41 MASTER SWITCH
Switch provided to turn on the power supply. The switch has a two-part design. The left switch actuates a relay to connect the battery to the power distribution system (battery master switch). The right switch directly provides excitating current to the alternator (alternator master switch). It is not possible to turn on the alternator without the battery or to turn off the battery without the alternator.

42 AIRCRAFT CLOCK (if installed)
Electrically operated aircraft clock, either with or without stop-watch feature, as desired.

43 AVIONCS MASTER SWITCH (if installed)
Switch provided to turn on/off the power supply of the avionics. Throw the switch up to turn it ON.

44 EMERGENCY AVIONICS SWITCH (if installed)
If the avioncs master switch has failed the avionics may be provided with current by placing this switch in the ON position. Throw the switch up to turn it ON.

45 ANNUNCIATOR LIGHT DIMMING SWITCH
Switch provided to dim the annunciator lights including landing gear position lights but excluding the red warning lights for oil pressure, alternator, stall warning and landing gear. Depress the switch to operate it. The switch remains in the depressed position to dim the a.m. annunciator lights (night flying). Depress the switch a second time to reset the switch and to ensure adequate illumination for daylight operation.

46 ANNUNCIATOR LIGHT TEST KEY
Key provided to check all annunciator lights including landing gear position lights and the stall warning horn for proper operation. Depress the key and keep it depressed. The annunciator lights will light up (undimmed) and the stall warning horn will sound. Control lights which do not light up are considered to have failed.

47 ALTERNATOR WARNING LIGHT
will illuminate when the alternator fails. Colour: red.

48 INSTRUMENTS DIMMING SWITCH
Switch provided to turn on and dim instrument lighting. Clockwise rotation ⇨ Increased brightness; counterclockwise rotation beyond the detent ⇨ OFF.

49 INSTRUMENT PANEL DIMMING SWITCH
Switch provided to turn on and dim instrument panel lighting, located below the padded glareshield. Clockwise rotation ⇨ increased brightness; counterclockwise rotation beyond the detent ⇨ OFF.

50 AVIONICS DIMMING SWITCH
Switch provided to turn on and dim the avionics control units Clockweise rotation ⇨ increased brightness; counterclockwise rotation beyond the detent ⇨ OFF.

51 PITOT HEATING ANNUNCIATOR LIGHT (if installed)
illuminates as long as the pitot heating is operated. Colour: white.

52 AVIONICS STACK
The avionics stack provides sufficient space for avionics installation. The equipment installed has been described in Section IX.

53 CIRCUIT BREAKER PANEL
The circuit breaker panel includes the push/pull circuit breakers only.

a POWER PLANT BUS CIRCUIT BREAKER PANEL
This panel contains the circuit breakers of those loads being supplied by the power plant bus (see also List of Circuit Breakers on page 7-75).

b SYSTEMS BUS CIRCUIT BREAKER PANEL
This panel contains the circuit breakers of those loads being supplied by the systems bus (see also List of Circuit Breakers on page 7-75).

c AVIONICS BUS CIRCUIT BREAKER PANEL

This panel contains the circuit breakers of those loads being supplied by the avionics bus (see also List of Circuit Breakers on page 7-75).

54 FAN SWITCH

Rocker-type switch provided to switch on the cabin heating and ventilating fan motor (two speeds).

55 FRESH AIR CONTROL

Lever provided to regulate the quantity of fresh air to the cabin.

56 WARM AIR CONTROL

Lever provided to regulate the quantity of warm air to the cabin.

57 AIR DISTRIBUTION CONTROL

Lever provided to regulate the air distribution to the cabin.

58 PITOT HEAT SWITCH BREAKER (if installed)

The electrical pitot heating is switched on by pressing the upper part of the rocker. In case of excessive current (short circuit), the switch breaker automatically returns to the OFF position.

59 ASH TRAY

60 HANDHELD MICROPHONE (if installed)

61 PARKING BRAKE

Control lever provided to set the parking brake.

LANDING GEAR AND HYDRAULIC SYSTEM

LANDING GEAR

The airplane is equipped with a retractable tricycle landing gear provided with nosewheel steering and self-adjusting disk brakes for each main wheel. The landing gear retraction system is electrically controlled and hydraulically actuated. Shock absorption is provided on each gear by an air-over-oil shock strut. The shock absorber element of the nose gear is located in the straight strut while the shock absorber elements of the main gear are separate elements located between main gear strut and trailing link.

The nosewheel steering system is operated by the rudder pedals and actuated by control rods.

Rudder control and nosewheel steering system are disconnected automatically while retracting the landing gear. The nose gear is reset to neutral by means of a centering device. The landing gear may be extended at any pedal position. The position of the nose wheel is automatically matched to the respective pedal position.

LANDING GEAR RETRACTION AND EXTENSION SYSTEM

Fig. 7-4 shows the hydraulic system schematic of the landing gear retraction and extension system. The letters in square brackets, appearing in the following chapters, refer to Fig. 7-4.

Pressure Pipe
Return Pipe
Suction Pipe
(the shown condition is "Retraction")

N : Hydraulic Cylinder Nose
L/H : Hydraulic Cylinder Left Hand
R/H : Hydraulic Cylinder Right Hand
E : Emergency Extension Valve
PS : Pressure Switch
C : Change-over Valve
H : Hydraulic Unlock
R : Relief Valve
PC : Pressure Control Valve
S : Suction Valve
F : Filter
P : Pump
M : DC Motor
SV : Safety Valve
T : Hydraulic Reservoir

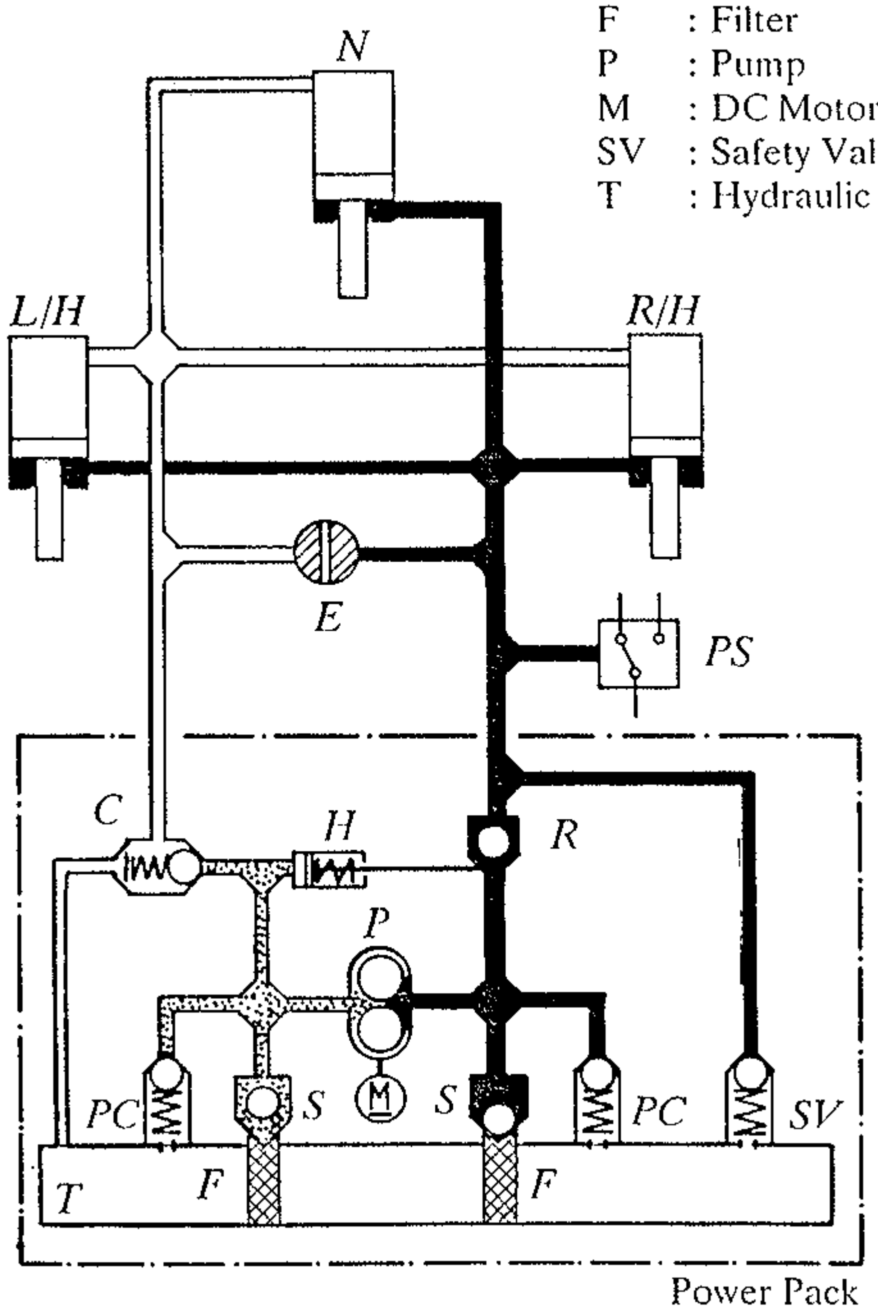


Figure 7-4: Hydraulic System Schematic of the Landing Gear Retraction and Extension System

MODE OF OPERATION

To operate the landing gear, slightly pull out the landing gear switch located on the left side of the instrument panel to pass the latch. To retract the landing gear, throw the switch up to turn on the motor [M] of the electrically operated hydraulic system and to retract the hydraulic cylinders [N, L/H, R/H] located on the landing gears. The hydraulic cylinders first release the locks and then retract the landing gears into their wheel wells by means of the drag and side struts. When reaching the upstop bumpers, the position indicator limit switches will be activated. Hydraulic pressure continues to rise until a pressure switch [PS] cuts the hydraulic system off. A relief valve [R] incorporated in the hydraulic system provides for maintaining pressure. If the holding pressure drops below a set value the pressure switch will re-engergize the hydraulic system for a short period of time. At normal maintenance condition, cut-in will take place approximately every 10 minutes and is audible to the pilot (max. cut-in time 5 seconds).

If the landing gear switch is thrown down to extend the landing gear hydraulic flow is to the opposite direction and the relief valve [R] is opened. Hydraulic fluid is now entering the lines opposite to the retraction direction and cause the landing gears to extend until the drag and side struts have reached a straight position. Downlocks hold the landing gear in the extended position and actuate the extension limit switches to cut off the hydraulic system and to turn on the position indicator lights.

Note

It is recommended to not simultaneously operate the landing gear retraction and extension system and the wing flap system to keep the load of the electrical power supply as low as possible.

The single-piece main gear doors are linked to their respective landing gear by control rods, retracting and extending with each landing gear. The main wheels are not covered when retracted.

WARNING

Before operating the landing gear emergency extension lever, pull the hydraulic system circuit breaker in any case. Landing gear emergency extension supported by a working hydraulic system might lead to a high extension speed and thus to landing gear and airframe damage.

BRAKE SYSTEM

HYDRAULIC BRAKE SYSTEM

Fig. 7-5 shows the brake system schematic.

The two main wheels are provided with hydraulically actuated, self-adjusting single disk brakes using single recoil piston wheel cylinders which are actuated by independent master cylinders being attached to the rudder pedals. The brakes can be operated by pushing the upper part of the pedals with the tiptoes. Pushing the lower part of the pedals serves for rudder control and nosewheel steering.

The master cylinders attached to the copilot's rudder pedals are supplied by a joint hydraulic reservoir located under the engine cowling and mounted to the fire wall. They deliver hydraulic pressure through the master cylinders of the pilot's rudder pedals. If the brakes are operated simultaneously from both the pilot's and the copilot's pedals the higher of the two pressures generated is applied to the brakes.

PARKING BRAKE SYSTEM

A parking brake valve is located in the lines between the master cylinders and the wheel cylinders to hold pressure on the brake assemblies until released. The parking brake is set by applying the brake and then pulling the parking brake button. To release the parking brake, push the parking brake button in.

The parking brake is provided with a latch to avoid accidental parking brake operation. The latch is released by pressing a button incorporated in the parking brake button.

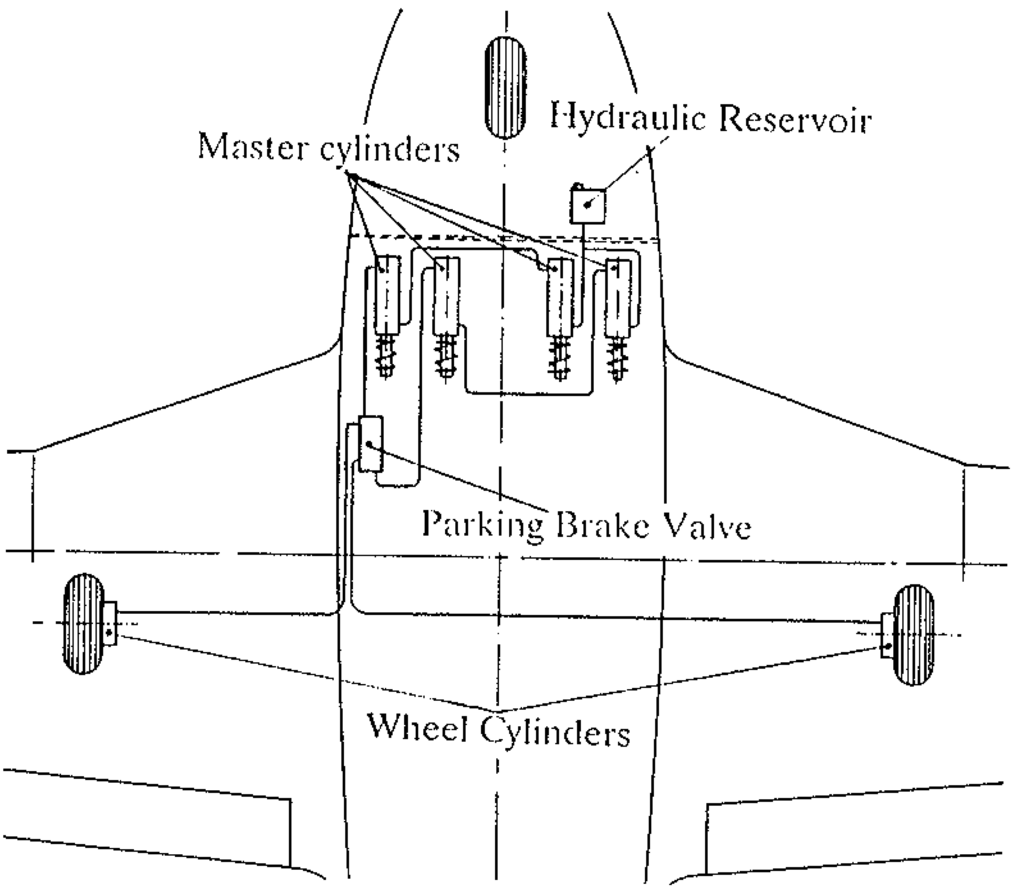


Figure 7-5: Brake System Schematic

The nose gear doors have a three-piece design. The front door is linked to the nose gear by two control rods and moving with the nose gear. A linkage connects the two side doors to the nose gear and shuts the doors shortly before the upstop bumpers will be reached. In the extended position, the doors are secured by means of a spring-loaded linkage.

LANDING GEAR POSITION AND WARNING LIGHTS

Three green annunciator lights have been installed above the landing gear switch, in accordance with the triangular arrangement of the landing gear. They will illuminate only if the respective landing gear is fully extended and locked.

The red warning light located beneath the annunciator lights will illuminate while the landing gears are retracted or extended. The warning light is turned on by the first of the gears leaving its stop and turned off by the last of gears reaching its stop.

A test key located in the annunciator panel is provided to check the annunciator lights. In addition, the green annunciator lights can be dimmed by means of the dimming switch in the annunciator panel.

WEIGHT-ON-WHEEL SAFETY SWITCH

During ground operation, accidental gear retraction is prevented by a weight-on-wheel safety switch located on each of the main landing gears. The electrically actuated gear retraction control is activated only after the two gears have lift off the ground. Now the landing gear switch may be moved to the UP position.

If the landing gear switch is placed unintentionally in the UP position while the airplane is still on the ground (additionally impeded by a latch) an acoustic warning will be released in case the master switch is in the ON position.

LANDING GEAR WARNING HORN

Apart from the acoustic warning during ground operation, there is a warning through the same warning horn if the landing gear is not extended during flight. The warning horn is activated if:

1) the throttle is retarded to idle. The ALARM pushbutton located on the control stick is provided to deactivate the warning horn. Increased power will stop the warning and cancel the deactivation function.

2) the wing flaps are lowered past the takeoff position (15°). This warning function cannot be deactivated.

LANDING GEAR EMERGENCY EXTENSION VALVE

If electrical control and/or hydraulic actuation of the landing gear system have failed landing gear emergency extension is caused by a valve [E] located between the retraction and extension lines. The holding pressure required for keeping the landing gear in the retracted position is released thus causing the hydraulic fluid to flow to the extension side of the hydraulic cylinder during gear extension.

Landing gear emergency extension is supported by the weight of the landing gears and supporting gas springs. As the nose gear has to bear airstream and propeller slipstream during extension it may be necessary to reduce airspeed and power (see Section III). If electrical power supply is working the functions of the annunciator and warning lights remain unchanged.

Before operating the emergency extension lever located on the left side of the center pedestal (lever to be pulled back/up), pull the hydraulic system circuit breaker in any case because landing gear emergency extension supported by a working hydraulic system might lead to a high extension speed and thus to landing gear and airframe damage.

CABIN FEATURES

DOORS

The airplane is provided with two lockable, outward opening cabin doors, one on each side. In the opened position, the doors are secured by a gas spring. The door locking mechanism can be operated from outside and inside. The exterior door handles are fitted with locks and keys.

Operation from outside:

To open the door, pull the door handle out of the grip hole and turn it forward thus releasing the two locking pins located at the front and rear edge of the door. Now open the door slowly outward. To shut the door, press it against the rubber sealing, turn the door handle backward and push it into the grip hole.

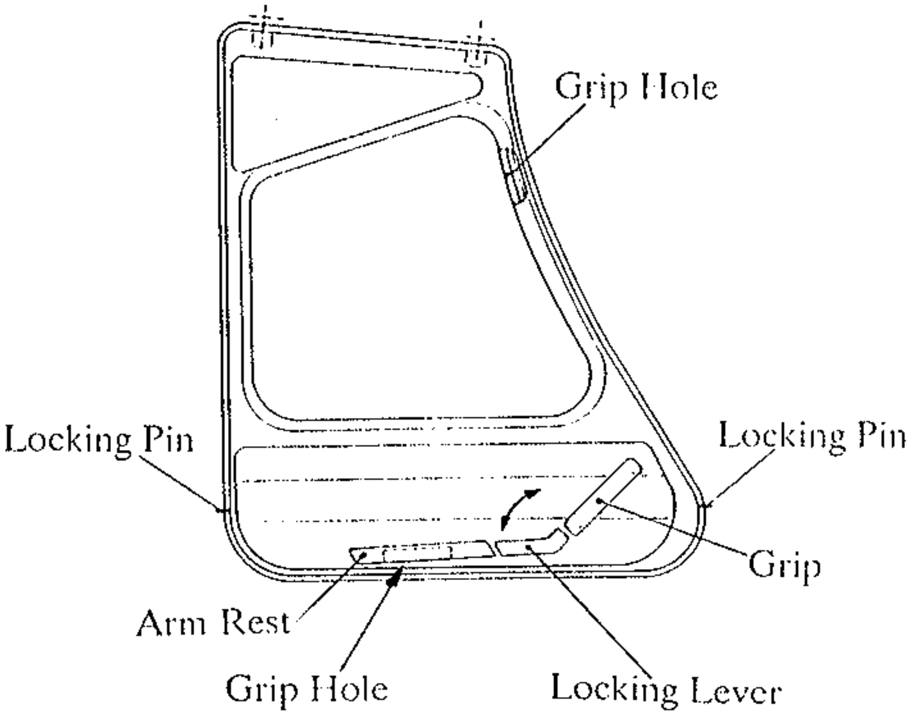


Figure 7-6: Inner Side of Cabin Doors

Operation from inside:

To open the door, turn the locking lever located between arm rest and handle up/forward (see Fig. 7-6). Then use the handle to open the door outward. As the handle cannot be reached from normal seating position with the door open, there is a grip hole located at the interior front edge of the door provided to pull the door shut. To lock the door, put one hand in the grip hole beneath the arm rest, pull the door completely into the door frame and use the other hand to turn the locking lever backward/down.

CAUTION

If the front handle is used to pull the door shut and to lock it it might be possible that the rear locking pin does not latch and that the door is not locked completely.

Emergency Exit:

In case of nosing over after landing on a soft surface it might happen that the doors cannot be opened sufficiently to leave the airplane. In this case, use the baggage compartment door for emergency exit. The headrests can be removed to ensure access to the baggage compartment door from all seats. If the baggage compartment door cannot be opened by means of the operating button located in the left door frame give a kick to the door-lock area.

SEATS

The seats of the R90-230 RG have an ergonomic design, allowing non-fatiguing endurance flights. The seat structure consists of a tubular steel frame and a seat pan made of composite material. All the seats are equipped with headrests. Three different positions are provided to adjust the headrests.

FRONT SEATS

The adjustable front seats are attached to inclined seat rails to match the seating position with the individual height (see Fig. 7-7). Release the seat locking by using handle A, located below the forward face of the seat near the bulge in the forward face. Lift the handle to release and adjust the seat. Eleven positions are available for seat adjustment.

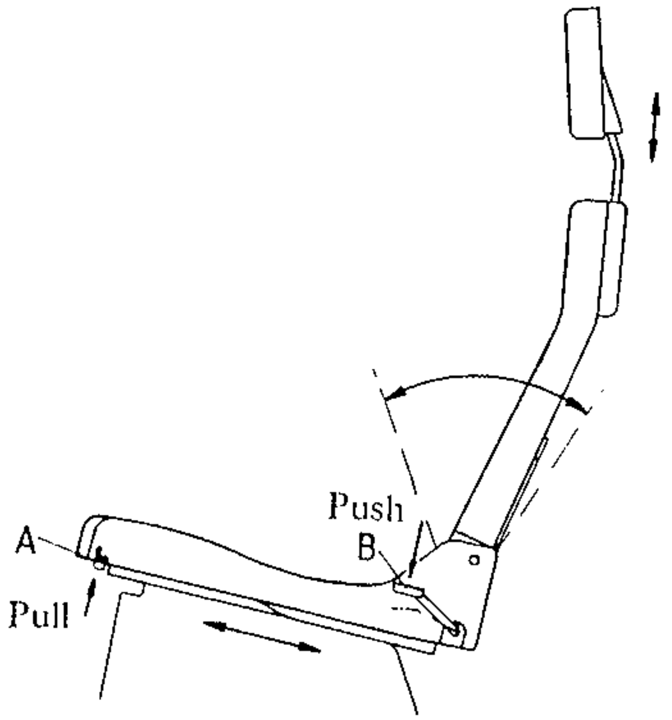


Figure 7-7: Adjustment of the Front Seats

CAUTION

Take one hand to hold on to the window bar while adjusting the seat. Holding on the instrument panel, the center pedestal or the unlocked door might cause damage.

The seat is locked in the desired position by releasing the handle. The seat must lock audibly and perceptibly.

WARNING

Check proper seat locking by jerking the seat fore and aft.

Two positions are provided to recline the seat back which may be adjusted by pushing down handle B. In addition, the seat backs of the front seats may be folded down for ease in boarding and deplaning passengers sitting in the rear seats. The seats are folded down by pushing down handle B with one hand and folding down the seat back with the other.

REAR SEATS

The rear seats are not adjustable.

If required, the rear seats can be easily removed by unscrewing the rear knurled-head screws (from the baggage compartment side). Move the rear seat slightly forward and lift it. Load and unload the disassembled rear seat through the baggage compartment.

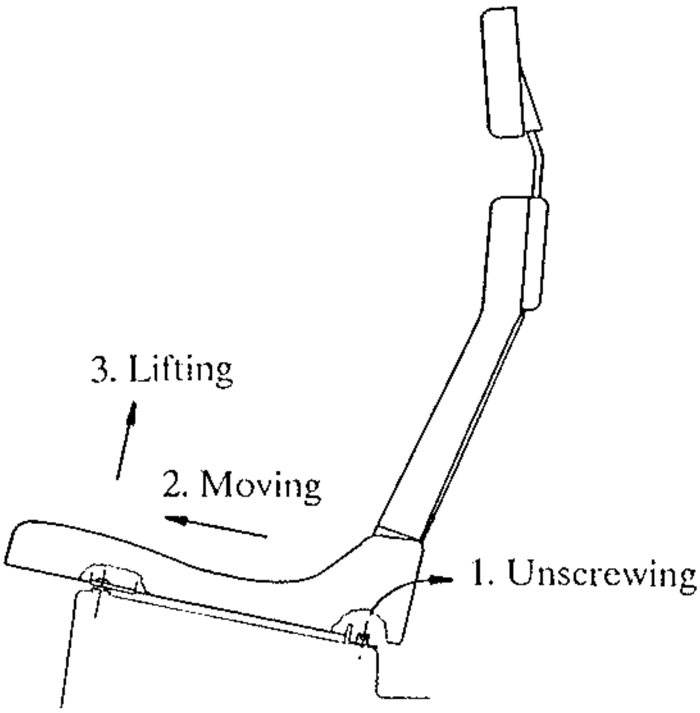


Figure 7-8: Removing of the Rear Seats

SEAT BELTS AND SHOULDER HARNESSES

Each seat is equipped with a three-point harness consisting of a two-piece seat belt and a shoulder harness. When sitting in the adjustable front seats, select your individual seating position before putting on the seat belts and shoulder harnesses. Select a seating position which allows the cockpit crew to carry out the work necessary for the conduct of the flight with seat belts and shoulder harness fastened. Seat belts and shoulder harnesses must be put on during takeoff, landing and in turbulence.

CAUTION

It is recommended to keep seat belts and shoulder harness fastened during the entire flight.

Put on and remove the three-point harness as follows (see Fig. 7-9):
- The inboard seat belt (1) attaches permanently to the seat belt turning lock (B) with a fitting.
  The belt system is locked by slipping the fittings (2) and (3) into the openings provided in the turning lock until they lock with an audible click.
- The outboard seat belt and the shoulder harness are adjustable in length. Adjust seat belt and shoulder harness to place the seat belt around the pelvic bones and fasten them.
- Open the belt system by turning the lock (B) to either the right or the left side.
- The shoulder harness (3) may be released separately by pushing the black releasing device (A), located on the upper part of the turning lock, forward. The seat belt remains locked.

If some of the seats are not occupied, secure the seat belts and shoulder harnesses of the respective seats as follows:

Lock seat belt (2) and shoulder harness (3) by means of the turning lock (B) and adjust them to minimum length.

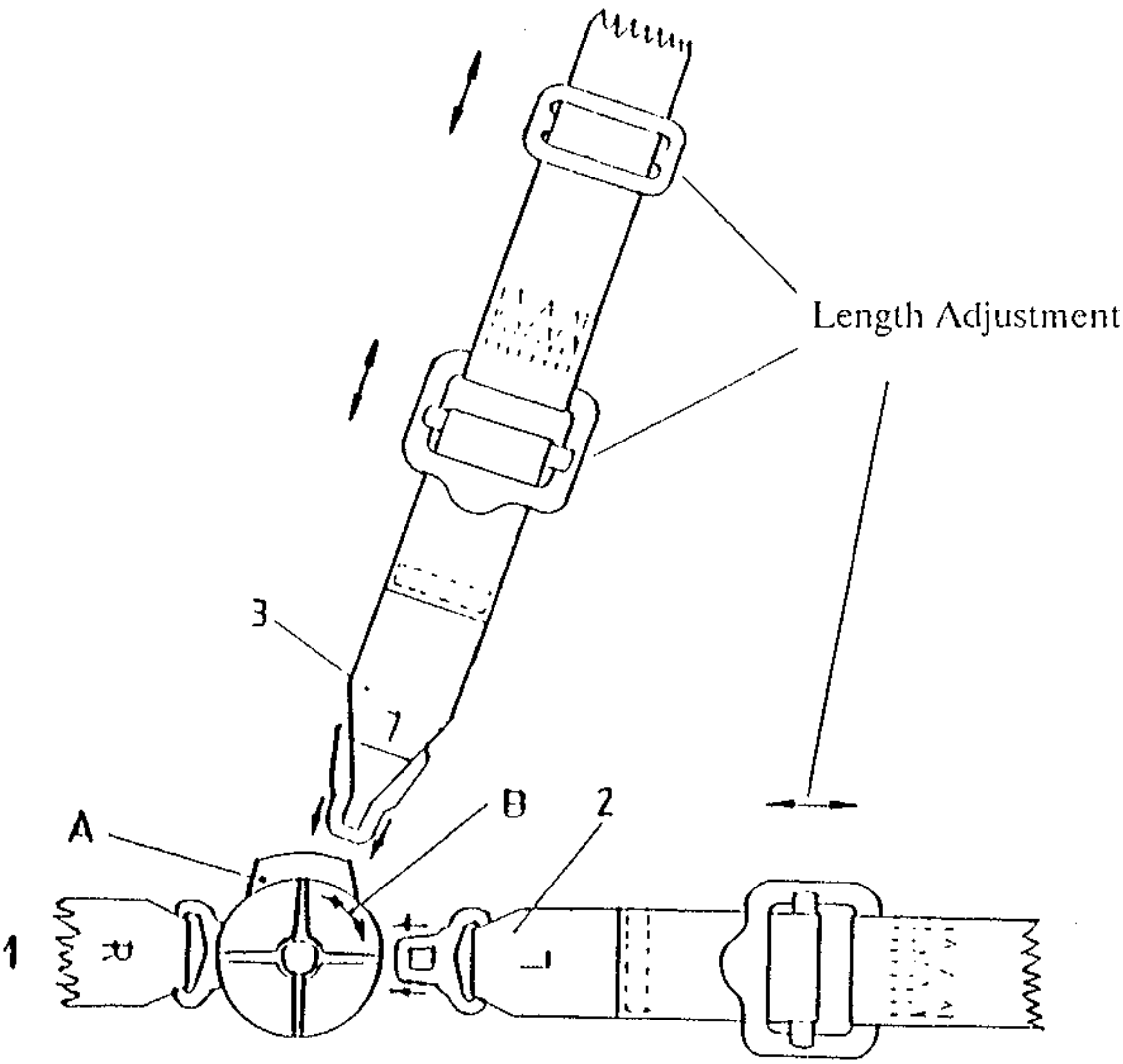


Figure 7-9: Three Point Harness (left side shown)

BAGGAGE COMPARTMENT

The baggage compartment has a capacity of approximately 28.3 ft^3 (0.8 m^3). The surface area is 9.7 ft^2 (0.9 m^2). Access width is 15.4 in (39 cm) and access height is 18.9 in (48 cm).

WARNING

Maximum permissible baggage weight is 110 lbs (50 kg).

From the outside, the baggage compartment is easily accessible through the baggage compartment door, located on the left side of the fuselage. A pushbutton located in the frame of the left cabin door is provided to open the baggage compartment. Lock the door by simply pushing it shut. A light is mounted to the aft cabin bulkhead and switched on by moving its cover backward.

CAUTION

Always turn off the baggage compartment light after use because it is supplied with power even with the master switch turned off.

The standard equipment of the airplane includes 6 restraint grommets. 12 screwing points are provided to fix the grommets. When loading objects, ensure that adequate protection is available to prevent damage. When loading the maximum permissible baggage weight of 110 lbs (50 kg), at least 4 grommets are required to lash the baggage.

CAUTION

- Use a restraint strap of at least 0.7 in (18 mm) of width and bearing a minimum of 900 lbs (4000 N) of tearing strength.
- Ensure that all objects are lashed tight.
- The transportation of hazardous material is not permitted.
- When loading, comply with the permissible center of gravity and baggage weight limits.
- Under no circumstances, allow the loading of people or animals in the baggage compartment.

POWER PLANT

The airplane is equipped with one Textron-Lycoming reciprocating engine of type IO-540-C4D5. The engine is an air-cooled, 6-cylinder, horizontally opposed, direct-drive, fuel-injected engine. Due to noise reasons, the maximum rated power has been reduced to 231 hp (172 kW) at 2400 RPM. The engine is attached to the steel-tube structure of the engine mounting by four shock mounts. The engine cowling has a two-piece design (upper and lower cowling) and is made of composite material.

The engine must be operated with 100/130 octane aviation grade fuel only. Approved aviation fuel grades and colours are AVGAS 100 (green) and AVGAS 100LL (blue). Only use Textron-Lycoming approved aviation grade engine oils to provide lubrication to the engine (see Section I).

Each airplane is provided with an Operator's Manual, issued by Textron-Lycoming. Refer to this manual for technical data and detailed information on the engine.

ENGINE CONTROLS

The top of the center pedestal, located between the pilot's and copilot's seat, contains all engine controls. Engine control arrangement can be seen from Fig. 7-2.

Functioning and operation of the engine controls will be described by the following (from the left to the right in the order of their arrangement on the center pedestal):

- Throttle Control Lever Friction Lock

  On the left side of the center pedestal, a friction lock is provided to prevent the primary engine controls (throttle, mixture and propeller control) from creeping once they have been set. Turning the lever clockwise increases the control force, turning it counterclockwise will decrease the control force. If further setting is required the locking handle has to be moved to the opposite side of the shaft allowing another half turn.

- Throttle Control

  The throttle control lever is identified by its large black knob. The lever is used to increase and decrease engine power. Moving the control lever forward (in flight direction) increases engine power, moving it backward will decrease engine power. A push and pull cable attaches the throttle control lever to the fuel injection system.

- Propeller Control Lever

  The propeller control lever is identified by its blue star-shaped knob. Moving the lever forward (in flight direction) increases the propeller speed, moving it backward will decrease the propeller speed. A push and pull cable attaches the propeller control lever to the propeller governor located at the engine.

- Mixture Control Lever

The mixture control lever is identified by its red star-shaped knob. Moving the lever forward (in flight direction) increases the fuel-to-air ratio; moving it backward will decrease the fuel-to-air ratio. The lever guide is provided with an intermediate gate. Shut down the engine by slighly pushing the lever to the left and pulling it back to the rear detent. The gate is provided to prevent accidental engine shutdown. A push and pull cable attaches the mixture control lever to the fuel injection system.

- Cowl Flap Control

The cowl flap control is located on the right side next to the center pedestal. Press the black button located in the center of the control lever to release and move it. Pull out the control lever to open the cowl flaps and push it in to close them. In addition, fine adjustment is possible. Rotating the control clockwise will cause a slight adjustment of the cowl flaps towards the closed position. Rotating it counterclockwise will cause a slight adjustment towards the opened position. The cowl flap control is actuated by a push and pull cable, levers, and control rods.

ENGINE INSTRUMENTS

Fig. 7-2 shows the engine instrument arrangement.

- Tachometer

The tachometer is an electromechanical gauge, driven by a shaft at half of the crankshaft rational speed. The shaft must be laid free from buckling. The normal operating range of the engine is 1800 to 2400 RPM, marked on the gauge by a green arc. Never exceed the upper limit of 2400 RPM (red radial).
Optionally the aircraft can be equipped with an electronical tachometer as a primary instrument. Further information about this tachometer is given in section IX.

- Multifunction Gauge

The multifunction gauge is located on the right instrument panel and consists of six indicators designed as moving-coil instruments. The four gauges, described by the following, are engine instruments:

Oil Pressure Gauge

The oil pressure signal is generated by an electrical oil pressure transmitter located at the engine. The oil pressure is indicated in bar. The normal operating range is 3.8 to 6.6 bar (55 to 95 psi) (green arc). The caution ranges are 1.7 to 3.8 bar (25 to 55 psi) and 6.6 to 7.9 bar (95 to 115 psi)(yellow arcs). Oil pressure may not exceed the upper limit of 7.9 bar (115 psi) (red radial) or fall below the lower limit of 1.7 bar (25 psi) (red radial).

Oil Temperature Gauge

The oil temperature signal is generated by an electrical oil temperature transmitter located at the engine. The oil temperature is indicated in °C. The normal operating range is 74° to 93°C (165° to 200°F) (green arc). The caution ranges are 40° to 74°C (104° to 165°F) and 93° to 118°C (200° to 245°F) (yellow arcs). Oil temperature may not exceed the upper limit of 118°C (245°F) (red radial).

Cylinder Head Temperature Gauge

The cylinder head temperature signal is generated by an electrical temperature transmitter located in cylinder No. 5. The cylinder head temperature (CHT) is indicated in °C. The normal operating range is 130° to 224°C (266° to 435°F) (green arc). The caution range is 224° to 260°C (435° to 500°F) (yellow arc). Cylinder head temperature may not exceed the upper limit of 260°C (500°F) (red radial).

Fuel Pressure Gauge

The fuel pressure signal is generated by an electrical pressure transmitter located between the mechanical fuel pump and the fuel injection system. Fuel pressure is indicated in bar. The normal operating range is 0.96 to 3.1 bar (14 to 45 psi) (green arc). The caution range is 0.83 to 0.96 bar (12 to 14 psi) (yellow arc). Fuel pressure may not fall below the lower limit of 0.83 bar (12 psi) (red radial) or exceed the upper limit of 3.1 bar (45 psi) (red radial).

- Combination Gauge

The combination gauge indicates manifold pressure and fuel flow. It is located in the right instrument panel. The manifold pressure is indicated on the left side of the circular gauge while the fuel flow is indicated on the right.

Manifold pressure is indicated in "inches of Mercury" (inHg). It is measured through a hose line located in the intake manifold of cylinder No. 5. The static pressure required for this measurement is taken from the engine compartment by means of hose lines. The normal operating range, allowing all normal engine speeds, is 15 to 25 inHg (green arc). The caution range is 25 to 29.4 inHg (yellow arc) and allows certain engine speeds only, dependent on the manifold pressure (see Section II). The upper limit of 29.4 inHg (red radial) may not be exceeded.

Fuel flow is indicated in US gallons per hour. The fuel flow gauge consists of a fuel pressure gauge. The scale of the fuel pressure gauge is designed to indicate the relation between the fuel pressure in the flow distributor and the fuel flow. The fuel pressure is measured through a fuel pressure measuring hose line attached to the flow distributor of the engine. The measuring range is 0 to 26 US gallons per hour.

- Exhaust Gas Temperature Gauge

The exhaust gas temperature gauge (EGT) is located in the upper left instrument panel. A thermocouple element, located at the exhaust pipe of cylinder No. 6, supplies the required signal. The exhaust gas temperature gauge serves for indicating temperature changes relative to mixture leaning. The exhaust gas temperature is indicated in °F of relative temperature. One graduation on the scale corresponds to 25°F.

- Oil Pressure Warning Light

The red oil pressure warning light (OIL PRESS) is located in the annunciator panel. It will illuminate when the oil pressure drops below the minimum permissible value. The warning light is actuated by a contact attached to the electrical oil pressure transmitter.

OPERATION AND CARE

- Break-in

New engines have been already run-in and tested at the Textron-Lycoming factory. Additional break-in time is not required. For faster ring seating and stabilization of oil consumption, straight aviation mineral oil has to be used during the first 50 hours of operation. The usual 50 h engine inspection has to be performed after the first 25 hours of operation. This must include draining of the engine oil, cleaning of the oil screen located in the oil sump and changing of the oil filter which is mounted to the rear of the engine. The oil sump has to be refilled again with straight aviation mineral oil. After the first 50 hours of operation, the mineral oil has to be drained and replaced by ashless dispersant aviation oil (HD oil) (see also Section I).

- Operation

During operation, the engine may never exceed the propeller speed and power limits specified in the airplane flight manual to ensure maximum operating reliability and a long engine life. In addition to proper engine operation, preflight checks and periodical inspections, any signs of trouble or leakage must be examined and eliminated. During inspection, special attention must be given to the fuel and engine oil system. Any trouble detected must be eliminated before the next flight. If the oil quantity measured during the preflight check is less than 9 qt (8.5 l) add aviation oil. For maintenance of the engine oil system, adhere to the detailed information provided in the applicable airplane maintenance manual and the applicable Lycoming engine operating manual.

ENGINE OIL SYSTEM

The engine installed in the airplane has a wet sump type lubricating system which to a large extent is an integral portion of the certificated Lycoming engine thus including oil pump, oil filter, pressure relief and thermally operated valves as well as the oil pan. Only the oil cooler including the connecting hose lines and the electrically operated oil pressure and temperature transmitters are provided by the airplane manufacturer. The oil cooler is attached to the cooling baffles located on the right side of the engine rear. The oil drain screws are easily accessible and located on the lower surface of the oil pan. The crankcase is vented through the lefthand cowl flap opening by a breather hose connected to an oil separator which is attached to the left side of the fire wall. The oil separator is equipped with a hose to allow the separated oil to flow back to the crankcase. The oil level of the engine is checked by using a dip stick. Check the oil level during each preflight check. During operation, monitor the engine oil system by watching the oil pressure and temperature readings.

IGNITION SYSTEM

The engine is equipped with a dual ignition system. The two ignition systems are entirely independent from each other. Each system is actuated by a Slick 6251 magneto with impulse coupling. The magneto system provides for proper ignition even if the electrical system (alternator and/or battery) has failed. The two ignition systems are provided with screened lines and a spark plug in each cylinder. The left magneto fires the lower spark plugs of cylinders 2, 4, and 6 and the upper spark plugs of cylinders 1, 3, and 5 while the right magneto fires the spark plugs on the opposite side. Ignition is switched on by the ignition key provided with the following positions: OFF, R(ight magneto), L(eft magneto), BOTH and START.

AIR INDUCTION SYSTEM

Engine induction air is taken in through the air inlet, located in the engine cowling below the propeller spinner, at which point it passes through the air duct and the air filter. The induction air filter housing is attached to the fuel injection system. The air passes through a foam filter and then directly into the fuel injection system. If the foam filter is clogged or frozen a spring-loaded flap, located in the air filter housing, will open to provide an emergency air induction source to take preheated, unfiltered air in from the engine compartment for mixture preparation. If the air is taken in from the emergency air induction source, there will be a decrease in power of approximately 14 %. Use an air inlet cover to prevent air inlet, air duct and air filter from being clogged while the airplane is not operated.

Note

Use the air inlet cover whenever the airplane is not operated because the air inlet may be checked thoroughly for foreign bodies only if the engine cowling has been removed.

EXHAUST SYSTEM

The exhaust system consists of stainless steel parts only. It is located below the engine and covered by the engine cowling. The exhaust manifolds of the six cylinders end in a muffler assembly. The exhaust manifolds are interconnected by sleeve joints, serving to compensate thermal expansion. From the muffler assembly, the exhaust gases are routed to the outside through two tail pipes. A heat exchanger is attached to the upper surface of the muffler assembly and provided to supply the cabin heating system with warm air.

FUEL INJECTION SYSTEM

Fuel is supplied to the Textron Lycoming IO-540-C4D5 engine using a mechanically operated Bendix-RSA-5AD1 injection system. The fuel injection system consists of an air flow measuring system and the fuel control system. Butterfly valve, servo valve, and fuel valve are located in a cast housing below the engine. The rate of air flow is measured inside the cast housing and a servo valve is controlled to convert differential air pressure into a differential fuel pressure. After the air has passed the cast housing and the induction air collector, it is routed to the cylinders. The dosed fuel flows from the fuel distributor, located above the engine, to the fuel injection nozzles of the different cylinders. Fuel and air are mixed inside the cylinders.

ENGINE COOLING

The engine is cooled by ram air, provided by two ram air inlets located on the right and left side of the propeller spinner and routing the ram air into the engine compartment. The engine compartment is divided by the engine-mounted cooling baffles into an upper, cool section and a lower, warm section. In the cool section, the cooling air circulates around the engine and - guided by the cooling baffles - between the cylinders to absorb heat. One portion of the cooling air passes the oil cooler and another small portion of the cooling air is routed through hoses to the alternator and the battery to cool them. The heated cooling air enters the warm section of the engine compartment to circulate around and to cool the exhaust system. The two cooling air outlets are provided with manually operated cowl flaps and located at the lower engine cowling. At this point, the heated air is leaving the system. The cooling baffles are equipped with weathertight rubber sections located at the engine contact point. Damage to the cooling baffles or the rubber sections may lead to reduced cooling and thus to engine overheating.

PROPELLER

The airplane is equipped with a 4-bladed, hydraulically operated Mühlbauer mt constant speed propeller, Model No. MTV-14-B. The diameter is 74.8 in (1.90 m).

Pitch is controlled by a propeller governor. The governor serves for maintaining the selected propeller speed when airspeed or power are changed. The governor is connected to the oil circulation of the engine and increases the engine oil pressure for pitch control. Increased pressure is required to increase the pitch while pressure reduction will result in a decrease in pitch.

The propeller is provided with high pitch and low pitch stops. In case of governor oil pressure loss, the propeller blades will move automatically to the low pitch stop (see Section III).

The propeller blades are made in wood composite construction with a fibre-reinforced plastic coat and superrefined steel edge protection.

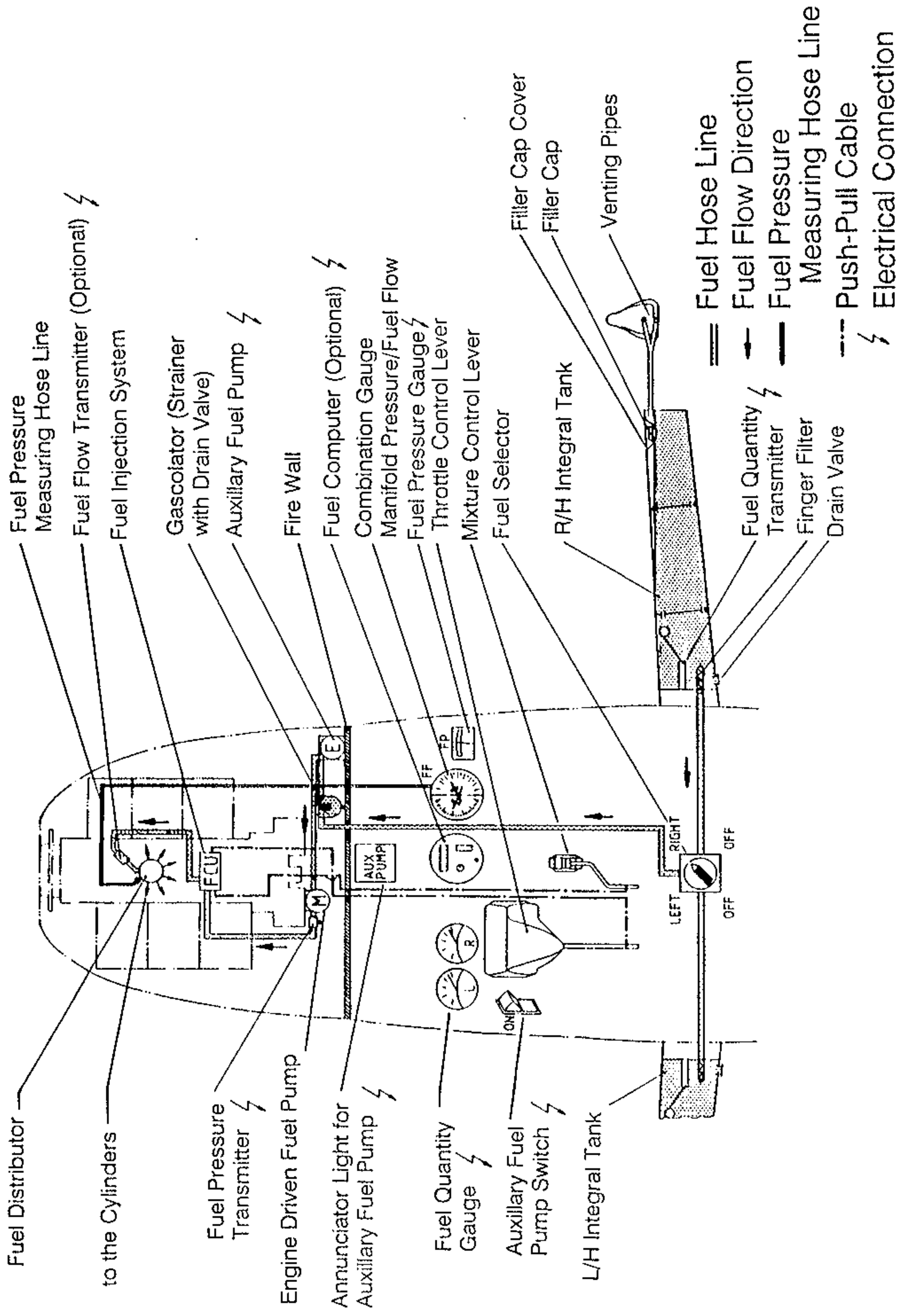
FUEL SYSTEM

The fuel tanks are located in the wings. The inward third of each wing is provided with a tank designed as an integral tank (see Fig. 7-10). Overall fuel capacity of each integral tank is 33 US gal./125 l. Usable fuel is 31 US gal./118 l (unusable fuel: 1.8 US gal./7 l).

The integral tank selected by means of the fuel selector supplies the fuel injection system with fuel through a finger-type filter, the fuel selector valve, a gascolator, an electrically actuated auxillary fuel pump and an engine-driven, mechanically actuated fuel pump. The fuel injection system supplies the flow distributor of the engine with the amount of fuel according to the respective power (THROTTLE) and mixture (MIXTURE) setting. If the optional fuel computer is used, a fuel flow transmitter is installed between the fuel injection system and the flow distributor. From the flow distributor, the fuel flows to the cylinders.

FILLER CAPS

Each integral tank is provided with a filler cap located in the top surface of the wing. The filler cap is faired with a filler cap cover. Open the filler cap covers by pulling the release button located inside the cabin below the cabin door. Do not pull the button during flight. Below the filler cap cover, there is the filler cap, provided to lock the integral tank. Open the cap by a 1/4 turn to the left.


Fuel Pressure Measuring Hose Line
Fuel Flow Transmitter (Optional)
Fuel Injection System
Gascolator (Strainer with Drain Valve)
Auxiliary Fuel Pump
Fire Wall
Fuel Computer (Optional)
Combination Gauge Manifold Pressure/Fuel Flow
Fuel Pressure Gauge
Throttle Control Lever
Mixture Control Lever
Fuel Selector
R/H Integral Tank
Filler Cap Cover
Filler Cap
Venting Pipes
Fuel Quantity Transmitter
Finger Filter
Drain Valve
Fuel Hose Line
Fuel Flow Direction
Fuel Pressure Measuring Hose Line
Push-Pull Cable
Electrical Connection
Fuel Distributor
to the Cylinders
Fuel Pressure Transmitter
Engine Driven Fuel Pump
Annunciator Light for Auxiliary Fuel Pump
Fuel Quantity Gauge
Auxiliary Fuel Pump Switch
L/H Integral Tank
FCU
M
E
AUX PUMP
FF
FP
RIGHT
LEFT
OFF
OFF
ON
L
R

WARNING

Always ground the airplane using the ground connecting point before refueling. Exhaust tail pipes and landing gear are not appropriate for grounding the aircraft.

The ground connecting point is located at the right side of the engine cowling and marked by a corresponding symbol.

To lock the integral tank, insert the filler cap and lock it by a 1/4 turn to the right. Check the filler cap for proper seating. To lock the filler cap cover, press it down until it engages perceptibly and with an audible click.

A drain line has been installed near the filler neck to drain off the fuel being spilt while refueling and the water which penetrated through the filler cap covers. Fuel and water are drained off through an outlet located on the lower surface of the wing.

FUEL FILTER AND DRAIN VALVES

The fuel drain points of the integral tanks are provided with a finger filter. In addition, a gascolator has been mounted to the engine side of the fire wall.

Below the drain point of each integral tank, a drainable fuel tank sump is located on the front lower surface of the wing. The fuel tank sump is provided with a drain valve to take fuel samples from outside. At least before the first flight of the day, use a transparent container (fuel sampler) to take fuel samples from the drain valves. Check the fuel samples for proper grade of fuel (colour: blue or green), water and contaminants. If water or contamination is detected, continue draining until all water and contamination have been removed.

Open the inspection cover and pull the knob of the fuel strainer drain to take a fuel sample from the gascolator. The drained fuel is discharged through a line at the lower part of the fire wall near the righthand nosewheel door.

WARNING

After draining has been completed, check all fuel drain valves for tightness.

To avoid fuel contamination, refuel the airplane at fuel stations only that are provided with filters to remove contaminants and water from the fuel. If there is not any filter available use a dry and clean chamois leather to filter the fuel. It is recommended to top up the integral tanks after the last flight of the day to avoid the formation of condensed water.

FUEL SELECTOR

The fuel selector is located on the center pedestal, aft of the engine controls. The selector switch provides for switching the fuel selector valve to the right integral tank (RIGHT), to the left integral tank (LEFT) and to two "integral tanks closed" (OFF) positions. Each of the four possible switch positions has a perceptible detent. In case of engine fire, the fuel selector valve serves as fire cock by switching it to the OFF position.

During cruise, switch frequently between right and left fuel tank in order to avoid higher asymmetric fuel loading. It is recommended to change the selected fuel tank every 30 minutes.

WARNING

Maximum permissible fuel asymmetry between lefthand and righthand integral tank is 13 US Gal (50 l).

AUXILIARY FUEL PUMP

The electrically driven auxiliary fuel pump is located on the right side of the firewall in the engine compartment. It is actuated by a switch (AUX PUMP) on the left instrument panel. An annunciator light (AUX PUMP) located on the annunciator panel will illuminate when the auxiliary fuel pump is turned on. The auxiliary fuel pump is used for starting support, during takeoff and landing and after failure of the engine driven fuel pump.

TANK VENT SYSTEM

For each integral tank, a NACA inlet with two vent pipes is provided at the bottom side of the wing. These vent pipes may not be clogged and are to be checked during each preflight inspection. An obstructed vent pipe may cause fuel flow interruption and successive engine stoppage.

FUEL QUANTITY GAUGE

A fuel quantity gauge for each fuel tank is located on the lefthand instrument panel. The fuel quantity gauge receives an electrical signal from a float-controlled transmitter, installed in the fuel tank. These float-controlled fuel transmitters have been calibrated by the manufacturer. In case, a transmitter has to be recalibrated, recalibration must be performed by a qualified airplane mechanic according to the procedure described in the applicable maintenance manual.

It is the pilot's sole responsibility to make sure that there is enough fuel in the integral tanks for the safe conduct of the flight. Flight planning must consider a reserve fuel quantity sufficient for safe completion of the flight.

In addition to monitoring the fuel gauges, the fuel quantity has to be checked visually. A scale has been mounted to the filling hole which allows fuel quantity reading from 20 US Gal (80 l) onwards. The airplane must be in a level position when checking the fuel quantity.

In order to avoid high asymmetrical fuel loading, fuel has to be used alternatingly from both integral tanks. Maximum permissible fuel asymmetry is 13 US Gal (50 l). This requirement will be met in any case if switching to the fullest tank is performed when the difference between right and left tank reaches 1/4 of the total fuel capacity.

A fuel computer is offered as recommended optional equipment, providing very accurate monitoring of fuel flow and fuel quantity. If the fuel computer is installed comply with the operating instructions provided in Section IX "Supplements".

ELECTRICAL SYSTEM

Figure 7-11 shows the electrical system (power distribution) schematic. For location of control elements and instruments, refer to Fig. 7-3.

ALTERNATOR AND BATTERY

The 24 Volt direct current system is supplied by a battery and an engine driven alternator with integrated diodes. Power is supplied via the master switch to a collecting bus (powerplant/systems bus) which then provides power to the consumer devices via circuit breakers and switches. The battery has a capacity of 10 Ah and is the only power source available as long as the engine is not running. It is charged when the engine is running by a Prestolite alternator of 70 A maximum output current via a regulator with overcharging protection feature. An overvoltage protection system will switch off the alternator immediately as soon as voltage peaks occur. This protective function is cancelled by recycling the alternator switch and the alternator will be reset to work, provided there is no malfunction.

MASTER SWITCH

A dual red rocker-type switch, which is used to supply power to all electrical loads, is located in the left half of the instrument panel. The left part of the switch, labelled BAT, serves for switching on and off the total battery power supply of the distribution system. The right part, labelled ALT, is provided for switching the alternator. During normal operation, both parts are switched on. Switching is done by pressing down the upper half of the switch.

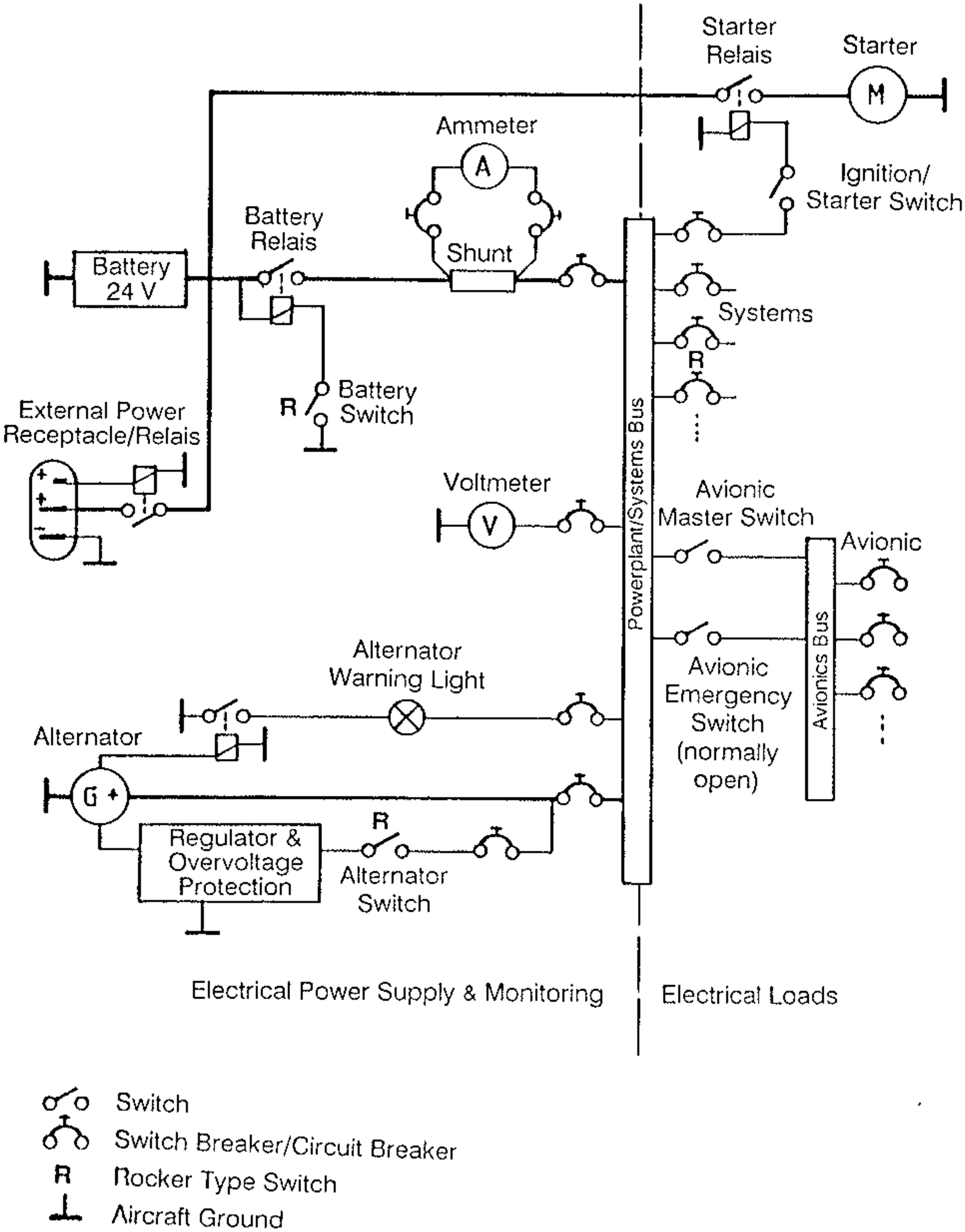


Fig. 7-11: Electrical System Schematic

In case of alternator failure, the ALT part of the master switch may be switched off solely. The battery will not be charged any more and any electrical loads, which are not essential, must be switched off.

In case of battery malfunctions, the ALT switch is forced to be switched off when switching off the BAT switch because power supply by the alternator alone is not possible.

POWER SUPPLY CONTROL INSTRUMENTS

The electrical power supply system is monitored by a voltmeter and an ammeter located in the multifunction gauge on the right half of the instrument panel (see Figure 7-2).

The ammeter indicates the total current between battery and the power plant/systems bus (except starting current). During normal operation, the battery will be charged, i.e. the indication is more or less in the positive (+) range, depending on the respective battery charging condition. If the alternator has failed or the electrical load exceeds the capability of the alternator the ammeter will show negative readings (-).

The voltmeter indicates the voltage of the powerplant/systems bus. During normal operation, the indication is within the green arc of the instrument (24 to 28 Volts), showing slightly higher voltages when the alternator output is added to the battery voltage. Exceeding the maximum allowable voltage (red line) is reliably prevented by an overvoltage protection system. During emergency operation with the battery only, the indicated voltage will decrease according to the connected load and time, with the voltmeter pointer in the yellow arc or even below.

ANNUNCIATOR LIGHTS

The alternator warning light is the only annunciator light which is important for monitoring the electrical system. If it does not illuminate the alternator is operating and generating a voltage output. This does not mean that the battery is being charged. The ammeter is the only means to check battery charging.

Any other annunciator lights indicate the function of consuming devices or operating conditions and are to be considered as loads with respect to the electrical power supply.

CIRCUIT BREAKERS AND SWITCH BREAKERS

All electrical systems in the airplane are protected by circuit breakers. Fuses are not used. Switch breakers are circuit breakers intended to be used for frequent switching and have a rocker-type design. Electrical devices having separate switches or usually no need to be switched are protected by push-to-reset-type circuit breakers. A listing of all push-to-reset-type circuit breakers is provided at the end of this section.

An avionics master switch is provided between the powerplant / systems bus and the avionics bus for joint switching of all avionics equipment. In parallel to the avionics master switch, an emergency avionics switch is installed. This emergency avionics switch is intended to be used as a "substitute" to the avionics master switch in case the master switch fails and the avionics are out of power supply. Avionics master switch and emergency avionics switch provide power to any electrical device protected by a circuit breaker located in the lower circuit breaker panel (avionics bus) (see listing at the end of this section).

EXTERNAL POWER RECEPTACLE

An external power receptacle is available as optional equipment and provides external power supply during ground checks and engine starting. The receptacle has a three-pin design according to MIL standard and is protected against interchanging of polarity. It will be connected to the battery via a relais as soon as external power is supplied. For engine start up using external power proceed as follows: connect external power cable to the aircraft, power on external power source, switch on aircraft master switch and start engine as usual. Disconnect external power cable when engine is running.

The external power supply may only remain connected to the aircraft as long as absolutely necessary, because longer periods can lead to uncontrolled charging or discharging of the battery.

LIGHTING SYSTEM

INTERNAL LIGHTING

The entire internal lighting system consists of instrument lights, instrument panel lighting, avionics lighting, overhead red lights, overhead reading lights, and a baggage compartment light. Dimming controls and switches jointly control the different lights and provide for individual brightness control:

- Instrument Lights
  A combined rotary switch/dimmer is located below the annunciator panel and controls the post lights of the externally lighted instrument as well as the lighting of the internally lighted instruments (Marking: INSTR).

- Instrument Panel Lighting:
  A combined rotary switch/dimmer is located below the annunciator panel and controls the instrument panel lighting installed below the glareshield. (Marking: PANEL).

- Avionics Lighting
  A combined rotary switch/dimmer is located below the annunciator panel and controls the integral lighting of the installed avionics equipment. (Marking: AVIONIC).

- Overhead Red Lights
  A combined rotary switch/dimmer is located on the overhead panel between the front seats and controls two adjustable overhead red lights. (see Figure 7-12).

- Overhead Reading Lights
  A combined rotary switch/dimmer and two pushbutton switches are located on the overhead panel between the front seats and control the two adjustable white overhead reading lights. (see Figure 7-12).

- Baggage Compartment Light
  The baggage compartment light is mounted to the aft cabin bulkhead and switched on and off by moving its cover forward and backward. Always turn off the baggage compartment light after use because it is supplied with power even with the master switch turned off.

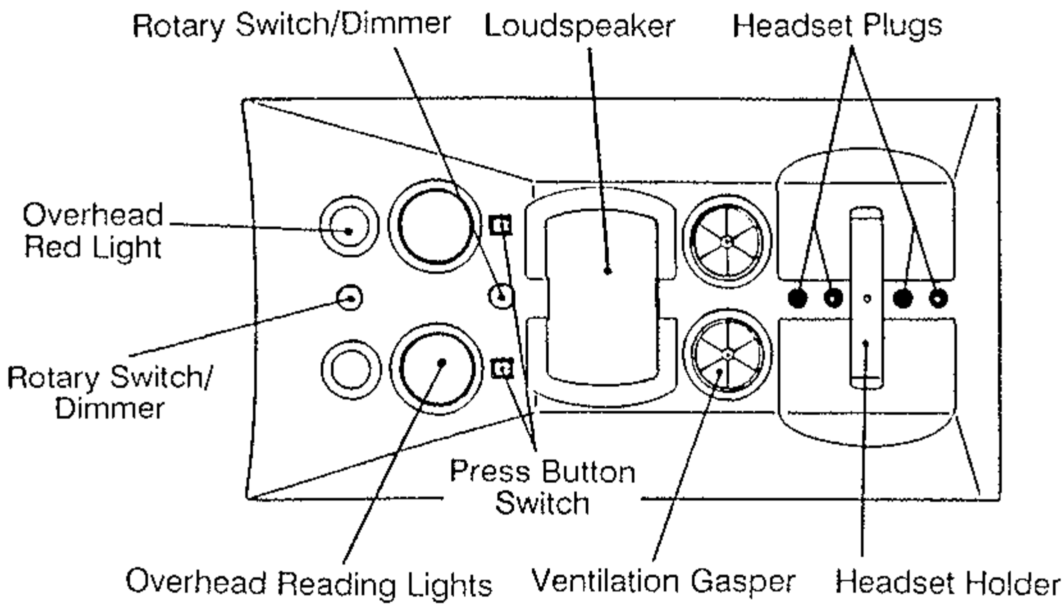


Figure 7-12: Front Overhead Panel

EXTERNAL LIGHTING

The airplane is provided with the following external lighting:

- Navigation Lights
  located in the left and right wing tips, covered by aerodynamically shaped transparent caps, and a tail light mounted at the upper aft end of the rudder. The navigation lights are switched on by a common switch breaker (NAV LIGHT).

- Anti-Collision Lights
  located in the left and right wing tips, covered together with the navigation lights by transparent caps, as well as an anti-collision light mounted at the top of the vertical tail which is covered at its front part, to prevent dazzling effects from wing and propeller back side.

  The three associated power supply units are located beneath the lights and synchronized by connecting wiring. The three anti-collision lights are jointly activated by the ACL switch breaker.

- Landing Light
  installed in the outer lefthand wing leading edge below a transparent cover. A second landing light in the right wing is available as optional equipment. In this case, both landing lights are jointly actuated by the LAND LIGHT switch breaker.

- Taxi Light
  installed in the outer lefthand wing leading edge below a transparent cover. A second taxi light in the right wing is available as optional equipment. In this case, both taxi lights are jointly actuated by the TAXI LIGHT switch breaker.

All switch breakers for external lighting are located on the center instrument panel below the avionics stack.

PITOT/STATIC PRESSURE SYSTEM

Pitot pressure is supplied from the pitot tube mounted to the left wing (see Figure 7-13). The pitot tube is connected to the airspeed indicator on the lefthand side of the instrument panel.

Static pressure is obtained by two external static pressure ports located on the right and left side of the rear fuselage (see Figure 7-13), thus partially eliminating the asymmetrical effects of propeller slipstream and small sideslip. The static pressure hose is leading from the static ports along the fuselage roof to the baggage compartment bulkhead and below baggage compartment floor and door frame to the "Alternate Static" selector switch, located on the left lining of the center pedestal. The selector switch is connected to the respective instruments (altimeter, airspeed indicator and vertical speed indicator).

The static source may be switched from the normal external ports to cabin pressure by means of the "Alternate Static" selector switch. This might become necessary in case the normal system is clogged or frozen. When switching to cabin pressure, the normal static system is disconnected.

Due to the airflow along the fuselage, the cabin pressure is depending on airspeed and deviating from the precise outside static pressure. Therefore, airspeed indicator and altimeter will show inaccurate readings with the alternate static system selected. Calibration tables for converting indicated values to actual values are provided in Section V.

A heated pitot tube, which may be heated electrically in icing conditions, is available as optional equipment. The respective switch breaker (PITOT HEAT) is located below the avionics stack. A light in the annunciator panel indicates that the pitot heat is switched on. On the ground, pitot heat may be operated for a short period of time only for checking purposes because overheating may occur due to the lack of cooling airflow.

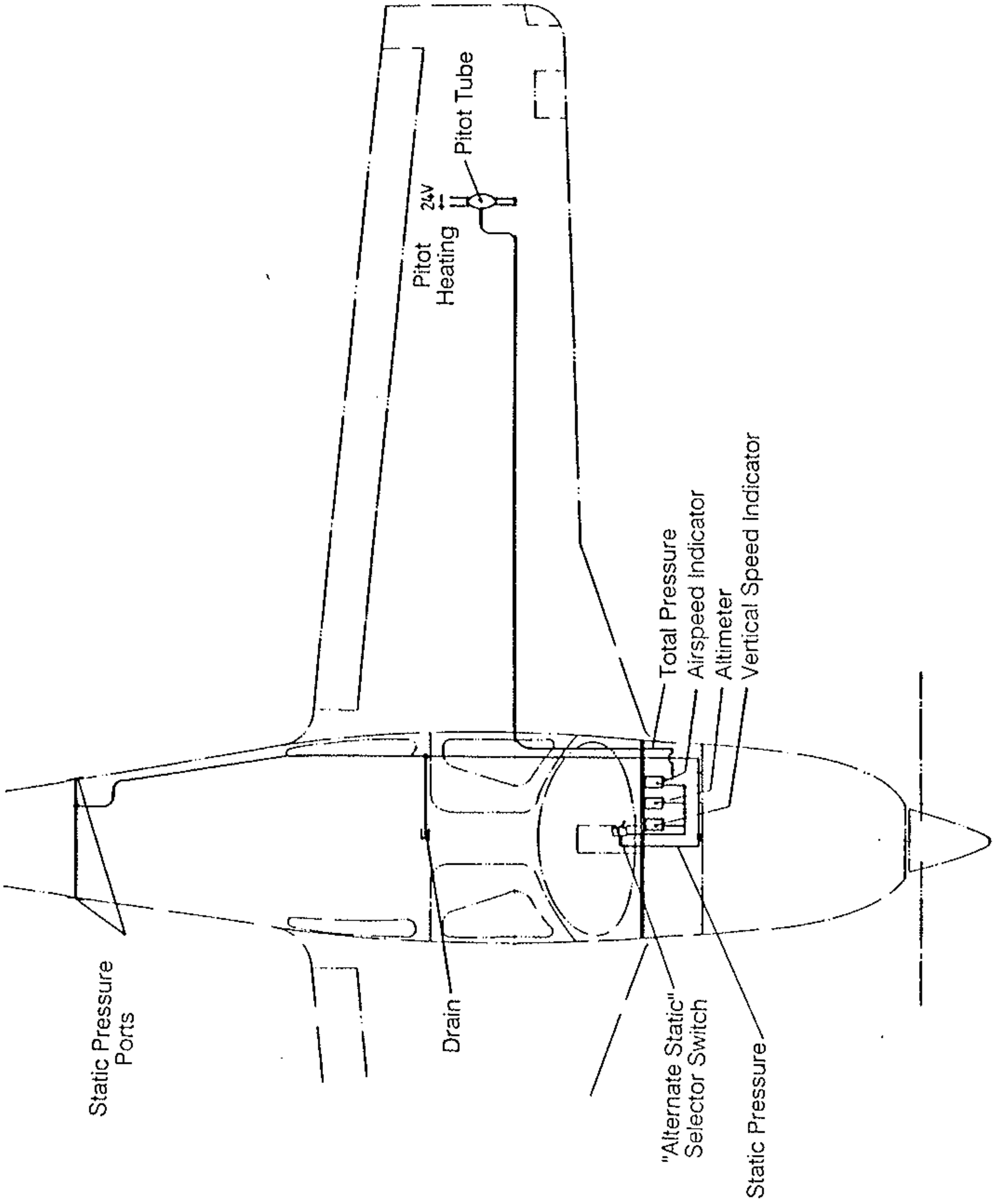


Fig. 7-13: Pitot/Static Pressure System Schematic

VACUUM SYSTEM

A vacuum system may be installed as optional equipment, providing a source of vacuum to the artificial horizon and/or the directional gyro (see Figure 7-14). The engine mounted vacuum pump is permanently driven by the engine. The vacuum pump pulls a vacuum on the driven instruments via a pressure relief valve. The inlet air side of the gyros is connected to a vacuum air filter. A "Suction" gauge on the left half of the instrument panel indicates the differential pressure applied to the artificial horizon. If the pointer of the suction gauge is out of the green arc with the engine running the system is defective and the readings of the connected instruments are unreliable.

CAUTION

If the engine is not running the vacuum system, including any connected instrument, is inoperable.

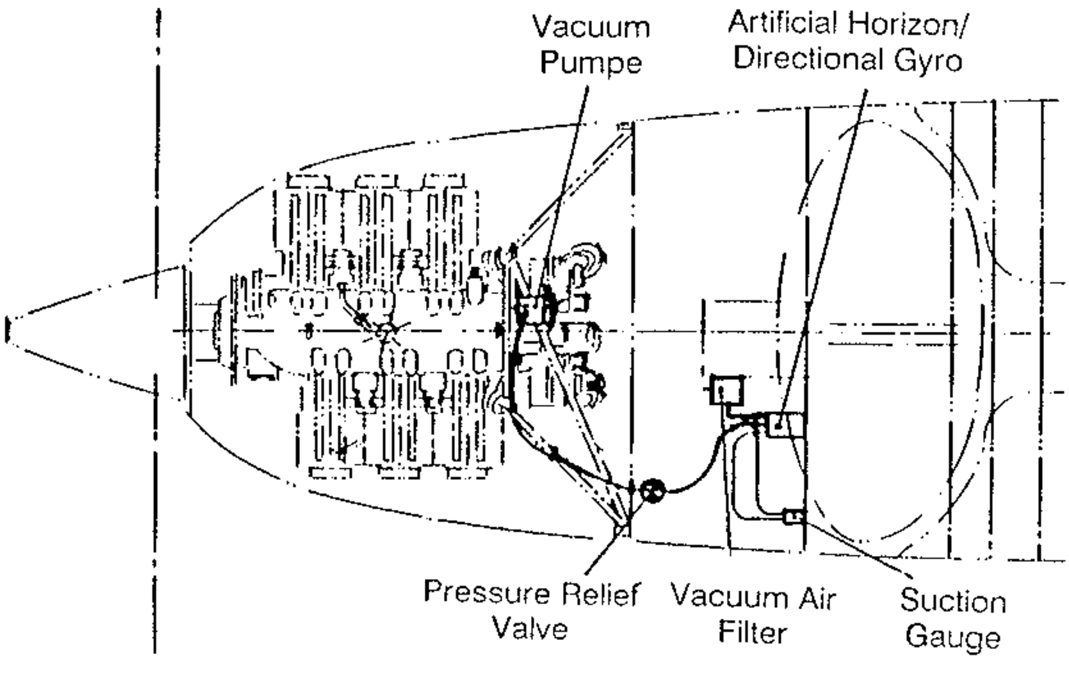


Figure 7-14: Vacuum System Schematic

STALL WARNING SYSTEM

The airplane is equipped with an acoustic and visual stall warning system, i.e. a warning horn and a warning light. The warning horn is located behind the left half of the instrument panel and the red warning light is part of the annunciator panel (Marked: STALL WARN).

A stall warning transmitter vane is mounted to the leading edge of the right wing. The stall warning transmitter switch is adjusted to close an electrical circuit, activating the stall warning 5 to 10 kts above stall speed in any flight configuration. The stall warning remains activated until the airspeed is returned to the safe operating range.

WARNING

The stall warning system is inoperable with the master switch turned off.

CABIN HEATING AND VENTILATION

The cabin heating and ventilation system can be adjusted individually for any seat (see Figure 7-15). The control unit on the right side of the instrument panel consists of three sliding levers for heater adjustment.

The sliding control "▲ ▼" provides for directing the total airflow to either the windshield or the leg room and rear seats. The sliding control "FRESH" adjusts the ventilating fresh air supply from the righthand NACA intake. The sliding control "WARM" regulates heated air supply from the heat exchanger.

CAUTION

In case of engine fire, the sliding control "WARM" must be placed in the "CLOSED" position in order to prevent smoke and gases coming out of the engine compartment from entering the cabin.

The front seat area is heated by warm air from the leg room- and windshield defroster outlets. For the rear seats, outlets on each sidewall are provided. Maximum heating can be achieved by setting the sliding control "▲ ▼" to "▼" and closing the "FRESH"-air valve.

Cabin ventilation may be adjusted independantly from cabin heating. For each front seat, one fresh air gasper on the instrument panel and one overhead gasper is provided. The ventilation airflow from the swivelling gaspers may be adjusted by turning the outer ring of the vent from "closed" to "full open". For each of the rear seats, an overhead gasper is provided.

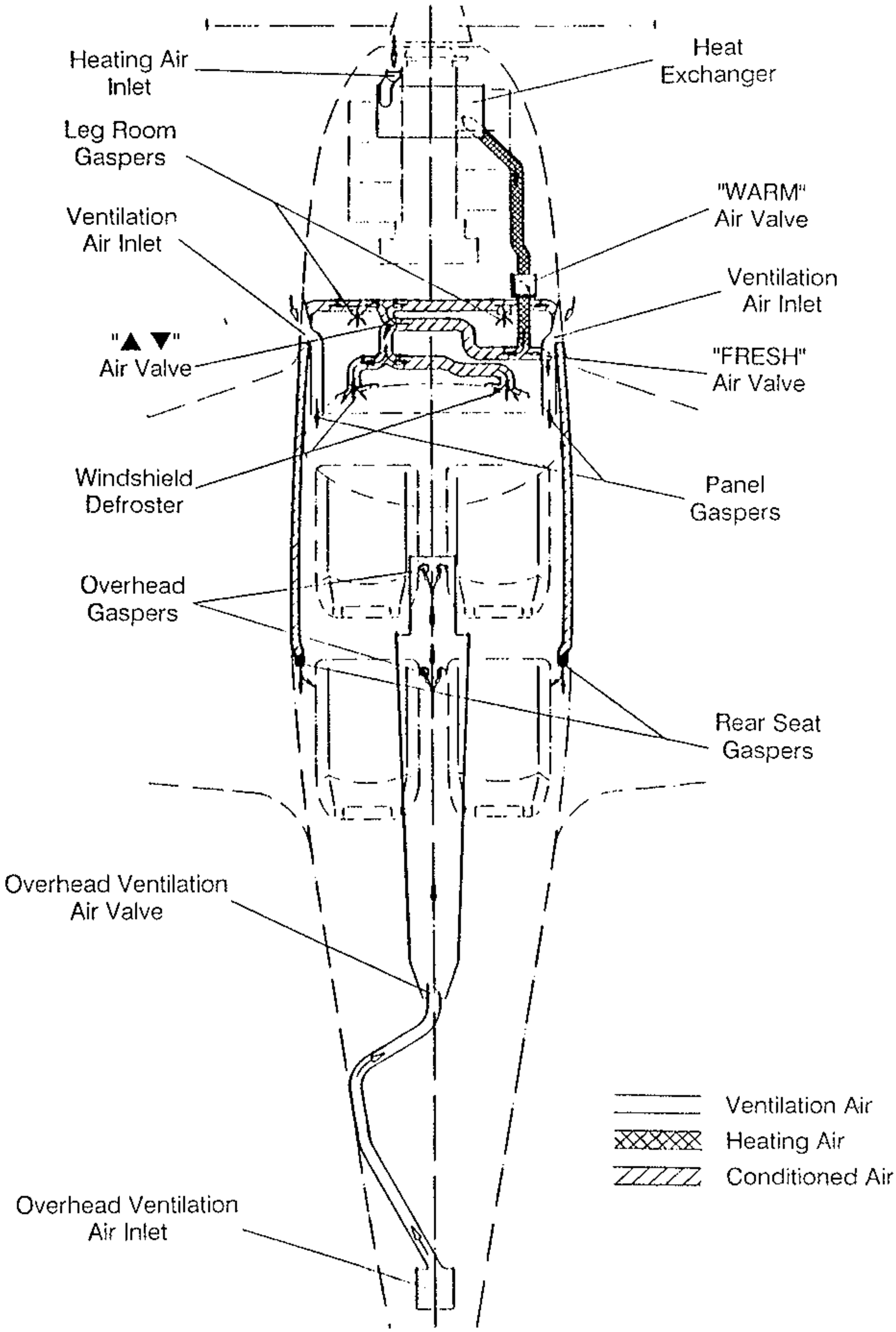


Fig. 7-15: Cabin Heating and Ventilation System Schematic

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LIST OF CIRCUIT BREAKERS

Designation	Protected Units
Power Plant Bus	
BAT	Battery
ALT	Alternator incl. Excitation
ALT EXITAT	Alternator Excitation
STARTER	Starter Relais
INST1	Alternator Warning Light, Fuel Gauge -left tank, Multifunction Gauge: Oil Pressure and Temperature, Cylinder Head Temperature, Fuel Pressure
INST2	Oil Pressure Warning Light, Fuel Gauge- right tank, Trim and Wing Flap Position Indicator
FUEL PUMP	El. Fuel Pump incl. Annunciator Light
VOLT	Voltmeter
AMP	Ammeter
AMP	Ammeter
Systems Bus	
GEAR ACT	Hydraulic Power
GEAR RELAY	Landg. Gear Control incl. Annunc. Lights and Warng. Horn
FLAP ACT	Wing Flap Motor
STALL WARN	Stall Warning incl. Annunciator Light and Warning Horn
INT LIGHT	Avionics, Instrument and Panel Lighting
DOME LIGHT	Overhead Cockpit Illumination and Ventilation Blower
TURN/SLIP	Turn and Bank Indicator
TEST FUNC	Test Function of Annunciator Lights incl.Stall Warning Horn
Avionics Bus	
AUDIO SPKR	Audio Control Panel - Loudspeaker
AUDIO HEADPH	Audio Control Panel -Headsets
COM/NAV 1	Com/Nav No. 1 incl. HSI
COM/NAV 2	Com/Nav No. 2
DME	DME Receiver and Indicator
ADF	ADF Receiver
XPDR	Transponder
ENC ALT	Encoding Altimeter
GYRO	Slaved Directional Gyro
HORIZ	Artificial Horizon
AVIONIC BLOWER	Avionics Blower
RMI	Radio Magnetic Indicator
FUEL COMPUT	Fuel Computer
A/P	Autopilot System incl. PITCH/TRIM Circuit Breaker
ALERTER	Acoustic Autopilot Warning System
PITCH/TRIM	Elecrical Elevator Trim System

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