SECTION 7
AIRPLANE & SYSTEMS DESCRIPTIONS

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INTRODUCTION

This section provides description and operation of the airplane and its systems. Some equipment described herein is optional and may not be installed in the airplane. Refer to Section 9, Supplements, for details of other optional systems and equipment.

AIRFRAME

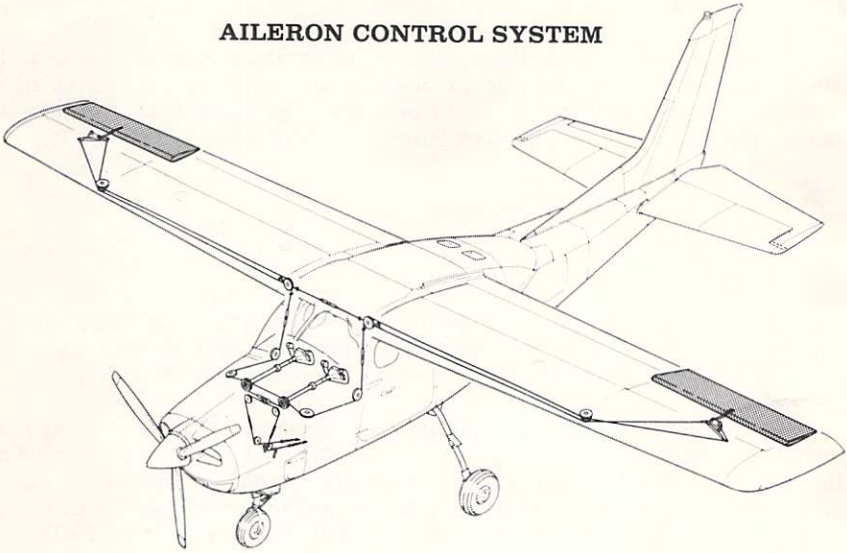
The airplane is an all-metal, six-place, high-wing, single-engine airplane equipped with a cabin pressurization system and retractable tricycle landing gear and designed for general utility purposes.

The construction of the fuselage is a conventional formed sheet metal bulkhead and skin design referred to as semimonocoque. Incorporated into the fuselage structure are a large cabin door opening, an emergency exit door opening and a baggage door opening. Major items of structure include a forward carry-through spar and a forged aluminum main carry-through spar to which the wings are attached. The lower aft portion of the fuselage center section contains the forgings and structure for the retractable main landing gear.

The full cantilever wings have integral fuel tanks and are constructed of a forward spar, main spar, conventional formed sheet metal ribs and aluminum skin. The integral fuel tanks are formed by the forward spar, two sealing ribs, and an aft fuel tank spar forward of the main spar. The Frise-type ailerons and single-slot type flaps are of conventional formed sheet metal ribs and smooth aluminum skin construction. The ailerons are equipped with ground adjustable trim tabs on the inboard end of the trailing edge, and balance weights in the leading edges.

The empennage (tail assembly) consists of a conventional vertical stabilizer, rudder, horizontal stabilizer, and elevator. The vertical stabilizer consists of a forward and aft spar, formed sheet metal ribs and reinforcements, four skin panels, formed leading edge skins, and a dorsal. The rudder is constructed of a forward and aft spar, formed sheet metal ribs and reinforcements, and a wrap-around skin panel. The top of the rudder incorporates a leading edge extension which contains a balance weight. The horizontal stabilizer is constructed of a forward and aft spar, ribs and stiffeners, center upper skin panel, and two left and two right wrap-around skin panels which also form the leading edges. The horizontal stabilizer also contains the elevator trim tab actuator. Construction of the elevator consists of a forward and aft spar, ribs, torque tube and bellcrank, left upper and lower skin panels, a formed one-piece left trailing edge, right

AILERON CONTROL SYSTEM



RUDDER AND RUDDER TRIM CONTROL SYSTEMS

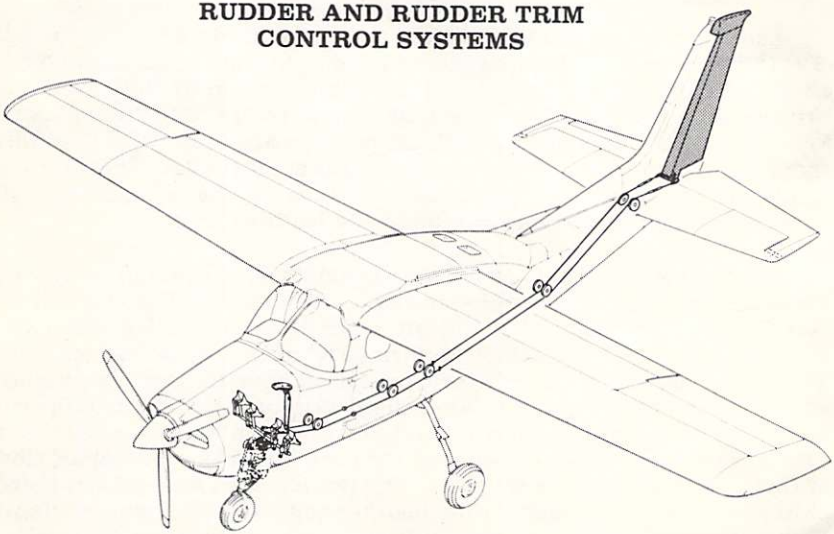
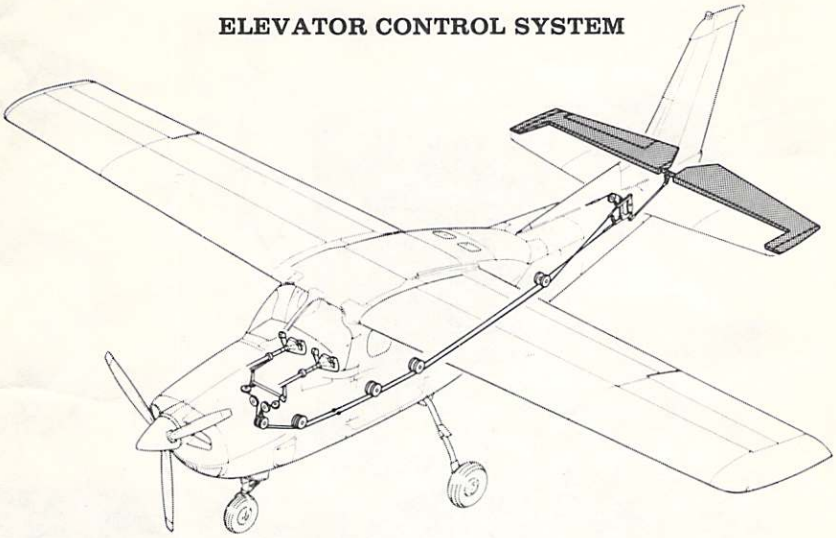


Figure 7-1. Flight Control and Trim Systems (Sheet 1 of 2)



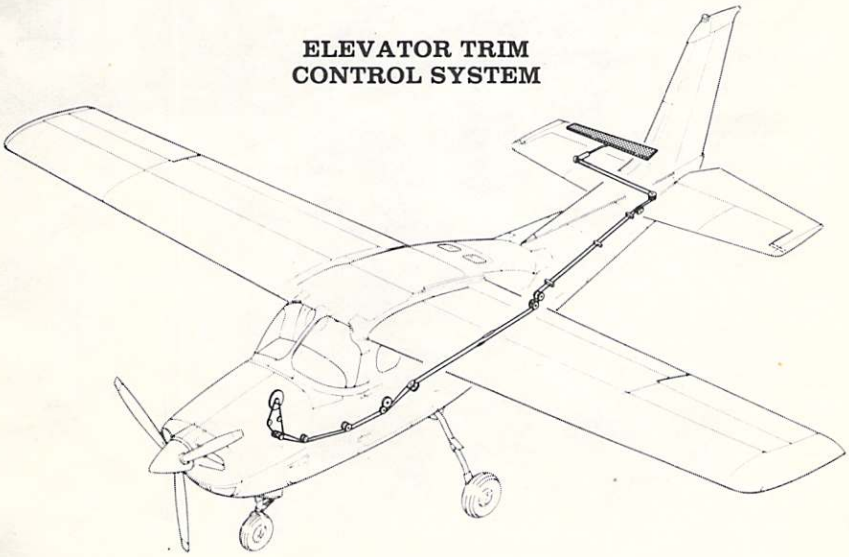


Figure 7-1. Flight Control and Trim Systems (Sheet 2 of 2)

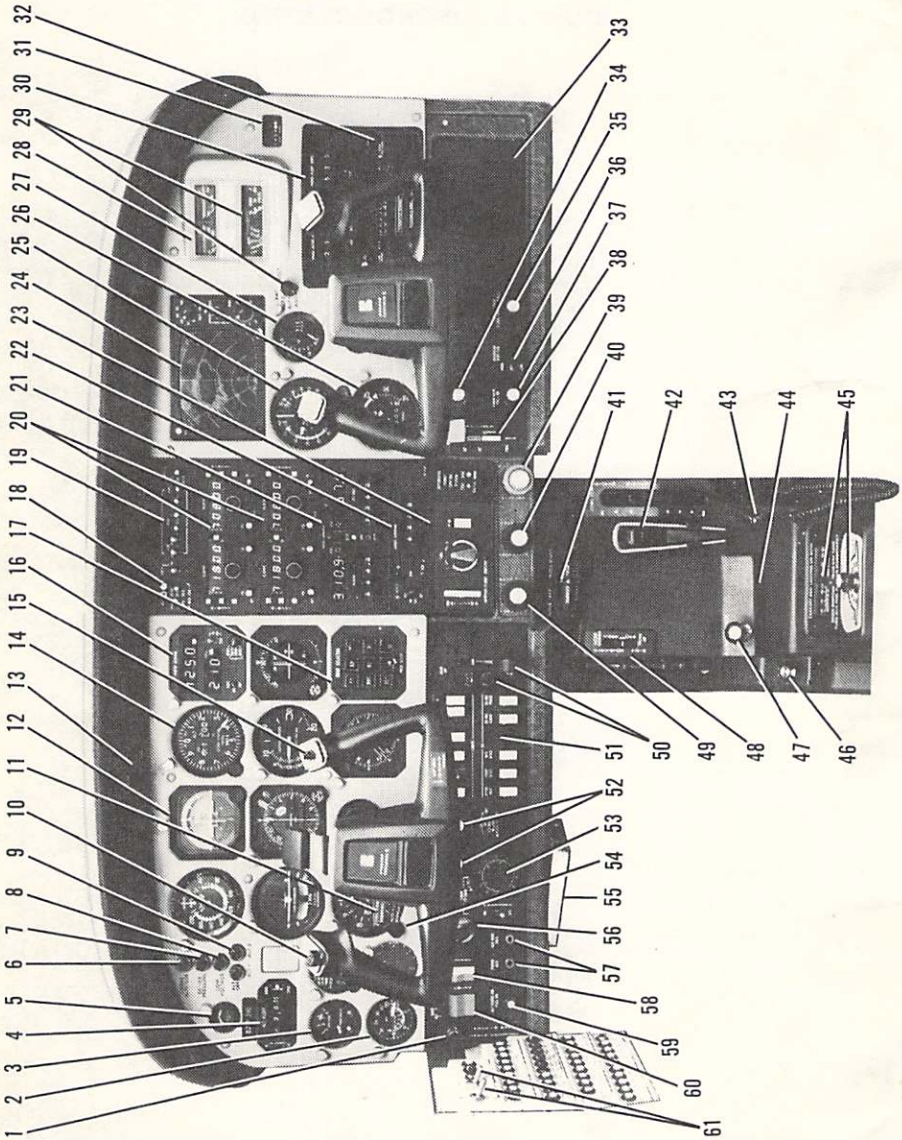


Figure 7-2. Instrument Panel (Sheet 1 of 2)

1. Pressurization Dump Valve Control Handle
2. Pressurization System Instruments
3. Suction Gage
4. Altitude Alerter
5. Propeller Anti-Ice Ammeter
6. Cabin Altitude Warning Light
7. Wing De-Ice Pressure Indicator Light
8. Low-Voltage Warning Light
9. Alternator Out Warning Lights
10. Electric Elevator Trim, IFCS Go-Around and Microphone Switches
11. Autopilot and Electric Elevator Trim Disengage Switches
12. Flight Instrument Group
13. Map Light and Switch
14. Encoding Altimeter
15. IFCS Pitch Synchronizer and Transponder Ident Switches
16. DME
17. IFCS Mode Selector
18. Marker Beacon Indicator Lights and Switches
19. Audio Control Panel
20. NAV/COM Radios
21. Area Navigation (RNAV) Radio
22. Transponder
23. Autopilot Control Unit
24. Weather Radar
25. Manifold Pressure/Fuel Flow Indicator
26. Tachometer
27. Turbine Inlet Temperature Gage
28. Cylinder Head Temperature and Oil Temperature Gages
29. Volt-Ammeter With Selector Switch and Oil Pressure Gage
30. ADF Radio
31. Flight Hour Recorder
32. AM/FM Cassette Stereo Entertainment Center
33. Map Compartment
34. Defrost Control
35. Cabin Air Selector Control
36. Cabin Air Ventilation Fan Switch
37. Cabin Heat Control
38. Wing Flap Switch Lever and Position Indicator
39. Mixture Control
40. Propeller Control
41. Rudder Trim Control Wheel and Position Indicator
42. Microphone
43. Cowl Flap Control Lever
44. Fuel Selector Light
45. Fuel Selector Valve Handle and Fuel Quantity Indicators
46. Fuel On-Off Valve
47. Primer
48. Elevator Trim Control Wheel and Position Indicator
49. Throttle (With Friction Lock)
50. Landing Gear Lever and Position Indicator Lights
51. Electrical Switches
52. Instrument and Control Panel Lighting Controls
53. Pressurization System Controls
54. Secondary Altimeter
55. Parking Brake Handle
56. Ignition Switch
57. Phone and Auxiliary Mike Jacks
58. Auxiliary Fuel Pump Switch
59. Static Pressure Alternate Source Valve
60. Master Switch
61. Avionics Power and Avionics Standby Power Switches

Figure 7-2. Instrument Panel (Sheet 2 of 2)

upper and lower skin panels, and right inboard and outboard formed trailing edges. The elevator trim tab consists of a bracket assembly, hinge half, and a wrap-around skin panel. Both elevator tip leading edge extensions incorporate balance weights.

FLIGHT CONTROLS

The airplane's flight control system (see figure 7-1) consists of conventional aileron, elevator and rudder control surfaces. The control surfaces are manually operated through mechanical linkage using a control wheel for the ailerons and elevator, and rudder/brake pedals for the rudder. The elevator control system is equipped with a downspring, and an aileron-rudder interconnect is incorporated to provide improved stability in flight.

Extensions are available for the rudder/brake pedals. They consist of a rudder pedal face, two spacers and two spring clips. To install an extension, place the clip on the bottom of the extension under the bottom of rudder pedal and snap the top clip over the top of the rudder pedal. Check that the extension is firmly in place. To remove the extensions, reverse the above procedures.

TRIM SYSTEMS

Manually-operated rudder and elevator trim is provided (see figure 7-1). Rudder trimming is accomplished through a bungee connected to the rudder control system and a trim control wheel mounted on the control pedestal. Rudder trimming is accomplished by rotating the horizontally mounted trim control wheel either left or right to the desired trim position. Rotating the trim wheel to the right will trim nose-right; conversely, rotating it to the left will trim nose-left. Elevator trimming is accomplished through the elevator trim tab by utilizing the vertically mounted trim control wheel. Forward rotation of the trim wheel will trim nose-down; conversely, aft rotation will trim nose-up. The airplane may also be equipped with an electric elevator trim system. For details concerning this system, refer to Section 9, Supplements.

INSTRUMENT PANEL

The instrument panel (see figure 7-2) is designed around the basic "T" configuration. The gyros are located directly in front of the pilot and arranged vertically. The airspeed indicator and altimeter are located to the left and right of the gyros respectively. The remainder of the flight

instruments are located around the basic "T". Avionics equipment is stacked slightly right of the panel centerline, with the right side of the panel containing the manifold pressure/fuel flow indicator, turbine inlet temperature gage, AM/FM cassette stereo player (if installed), map compartment, and space for additional instruments and avionics equipment. The top right hand corner of the panel contains the engine instrument cluster angled slightly toward the pilot. A switch and control panel, at the lower edge of the instrument panel, contains most of the switches and controls necessary to operate the airplane. The left side of this panel contains the master switch, auxiliary fuel pump switch, ignition switch, light intensity controls, electrical switches, cabin pressurization controls, static pressure alternate source valve, and the landing gear lever with its associated gear position indicator lights.The lower center portion of the instrument panel contains the throttle, propeller control, and mixture control. The right side of the panel contains the wing flap switch lever and indicator, cabin heating and ventilating controls and the cigar lighter. A pedestal, extending from the lower center area of the instrument panel to the floorboard, contains the elevator and rudder trim control wheels, cowl flap control lever, engine primer and microphone bracket. The fuel fuel on-off valve control, selector valve handle and adjacent fuel quantity indicators are located at the base of the pedestal. A parking brake handle is mounted under the switch and control panel in front of the pilot. All electrical equipment circuit breakers, including an avionics power switch type circuit breaker are located on a separate circuit breaker panel mounted on the left cabin sidewall adjacent to the pilot.

For details concerning the instruments, switches, circuit breakers and controls on this panel, and the circuit breaker panel, refer in this section to the description of the systems to which these items are related.

GROUND CONTROL

Effective ground control while taxiing is accomplished through nose wheel steering by using the rudder pedals; left rudder pedal to steer left and right rudder pedal to steer right. When a rudder pedal is depressed, a spring-loaded steering bungee (which is connected to the nose gear and to the rudder bars) will turn the nose wheel through an arc of approximately 14.5° each side of center. By applying either left or right brake, the degree of turn may be increased up to 35° each side of center.

Moving the airplane by hand is most easily accomplished by attaching a tow bar to the nose gear strut. If a tow bar is not available, or pushing is required, use the main landing gear struts as push points. Do not use the vertical or horizontal tail surfaces to move the airplane. If the airplane is

to be towed by vehicle, never turn the nose wheel more than 35° either side of center or structural damage to the nose gear could result.

The minimum turning radius of the airplane, using differential braking and nose wheel steering during taxi, is approximately 26 feet 11 inches.

WING FLAP SYSTEM

The wing flaps are of the large span, single-slot type (see figure 7-3), and are extended or retracted by positioning the wing flap switch lever on the instrument panel to the desired flap deflection position. The switch lever is moved up or down in a slotted panel that provides mechanical stops at the 10° and 20° positions. For flap settings greater than 10°, move the switch lever to the right to clear the stop and position it as desired. A scale and pointer on the left side of the switch lever indicates flap travel in degrees. The wing flap system circuit is protected by a 10-ampere circuit breaker, labeled FLAP, on the left sidewall circuit breaker panel.

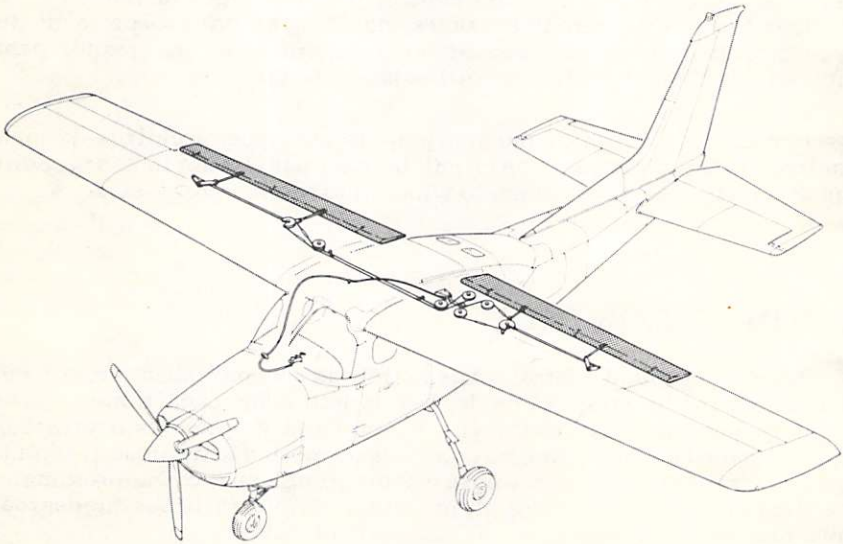


Figure 7-3. Wing Flap System

LANDING GEAR SYSTEM

The landing gear is a retractable, tricycle type with a steerable nose wheel and two main wheels. Shock absorption is provided by the tubular spring-steel main landing gear struts and the air/oil nose gear shock strut. Each main wheel is equipped with a hydraulically actuated disc-type brake on the inboard side of the wheel.

Landing gear extension, retraction, and down lock operation is accomplished by hydraulic actuators powered by an electrically-driven hydraulic power pack (see figure 7-8). The power pack assembly is housed within the control pedestal. Hydraulic system fluid level may be checked by utilizing the dipstick/filler cap, on the power pack, behind a snap-out cover panel on the right side of the control pedestal. The system should be checked at 25-hour intervals. If the fluid level is at or below the ADD line on the dipstick, hydraulic fluid (MIL-H-5606) should be added. Nose gear wheel and strut door operation is accomplished mechanically.

Power pack operation is initiated by a landing gear lever, and is turned off by a pressure switch. Two position indicator lights are provided to show landing gear position. The landing gear system is also equipped with a nose gear safety switch, an emergency extension hand pump, and a gear-up warning system.

LANDING GEAR LEVER

The landing gear lever, mounted to the left of the engine controls, has two positions (up labeled GEAR UP and down labeled GEAR DOWN) which give a mechanical indication of the gear position selected. From either position, the lever must be pulled out to clear a detent before it can be repositioned. Moving the lever out of the GEAR DOWN detent will start the hydraulic power pack. Positioning the lever in the GEAR UP position will direct hydraulic pressure to retract the landing gear. Operation of the landing gear system to extend the gear will not begin until the landing gear lever is repositioned in the GEAR DOWN detent.

LANDING GEAR POSITION INDICATOR LIGHTS

Two position indicator lights, mounted adjacent to the landing gear lever, indicate that the gear is either up or down and locked. The lights are the press-to-test type. The gear-down indicator light (green) has two positions; with the light pushed in half way (throttle retarded and master switch on) the gear warning system should be heard intermittently on the airplane speaker, and with the light pushed full in, it should illuminate. The gear-up indicator light (amber) has only one test position; with the light pushed full in, it should illuminate. The indicator lights contain dimming shutters for night operation.

LANDING GEAR OPERATION

To retract or extend the landing gear, pull out on the gear lever and move it to the desired position. After the lever is positioned, the electrically-driven hydraulic power pack will create pressure in the system and the landing gear will be actuated to the selected position.

CAUTION

If for any reason the hydraulic pump continues to run after gear cycle completion (up or down), the 35-amp "pull-off" type circuit breaker, labeled GEAR PUMP should be pulled out. This will shut off the hydraulic pump motor and prevent damage to the pump and motor. Refer to Section 3 for complete emergency procedures.

During a normal cycle, the gear stops full up or locks down and the position-indicator light (amber for up and green for down) comes on. When the light illuminates, hydraulic pressure will continue to build until a pressure switch turns off the hydraulic pump. The gear is held in the full up position by hydraulic pressure. If the system pressure drops below minimum, the power pack pressure switch will turn the power pack on and return the pressure to maximum except when the nose gear safety (squat) switch is open. During cruising flight with the landing gear retracted, automatic cycling on of the hydraulic pump motor to restore system pressure bleed down may normally occur up to twice per hour. More frequent cycling is an indication of abnormal pressure loss and the cause of this condition should be investigated.

A landing gear safety (squat) switch, actuated by the nose gear strut, electrically prevents inadvertent retraction by the electrically-driven hydraulic power pack whenever the nose gear strut is compressed by the weight of the airplane. When the nose gear is lifted off the runway during takeoff, the squat switch will close, causing the power pack to operate for 1 to 2 seconds which will return system pressure to maximum in the event pressure has dropped.

A "pull-off" type circuit breaker, mounted on the left sidewall circuit breaker panel, should be used for safety during maintenance. With the circuit breaker pulled out, landing gear operation by the gear motor cannot occur. After maintenance is completed, and prior to flight, the circuit breaker should be pushed back in.

WARNING

Safety placards are installed in the nose wheel well to

warn against any maintenance in this area with the circuit breaker pushed in.

EMERGENCY HAND PUMP

A hand-operated hydraulic pump, located between the two front seats, is provided for manual extension of the landing gear in the event of a hydraulic or electrical system failure. The landing gear cannot be retracted with the hand pump. To utilize the pump, extend the handle forward and pump vertically. For complete emergency procedures, refer to Section 3.

For practice manual gear extensions, pull out the GEAR PUMP circuit breaker before placing the landing gear lever in the GEAR DOWN position. After the practice manual extension is completed, push the circuit breaker in to restore normal gear operation.

LANDING GEAR WARNING SYSTEM

The airplane is equipped with a landing gear warning system designed to help prevent the pilot from inadvertently making a wheels-up landing. The system consists of a throttle-actuated switch which is electrically connected to a dual warning unit. The warning unit is connected to the airplane speaker.

When the throttle is retarded below approximately 15 inches of manifold pressure (master switch on), the throttle linkage will actuate a switch which is electrically connected to the gear warning portion of a dual warning unit. If the landing gear is retracted (or not down and locked), an intermittent tone will be heard on the airplane speaker. The system may be checked for correct operation before flight by retarding the throttle to idle and depressing the green gear-down position indicator light half way in. With the indicator light depressed as described, an intermittent tone should be heard on the airplane speaker.

BAGGAGE COMPARTMENT

The baggage compartment consists of the area immediately behind the aft cabin pressure bulkhead and extends rearward to a removable baggage compartment wall in the tailcone. Access to the baggage compartment is gained through a lockable baggage door on the left side of the airplane. When loading the airplane, no material which might be hazardous to the airplane or occupants should be placed in the baggage compartment. For baggage area and door dimensions, refer to Section 6.

SEATS

The seating arrangement consists of four separate adjustable seats and a one-piece fixed seat. The pilot's seat is a six-way adjustable seat, and the front and center passengers seats are four-way adjustable. The front passenger's seat is also available in the six-way adjustable configuration. The two aft passengers utilize a one-piece fixed seat.

The six-way adjustable pilot's seat may be moved forward or aft, adjusted for height and the seat back angle is infinitely adjustable. A lever on the left forward side of the seat controls a special aft position stop, enabling the seat to be positioned aft far enough for easy entry of the copilot and pilot. Once the copilot and pilot are aboard, position the pilot's seat forward by lifting the tubular handle, under the center of the seat bottom, and slide the seat into position; then release the handle and check that the seat is locked in place. Raise or lower the seat by rotating a large crank under the right corner of the seat. Seat back angle is adjustable by rotating a small crank under the left corner of the seat. The seat bottom angle will change as the seat back angle changes, providing proper support. The seat back will also fold full forward. If the front passenger's seat is six-way adjustable, it will function the same as the pilot's seat except the height adjusting and back reclining cranks will be opposite the respective adjustment cranks of the pilot's seat.

Six-way adjustable seats may be equipped with variable lumbar supports located inside the lower seat backs. The firmness of the lower seat back may be controlled by utilizing a button located on the lower inboard side of the seat back. After adjusting the seat back to a comfortable position, move forward on the seat to remove all the weight from the seat back. Hold the button in until the support fully inflates, release the button, and lean back in the seat. If the support is too firm, hold the button in until the desired amount of firmness is obtained.

The four-way adjustable front and center passenger's seats may be moved forward and aft, and the seat back angle is infinitely adjustable. Position the seat by lifting up on the tubular handle under the center of the seat bottom of the front passenger's seat, or the handle under the inboard corner of the center passenger's seats, and slide the seat into position; then release the handle and check that the seat is locked in place. The seat back angle of either front or center passenger seats may be adjusted by rotating a crank under the outboard corner of the seat. The seat bottom angle will change as the seat back angle changes, providing proper support. The seat backs will also fold full forward.

The aft passengers' seats consist of a fixed position one-piece seat bottom and a one-piece seat back rigidly attached to the aft cabin pressure bulkhead. The space under the aft seat is used to house some avionics

components and to provide an exit passage for the cabin air. Consequently, the space around the seat bottom should not be blocked closed.

Headrests are available for any of the seat configurations except the aft passengers' fixed seat. To adjust the headrest, apply enough pressure to it to raise or lower it to the desired level. The headrest may be removed at any time by raising it until it disengages from the top of the seat back.

Removable arm rests are available for the inboard sides of the pilots, front passenger's and center passengers' seats.

SEAT BELTS AND SHOULDER HARNESSES

All seat positions are equipped with seat belts (see figure 7-4). The pilot's and front passenger's seats are also equipped with separate shoulder harnesses; separate shoulder harnesses are available for the remaining seat positions and can be readily installed to attach points furnished in the airplane. Integrated seat belt/shoulder harnesses with inertia reels can be furnished for the pilot's and front passenger's seat positions if desired.

SEAT BELTS

The seat belts used with the pilot, front passenger, and center passenger seats are attached to fittings on the floorboard. The buckle half is inboard of each seat and the link half is outboard of each seat. The belts for the aft seat are attached to the seat frame, with the link halves on the left and right sides of the seat bottom, and the buckles at the center of the seat bottom.

To use the seat belts for the front and center seats, position the seat as desired, and then lengthen the link half of the belt as needed by grasping the sides of the link and pulling against the belt. Insert the lock and belt link into the buckle. Tighten the belt to a snug fit. Seat belts for the aft seat are used in the same manner as the belts for the front and center seats. To release the seat belts, grasp the top of the buckle opposite the link and pull upward.

SHOULDER HARNESSES

Each front seat shoulder harness is attached to a rear doorpost above the window line and is stowed behind a stowage sheath above the cabin door. To stow the harness, fold it and place it behind the sheath. When shoulder harnesses are furnished for the remaining seats, they are attached above and aft of the side windows. Each harness is stowed behind

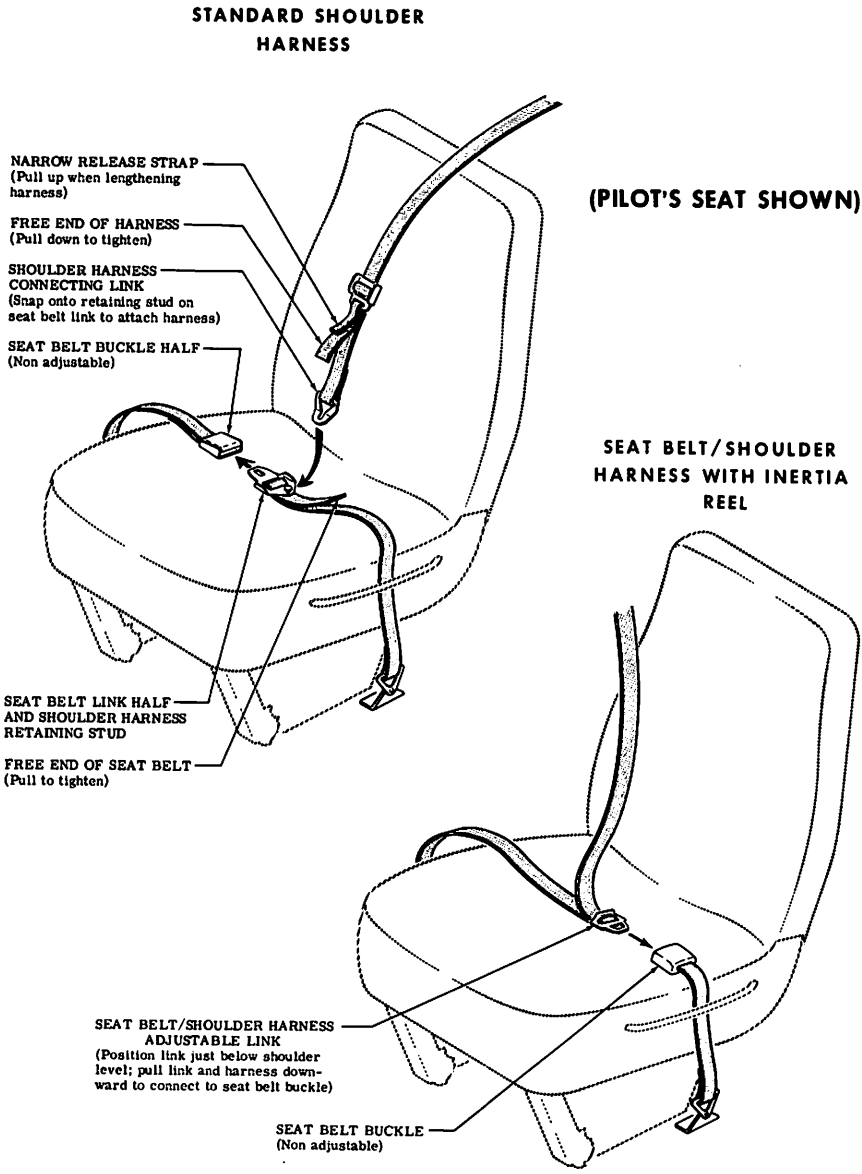


Figure 7-4. Seat Belts and Shoulder Harnesses

a stowage sheath above the side windows.

To use the shoulder harness, fasten and adjust the seat belt first. Lengthen the harness as required by pulling on the connecting link on the end of theharness and the narrow release strap. Snap the connecting link firmly onto the retaining stud on the seat belt link half. Then adjust to length. A properly adjusted harness will permit the occupant to lean forward enough to sit completely erect, but prevent excessive forward movement and contact with objects during sudden deceleration. Also, the pilot will want the freedom to reach all controls easily.

Removing the shoulder harness is accomplished by pulling upward on the narrow release strap, and removing the harness connecting link from the stud on the seat belt link. In an emergency, the shoulder harness may be removed by releasing the seat belt first and allowing the harness, still attached to the link half of the seat belt, to drop to the side of the seat.

INTEGRATED SEAT BELT/SHOULDER HARNESSES WITH INERTIA REELS

Integrated seat belt/shoulder harnesses with inertia reels are available for the pilot and front seat passenger. The seat belt/shoulder harnesses extend from inertia reels located in the cabin top structure, through the overhead console marked PILOT and COPILOT, to attach points inboard of the two front seats. A separate seat belt half and buckle is located outboard of the seats. Inertia reels allow complete freedom of body movement. However, in the event of a sudden deceleration, they will lock automatically to protect the occupants.

To use the seat belt/shoulder harness, position the adjustable metal link on the harness just below shoulder level, pull the link and harness downward, and insert the link in the seat belt buckle. Adjust belt tension across the lap by pulling upward on the shoulder harness. Removal is accomplished by releasing the seat belt buckle, which will allow the inertia reel to pull the harness inboard of the seat.

ENTRY DOOR

Entry to, and exit from the airplane is accomplished through a cabin entry door on the left side of the fuselage at the pilot's position (refer to Section 6 for cabin and cabin door dimensions). The door incorporates a recessed exterior door handle with separate key operated lock, a conventional interior door handle with separate push-pull type safety lock and the internal linkage necessary to connect both handles to several pin-type lock devices spaced around the edge of the door. Additional features of the

door include a door stop mechanism, an emergency external push button-type handle safety lock release and two windows one of which may be opened at anytime except when the cabin is pressurized.

To open the door from outside the airplane, utilize the recessed door handle near the center of the door. Grasp the handle through the cutout provided, pull it out from its recess and rotate it clockwise approximately 90 degrees. A placard, labeled OPEN and CLOSE on the door adjacent to the handle indicates the direction of handle rotation.

From inside the airplane, the door is closed or opened using the conventional inside door handle and the push-pull safety lock. The inside door handle has three positions as indicated by a placard directly behind the handle. The positions are labeled OPEN, CLOSE, and LOCK with the handle spring loaded to the CLOSE position. To close the door after entering the airplane, position the door approximately four or five inches from the fully closed position; then using the assist strap handle, briskly pull the door shut and rotate the inside handle clockwise (full forward) to the LOCK position.

NOTE

> Due to the sealing characteristics of the cabin and entry door, it is necessary to close and lock the door from the inside with the openable window open.

Lock the handle by pushing the handle safety lock flush with the door. In this position the red and white visual indicator on the lock should NOT be visible. The cabin entry door should be locked prior to flight and should not be opened during flight. The handle safety lock should be locked after takeoff (but before pressurizing) and unlocked before landing if passengers are not familiar with its necessary unlocked position to enable opening the door.

CAUTION

> Refer to Section 3 for proper operational procedures to be followed if the entry door should inadvertently open in flight.

Exit from the airplane is accomplished by lifting out on the safety lock (visual indicator exposed), rotating the inside door handle full aft to the OPEN position and pushing the door open. To lock the airplane after exiting, ensure the emergency exit door is closed and locked, close and lock the entry door using the exterior handle, then lock the handle in its recess by use of the ignition key in the key lock.

The exterior emergency push button-type lock release located just forward of the exterior door handle operates in conjunction with the interior safety lock and is used whenever it is desired to open the door from outside the airplane while the interior door handle and safety lock are in their locked positions. Depressing the pushbutton releases the interior safety lock and allows the exterior door handle to function normally to open the door.

An openable window is incorporated into the entry door and may be opened in flight (unpressurized only), if required. The window pivots inboard at its forward edge then slides aft along upper and lower tracks to increase the amount of opening. To open the window, rotate the detent equipped latch counterclockwise until it is clear of its striker, then pull the window inboard until it locks at a slight angle. From this position, the window can be pushed aft as desired. To close the window, reverse this procedure.

EMERGENCY EXIT DOOR

The airplane is provided with an emergency exit door for use in exiting the airplane in the event the cabin door cannot be used. The door, which is located on the right side of the fuselage, is hinged at the top and incorporates a hydraulically-loaded telescoping door lift mechanism attached to the aft edge of the door. Once the door has been opened and pushed outboard a few inches, the door lift automatically lifts it gently to the fully open position.

The emergency exit door is opened or closed from inside the airplane only, utilizing the recessed interior "D" handle and the assist strap. The handle has three positions as indicated by a placard above the handle labeled OPEN, CLOSE and LATCH, and is spring loaded to the CLOSE position. To close the door, pull the door shut and rotate the "D" handle clockwise to the horizontal or LATCH position. From this position, the "D" handle is pushed flush with the door to lock it, as indicated by a placard behind the handle labeled PUSH FLUSH TO LOCK. To open the door, lift the "D" handle out of its recess and rotate it fully counterclockwise to the OPEN position; then push out on the door. The emergency exit door should be locked prior to flight and should not be opened in flight.

CAUTION

Refer to Section 3 for proper operational procedures if the emergency exit door should inadvertently open in flight.

CABIN WINDOWS

The airplane is equipped with a one-piece windshield reinforced with a metal center strip. There are eight cabin side windows. Except for one openable window in the cabin entry door, all windows are the fixed type. In addition, two smaller fixed windows equipped with retractable shades are located in the fuselage top just above the aft passengers' seat.

CONTROL LOCKS

A control lock is provided to lock the aileron and elevator control surfaces to prevent damage to these systems by wind buffeting while the airplane is parked. The lock consists of a shaped steel rod and flag. The flag identifies it as a control lock and cautions about its removal before starting the engine. To install the control lock, align the hole on the right side of the pilot's control wheel shaft with the hole in the right side of the shaft collar on the instrument panel and insert the rod into the aligned holes. Installation of the lock will secure the ailerons in a neutral position and the elevators in a slightly trailing edge down position. Proper installation of the lock will place the flag over the ignition switch. In areas where high or gusty winds occur, a control surface lock should be installed over the vertical stabilizer and rudder. The control lock and any other type of locking device should be removed prior to starting the engine.

ENGINE

The airplane is powered by a horizontally-opposed, six-cylinder, overhead-valve, turbocharged, air-cooled, fuel injected engine equipped with a wet sump oil system and remotely mounted oil cooler. The engine is a Continental Model TSIO-520-AF and is rated at 310 horsepower at 2700 RPM and 35.5 inches of manifold pressure for five minutes, and at 285 horsepower at 2600 RPM and 34.5 inches of manifold pressure continuous. Major accessories include a propeller governor on the front of the engine and dual magnetos, starter, belt-driven alternator, and full flow oil filter on the rear of the engine. Other major accessories include a turbocharger, connected to the induction air and exhaust systems, with associated components, and connections to the cabin pressurization system. Provisions are also made for a second belt-driven alternator and an air conditioner compressor on the front, of the engine and either one or two vacuum pumps on the rear of the engine.

ENGINE CONTROLS

Engine manifold pressure is controlled by a throttle located on the lower center portion of the instrument panel. The throttle operates in a

conventional manner; in the full forward position, the throttle is open, and in the full aft position, it is closed. A friction lock, which is a round knurled disk, is located at the base of the throttle and is operated by rotating the lock clockwise to increase friction or counterclockwise to decrease it. The throttle linkage is designed to mechanically actuate a microswitch electrically connected to the landing gear warning system. The switch will cause a warning tone to sound anytime the throttle is retarded with the landing gear retracted, with less than approximately 15 inches of manifold pressure.

The mixture control, mounted above the right corner of the control pedestal, is a red knob with raised points around the circumference and is equipped with a lock button in the end of the knob. The rich position is full forward, and full aft is the idle cut-off position. For small adjustments, the control may be moved forward by rotating the knob clockwise, and aft by rotating the knob counterclockwise. For rapid or large adjustments, the knob may be moved forward or aft by depressing the lock button in the end of the control, and then positioning the control as desired.

ENGINE INSTRUMENTS

Engine operation is monitored by the following instruments: oil pressure gage, oil temperature gage, cylinder head temperature gage, tachometer, manifold pressure/fuel flow indicator and a turbocharger turbine inlet temperature gage.

The oil pressure gage, located on the right side of the instrument panel, is operated by oil pressure. A direct pressure oil line from the engine delivers oil at engine operating pressure to the oil pressure gage. Gage markings indicate that minimum idling pressure is 10 PSI (red line), the normal operating range is 30 to 60 PSI (green arc), and maximum pressure is 100 PSI (red line).

Oil temperature is indicated by a gage adjacent to the oil pressure gage. The gage is operated by an electrical-resistance type temperature sensor which receives power from the airplane electrical system. Gage markings indicate the normal operating range (green arc) which is 100°F (38°C) to 240°F (116°C), and the maximum (red line) which is 240°F (116°C).

The cylinder head temperature gage, adjacent to the oil temperature gage, is operated by an electrical-resistance type temperature sensor on the engine and is powered by the airplane electrical system. Gage markings indicate the normal operating range (green arc) which is 200°F (93°C) to 460°F (238°C) and the maximum (red line) which is 460°F (238°C).

The engine-driven mechanical tachometer is located on the lower right side of the instrument panel. The instrument is calibrated in increments of

100 RPM and indicates both engine and propeller speed. An hour meter below the center of the tachometer dial records elapsed engine time in hours and tenths. Instrument markings include a normal operating range (green arc) of 2200 to 2500 RPM, a five minute maximum power range (yellow arc) of 2600 to 2700 RPM, and a maximum (red line) of 2700 RPM.

The manifold pressure gage is the left half of a dual-indicating instrument mounted above the tachometer. The gage is direct reading and indicates induction air manifold pressure in inches of mercury. It has a normal operating range (green arc) of 15 to 30 inches of mercury, a five minute maximum power range (yellow arc) of 34.5 to 35.5 inches of mercury, and a maximum (red line) of 35.5 inches of mercury.

The fuel flow indicator is the right half of a dual-indicating instrument mounted above the tachometer. The indicator is a fuel pressure gage calibrated to indicate the approximate pounds per hour of fuel being metered to the engine. The normal cruise range (green arc) is from 36 to 125 PPH, the normal climb range (white arc) is from 125 to 168 PPH, the minimum fuel flow for takeoff power (white triangle) is 180 PPH, the minimum (red line) is 3.0 PSI, and the maximum (red line) is 186 PPH (19.5 PSI).

A turbocharger turbine inlet temperature (TIT) indicator is provided for the airplane and is located on the right side of the instrument panel. A thermocouple probe in the exhaust manifold at the inlet to the turbocharger measures exhaust gas temperature and transmits it to the indicator. The indicator serves as a visual aid to the pilot in adjusting cruise mixture up to 80% power, since exhaust gas temperature varies with fuel-to-air ratio, power, and RPM. The normal operating range (green arc) is from 1300 to 1650 degrees F. The maximum limit (red line) is 1650 degrees F.

NEW ENGINE BREAK-IN AND OPERATION

The engine underwent a run-in at the factory and is ready for the full range of use. It is, however, suggested that cruising be accomplished at 70% to 80% power until a total of 50 hours has accumulated or oil consumption has stabilized. This will ensure proper seating of the rings.

The airplane is delivered from the factory with corrosion preventive oil in the engine. If, during the first 25 hours, oil must be added, use only aviation grade straight mineral oil conforming to Specification No. MIL-L-6082.

ENGINE OIL SYSTEM

Oil for engine lubrication, propeller governor operation, and turbocharger lubrication and system control is supplied from a sump on the

bottom of the engine. The capacity of the engine sump is 10 quarts (one additional quart is contained in the engine oil filter). Oil is drawn from the sump through a filter screen on the end of a pickup tube to the engine-driven oil pump. Oil from the pump passes through the full flow oil filter to the turbocharger system controls, a pressure relief valve at the rear of the right oil gallery, and a thermostatically controlled, remotely mounted, oil cooler. Oil from the cooler is then circulated to the left gallery and propeller governor. The engine parts and turbocharger are then lubricated by oil from the galleries. After lubricating the engine, the oil returns to the sump by gravity, and oil from the turbocharger is returned to the sump by a scavenger pump. The oil filter adapter is equipped with a bypass valve which will cause lubricating oil to bypass the filter in the event the filter becomes plugged, or the oil temperature is extremely cold.

An oil dipstick is located at the rear of the engine on the left side, and an oil filler tube is on top of the crankcase near the front of the engine. The dipstick and oil filler are accessible through doors on the engine cowling. The engine should not be operated on less than 7 quarts of oil. To minimize loss of oil through the breather, fill to 8 quarts for normal flights of less than three hours. For extended flight, fill to 10 quarts (dipstick indication only). For engine oil grade and specifications, refer to Section 8 of this handbook.

IGNITION-STARTER SYSTEM

Engine ignition is provided by two engine-driven magnetos and two spark plugs in each cylinder. The right magneto fires the lower left and upper right spark plugs, and the left magneto fires the lower right and upper left spark plugs. Normal operation is conducted with both magnetos due to the more complete burning of the fuel-air mixture with dual ignition.

Ignition and starter operation is controlled by a rotary type switch located on the left switch and control panel. The switch is labeled clockwise, OFF, R, L, BOTH, and START. The engine should be operated on both magnetos (BOTH position) except for magneto checks. The R and L positions are for checking purposes and emergency use only. When the switch is rotated to the spring-loaded START position, (with the master switch in the ON position), the stacter contactor is energized and the starter will crank the engine. When the switch is released, it will automatically return to the BOTH position.

AIR INDUCTION SYSTEM

The engine air induction system receives ram air through a flush aerodynamic scoop which is an integral part of the lower right removable cowl. Air enters the scoop and flows through a removable air filter mounted adjacent to the scoop. The filter removes dust and other foreign

matter from the induction air. However, in the event the airscoop or the induction air filter become blocked, suction created by the engine will open an alternate air door allowing air to be admitted from a series of louvered openings located immediately aft of the main induction air scoop. This air bypasses the air filter and will result in a decrease of up to 10 inches Hg manifold pressure from a cruise power setting. Maximum continuous manifold pressure (34.5 inches Hg) can be maintained with throttle and / or RPM adjustment up to 13,000 feet under warm day conditions with the alternate air door open. A flexible duct connects the air filter airbox to the turbocharger compressor inlet. As the air passes through the compressor, it is pressurized, then ducted through a fuel/ air control unit and induction manifold to the cylinders.

EXHAUST SYSTEM

Exhaust gas from each cylinder passes through riser assemblies into an exhaust manifold which discharges the gas into the turbine section of the turbocharger. After the exhaust gas has passed through the turbine, it is vented overboard through a tailpipe. A waste gate is incorporated into the exhaust manifold, and controls the amount of exhaust gas to the turbine by venting excess gas to the tailpipe through a bypass.

FUEL INJECTION SYSTEM

The engine is equipped with a fuel injection system. The system is comprised of an engine-driven fuel pump, fuel/air control unit, fuel manifold, fuel flow indicator, and air-bleed type injector nozzles.

Fuel is delivered by the engine-driven fuel pump to the fuel/ air control unit behind the engine. The fuel/ air control unit correctly proportions the fuel flow to the induction air flow. After passing through the control unit, induction air is delivered to the cylinders through intake manifold tubes, and metered fuel is delivered to a fuel manifold. The fuel manifold, through spring tension on a diaphragm and valve, evenly distributes the fuel to an air-bleed type injector nozzle in the intake valve chamber of each cylinder. A pressure line is also attached to the fuel manifold, and is connected to a fuel flow indicator on the instrument panel.

COOLING SYSTEM

Ram air for engine cooling enters through two intake openings in the front of the engine cowling. The cooling air is directed around the cylinders and other areas of the engine by baffling, and is then exhausted through cowl flaps on the lower aft edge of the cowling. An airscoop on the lower left side of the cowling directs ram air to the remotely mounted oil cooler. After passing through the oil cooler, ram air is exhausted through the cowl flaps. The cowl flaps are mechanically operated from the cabin by means of

a cowl flap lever on the right side of the control pedestal. The pedestal is labeled COWL FLAP, OPEN, CLOSED. During takeoff and high power operation, the cowl flap lever should be placed in the OPEN position for maximum cooling. This is accomplished by moving the lever to the right to clear a detent, then moving the lever up to the OPEN position. Anytime the lever is repositioned, it must first be moved to the right. While in cruise flight, cowl flaps should be adjusted to keep the cylinder head temperature within the normal operating range (green arc). During extended let-downs, it may be necessary to completely close the cowl flaps by pushing the cowl flap lever down to the CLOSED position.

TURBOCHARGING SYSTEM

Because the engine is turbocharged, some of its characteristics are different from a normally aspirated engine. The following information describes the system and points out some of the items that are affected by turbocharging. Section 4 contains the normal operating procedures for the turbocharged engine.

The following steps, when combined with the turbocharger system schematic (figure 7-5), provide a better understanding of how the turbocharger system works. The steps follow the induction air as it enters and passes through the engine until it is expelled as exhaust gases.

1. Engine induction air is taken in through an opening in the lower right cowling, ducted through a filter and into the compressor where it is compressed.
2. The pressurized induction air then passes through the throttle body and induction manifold into the cylinders.
3. The air and fuel are burned and exhausted to the supercharger turbine.
4. The exhaust gases drive the turbine which, in turn, drives the compressor, thus completing the cycle.

The compressor has the capability of producing manifold pressures in excess of the 5 minute takeoff maximum of 35.5 inches Hg. In order not to exceed 35.5 inches of manifold pressure, a waste gate is used so that some of the exhaust will bypass the turbine and be vented into the tailpipe.

It can be seen from studying Steps 1 through 4 that anything that affects the flow of induction air into the compressor or the flow of exhaust gases into the turbine will increase or decrease the speed of the turbocharger. This resultant change in flow will have no effect on the engine if the waste gate is still open since the waste gate position is changed to hold the relationship between compressor discharge pressure and manifold pressure near optimum. The waste gate controller automatically changes compressor discharge pressure approximately 70% of the amount of

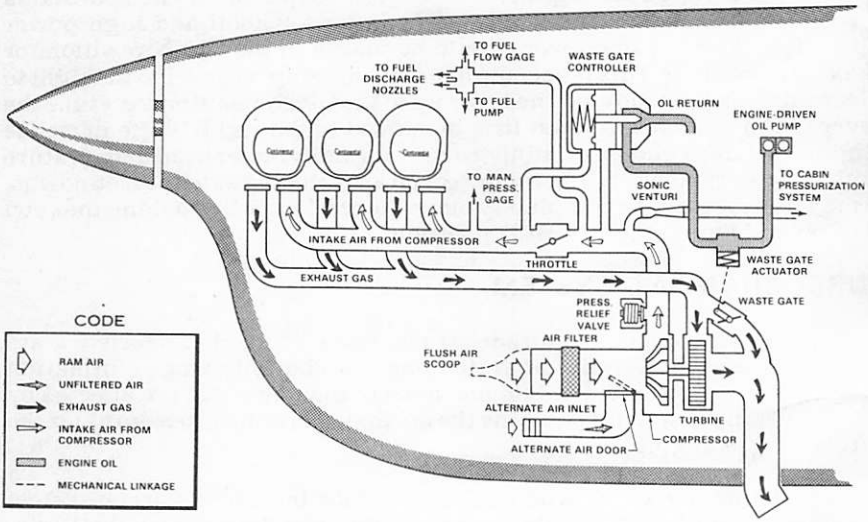


Figure 7-5. Turbocharger System

change desired in manifold pressure as controlled by the throttle. This relationship minimizes the work done by the turbocharger in producing a given manifold pressure while still maintaining adequate compressor discharge pressure for cabin pressurization bleed.

At high altitude, part throttle, or low RPM, the exhaust flow is not capable of turning the turbine and compressor fast enough to maintain maximum compressor discharge pressure, and the waste gate will close to force all of the exhaust flow through the turbine.

When the waste gate is fully closed, any change in turbocharger speed will mean a change in engine operation. Thus, any increase or decrease in turbine speed will cause an increase or decrease in manifold pressure and fuel flow. If turbine speed increases, the manifold pressure increases; if the turbine speed decreases, the manifold pressure decreases. Since the compression ratio approaches 3 to 1 at high altitude, any change in exhaust flow to the turbine or ram induction air pressure will be magnified proportionally by the compression ratio and the change in flow through the exhaust system.

MANIFOLD PRESSURE VARIATION WITH ENGINE RPM

When the waste gate is open, the turbocharged engine will react the same as a normally aspirated engine when the engine RPM is varied. That

is, when the RPM is increased, the manifold pressure will decrease slightly. When the engine RPM is decreased, the manifold pressure will increase slightly.

However, when the waste gate is closed, manifold pressure variation with engine RPM is just the opposite of the normally aspirated engine. An increase in engine RPM will result in an increase in manifold pressure, and a decrease in engine RPM will result in a decrease in manifold pressure.

MANIFOLD PRESSURE VARIATION WITH ALTITUDE

At full throttle, the turbocharger has the capability of maintaining the maximum continuous manifold pressure of 34.5 inches Hg to well above 18,000 feet depending on engine and atmospheric conditions. However, engine operating limitations establish the maximum manifold pressure that may be used. Manifold pressure must be reduced above 18,000 feet, as noted on the operating placard in the airplane (subtract 1 inch Hg from 34.5 inches for each 1000 feet above 18,000 feet).

At part throttle, the turbocharger is capable of maintaining cruise climb power of 2500 RPM and 30 in. Hg from sea level to 17,000 feet in standard temperatures, and from sea level to 12,000 feet under hot day conditions without changing the throttle position, once the power setting is established after takeoff. This climb power setting is maintained above these altitudes by advancing the throttle as necessary to maintain 30 inches of manifold pressure in the same manner as a normally aspirated engine during climb.

MANIFOLD PRESSURE VARIATION WITH AIRSPEED

When the waste gate is closed, manifold pressure will vary with variations in airspeed. This is because the compressor side of the turbocharger operates at pressure ratios of up to 3 to 1 and any change in pressure at the compressor inlet is magnified at the compressor outlet with a resulting effect on the exhaust flow and turbine side of the turbocharger.

FUEL FLOW VARIATIONS WITH CHANGES IN MANIFOLD PRESSURE

The engine-driven fuel pump output is regulated by engine speed and compressor discharge pressure. Engine fuel flow is regulated by fuel pump output and the metering effects of the throttle and mixture control. When the waste gate is open, fuel flow will vary directly with manifold pressure, engine speed, mixture, or throttle control position. In this case, manifold pressure is controlled by throttle position and the waste gate

controller, while fuel flow varies with throttle movement and manifold pressure.

When the waste gate is closed and manifold pressure changes are due to turbocharger output, as discussed previously, fuel flow will follow manifold pressure even though the throttle position is unchanged. This means that fuel flow adjustments required of the pilot are minimized to (1) small initial adjustments on takeoff or climb-out for the proper rich climb setting, (2) lean-out in cruise, and (3) return to full rich position for approach and landing.

MANIFOLD PRESSURE VARIATION WITH INCREASING OR DECREASING FUEL FLOW

When the waste gate is open, movement of the mixture control has little or no effect on the manifold pressure of the turbocharged engine.

When the waste gate is closed, any change in fuel flow to the engine will have a corresponding change in manifold pressure. That is, increasing the fuel flow will increase the manifold pressure and decreasing the fuel flow will decrease the manifold pressure. This is because an increased fuel flow to the engine increases the mass flow of the exhaust. This turns the turbocharger faster, increasing the induction air flow and raising the manifold pressure.

MOMENTARY OVERSHOOT OF MANIFOLD PRESSURE

Under some circumstances (such as rapid throttle movement, especially with cold oil), it is possible that the engine can be overboosted slightly above the maximum five minute takeoff manifold pressure of 35.5 inches. This would most likely be experienced during the takeoff roll or during a change to full throttle operation in flight. The induction air pressure relief valve will normally limit the overboost to 2 to 4 inches.

A slight overboost of 2 to 4 inches of manifold pressure is not considered detrimental to the engine as long as it is momentary. No corrective action is required when momentary overboost corrects itself and is followed by normal engine operation. However, if overboosting of this nature persists when oil temperature is normal or if the amount of overboost tends to exceed 4 inches or more, the throttle should be retarded to eliminate the overboost and the controller system, including the waste gate and relief valve, should be checked for necessary adjustment or replacement of components.

ALTITUDE OPERATION

Because a turbocharged airplane will climb faster and higher than a

normally aspirated airplane, fuel vaporization may be encountered. When fuel flow variations of ± 5 PPH or more are observed (as a "nervous" fuel flow needle), placing the auxiliary fuel pump switch in the ON position will control vapor. However, it will also increase fuel flow, making it necessary to adjust the mixture control for the desired fuel flow. The auxiliary fuel pump should be left on for the remainder of the climb. It can be turned off whenever fuel flow will remain steady with it off, and the mixture must be readjusted accordingly. The auxiliary fuel pump should be turned off and the mixture reset prior to descent.

HIGH ALTITUDE ENGINE ACCELERATION

The engine will accelerate normally from idle to full throttle with full rich mixture at any altitude below 20,000 feet. At higher altitudes, it is usually necessary to lean the mixture to get smooth engine acceleration from idle to maximum power.

PROPELLER

The airplane has an all-metal, three-bladed, constant-speed, governor-regulated propeller. A setting introduced into the governor with the propeller control establishes the propeller speed, and thus the engine speed to be maintained. The governor then controls flow of engine oil, boosted to high pressure by the governing pump, to or from a piston in the propeller hub. Oil pressure acting on the piston twists the blades toward high pitch (low RPM). When oil pressure to the piston in the propeller hub is relieved, centrifugal force, assisted by an internal spring, twists the blades toward low pitch (high RPM).

A control knob on the lower center portion of the instrument panel is used to set the propeller and control engine RPM as desired for various flight conditions. The knob is labeled PROP PITCH PUSH INCR RPM. When the control knob is pushed in, blade pitch will decrease, giving a higher RPM. When the control knob is pulled out, the blade pitch increases, thereby decreasing RPM. The propeller control knob is equipped with a vernier feature which allows slow or fine RPM adjustments by rotating the knob clockwise to increase RPM, and counterclockwise to decrease it. To make rapid or large adjustments, depress the button on the end of the control knob and reposition the control as desired.

FUEL SYSTEM

The fuel system (see figure 7-6) consists of two vented integral fuel

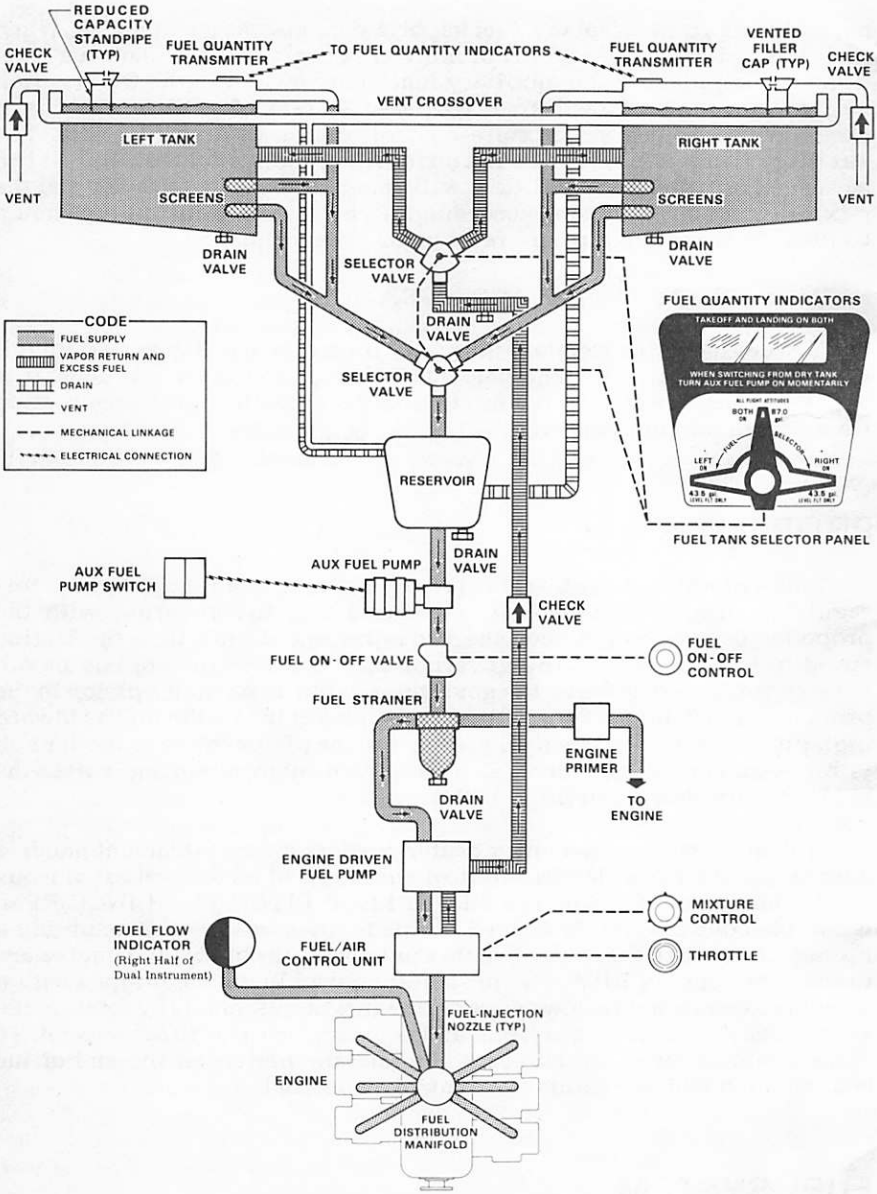


Figure 7-6. Fuel System

tanks (one in each wing), a fuel selector valve, a reservoir, an auxiliary fuel pump, an on-off valve, fuel strainer, engine-driven fuel pump, fuel/air control unit, fuel manifold, and fuel injection nozzles.

NOTE

Unusable fuel is at a minimum due to the design of the fuel system. However, when the fuel tanks are 1/4 full or less, prolonged uncoordinated flight such as slips or skids can uncover the fuel tank outlets, possibly causing fuel starvation and engine stoppage. Therefore, with low fuel reserves, do not allow the airplane to remain in uncoordinated flight for periods in excess of 30 seconds with fuel selector in LEFT ON or RIGHT ON or one minute with the selector in BOTH ON position.

Fuel flows by gravity from the two integral tanks (one in each wing) to a two-segment, three-position selector valve labeled LEFT ON, RIGHT ON, and BOTH ON.

NOTE

When the fuel selector valve handle is in the BOTH ON position in cruising flight, unequal fuel flow from each tank may occur if the wings are not maintained exactly level. Resulting wing heaviness can be alleviated gradually by turning the selector valve handle to the tank in the "heavy" wing.

With the selector valve handle in one of the designated detent positions, fuel from either the left tank, right tank or both tanks flows through the selector valve, the reservoir, through a bypass in the auxiliary fuel pump (when it is not in operation), and through a fuel on-off valve to a strainer mounted on the firewall. The engine-driven fuel pump delivers fuel from the strainer to the fuel/air control unit where it is metered and directed to a manifold which distributes it to each cylinder. Vapor and excess fuel from the engine-driven fuel pump and the fuel/air control unit are returned by way of a separate segment of the selector valve to the wing fuel tank or tanks, as selected by the fuel selector handle position.

A two-position on-off valve is incorporated into the fuel system to be used for maintenance purposes or in the event of emergency conditions requiring fuel flow to be shut off. A push-pull control, labeled FUEL VALVE PUSH ON and located on the lower left side of the pedestal, controls valve position. Under normal conditions the control is pushed fully in to the ON position. To shut off fuel flow, the control is pulled fully out.

FUEL QUANTITY DATA (U.S. GALLONS)				
FUEL TANKS	FUEL LEVEL (QUANTITY EACH TANK)	TOTAL FUEL	TOTAL UNUSABLE	TOTAL USABLE ALL FLIGHT CONDITIONS
STANDARD	FULL (45)	90	3	87
	REDUCED (33.5)	67	3	64

Figure 7-7. Fuel Quantity Data

Fuel system venting is essential to system operation. Complete blockage of the venting system will result in decreasing fuel flow and eventual engine stoppage. Venting is accomplished by vent lines, one from each fuel tank, which are equipped with check valves, and a tank interconnect vent line. The fuel filler caps are equipped with vacuum operated vents which open, allowing air into the tanks, should the fuel tank vent lines become blocked.

Fuel quantity is measured by two float-type fuel quantity transmitters (one in each tank) and indicated by two electrically-operated fuel quantity indicators on the lower portion of the pedestal adjacent to the fuel selector valve handle. The indicators are marked in pounds (top scale) and gallons (bottom scale) with a red line indicating an empty tank. When an indicator shows an empty tank, approximately 1.5 gallon remains in the tank as unusable fuel. The indicators cannot be relied upon for accurate readings during skids, slips or unusual attitudes. Maximum indicator travel is reached with 40 to 41 gallons in the tank. Therefore, indications at the right end of the scale (40 gallons to F) should be verified by visual inspection of the tanks if a short field takeoff or a long range flight is planned. If both indicator pointers should rapidly move to a zero reading, check the cylinder head temperature and oil temperature gages for operation. If these gages are not indicating, an electrical malfunction has occurred.

The airplane may be serviced to a reduced capacity to permit heavier cabin loadings. This is accomplished by filling each tank to the bottom edge of the fuel filler neck, thus giving a reduced fuel load of 201 pounds in each tank (192 pounds usable in all flight conditions).

Before refueling or when the airplane is parked on a slope, place the fuel selector handle in the LEFT ON or RIGHT ON position, whichever corresponds to the low wing. This action minimizes crossfeeding from the fuller tank and reduces fuel seepage from the wing tank vents.

The auxiliary fuel pump switch is located on the left side of the instrument panel and is a yellow and red split-rocker type switch. The yellow right half of the switch is labeled START, and its upper ON position is used for normal starting, minor vapor purging and continued engine operation in the event of an engine-driven fuel pump failure. With the right half of the switch in the ON position, the pump operates at one of two flow rates that are dependent upon the setting of the throttle. With the throttle open to a cruise setting, the pump operates at a high enough capacity to supply sufficient fuel flow to maintain flight with an inoperative engine-driven fuel pump. When the throttle is moved toward the closed position (as during letdown, landing, and taxiing), the fuel pump flow rate is automatically reduced, preventing an excessively rich mixture during these periods of reduced engine speed.

NOTE

If the engine-driven fuel pump is functioning and the auxiliary fuel pump switch is placed in the ON position, an excessively rich fuel/air ratio is produced unless the mixture is leaned. Therefore, this switch should be turned off during takeoff (unless takeoff fuel flow is slightly deficient) and during landing.

NOTE

If the auxiliary fuel pump switch is accidentally placed in the ON position with the master switch on and the engine stopped, the intake manifolds will be flooded.

The red left half of the switch is labeled EMERG, and its upper HI position is used in the event of an engine-driven fuel pump failure during takeoff or high power operation. The HI position may also be used for extreme vapor purging. Maximum fuel flow is produced when the left half of the switch is held in the spring-loaded HI position. In this position, an interlock within the switch automatically trips the right half of the switch to the ON position. When the spring-loaded left half of the switch is released, the right half will remain in the ON position until manually returned to the off position.

Under hot day-high altitude conditions, or conditions during a climb that are conducive to fuel vapor formation, it may be necessary to utilize the auxiliary fuel pump to attain or stabilize the fuel flow required for the type of climb being performed. In this case, turn the auxiliary fuel pump on and adjust the mixture to the desired fuel flow. If fluctuating fuel flow (greater than 5 PPH is observed during climb or cruise at high altitudes on hot days, place the auxiliary fuel pump switch in the ON position to clear the fuel system of vapor. The auxiliary fuel pump may be operated

continuously in cruise, if necessary, but should be turned off prior to descent. Each time the auxiliary fuel pump switch is turned on or off, the mixture should be readjusted.

If it is desired to completely exhaust a fuel tank quantity in flight, the auxiliary fuel pump will be needed to assist in restarting the engine when fuel exhaustion occurs. Therefore, it is recommended that proper operation of the auxiliary fuel pump be verified prior to running a fuel tank dry by turning the auxiliary fuel pump ON momentarily and checking for a slight rise in fuel flow indication.

To ensure a prompt engine restart in flight after running a fuel tank dry, immediately switch to the tank containing fuel or to BOTH ON at the first indication of fuel pressure fluctuation and/or power loss. Then place the right half of the auxiliary fuel pump switch in the ON position momentarily (3 to 5 seconds) with the throttle at least 1/2 open. Excessive use of the ON position at high altitude and full rich mixture can cause flooding of the engine as indicated by a short (1 to 2 seconds) period of power followed by a loss of power. This can later be detected by a fuel flow indication accompanied by a lack of power. If flooding does occur, turn off the auxiliary fuel pump switch, and normal propeller windmilling should start the engine in 1 to 2 seconds. At high altitudes, leaning the mixture will hasten restarting.

If the propeller should stop (possible at very low airspeeds) before the tank containing fuel is selected, place the auxiliary fuel pump switch in the ON position and advance the throttle promptly until the fuel flow indicator registers approximately 1/2 way into the green arc for 1 to 2 seconds duration. Then retard the throttle, turn off the auxiliary fuel pump, and use the starter to turn the engine over until a start is obtained.

The fuel system is equipped with drain valves to provide a means for the examination of fuel in the system for contamination and grade. The system should be examined before the first flight of every day and after each refueling, by using the sampler cup provided to drain fuel from the wing tank sumps, and by utilizing the fuel strainer quick-drain on the lower left side of the engine cowling. Quick-drain valves are also provided for the fuel reservoir and the vapor return line. These valves are located beneath the cabin. The fuel tanks should be filled after each flight to minimize condensation.

HYDRAULIC SYSTEM

Hydraulic power (see figure 7-8) is supplied by an electrically-driven hydraulic power pack located behind the control pedestal. The power

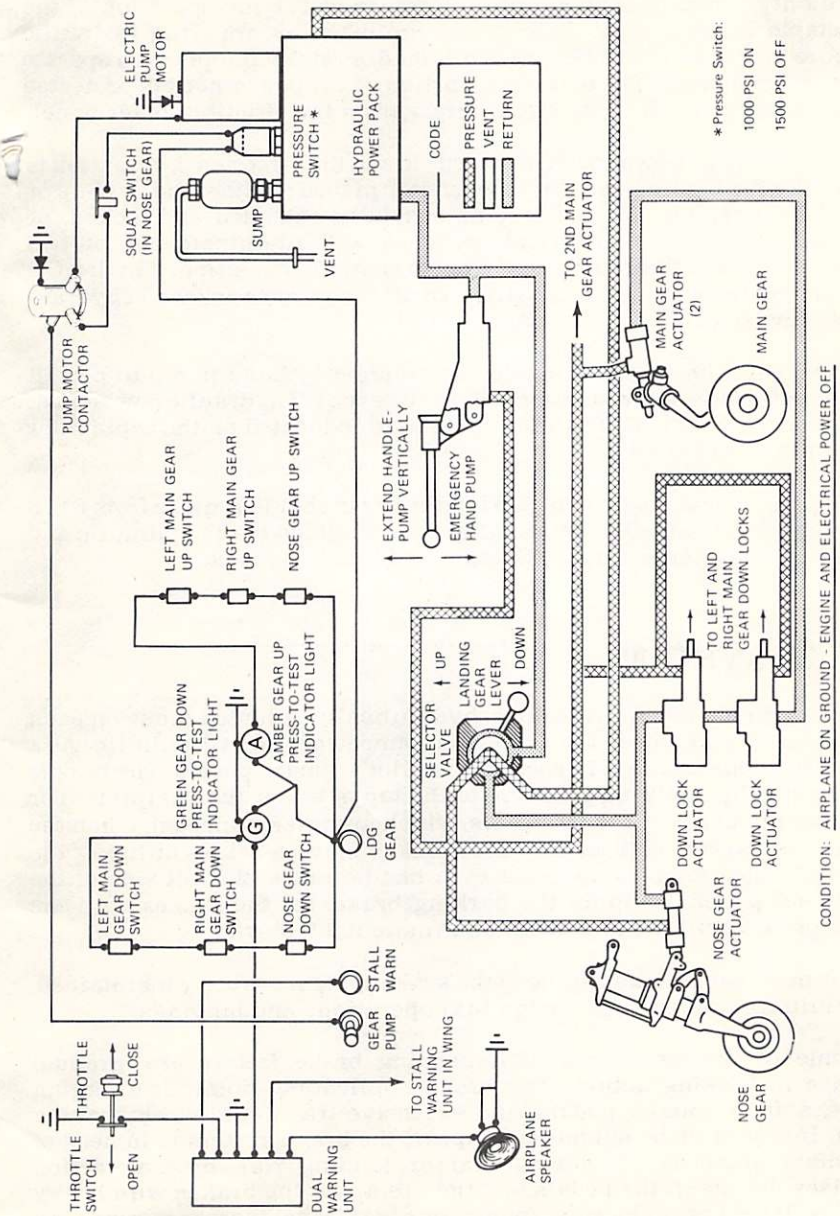


Figure 7-8. Hydraulic System

pack's only function is to supply hydraulic power for operation of the retractable landing gear. This is accomplished by applying hydraulic pressure to actuator cylinders which extend or retract the gear and operate the gear down locks. The electrical portion of the power pack is protected by a 35-amp "pull-off" type circuit breaker on the circuit breaker panel.

The hydraulic power pack is turned on, and the direction of actuation is selected by the landing gear lever when it is placed in either the gear-up or gear-down position. When the gear has fully extended and locked, or retracted, a series of electrical switches will illuminate one of two indicator lights on the instrument panel to show gear position. A hydraulic pressure switch will automatically turn off the power pack when hydraulic pressure reaches a preset value.

The hydraulic system includes an emergency hand pump to permit manual extension of the landing gear in the event of hydraulic power pack or electrical system failure. The hand pump is located on the cabin floor between the front seats.

During normal operations, the landing gear should require from 6 to 8 seconds to fully extend or retract. For malfunctions of the hydraulic and landing gear systems, refer to Section 3 of this handbook.

BRAKE SYSTEM

The airplane has a single-disc, hydraulically-actuated brake on each main landing gear wheel. Each brake is connected, by a hydraulic line, to a master cylinder attached to each of the pilot's rudder pedals. The brakes are operated by applying pressure to the top of either the left (pilot's) or right (copilot's) set of rudder pedals, which are interconnected. When the airplane is parked, both main wheel brakes may be set by utilizing the parking brake which is operated by a handle under the left side of the instrument panel. To apply the parking brake, set the brakes with the rudder pedals, pull the handle aft and rotate it 90° down.

For maximum brake life, keep the brake system properly maintained, and minimize brake usage during taxi operations and landings.

Some of the symptoms of impending brake failure are: gradual decrease in braking action after brake application, noisy or dragging brakes, soft or spongy pedals, and excessive travel and weak braking action. If any of these symptoms appear, the brake system is in need of immediate attention. If, during taxi or landing roll, braking action decreases, let up on the pedals and then re-apply the brakes with heavy pressure. If the brakes become spongy or pedal travel increases, pumping

the pedals should build braking pressure. If one brake becomes weak or fails, use the other brake sparingly while using opposite rudder, as required, to offset the good brake.

ELECTRICAL SYSTEM

The airplane is equipped with a 28-volt, direct-current electrical system (see figure 7-9). The system uses a battery located on the forward side, upper left portion, of the firewall, as the source of electrical energy and a belt-driven, 60-amp alternator (or 95-amp, if installed) to maintain the battery's state of charge. Power is supplied to most general electrical and all avionics circuits through the primary bus bar and the avionics bus bar, which are interconnected by an avionics power switch. The primary bus is on anytime the master switch is turned on, and is not affected by starter or external power usage. Both bus bars are on anytime the master and avionics power switches are turned on. The airplane may be equipped with a dual alternator electrical system. Details of this equipment are presented in Section 9, Supplements.

CAUTION

Prior to turning the master switch on or off, starting the engine or applying an external power source, the avionics power switch, labeled AVN PWR should be turned off to prevent any harmful transient voltage from damaging the avionics equipment.

MASTER SWITCH

The master switch is a split-rocker type switch labeled MASTER, and is ON in the up position and off in the down position. The right half of the switch, labeled BAT, controls electrical power to the airplane through the primary bus bar. The left half, labeled ALT, controls the alternator.

Normally, both sides of the master switch should be used simultaneously; however, the BAT side of the switch could be turned on separately to check equipment while on the ground. To check or use avionics equipment or radios while on the ground, the avionics power switch must also be turned on. The ALT side of the switch, when placed in the off position, removes the alternator from the electrical system. With this switch in the off position, the entire electrical load is placed on the battery. Continued operation with the alternator switch in the off position will reduce battery power low enough to open the battery contactor, remove power from the alternator field, and prevent alternator restart.

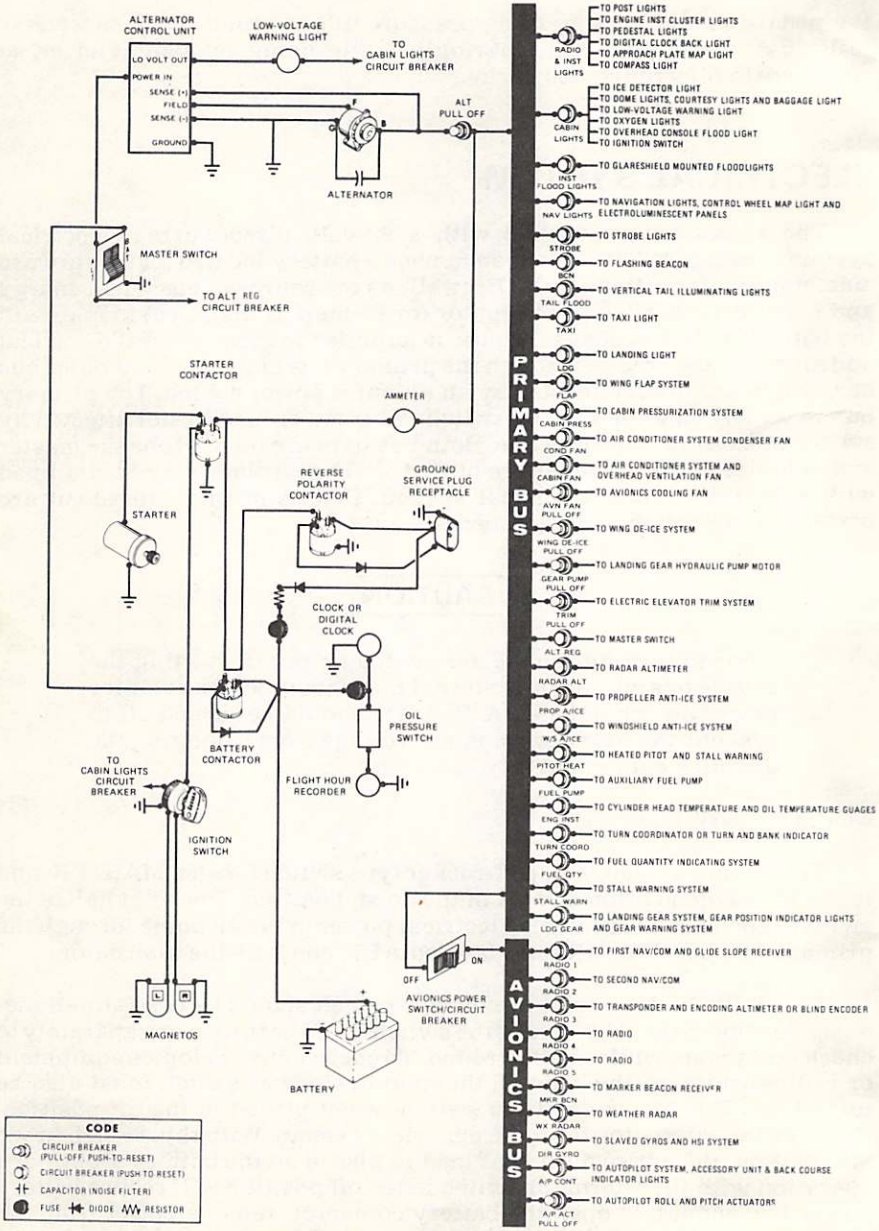


Figure 7-9. Electrical System

AVIONICS POWER SWITCH

Electrical power from the airplane primary bus to the avionics bus (see figure 7-9) is controlled by a rocker-type circuit breaker-switch labeled AVN PWR. The switch is located on the left sidewall circuit breaker panel and is ON in the forward position and OFF in the aft position. With the switch in the OFF position, no electrical power will be applied to the avionics equipment, except the Bonzer radar altimeter (if installed), regardless of the position of the master switch or the individual equipment switches. The avionics power switch also functions as a circuit breaker. If an electrical malfunction should occur and cause the circuit breaker to open, electrical power to the avionics equipment will be interrupted and the switch will automatically move to the OFF position. If this occurs, allow the circuit breaker approximately two minutes to cool before placing the switch in the ON position again. If the circuit breaker opens again, do not reset it. The avionics power switch should be placed in the OFF position prior to turning the master switch on or off, starting the engine, or applying an external power source, and may be utilized in place of the individual avionics equipment switches.

AMMETER

The ammeter, located on the upper right side of the instrument panel, indicates the amount of current, in amperes, from the alternator to the battery or from the battery to the airplane electrical system. When the engine is operating and the master switch is turned on, the ammeter indicates the charging rate applied to the battery. In the event the alternator is not functioning or the electrical load exceeds the output of the alternator, the ammeter indicates the battery discharge rate.

ALTERNATOR CONTROL UNIT AND LOW-VOLTAGE WARNING LIGHT

The airplane is equipped with a combination alternator regulator high-low voltage control unit mounted on the cabin side of the firewall and a red warning light, labeled LOW VOLTAGE, near the upper left corner of the instrument panel.

In the event an over-voltage condition occurs, the alternator control unit automatically removes alternator field current which shuts down the alternator. The battery will then supply system current as shown by a discharge rate on the ammeter. Under these conditions, depending on electrical system load, the low-voltage warning light will illuminate when system voltage drops below normal. The alternator control unit may be reset by turning the master switch off and back on again. If the warning light does not illuminate, normal alternator charging has resumed; however, if the light does illuminate again, a malfunction has occurred,

and the flight should be terminated as soon as practicable.

NOTE

> Illumination of the low-voltage light and ammeter discharge indications may occur during low RPM conditions with an electrical load on the system, such as during a low RPM taxi. Under these conditions, the light will go out at higher RPM. The master switch need not be recycled since an over-voltage condition has not occurred to de-activate the alternator system. Momentary illumination of the low-voltage warning light and/or ammeter needle deflection may also occur during startup of the landing gear system hydraulic pump motor.

The warning light may be tested by turning on the landing lights and momentarily turning off the ALT portion of the master switch while leaving the BAT portion turned on.

CIRCUIT BREAKERS AND FUSES

Most of the electrical circuits in the airplane are protected by "push-to-reset" type circuit breakers mounted on a single circuit breaker panel on the left cabin sidewall between the forward doorpost and the instrument panel. Six "pull off" type circuit breakers on this panel protect the alternator output landing gear system hydraulic pump motor, avionics cooling fan, wing and stabilizer de-ice system, electric elevator trim system, and the autopilot pitch and roll actuators. All of the avionics circuits are protected by circuit breakers grouped together in the lower portion of the circuit breaker panel and also by a rocker-type circuit breaker switch labeled AVN PWR. Fuses protect the cigar lighter circuit, the battery contactor closing circuit (when used with external power), and the digital clock and flight hour recorder circuits.

GROUND SERVICE PLUG RECEPTACLE

A ground service plug receptacle may be installed to permit the use of an external power source for cold weather starting and during lengthy maintenance work on the airplane electrical system. Details of the ground service plug receptacle are presented in Section 9, Supplements.

LIGHTING SYSTEMS

EXTERIOR LIGHTING

Conventional navigation lights are located on the wing tips and tail

stinger, and dual landing lights are installed in the cowl nose cap. Additional lighting is available and includes a strobe light on each wing tip, a flashing beacon on top of the vertical stabilizer, a courtesy light under the left wing, just outboard of the cabin door, and vertical tail illumination lights mounted on the top of each horizontal stabilizer. Details of the strobe light system are presented in Section 9, Supplements. The courtesy light is operated by the dome light switch in the overhead console. All exterior lights, except the courtesy light, are controlled by rocker type switches on the left switch and control panel. The switches are ON in the up position and off in the down position.

The flashing beacon should not be used when flying through clouds or overcast; the flashing light reflected from water droplets or particles in the atmosphere, particularly at night, can produce vertigo and loss of orientation.

INTERIOR LIGHTING

Instrument and control panel lighting is provided by flood and integral lighting, with electroluminescent lighting, post lighting and a white flood light also available. Rheostats and control knobs, located on the left switch and control panel, control the intensity of all lighting, except the white flood light. The following paragraphs describe the various lighting systems and their controls.

Switches and controls on the lower part of the instrument panel and the marker beacon/audio control panel may be lighted by electroluminescent panels which do not require light bulbs for illumination. To utilize this lighting, turn on the NAV light switch and adjust light intensity with the small (inner) control knob of the concentric control knobs labeled EL PANEL, ENG-RADIO.

Instrument panel flood lighting consists of six red flood lights on the underside of the glare shield, up to eight red post lights for the bottom and left and right instruments, and two red flood lights in the forward part of the overhead console (one, if an air conditioner is installed). All of these lights are utilized by adjusting light intensity with the large (outer) control knob of the concentric control knobs labeled POST, FLOOD.

The instrument panel may be equipped with post lights which are mounted at the edge of each instrument and provide direct lighting. To operate the post lights, adjust light intensity with the small (inner) control knob of the concentric control knobs labeled POST, FLOOD. To combine post and flood lighting, adjust flood light intensity with the large (outer) control knob.

The engine instrument cluster, radio equipment, digital clock, and

magnetic compass have integral lighting and operate independently of post or flood lighting. The light intensity of the instrument cluster, magnetic compass, digital clock, and radio equipment lighting is controlled by the large (outer) control knob of the concentric control knobs labeled EL PANEL, ENG-RADIO. If the airplane is equipped with avionics incorporating incandescent digital readouts, the ENG-RADIO (large outer) control knob controls the light intensity of the digital readouts. For daylight operation, the control knob should be rotated full counterclockwise to produce maximum light intensity for the digital readouts only. Clockwise rotation of the control knob will provide normal variable light intensity for nighttime operation.

If the airplane is equipped with a Cessna 400B Integrated Flight Control System, individual dimming control of both the white and the green Mode Selector panel lamps is provided by the concentric control knobs labeled IFCS, WHITE, GREEN. A push-to-test feature is incorporated into the small (inner) knob to test for proper green mode selector lamp operation.

The control pedestal has integral lights and one white post light for the fuel on-off valve. If an air conditioner is installed, the overhead console is illuminated by post lights. Pedestal and console light intensity is controlled by the large (outer) control knob of the concentric control knobs labeled POST, FLOOD.

Map lighting is provided by overhead console map lights and a glare shield mounted map light. The airplane may also be equipped with a control wheel map light. The overhead console map lights (not installed if an air conditioner is installed) operate in conjunction with instrument panel flood lighting and consist of two openings just aft of the red instrument panel flood lights. The map light openings have sliding covers controlled by small round knobs which uncover the openings when moved toward each other. The covers should be kept closed unless the map lights are required. A map light and toggle switch, mounted in front of the pilot on the underside of the glare shield, is used for illuminating approach plates or other charts when using a control wheel mounted approach plate holder. The switch is labeled MAP LIGHT ON, OFF and light intensity is controlled by the POST, FLOOD control knob. A map light mounted on the bottom of the pilot's control wheel illuminates the lower portion of the cabin in front of the pilot, and is used for checking maps and other flight data during night operation. The light is utilized by turning on the NAV LIGHTS switch, and adjusting light intensity with the rheostat control knob on the bottom of the control wheel.

The airplane is equipped with a dome light aft of the overhead console, and a baggage compartment light above the baggage area. The lights are operated by a slide-type switch adjacent to the dome light.

The most probable cause of a light failure is a burned out bulb; however, in the event any of the lighting systems fail to illuminate when turned on, check the appropriate circuit breaker. If the circuit breaker has opened (white button popped out), and there is no obvious indication of a short circuit (smoke or odor), turn off the light switch of the affected lights, reset the breaker, and turn the switch on again. If the breaker opens again, do not reset it.

PRESSURIZATION SYSTEM

The airplane is equipped with a cabin pressurization system (refer to figure 7-10) to permit flight at high altitude without the need for oxygen masks and to increase passenger comfort. The system functions by utilizing bleed air from the engine turbocharger system and is designed to provide maximum pressure differential of 3.35 PSI. This differential allows an approximate 12,100 foot cabin altitude when the airplane is operating at its placarded maximum operating altitude of 23,000 feet. The system operates in conjunction with the cabin heating, ventilating and defrosting system by using the same ducts and outlets in the cabin.

When the pressurized mode is selected by the pilot, heated bleed air from the engine turbocharger compressor is ducted through a sonic venturi (flow limiter) to a heat exchanger. At the heat exchanger, the bleed air is either cooled or heated further, depending on the position of the cabin heat control on the instrument panel. If cool pressurized air is desired (cabin heat control pushed full in), ram air from an airscoop on the lower left side of the engine cowling is directed through the heat exchanger, cooling the bleed air. If heated air is desired, the cabin heat control is pulled out. This action closes the heat exchanger to cool ram air flow, and instead, allows ram air heated by passing through a shroud over the engine left exhaust manifold to pass through the heat exchanger. Bleed air is, therefore, further heated by this hotter air as it flows through the heat exchanger. Refer to the Cabin Heating, Ventilating and Defrosting System paragraphs in this section for further operating details.

From the heat exchanger, the bleed air is ducted to a dump valve chamber on the engine side of the firewall. This chamber houses the pressurization system dump valve and the cabin pressure check valve. In order for pressurized bleed air to enter the cabin from the dump valve chamber, the dump valve control handle on the left side of the instrument panel must be pushed fully in, closing the valve. Otherwise, with the dump valve handle pulled out (dump valve open), bleed air will dump from the chamber to the inside of the engine compartment, flowing overboard through the cowl flaps. The other component of the dump valve chamber, the cabin pressure check valve, closes to prevent a sudden loss in cabin

SYSTEM SHOWN PRESSURIZED AND UNHEATED

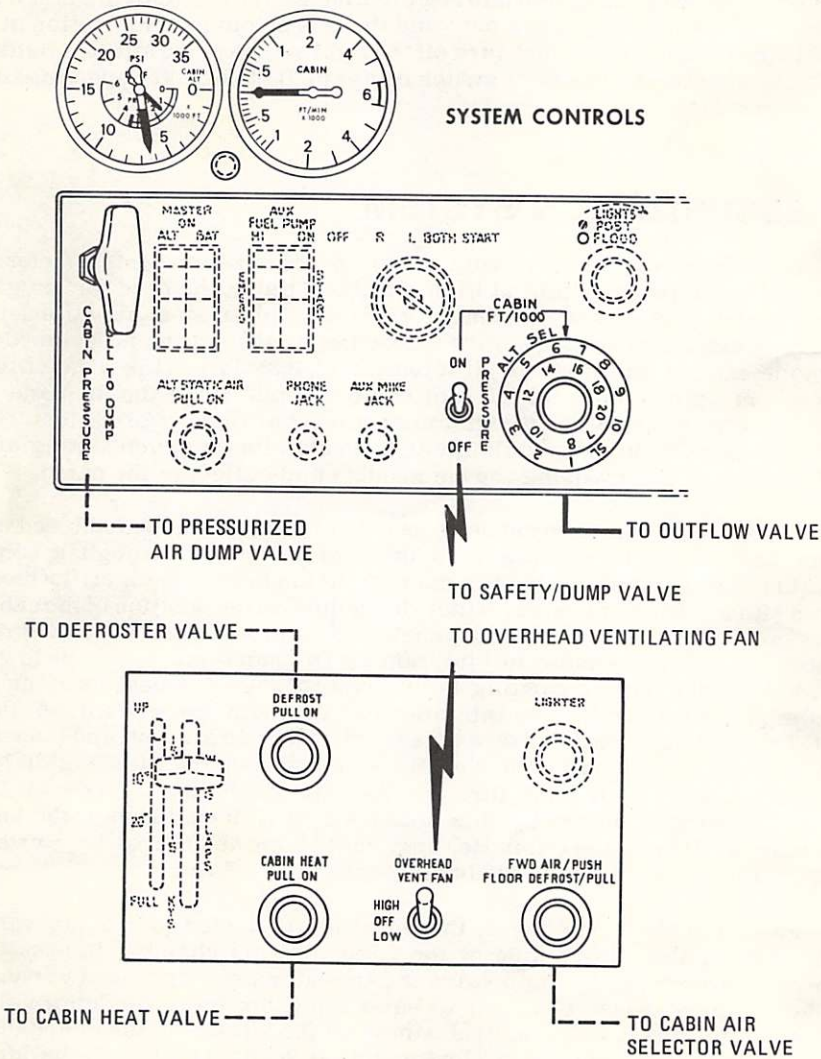


Figure 7-10. Pressurization System (Sheet 1 of 2)

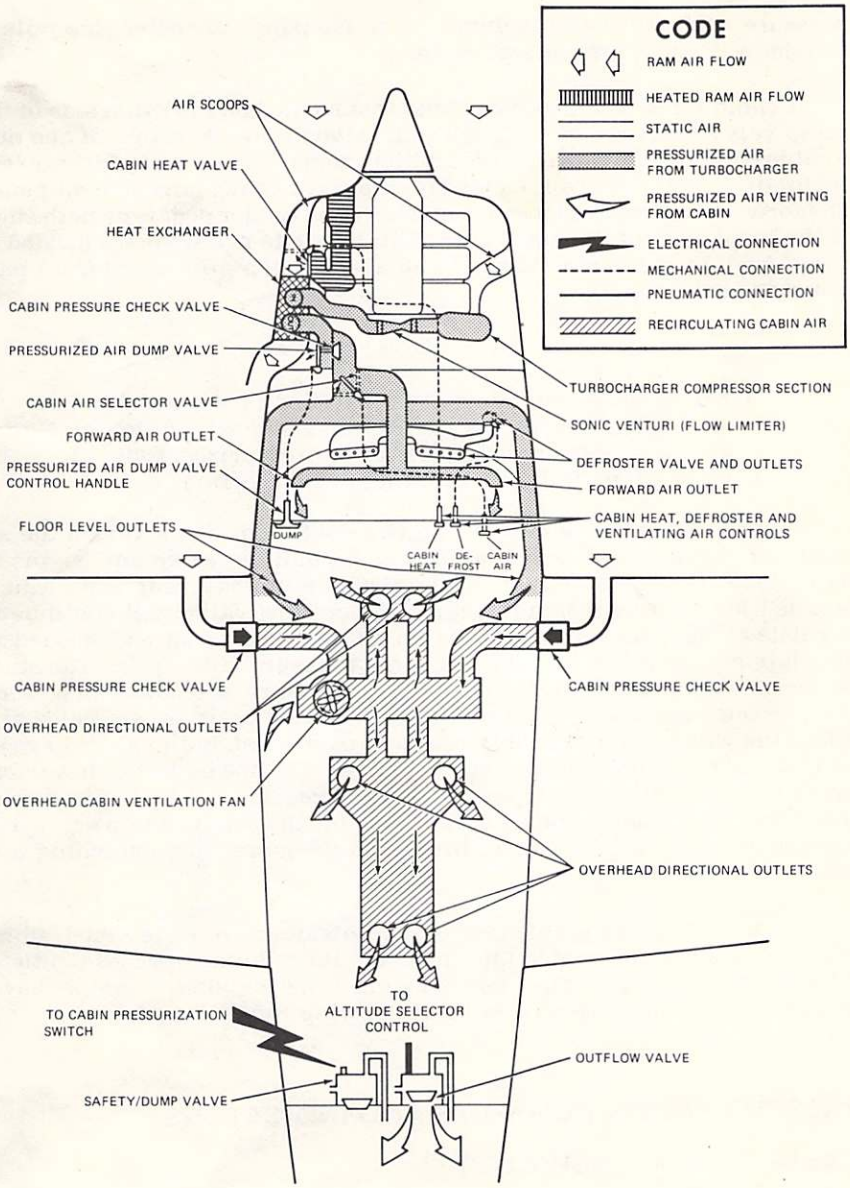


Figure 7-10. Pressurization System (Sheet 2 of 2)

pressure such as when the dump valve is opened or the engine fails to provide sufficient pressurized airflow.

A cabin air selector valve chamber is located on the cabin side of the dump valve chamber. The air selector valve allows selection of one or a combination of both pairs of fixed outlets depending on the position of the cabin air selector control, located on the right switch and control panel. The forward outlets are located just above the rudder pedals on both sides of the airplane and the other pair of outlets (one per side) are located at floor level on the cabin sidewalls adjacent to the pilot's and front passenger's positions.

NOTE

> A cabin air ventilation fan is installed in the overhead console to augment the distribution and circulation of cabin air, whether pressurized or unpressurized. Refer to the paragraph Cabin Ventilating in this section.

Cabin pressure is controlled by two dual-purpose valves on the aft cabin bulkhead. One valve functions as an outflow valve and begins to regulate airflow from the cabin as the airplane climbs through the altitude selected for pressurization to begin. The outflow valve will continue to regulate airflow until maximum cabin differential pressure is reached, at which point the valve will maintain this pressure differential. The other valve is a safety dump valve that contains an electric solenoid which, when the pressurization switch is placed in the OFF position, activates and opens the valve, dumping cabin pressure overboard. Both valves have as an integral part differential pressure valves. The one in the outflow valve prevents cabin differential pressure from exceeding 3.35 PSI. In the event this valve fails, another one incorporated into the safety/dump valve will actuate and prevent the cabin differential pressure from exceeding 3.50 PSI.

Anytime the cabin is not pressurized, outside air from the wing leading edge intakes may be vented into the cabin through the overhead outlets. When the cabin is pressurized however, the flow of outside air is stopped by two pressure check valves located at the wing roots.

PRESSURIZATION CONTROLS AND INDICATORS

CABIN ALTITUDE SELECTOR

A cabin altitude selector, labeled ALT SEL, is mounted on the lower left side of the instrument panel. This control is used in selecting the altitude at

which pressurization will begin and be maintained by controlling the outflow valve. The selector control knob has an outer scale marked SL, indicating sea level, and additional positions marked 1 through 10 indicating thousands of feet. An inner scale is included on this knob which reflects the airplane altitude in relation to the cabin altitude selected on the outer scale at maximum cabin pressure differential.

CABIN PRESSURIZATION SWITCH

A detent equipped switch adjacent to the cabin altitude selector turns the pressurization system on or off depending on its position. The detent requires that the switch be pulled out before repositioning, thus preventing inadvertent actuation. The two-position switch is labeled PRESSURE, and is ON in the up position and OFF in the down position. When the switch is placed in the ON position, electrical power to a solenoid in the safety-/dump valve is removed and the valve will close to permit pressurization. In the OFF position, electrical power is applied to the safety/dump valve solenoid, and the valve will open to prevent pressurization. Loss of electrical power, for any reason, will cause the safety/dump valve to close.

DUMP VALVE CONTROL HANDLE

A T-handle labeled, CABIN PRESSURE, PULL TO DUMP is located on the lower left side of the instrument panel, adjacent to the master switch. This handle is mechanically connected to the dump valve located in the dump valve chamber on the engine side of the firewall. When the handle is pulled, the dump valve opens and allows pressurization air to flow overboard. With the handle pushed in, pressurized air flows to the cabin through the air selector valve.

CABIN AIR SELECTOR CONTROL

A push-pull type control labeled, FWD AIR/PUSH, FLOOR DEFROST/PULL, permits incoming pressurized air to be directed to the two forward air outlets or to the two floor level outlets. With the cabin air selector control pushed fully in, pressurized air passes to the forward outlets. With the control pulled fully out, all airflow is diverted to the floor level outlets. A push-button type lock on the cabin air selector control allows positioning the control to any intermediate setting between full in and full out which results in pressurized airflow to both pairs of outlets.

CABIN RATE-OF-CLIMB INDICATOR

A cabin rate-of-climb indicator is located on the left side of the instrument panel above the dump valve control handle. The instrument is vented directly to the cabin and senses changes in pressure within the cabin to show cabin rate of climb or descent.

CABIN ALTITUDE/DIFFERENTIAL PRESSURE INDICATOR

This instrument, located adjacent to the cabin rate-of-climb indicator, shows both cabin altitude and differential pressure. It has two dials and two pointers. The outside dial indicates cabin altitude, and the inside dial indicates the pressure differential between cabin pressure and atmospheric pressure. The instrument is vented to the airplane cabin and the static air source.

CABIN ALTITUDE WARNING LIGHT

Anytime the cabin altitude exceeds 12,325 ± 175 feet, a barometric switch closes and illuminates a red press-to-test warning light labeled CABIN ALTITUDE. The light, located on the upper left corner of the instrument panel, indicates to the pilot that cabin altitude is too high and corrective action must be taken. Oxygen should be employed if available. If oxygen is not available, the airplane should be flown to a lower altitude. When the airplane descends to a cabin altitude of approximately 11,700 feet, the barometric switch opens and the warning light turns off.

PRESSURIZATION SYSTEM OPERATION

Operation of the pressurization system in the airplane is relatively simple, requiring few adjustments. Although the system is simple to operate, there are some things that should be remembered about flight with a pressurized cabin. These items, and the normal operating procedures for this system are covered in the following paragraphs. Pressurization emergencies are covered in Section 3 of this handbook.

Prior to starting the engine, set all of the pressurization controls for the anticipated flight. If the flight is to be made pressurized, place the dump valve control handle in the full in position, place the pressurization switch in the ON position, and set the cabin altitude selector approximately 1000 feet above the field elevation of the point of departure or the destination, whichever is the highest. No further adjustment of the system should be required.

NOTE

For improved cabin comfort on warm days, the cabin altitude selector should be set to 8000 to 10,000 feet, the dump valve control should not be pushed in, and the pressurization switch should not be turned on until approaching the set altitude. With the individual overhead outlets open, and the overhead ventilating fan set to HIGH, this procedure will allow maximum entry and circulation of the cooler ram air from the wing airscoops while

climbing or cruising through the warm lower altitudes. A similar procedure should be used for hot weather descents.

Because the cabin is sealed, supplying air into the cabin either by the wing ram air system (in flight) or by engine bleed air results in a small residual amount of cabin pressurization whenever the window is closed and the airplane altitude is lower than the altitude set on the cabin altitude selector. For example, cabin indicated altitude relative to airplane altitude will be up to 200 feet lower using ram air only, 350 feet lower using both ram and bleed air with the pressurization switch OFF (safety valve open), and 700 feet lower with the pressurization switch ON (safety valve closed).

NOTE

For improved cabin comfort during takeoff, the throttle should be advanced slowly or the pressurization switch should be turned off until after takeoff to avoid a sudden but natural slight increase in cabin pressure.

Pressurization will begin as the airplane climbs through the pressure altitude set on the cabin altitude selector. As the airplane passes through this altitude, the outflow valve will begin regulating air flow causing the cabin to start to pressurize. Pressurization can be verified by checking the cabin rate-of-climb indicator which should read zero at 700 feet above this altitude.

The cabin rate-of-climb indicator will continue to read zero until maximum cabin pressure differential (3.35 PSI) is attained. Maximum allowable cabin pressure differential is indicated by a red line on the cabin altitude/differential pressure indicator. Pressure differential is the difference between the pressure within the cabin and the atmospheric pressure at the altitude of the airplane.

NOTE

If the maximum cabin pressure differential stabilizes between 3.4 and 3.5 PSI, it may indicate that the safety valve is regulating cabin pressure instead of the outflow valve. In this case, monitor the pressure closely during the remainder of the flight, and upon landing, have the outflow valve and/or underseat and aft bulkhead air passages cleared. If pressure tends to go above 3.5 PSI, follow the overpressure emergency procedure checklist in Section3.

If the airplane continues to climb, exceeding the system's ability to maintain the selected cabin altitude, the cabin rate-of-climb indicator will begin to indicate a rate of climb which will be approximately 75% of that of

CABIN ALTITUDE vs. AIRPLANE ALTITUDE WITH 3.35 PSI DIFFERENTIAL	
AIRPLANE ALTITUDE IN FEET	CABIN ALTITUDE IN FEET
7000	Sea Level
8000	800
10,000	2400
12,000	4000
14,000	5500
16,000	7000
18,000	8500
20,000	10,000
23,000	12,100

Figure 7-11. Cabin Altitude Vs. Airplane Altitude

the airplane. Refer to figure 7-11 for the maximum altitude at which the airplane can be operated without causing the cabin altitude to climb above the altitude selected.

If, for any reason, the selected cabin altitude requires a change while enroute, consider carefully the current condition of the pressurization system and then act accordingly. If the system has not attained maximum pressure differential, any change in the cabin altitude selector (lower or higher) will result in a positive or negative cabin pressure change. If the cabin altitude selector is moved too rapidly, passenger discomfort can result. If a lower cabin altitude is desired, turn the cabin altitude selector to the desired setting VERY SLOWLY. Slow movement is important because the cabin pressure will respond almost instantly to cabin altitude selector movement. Should the pilot desire to increase the cabin altitude setting, again, turn the selector VERY SLOWLY.

When the pressurization system is operating at maximum pressure differential (3.35 PSI) with a cabin altitude equal to or above that selected, the cabin altitude selector may be repositioned as fast as desired to either a higher or lower altitude with no noticeable effect, provided the setting is not increased to a cabin altitude requiring less than the maximum allowable pressure differential to maintain, as described in the above paragraphs.

If the pilot climbs or descends, with the cabin pressure at maximum differential, the cabin altitude will follow at a proportional rate. In a descent, the cabin rate of climb will show a rate of descent about 75% of that

of the airplane until the airplane descends through the altitude at which the system can maintain the selected cabin altitude (the point at which cabin pressure differential begins to drop). At this point, cabin rate of climb will return to zero and the selected cabin altitude will be maintained.

Pressurization or re-pressurization of the airplane during flight should be accomplished with care to avoid passenger discomfort. If the airplane is at or below 10,000 feet, set the cabin altitude selector to 10,000 feet on the outer scale. After setting the selector, push the dump valve control handle full in and place the cabin pressurization switch in the ON position. Once the system is set up for pressurization, VERY SLOWLY turn the cabin altitude selector to the desired cabin altitude. By turning the selector slowly, no passenger discomfort is likely as the cabin pressure increases to the desired level.

If the airplane is above 10,000 feet, set the cabin altitude selector to 10,000 feet on the outer scale and check that the dump valve control handle is pulled **full out**. Place the pressurization switch in the ON position and VERY SLOWLY push the dump valve control handle full in. Very slow movement will prevent any sudden pressure change in the cabin. When the cabin rate-of-climb indicator has stabilized, VERY SLOWLY adjust the cabin altitude selector to the desired cabin altitude.

As the pilot approaches his destination, or if he is required to descend to a lower altitude enroute, care must be taken to maintain proper pressurization system operation. Descents should be made with sufficient power to maintain cabin pressure. Minimum power required to maintain cabin pressure is a function of many variables including cabin and pressurization air supply leakage, turbocharger efficiency, induction and exhaust systems condition, altitude, OAT, and the cabin differential pressure. Generally, the minimum manifold pressure required to maintain pressurization increases as RPM is decreased. A manifold pressure increase may also be required if a lower cabin altitude is selected (on partial differential pressure), if descending to a lower altitude with maximum differential cabin pressure, or if descending into warm atmospheric OAT's compared to cooler OAT's in previous cruise. For example, at 2500 and 2200 RPM respectively, minimum manifold pressure at maximum cabin differential pressure at 17,000 feet will be approximately 18 in. and 22 in. Hg. By comparison, descending to 10,000 feet at maximum differential (selected cabin pressure 2500 feet or lower) would require increasing the manifold pressure to 24 or 25 in. Hg, depending upon RPM. However, with the cabin altitude selector set to a 5000 foot cabin altitude, the minimum manifold pressure required would be 21 to 23 in. Hg, depending on RPM. With a 7500 foot cabin altitude selected, the minimum manifold pressure required would reduce to 18 to 21 in. Hg. Prior to landing, the differential pressure indicator should be checked to assure the airplane is depressurized. The airplane is not approved for landing while

pressurized.

After completing a trip, it is good practice to leave the pressurization controls in the positions expected to be appropriate for the next flight.

CABIN HEATING, VENTILATING AND DEFROSTING SYSTEM

Air for cabin heating, ventilating and defrosting is supplied by the compressor section of the turbocharger. This heated air (bleed air) is first passed through the heat exchanger where it is either cooled or further heated and then is directed to the cabin through the same ducts and outlets used by the cabin pressurization system (refer to figure 7-10). In addition, fresh air for cabin ventilation can enter through the overhead outlet system from airscoops in both wing leading edges anytime the cabin is not pressurized. The cabin heating, ventilating and defrosting system controls are located on a comfort control panel on the right side of the avionics equipment at the lower edge of the instrument panel. All the controls are the push-pull type and feature a push-button type lock to allow intermediate settings. The following paragraphs describe these controls and their operation. An air conditioning system may be installed in the airplane. Details of this system are presented in Section 9, Supplements.

CABIN HEATING

A full range of control of heated cabin air temperature is available when utilizing the cabin heat control. The control, labeled CABIN HEAT, PULL ON, is connected mechanically to a valve which admits ram air to the pressurization system heat exchanger. The valve allows ram air at outside air temperature (OAT) or heated ram air from a shroud surrounding the exhaust manifold to pass through the heat exchanger. Since the heat exchanger governs the temperature of bleed air fed to the cabin, the setting of the cabin heat control will determine the cabin air temperature. With the cabin heat control pushed fully in, the air entering the cabin will not be heated. However, pulling the control fully out results in maximum system heating of incoming cabin air.

Cabin heat can be selected with the pressurization system operating to pressurize the cabin or with the cabin unpressurized. To obtain heated air flow, first ensure the pressurized air dump valve control is pushed fully in, then select the desired cabin air temperature by pulling out the cabin heat control as required.

CABIN VENTILATING

Ventilating airflow into the airplane cabin while in unpressurized flight is available from any of three sources, simultaneously or individual-

ly: bleed air from the engine turbocharger compressor, ram air from an airscoop in each wing leading edge, or outside air from the openable window in the cabin entry door. When the airplane is on the ground, taxiing or parked, the cabin emergency exit door can be opened to provide an additional ventilating air source.

NOTE

The airplane is not approved for flight with the emergency exit door open.

Distribution of ventilating airflow from the wing ram airscoops or from recirculation of cabin air is provided by an overhead ventilating system in the overhead console. This system consists of six individually controllable outlets and a two-speed recirculating fan exhausting through the outlets into the cabin. Any time it is desired to increase cabin ventilating airflow, whether in pressurized or unpressurized flight, select one of the two positions of the ventilating fan switch, labeled OVERHEAD VENT FAN on the comfort control panel (figure 7-10). Ventilating airflow is then individually adjusted at each seat position using the overhead outlets.

To obtain maximum cabin ventilation during unpressurized flight, turn the pressurization switch off, open the individual overhead outlets, turn the overhead ventilating fan to HIGH, push the dump valve control handle full in, and adjust the cabin air selector control to obtain desired airflow distribution from the forward or floor outlets. In hot weather, use of only the ram air from the wing airscoops (dump valve pulled out) will provide the coolest cabin temperatures.

Whenever bleed air is being used for cabin ventilation (dump valve control pushed full in) with the cabin unpressurized (pressurization switch OFF), and the airplane altitude is higher than set on the cabin altitude selector, any loss of electrical power, such as turning off the master switch, will cause the cabin to pressurize. This occurs because the safety/dump valve requires electrical power to be held open. An electrical power loss for any reason allows the valve to close, causing the cabin to pressurize. Therefore, if the master switch must be turned off in flight, it is recommended that first the dump valve control be pulled full out or the cabin altitude selector be adjusted to a value greater than the airplane altitude. Under these circumstances, if ventilating air to the cabin is desired, ram air from the wing airscoops to the overhead outlets should be used.

WINDSHIELD DEFROSTING AND DEFOGGING

The airplane incorporates provisions for windshield defrosting or

defogging. Components include a valve, ducting, windshield outlets and a separate defroster control, labeled DEFROST, PULL ON. The defroster control is located directly above the cabin heat control and is mechanically connected to the defroster valve. This valve is located in the duct leading to the right side floor level outlet in the cabin. When the defrost control is pulled out, the valve opens to admit bleed air to a series of fixed outlets at the base of the windshield. The volume of airflow supplied to the windshield is, thus, controlled by the setting of the defrost control. However, the temperature of the air reaching the windshield will be the same as that supplied to the floor level outlets and this temperature is dependent on the setting of the cabin heat control. Therefore, to obtain heated airflow to the windshield for defrosting, the cabin heat control must be pulled to the on position.

PITOT-STATIC SYSTEM AND INSTRUMENTS

The pitot-static system supplies ram air pressure to the airspeed indicator and static pressure to the airspeed indicator, vertical speed indicator, airplane altimeter and the cabin altimeter/differential pressure indicator. The system is composed of a pitot tube mounted on the lower surface of the left wing, two external static ports, one on each side of the fuselage below the rear corners of the aft side windows, an alternate static source valve located on the lower left corner of the instrument panel, and the associated plumbing necessary to connect the instruments and sources.

The airplane may also be equipped with a pitot heat system. The system consists of a heating element in the pitot tube, a rocker-type switch labeled PITOT HEAT on the lower left side of the instrument panel, a 10-amp circuit breaker on the left sidewall circuit breaker panel, and associated wiring. When the pitot heat switch is turned on, the element in the pitot tube is heated electrically to maintain proper operation in possible icing conditions. Pitot heat should be used only as required.

A static pressure alternate source valve is installed on the lower left side of the instrument panel and can be used if the external static source is malfunctioning. This valve supplies static pressure from inside the baggage compartment.

If erroneous instrument readings are suspected due to water or ice in the pressure lines going to the standard external static pressure source, the alternate static source valve should be pulled on.

Pressures within the baggage compartment will vary. Refer to Sections 3 and 5 for the effect of varying pressures on airspeed and altimeter readings.

AIRSPEED INDICATOR

The airspeed indicator is calibrated in knots and miles per hour. Limitation and range markings (in KIAS) include the white arc (58 to 115 knots), green arc (73 to 167 knots), yellow arc (167 to 200 knots), and a red line (200 knots).

If a true airspeed indicator is installed, it is equipped with a rotatable ring which works in conjunction with the airspeed indicator dial in a manner similar to the operation of a flight computer. To operate the indicator, first rotate the ring until **pressure** altitude is aligned with outside air temperature in degrees Fahrenheit. Pressure altitude should not be confused with indicated altitude. To obtain pressure altitude, momentarily set the barometric scale on the altimeter to 29.92 and read pressure altitude on the altimeter. Be sure to return the altimeter barometric scale to the original barometric setting after pressure altitude has been obtained. Having set the ring to correct for altitude and temperature, read the true airspeed shown on the rotatable ring by the indicator pointer. For best accuracy, the indicated airspeed should be corrected to calibrated airspeed by referring to the Airspeed Calibration chart in Section 5. Knowing the calibrated airspeed, read true airspeed on the ring opposite the calibrated airspeed.

VERTICAL SPEED INDICATOR

The vertical speed indicator depicts airplane rate of climb or descent in feet per minute. The pointer is actuated by atmospheric pressure changes resulting from changes of altitude as supplied by the static source.

ALTIMETER

Airplane altitude is depicted by a barometric type altimeter. A knob near the lower left portion of the indicator provides adjustment of the instrument's barometric scale to the current altimeter setting.

VACUUM SYSTEMS AND INSTRUMENTS

Two engine-driven vacuum systems are available and provide the suction necessary to operate the attitude indicator and directional indicator. One system (see figure 7-12) consists of a single vacuum pump on the engine, a vacuum relief valve and vacuum system air filter on the aft side of the firewall below the instrument panel, vacuum-operated instruments, and a suction gage on the left side of the instrument panel. The other vacuum system (see figure 7-13) offers a dual pump installation on the rear of the engine, two vacuum relief valves, a system air filter, a check valve

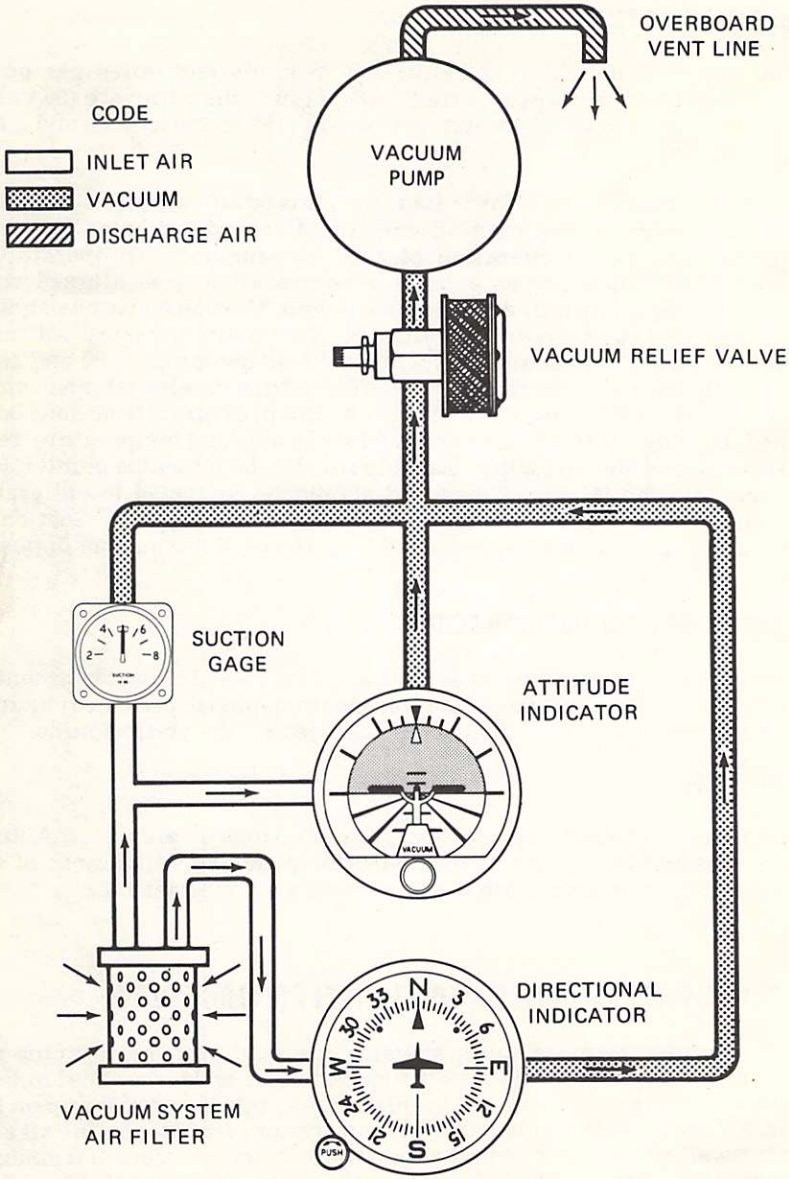


Figure 7-12. Single-Pump Vacuum System

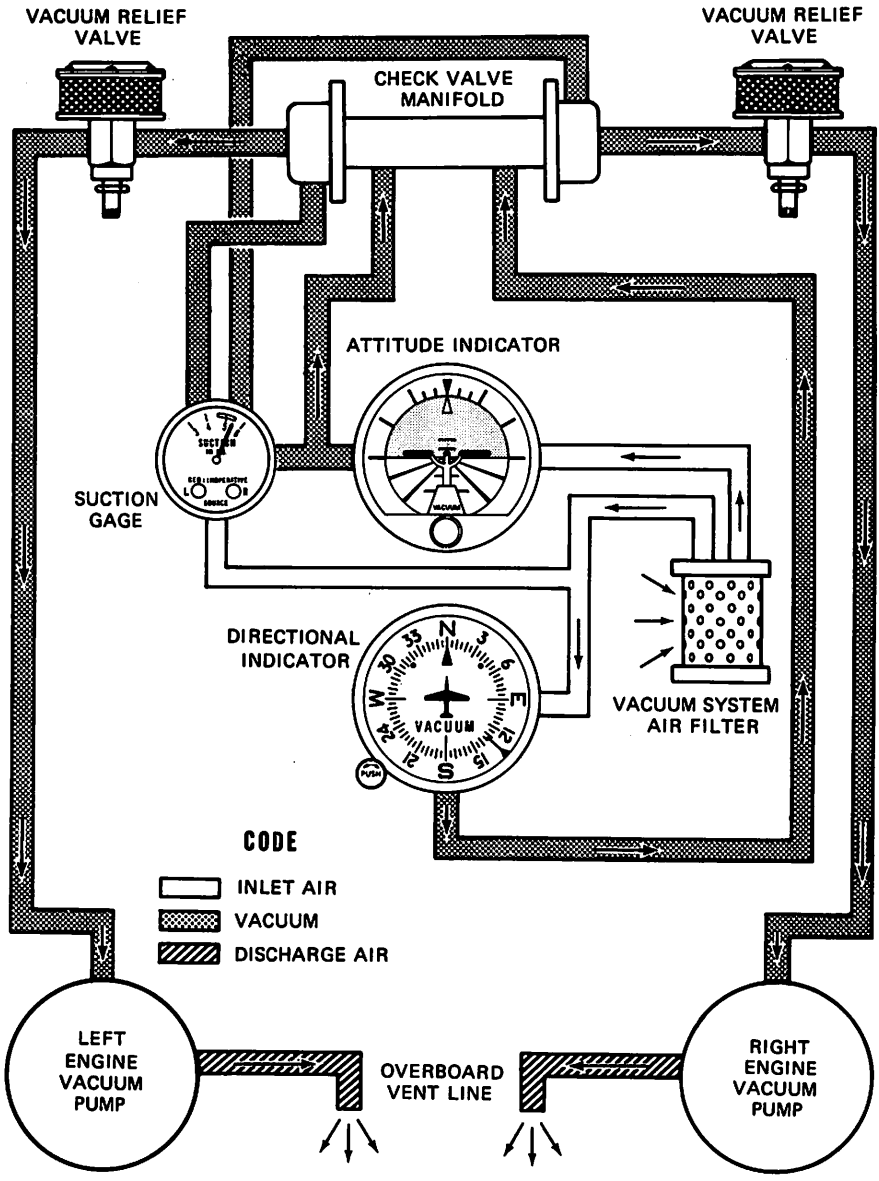


Figure 7-13. Dual-Pump Vacuum System

manifold, vacuum-operated instruments, and a suction gage, equipped with dual warning indicators labeled L and R, on the left side of the instrument panel.

ATTITUDE INDICATOR

An attitude indicator is available and gives a visual indication of flight attitude. Bank attitude is presented by a pointer at the top of the indicator relative to the bank scale which has index marks at 10°, 20°, 30°, 60°, and 90° either side of the center mark. Pitch and roll attitudes are presented by a miniature airplane superimposed over a symbolic horizon area divided into two sections by a white horizon bar. The upper "blue sky" area and the lower "ground" area have arbitrary pitch reference lines useful for pitch attitude control. A knob at the bottom of the instrument is provided for in-flight adjustment of the miniature airplane to the horizon bar for a more accurate flight attitude indication.

DIRECTIONAL INDICATOR

A directional indicator is available and displays airplane heading on a compass card in relation to a fixed simulated airplane image and index. The directional indicator will precess slightly over a period of time. Therefore, the compass card should be set in accordance with the magnetic compass just prior to takeoff, and occasionally re-adjusted on extended flights. A knob on the lower left edge of the instrument is used to adjust the compass card to correct for any precession.

SUCTION GAGE

A suction gage is located on the upper left side of the instrument panel when the airplane is equipped with a vacuum system. Suction available for operation of the attitude indicator and directional indicator is shown by this gage, which is calibrated in inches of mercury. The desired suction range is 4.6 to 5.4 inches of mercury. A suction reading out of this range may indicate a system malfunction or improper adjustment, and in this case, the indicators should not be considered reliable.

If the airplane is equipped with a dual vacuum pump system, the suction gage incorporates two red warning buttons, marked L and R, which extend visibly in the event either or both sources fail.

STALL WARNING SYSTEM

The airplane is equipped with a vane-type stall warning unit in the leading edge of the left wing. The unit is electrically connected to a dual

warning unit located above the cabin emergency exit door and behind the headliner. The vane in the wing senses the change in airflow over the wing, and operates the dual warning unit, which produces a continuous tone over the airplane speaker between 5 and 10 knots above the stall in all configurations.

If the airplane has a heated stall warning system, the vane-type unit in the wing leading edge is equipped with a heating element. The heated part of the system is operated by the PITOT HEAT switch, and is protected by the PITOT HEAT circuit breaker.

The stall warning system should be checked during the preflight inspection by momentarily turning on the master switch and actuating the vane in the wing. The system is operational if a continuous tone is heard on the airplane speaker as the vane is pushed upward.

AVIONICS SUPPORT EQUIPMENT

If the airplane is equipped with avionics, various avionics support equipment may also be installed. Equipment available includes two avionics cooling fans, microphone-headset installations and control surface static dischargers. The following paragraphs discuss these items. Description and operation of radio equipment is covered in Section 9 of this handbook.

AVIONICS COOLING FAN

An avionics cooling fan is provided anytime the airplane is equipped with a radio to supply cooling for prolonged avionic equipment life. The fan operates whenever the master switch is turned on. If a fan malfunction occurs, the fan can be shut off using the "pull-off" type circuit breaker labeled AVN FAN.

MICROPHONE-HEADSET INSTALLATIONS

Three types of microphone-headset installations are offered. The standard system provided with avionics equipment includes a hand-held microphone and separate headset. The keying switch for this microphone is on the microphone. Two optional microphone-headset installations are also available; these feature a single-unit microphone-headset combination which permits the pilot or front passenger to conduct radio communications without interrupting other control operations to handle a hand-held microphone. One microphone-headset combination is a lightweight type without a padded headset and the other version has a padded headset. The microphone-headset combinations utilize a remote keying switch

located on the left grip of the pilot's control wheel and, if an optional intercom system is installed, a second switch on the right grip of the front passenger's control wheel. The microphone and headset jacks are located on the lower left and right sides of the instrument panel. Audio to all three headsets is controlled by the individual audio selector switches and adjusted for volume level by using the selected receiver volume controls.

NOTE

When transmitting, with the hand-held microphone, the pilot should key the microphone, place the microphone as close as possible to the lips and speak directly into it.

STATIC DISCHARGERS

If frequent IFR flights are planned, installation of wick-type static dischargers is recommended to improve radio communications during flight through dust or various forms of precipitation (rain, snow or ice crystals). Under these conditions, the build-up and discharge of static electricity from the trailing edges of the wings, rudder, elevator, propeller tips, and radio antennas can result in loss of usable radio signals on all communications and navigation radio equipment. Usually the ADF is first to be affected and VHF communication equipment is the last to be affected.

Installation of static dischargers reduces interference from precipitation static, but it is possible to encounter severe precipitation static conditions which might cause the loss of radio signals, even with static dischargers installed. Whenever possible, avoid known severe precipitation areas to prevent loss of dependable radio signals. If avoidance is impractical, minimize airspeed and anticipate temporary loss of radio signals while in these areas.