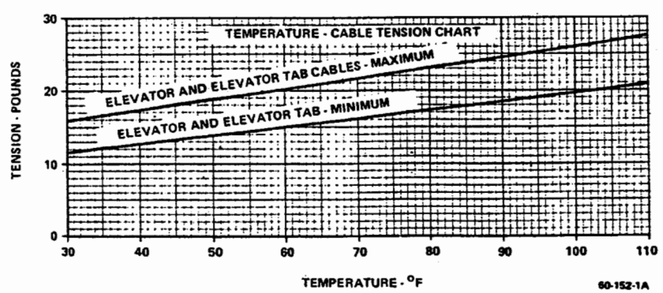


Rigging the Elevator Control System
Figure 201

RUDDER AND TAB - MAINTENANCE PRACTICES

RUDDER REMOVAL

a. Remove the tail cone and the access plate on the side of the fuselage beneath the RH horizontal stabilizer.

b. Detach the tail cone, disconnect the tail navigation light wire, and remove the tail cone.

c. Disconnect the rudder tab cables at the turnbuckles. Remove the tab cable retainer pins and fairlead.

d. Remove the four bolts securing the rudder torque tube and tab pulley bracket to the rudder bell crank.

e. Remove the upper and lower hinge bolts and remove the rudder.

RUDDER INSTALLATION

a. Align the holes in the rudder and stabilizer hinges and install the attaching bolts.

b. Secure the tab pulley bracket and rudder torque tube to the rudder bell crank with the four attaching bolts. Torque the bolts to 50-70 inch-pounds.

c. Connect the rudder tab cables at the turnbuckles and install the cable retainer pins and fairlead.

d. Check the travel of the tab and the rudder tab indicator to ensure that the tab and indicator agree.

e. Connect the tail light wires and install the tail cone. Reinstall the access plate beneath the RH horizontal stabilizer.

RUDDER CABLE REMOVAL
(Figure 201)

a. Remove the tail cone and the access plate on the side of the fuselage beneath the RH horizontal stabilizer.

b. Remove the pilot's and copilot's seat and the floorboards in the pilot's compartment.

c. Remove the forward passenger seats and the floorboards between the main and rear spar.

d. Remove the access panel in the floorboard aft of the rear spar.

e. Remove the cable retaining pins from the pulley brackets and the pressure seals from the rear pressure bulkhead.

f. Disconnect the rudder cables, in the aft fuselage, at the turnbuckles and connect lead lines to the forward cables.

g. Disconnect the forward cables at the bell crank and remove the cables through the pilot's compartment.

h. Disconnect the aft cables at the rudder bell crank and remove the cables.

RUDDER CABLE INSTALLATION

a. Route the aft rudder cables forward in the tail section and connect to the rudder bell crank.

b. Route the forward rudder cables aft from the pilot's compartment and connect the cables to the bell crank.

c. Install all cable retaining pins in the pulley brackets.

d. Using PD680 solvent (15, Chart 207, 91-00-00), clean the cables for the length of travel through the pressure seals. Lubricate to one inch beyond the cleaned area with MIL-G-23827 grease (11, Chart 207, 91-00-00).

e. Fill the pressure seals with MIL-G-23827 grease (11, Chart 207, 91-00-00). Install the seals.

f. Connect the cables to the turnbuckles in the aft fuselage and rig the cable system.

g. Install the aft floorboard access panel, the floorboards between the main and aft spar, and the forward seats.

h. Install the floorboards in the pilot's compartment, and the pilot's and copilot's seat.

i. Install the tail cone and the access plate beneath the RH horizontal stabilizer.

RUDDER CONTROL SYSTEM RIGGING
(Figure 201)

a. Remove the tail cone, the access plate on the side of the fuselage beneath the RH horizontal stabilizer, and the access panel in the aft floorboard.

b. Release the rudder pedal adjusting levers and place all pedals in the aft position.

c. Insert a 7/16-inch diameter rig pin through the upper arm of the pilot's rudder pedals to rig neutral on the rudder pedals. This will also bring the copilot's rudder pedals to the same adjustment as the pilot's pedals.

d. Place the rudder and rudder bell crank in the neutral position.

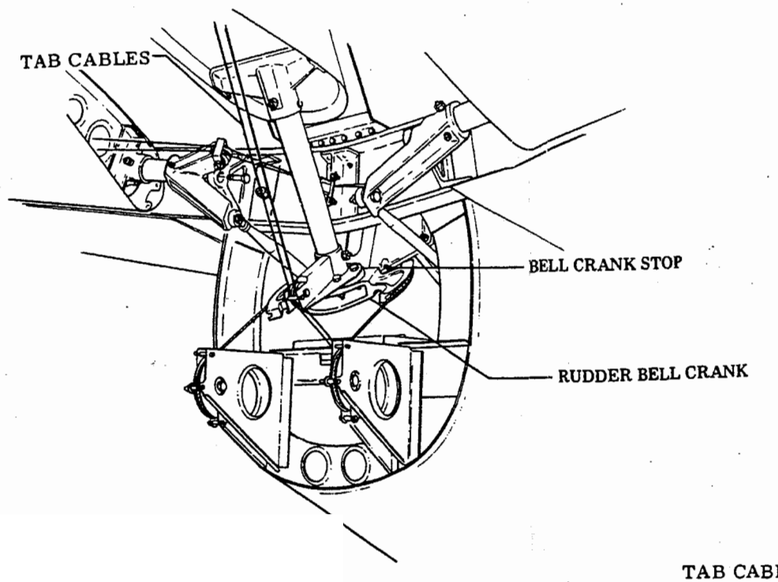
e. Rig the rudder cables to the proper tension as determined by reference to the Temperature-Cable Tension Chart. Safety wire the turnbuckles.

f. Remove the rig pin from the rudder pedals and adjust the stops for the rudder bell crank until the rudder has a travel of 33 to 35 degrees right and 28 to 30 degrees left from the centerline of the horizontal stabilizer in response to the corresponding movement of the rudder pedals.

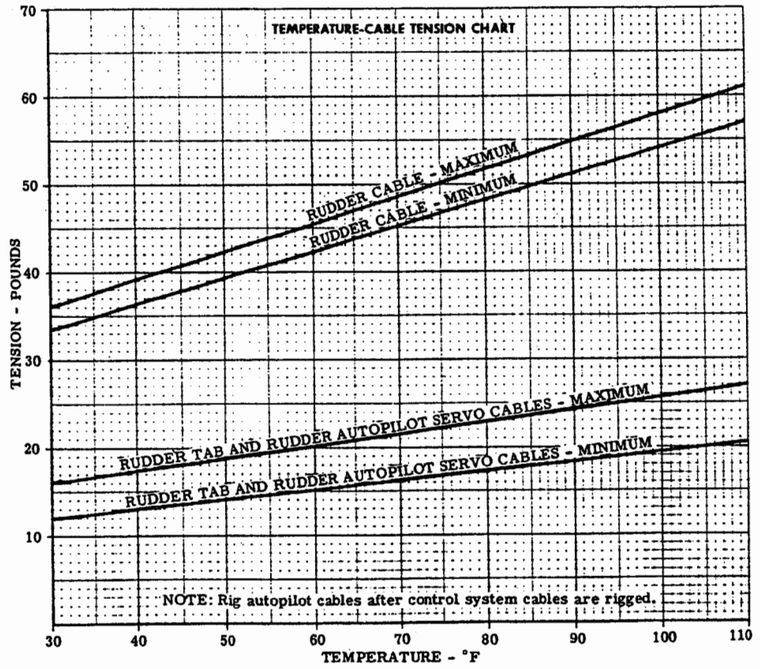
NOTE

With the rudder and steering system fully installed, properly rigged, and the nose wheel extended and off the ground, the force required for full right rudder deflection should not exceed 25 pounds as measured at the pivot point of each rudder pedal. The force required for full left rudder deflection, measured at the rudder pedal pivot point, should not exceed 23 pounds.

g. Install the tail cone, the access plate beneath the



RUDDER TRAVEL	RUDDER TAB TRAVEL
33° TO 35° RIGHT	19° TO 21° RIGHT
28° TO 30° LEFT	19° TO 21° LEFT



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Rigging the Rudder Control System
Figure 201

AILERON AND TAB - MAINTENANCE PRACTICES

AILERON REMOVAL

a. Disconnect the aileron tab push rod.
b. Support the aileron and remove the two attaching screws from the top and bottom of each hinge bracket.
c. Pull the aileron straight away from the wing to avoid damage to the attaching areas.
d. Remove the screws attaching the bonding cables to the aileron.

AILERON INSTALLATION

a. Attach the bonding cables to the aileron.
b. Place the aileron in position on the hinge brackets. Be sure the hinge bracket is in the proper place between the aileron skin and the reinforcing structure.
c. Install the upper and lower hinge bracket screws.
d. Connect the aileron tab push rod.

AILERON CONTROL CABLE REMOVAL
(Figure 201)

a. Remove the pilot's and copilot's seat, and the floorboards in the pilot's compartment.
b. Remove the forward passenger seats and the floorboards between the main and rear spar.
c. Remove the access plates, as necessary to gain access to the aileron cables and pulley brackets, on the lower trailing edge of the wings.
d. Remove all necessary cable retaining pins from the cable pulley brackets. Remove the pressure seals on each side of the fuselage.
e. Disconnect the forward aileron cables from the chain and cable assembly at the turnbuckles at the control column. Install lead lines to both aileron cables.
f. Paint one tooth of the control column sprocket and the corresponding link of the chain and cable assembly to insure proper alignment at installation.
g. Disconnect the forward aileron cables and the forward outboard wing cable at the turnbuckles in each wing. Identify and remove both forward cables.
h. Disconnect the balance cable at the turnbuckle in each wing. Connect a lead line to one end of the cable and remove the cable.
i. Disconnect the forward outboard, and the aft outboard cables at the bell crank in each wing. Identify and remove the cables.

AILERON CONTROL CABLE INSTALLATION

a. Connect the forward outboard, and the aft outboard cables to the bell crank in each wing. Route the cables inboard.
b. Route the balance cable through one wing, the fuselage, then through the opposite wing. Connect the balance cable and the aft outboard cables to the turnbuckles in each wing.
c. Route one end of the aileron cables outboard in each wing, and the other end forward to the control column. Connect the cables to the turnbuckles at the forward outboard cable in each wing.
d. Position the chain and cable assembly on the control column sprocket. Ensure that painted link and the corresponding painted tooth are aligned.
e. Connect the chain and cable assembly to both aileron cables at the turnbuckles at the control column.
f. Install all retaining pins in the pulley brackets.
g. Using PD680 solvent (15, Chart 207, 91-00-00), clean the cables for the length of travel through the pressure seals. Lubricate to one inch beyond the cleaned area with MIL-G-23827 grease (11, Chart 207, 91-00-00).
h. Fill the pressure seals with MIL-G-23827 grease (11, Chart 207, 91-00-00). Install the seals.
i. Rig the aileron control system.
j. Install the access plates on the lower trailing edge of the wings.
k. Install the floorboards and the forward passenger seats.
l. Install the floorboards and the pilot's and copilot's seats.

AILERON CONTROL SYSTEM RIGGING
(Figure 201)

a. The contour of the aileron must align with the contour of the wing within 1/16 inch (.0625 inch) on either or both sides.
b. Aileron and connecting linkage may have a maximum of 1/16 inch lost motion. Check for lost motion at the midpoint of the aileron trailing edge with the bell crank stationary.
c. The aileron is in neutral when its outboard trailing edge aligns with the trailing edge of the wing tip and its inboard end is parallel with the outboard end of the flap. A horizontal misalignment of plus or minus 3/16 inch is allowed between trailing edges of the aileron and wing tip. With the bell crank parallel to the wing rib, set the aileron in neutral by adjusting the length of the push-pull tube. Loosen the locknuts on both ends and turn the tube to shorten or lengthen.
d. Securely tighten the locknuts on all rod ends. Rig cable tension and adjust travel as noted on Aileron Rigging Illustration.
e. With the aileron system fully rigged, the maximum force required to move the control wheel 10 degrees to the left and right of neutral should not exceed 15 inch-pounds.

AILERON TRIM TAB CABLE REMOVAL
(Figure 201)

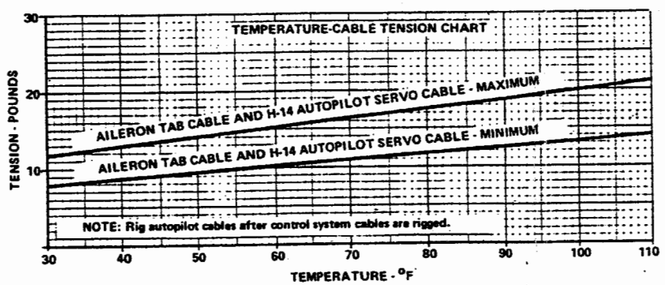
a. Remove the pilot's seat and the left floorboard.
b. Remove the lower forward upholstery panel on the

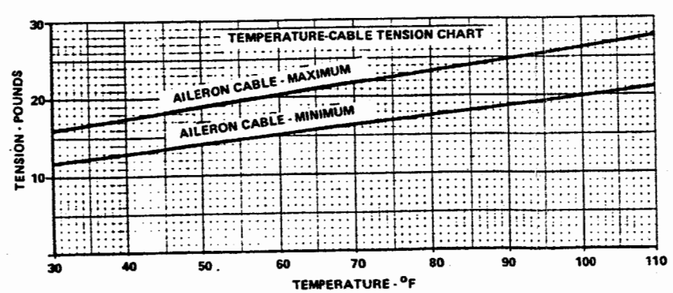
AILERON TRAVEL 25° ± 1° UP
15° ± 1° DOWN

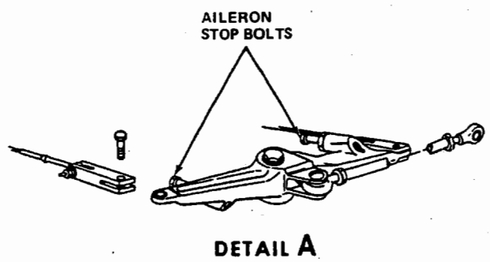
AILERON DEFLECTION
12° ± 1-1/2° ABOVE
7° ± 1-1/2° BELOW

AILERON TAB TRAVEL
10° ± 1° UP
10° ± 1° DOWN

WITH THE SYSTEM FULLY INSTALLED AND RIGGED. THE MAXIMUM FORCE REQUIRED TO MOVE THE CONTROL WHEEL 10° TO THE LEFT AND RIGHT OF NEUTRAL SHALL BE 15 INCH-POUNDS.







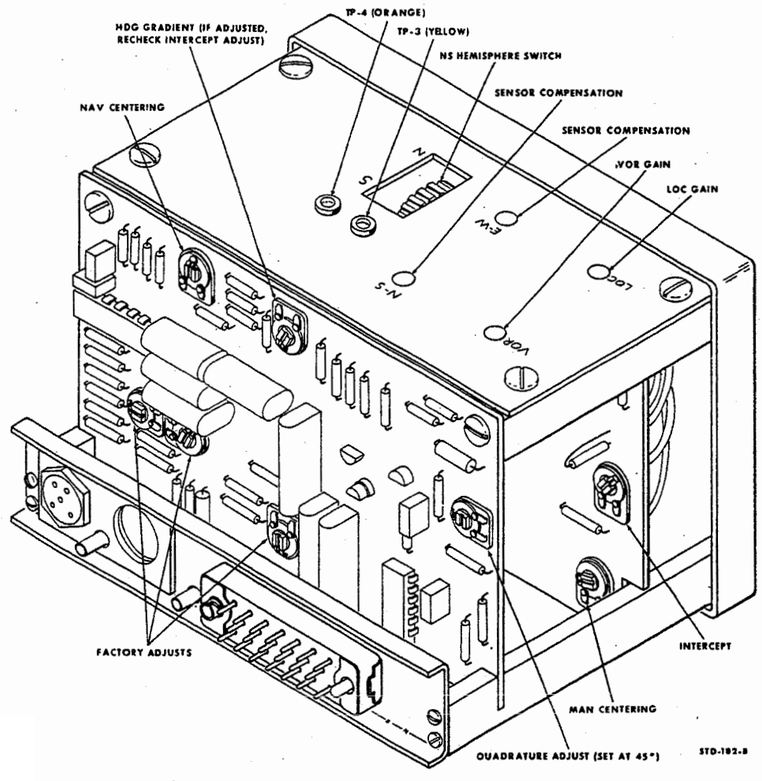
DETAIL A

Rigging the Aileron Control System
Figure 201

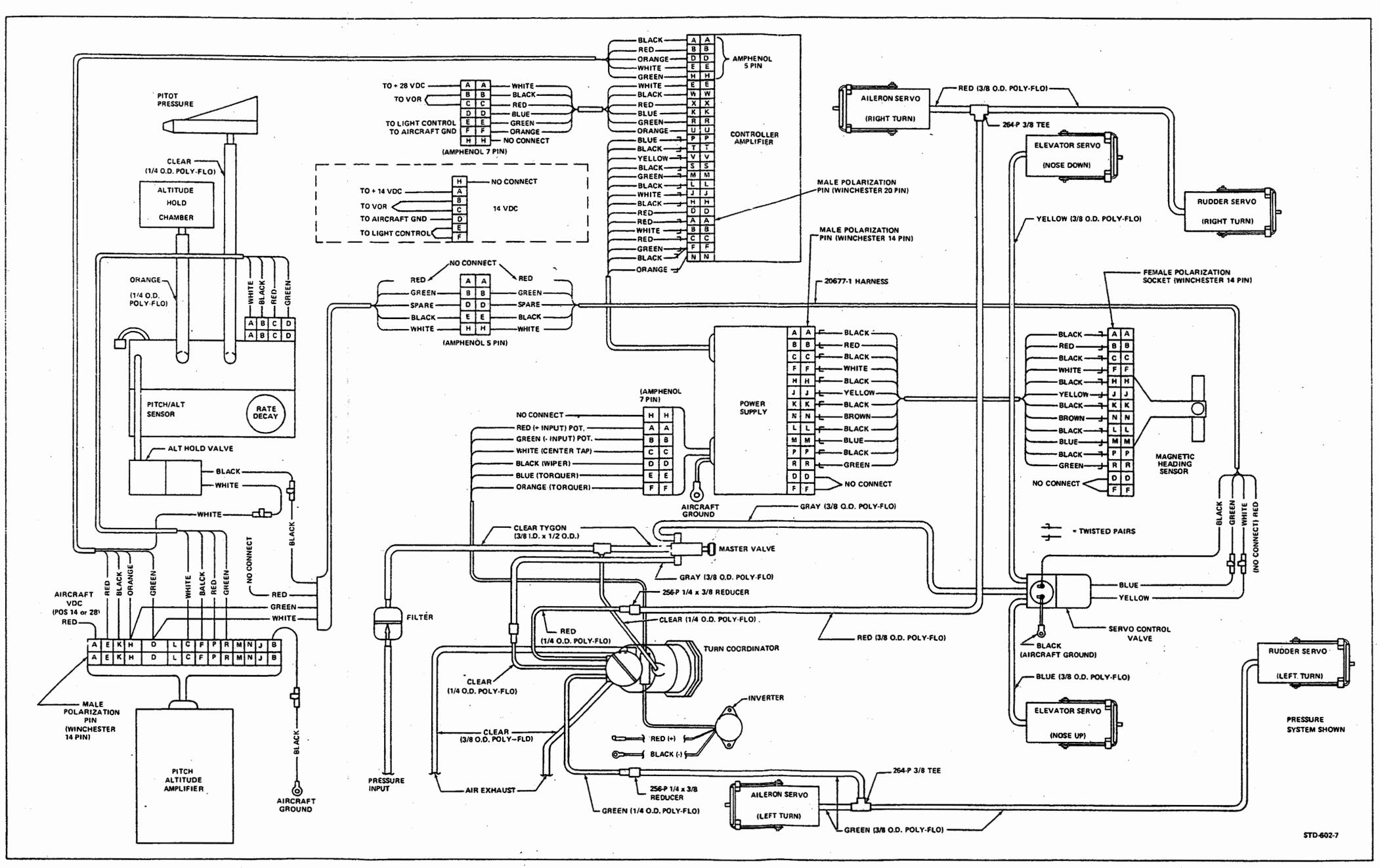
GENERAL - DESCRIPTION AND OPERATION

NEW-MATIC AUTOPILOT

The BEECHCRAFT New-matic autopilots operate on an electro-pneumatic concept. Electronic circuitry is used for navigational beam detections, magnetic heading direction, and turns. Pneumatic servos are used for the flight control actuators. The systems are completely non-tumbling. Yaw, roll and turn detection is made by a tilted gyro EVT turn coordinator (electrical vacuum torquing combination) mounted in the instrument panel. A dampened miniature aircraft serves as the instrument indicating arm. Any deviation from straight flight causes the rate gyro to move a pressure (or vacuum) valve which puts force into the aileron or rudder to return the aircraft to straight flight. 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 signal 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. An altitude hold sensing unit works in conjunction with the pitch control to sustain a given altitude.



Heading Loc/Navigation Coupler System Adjustments
Figure 1



New-matic Autopilot Block Diagram
Figure 2

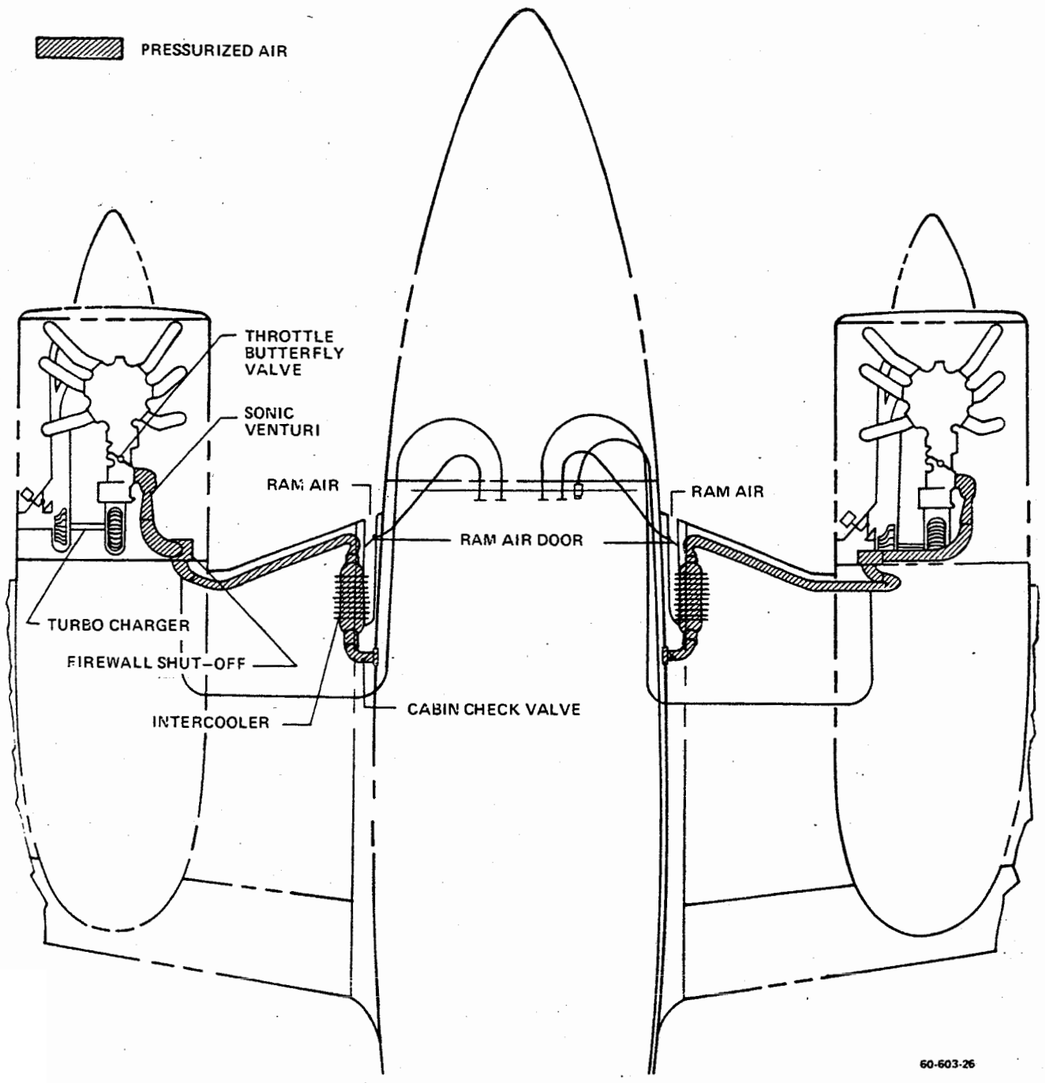
"END"

DISTRIBUTION - MAINTENANCE PRACTICES

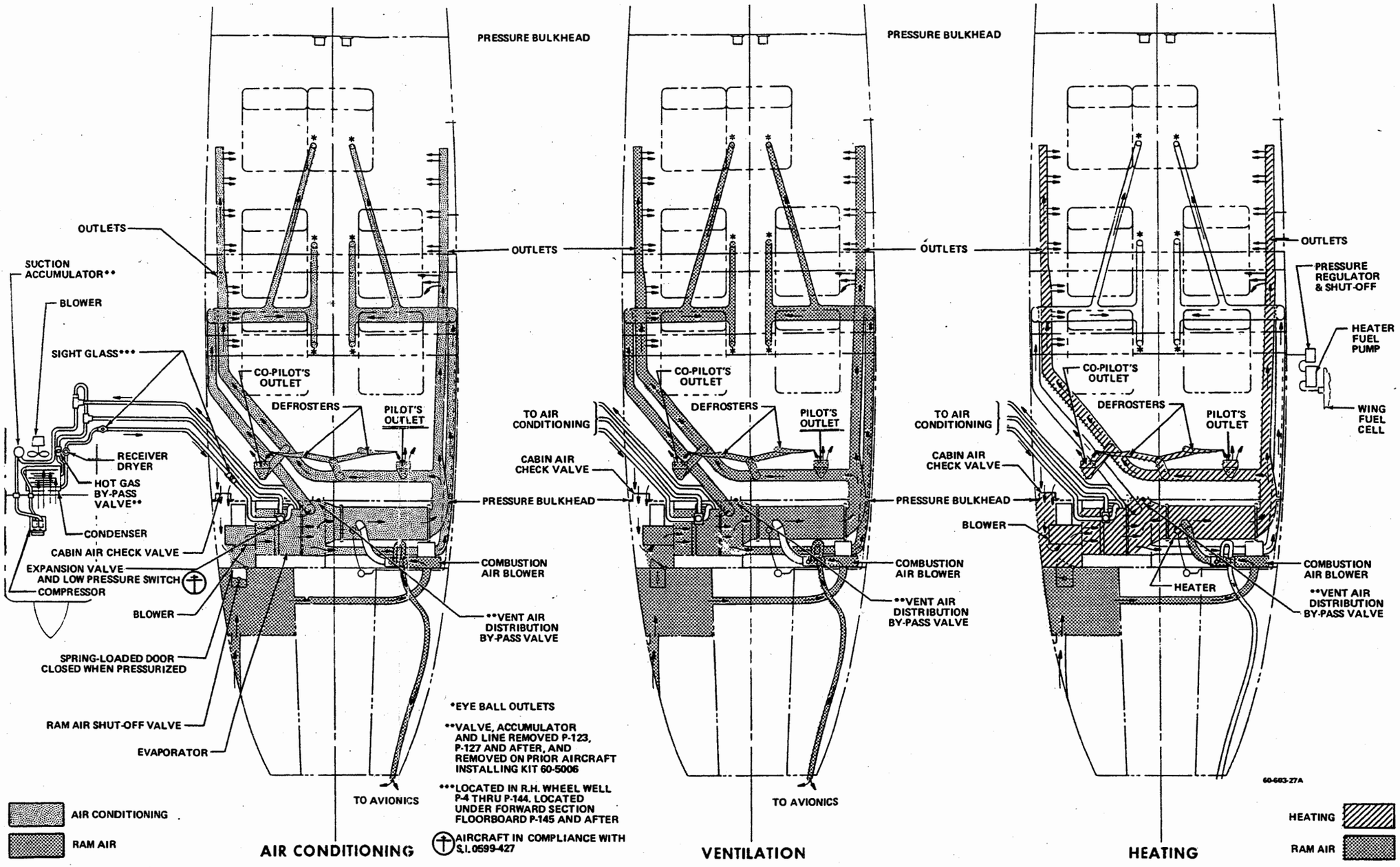
The functional diagrams on the following pages provide a detailed layout of the distribution system utilized by the various series of the Duke.

100-HOUR INSPECTION

Distribution Ducts - Check cabin hot and cold air outlet valves for condition, obstructions and proper operation; check heating and cooling ducts for condition and attachment.



Pressurization Distribution System
Figure 201



Air Conditioning Distribution System (P-3 thru P-246)
Figure 202