The Airplane
The PA-60-700 Aerostar Superstar is an all metal, six place, fully retractable tricycle gear, mid-wing, turbocharged and pressurized twin engine monoplane of semimonocoque construction.

Airframe
The airframe of the PA-60-700P Aerostar is of conventional all metal design utilizing no unusual materials or processes in its construction. Major structural components are of semi-monocoque design using relatively thick skins which results in more uniform contours and fewer stiffener type parts. All external skins are flush riveted, with the exception of portions of the control surfaces and flaps which employ a "low profile" type rivet.

Fuselage
The fuselage structure is designed to provide a uniform cross-section for the length of the cabin. The primary structure of the cabin area is frame-longeron, stiffened sheet aluminum with skin doublers and gussets at cutouts and discontinuities.

The cabin doors and emergency exit are reinforced for pressurization loads and incorporate redundant locking pins.

Windows and windshield are of "stretched" acrylic composition for resistance to cracking or crazing.

Wings
Flight and ground loads on the wings are carried primarily by two main spars which extend full span and are continued through the fuselage by means of a wing carry-thru center section. An additional rear spar extending from the wing root to tip provides mounting points for aileron and flap surfaces. The wings are attached to the fuselage-wing carry-thru structure by means of bolted multi-lug fittings. The wing structure outboard of the nacelle from the leading edge to the rear spar is sealed to provide integral fuel tanks. Wing flaps are of the single slotted fowler type, supported at three points. Each "Frise" type aileron has three hinge points.

Engine And Accessories

ENGINE COOLING
Two Avco Lycoming TIO-540-series, 350 horsepower engines are installed on the airplane. They are six cylinder, counter rotating, low compression, horizontally opposed, direct drive, turbocharged, wet sump, fuel injected engines, driving Hartzell three bladed, constant speed, full feathering propellers. Manually controlled cowl flaps are installed to provide control of engine cooling. Cooling air enters the cowling, is directed around the cylinders by cylinder baffles, through the accessory section, and is exhausted through exhaust chutes. Two blast tubes direct cooling air to the magnetos, one blast tube is on the rear baffle, and the second is on the intercooler ramp.
ENGINE OIL SYSTEM
The engine employs a full pressure, wet sump lubrication system. The sump is filled through a combination dipstick oil filler cap. Lubricating oil is drawn through the oil sump inlet (screen by the engine oil pump and directly to the oil cooler and a thermostatic bypass valve. When engine oil is cold, the thermostatic bypass valve will open allowing oil to flow directly to the full flow oil filter bypassing the cooler. As the oil warms up, the bypass valve will close thereby forcing more oil to circulate through the cooler prior to entering the oil filter. From the oil filter, the oil passes through an oil pressure relief valve which regulates system oil pressure. The regulated oil is then routed through the main oil galleries to the various engine bearings and piston oil cooling nozzles, valve mechanisms, and moving parts. Gravity returns the oil to the sump.

The turbochargers are also lubricated by the regulated oil from the engine system. Oil circulated through the turbochargers is returned to the sump by a scavenge pump attached to the hydraulic pump accessory pad. Oil from the oil pump is also supplied directly to the waste gate control system. The engine oil system utilizes a drawn cup type large capacity oil cooler mounted on the nose cowling for temperature control.

ENGINE INSTRUMENTS
Instrumentation for each engine consists of mechanical oil pressure; electrical oil temperature and electrical cylinder head temperature presented on a 3-way combination gauge; mechanical fuel pressure gauge; electrical turbine inlet temperature gauge; electrical fuel flow gauge; electric tachometer and a mechanical manifold pressure gauge. The gauges are placarded as to their operating limitations.

IGNITION SYSTEM
Each engine is equipped with a pressurized dual magneto ignition system. Compressed air is taken from the intercooler to pressurize the magnetos for smoother operation at high altitude. Additionally, a starting vibrator system simultaneously retards and intensifies the spark for easier starting. Each engine has its own combination ignition-starter switch. Moving the switch to the spring-loaded START position grounds the right magneto, activates the starter vibrator and engages the starter. When the engine starts, release the starter switch and allow it to spring back to the BOTH position, disengaging the operation of the starter vibrator system and un-grounding the right magneto.
AIR INDUCTION SYSTEM
Induction air is directed to the fuel injection regulator via two filtered turbocharger inlets mounted on the rear engine baffles which feed compressed (turbocharged) air to a common intercooler mounted on the engine firewall. The intercooler cools the compressed air prior to entering the fuel injection regulator by drawing free stream air from over the top of the engine and exhausting it through the intercooler tunnel exit over the trailing edge.

Two manually operated alternate air doors are located in the intercooler tunnel aft of the intercooler. Each door supplies warmed air to its respective turbocharger inlet downstream of the filter. There is no need to close the bleed air valve when alternate air is selected, since the alternate air is drawn from an uncontaminated source.

FUEL INJECTION SYSTEM
The engine is equipped with a Bendix RSA-IOEDI fuel injection system. An engine-drive fuel pump supplies fuel under pressure to the fuel injection regulator, which measures air flow and meters the correct proportion of fuel to a flow divider. The flow divider then directs the fuel to each of the individual cylinder injector nozzles. A fuel vent system provides a common reference vent pressure to the fuel pressure gauge, engine-driven fuel pump and injection nozzles. The vent source is taken downstream of the turbochargers to ensure proper vent pressure during turbocharger operation.

TURBOCHARGING SYSTEM
The Aerostar turbocharging system provides engine capability to maintain the takeoff manifold pressure of 42 in. Hg. to approximately 16,500 feet while providing bleed air to pressurize the cabin. (See BLEED AIR SYSTEM in this section.)

The turbochargers are exhaust-gas-driven engine accessories which raise the pressure of the induction air delivered to the engine. There is one turbocharger unit for each engine exhaust stack. Each unit consists of a compressor and a turbine connected by a common shaft; the compressor supplies pressurized air to the engine for high altitude operation and to the cabin for pressurization. The turbine utilizes the flow of exhaust gases to drive the compressor. Turbocharging is controlled by means of an automatic turbo control system consisting of a controller and an actuator. The controller senses the engine manifold pressure selected and compressor discharge pressure across a diaphragm then actuates an oil metering valve that regulates oil pressure within the actuator. The actuator is mechanically linked to the waste gate valves located in each exhaust stack. The pilot gives the controller the first input by moving the engine throttle control to the desired manifold pressure. As this is being accomplished the controller senses a ratio change between manifold pressure and compressor discharge pressure. The controller is programmed to maintain a certain ratio. Therefore, the internal diaphragm of the controller now moves seeking this ratio, which in turn moves the oil metering valve allowing oil pressure to reposition the waste gate valves. As the waste gate valves move toward the closed or open position, more or less exhaust gases are diverted to the turbochargers causing the turbines to increase or decrease RPM accordingly. Since the turbines are directly connected to the compressors, the compressor discharge pressure will raise or lower until the desired programmed pressure ratio is obtained. The actuator is provided with an oil drain line which returns engine oil to the sump in the event of leakage.
around the actuator piston seals. The actuator is also spring-loaded to move the waste gates to the non-turbocharged mode as a fail safe system.

ENGINE CONTROLS
All power plant controls are located on the control pedestal. The lever knobs are shaped to standard configurations, and the control levers are of different lengths so they can be readily identified by touch.

ENGINE OPERATION
Up to critical altitude the basic operational difference between a normally aspirated engine (non-turbocharged) and a turbocharged engine is that, with the turbochargers full throttle, manifold pressure does not decrease with altitude. Above the critical altitude of approximately 16,500 feet, the waste gates are fully closed causing certain operational characteristics which should be understood to fully utilize the capabilities of the turbocharging system:

- An increase in indicated airspeed will cause an increase in manifold pressure.
- Leaning the mixture will usually cause a decrease in manifold pressure.
- A decrease in RPM will usually cause a decrease in MAP.
- Large or sudden power reductions with rich mixtures can cause complete loss of engine power. Power changes, especially power reductions, should be made slowly adjusting mixture controls as necessary for smooth engine operation.

As discussed under Air Induction System, this airplane is equipped with a manual alternate air induction system. Operation with the alternate air open is identical to previously discussed operation because the engine is still being turbocharged even though power is less at any given power setting because the induction air is obtained from the warm intercooler tunnel.

PROPELLERS
Each engine drives a TR W Hartzell, three bladed, hydraulically operated, all metal, constant speed, full-feathering propeller. A TRW Hartzell governing system maintains selected engine RPM regardless of engine load or aircraft attitude. The governor is adjusted to provide 2500 RPM for takeoff, consequently during full power ground static run-up the engine RPM can be less.

LANDING GEAR
The hydraulic actuated tricycle landing gear is fully retractable and is of the air-oil oleo type. Wheel well doors fully enclose the landing gear when in the UP position, and partially enclose the wheel wells when the gear is DOWN to help protect those areas from water spray, mud splatter, and melting snow or ice thrown up during taxi, takeoff and landing.

The landing gear handle located on the pilot's lower instrument panel, is echanically linked to the landing gear control valve under the cabin floor. Selecting UP or DOWN position directs hydraulic system pressure to the appropriate side of the landing gear actuating cylinders and to the main landing gear door control valves.
A spring-loaded solenoid prevents the gear handle from being moved to the gear UP position when the airplane is on the ground. The solenoid is energized to unlock/the handle by a squat switch mounted on the nose landing gear scissors. The squat switch is activated as soon as the nose gear strut is extended such as rotation at takeoff, or if the airplane is on jacks. Should the solenoid fail, a small tab protruding through the handle slot allows the pilot to override the locking feature by pushing the tab to the right as the handle is activated to the gear UP position.

NOTE
The gear handle safety lock may also become unlocked if the airplane is sitting on the ground in an extreme tail-down position which could extend the nose gear strut and allow the squat switch to actuate the solenoid when the battery switch is turned ON.

The door control valves are actuated by the main landing gear, mechanical linkage and direct system pressure to the main landing gear door actuating cylinders to hold the wheel well doors closed when the gear is in either the fully extended or fully retracted position.

WARNING
The gear WILL NOT OPERATE in the event of a hydraulic system pressure loss.

Four landing gear position lights are installed on the left instrument panel. A single amber light is located above a cluster of three green lights. It illuminates and stays on when all three landing gears are in the full UP position. The three (3) green lights are arranged in a cluster with the nose gear light on top, and the respective main landing gear lights below. Each green light will only illuminate when its respective landing gear is in the full DOWN and LOCKED position. Time delay for a gear light to illuminate indicates the respective landing gear is in a transient position.
The nose landing gear green light is wired to a switch installed on the gear handle mechanism. For this light to illuminate, it is necessary that the nose landing gear be in the down and locked position and the gear handle also being the full DOWN position.

A landing gear warning horn will sound (if **Use Alarms** is selected on the Load Manager) if the landing gear is not down and locked and either throttle is retarded below approximately 15 inches MAP. The horn is activated by switches connected to both throttle controls.

The gear horn will also sound if the battery switch is ON and the gear handle is not in the full DOWN position when the airplane is on the ground. This system is activated by the nose landing gear switch.

A parallel electric circuit also connects the throttle gear horn switches to a red GEAR warning light on the annunciator panel which will illuminate if the throttles are below approximately 15 inches MAP and the gear is not down and locked.

**HYDRAULIC SYSTEM**
The hydraulic system provides pressure for operation of the landing gear, main landing gear doors, and flaps. A hydraulic reservoir located just aft of the baggage compartment supplies hydraulic fluid to the hydraulic pump installed on the right engine. An electrically operated shutoff valve is installed in the fluid supply line to shut off fluid flow to the engine driven pump in the event of an engine fire, and helps facilitate maintenance of the system.

The shutoff valve switch is located on the right instrument panel. Operation of the right engine with the hydraulic valve closed will cause damage to the hydraulic pump and the drive shaft to shear.
Hydraulic pressure is controlled by a pressure regulator, for an operating pressure of 1100-1400 PSI. A relief valve prevents system pressure from exceeding 1500 PSI. An accumulator dampens pressure pulsations in the system, and provides a limited pressure source, should the pump fail.

A direct read-out pressure gauge is located on the right instrument panel. In addition, an amber annunciator light is provided in the annunciator panel. An auxiliary hydraulic pump (electric motor pump) is also provided, and controlled with a selector switch labeled "AUX HYD; ARMED-OFF." It is located on the far right side of the instrument panel.

With the switch in the "ARM" position, in the event of right engine failure or failure of the primary, right engine driven hydraulic system pump, the auxiliary hydraulic pump will supply hydraulic system pressure as soon as that pressure drops below 700 PSI. The auxiliary pump will run, and the amber annunciator light will be on, until the hydraulic system pressure is increased to 1000 + PSI.

NOTES
The primary purpose of the auxiliary hydraulic system is to provide a means of extending landing gear and flaps in the event of a right engine failure or engine driven hydraulic pump failure. The auxiliary hydraulic pump is capable of retracting the landing gear, however the retraction time may be increased and the gear may not retract above the one engine inoperative best rate of climb speed of 116 KIAS.

Flight Control System

PRIMARY CONTROL SYSTEM
The Aerostar incorporates dual primary flight controls utilizing conventional elevator, aileron and rudder systems. The primary movable control surfaces are operated by means of a system of push-pull tubes, torque tubes and bell cranks. No cables are used anywhere in the system. All bearings within the system are permanently lubricated and require no servicing, and no adjustments should ever be required after the airplane leaves the factory.

TRIM CONTROL SYSTEM
The elevator and rudder trim tabs are operated by 28V D.C. trim motors which are actuated by pedestal mounted; spring-loaded rocker switches. The left elevator trim tab has an electric motor, the right elevator trim tab is driven by a flex cable.
interconnecting the two trim tabs. The aileron trim tab is a fixed type and is factory adjusted for level cruise flight. No readjustment of this tab should be required after the airplane leaves the factory.

**WING FLAPS**

Fowler type flaps are operated by two hydraulic actuating cylinders. The control valve for these cylinders is actuated by the panel mounted flap selector handle which allows selection of 0° to 45° as desired.

The wing flap handle on the instrument panel is mechanically linked to a control valve under the cabin floor, which, in turn, directs fluid to the flap actuating cylinders. A neutral position allows the pilot to stop the flaps at any intermediate setting. Flow control valves are installed in the flap system to provide equal fluid flow to the left and right flap actuators thereby ensuring symmetrical flap extension and retraction.

A restrictor is also located at each cylinder's downline port to prevent a rapid asymmetric condition from occurring should the downline rupture when the flaps are extended. Actuation of the flaps results in minimal trim change. The console mounted flap position indicator (above) receives electrical input from a flap's position transmitter linked to the flap actuator.

**Fuel System**

**FUEL TANKS**

Fuel is stored in large area, shallow, integral wet wing tanks located outboard of the engine nacelle, and a deep rectangular bladder-type fuselage fuel tank located between the rear cabin bulkhead and the forward bulkhead of the baggage compartment. Total capacity of this interconnected system is 215 gallons. Each wing fuel tank has a capacity of 65 gallons usable when fueled to 0.6 in. below filler neck. The fuselage fuel tank has a capacity of 85 gallons usable.

A multiple sump assembly is installed below the fuselage fuel tank. The center sump is the low point for the fuselage tank, and the two wing sumps are the low points of each wet wing tank. Each sump can be drained by depressing its respective drain valve located on the left lower side of the fuselage just aft of the wing.
Fuel is fed to the left and right wing sumps through check valve assemblies. There are two check valves installed in each sump. With the fuel selector in either wing or fuselage tank mode, the fuselage tank and wing tanks feed simultaneously through all check valves, each wing tank to its respective engine, with the fuselage tank feeding to both engines.

The check valves prevent backflow of fuel from one tank to another. A fuel strainer is located at the outlet of each wet wing tank on the inboard tank rib at approximately mid-chord and the fuselage tank strainer is located in the center sump. A fuel filter is installed in each engine fuel supply line between boost pump and engine.

**BOOST PUMPS**

One electric boost pump is installed in each engine fuel supply line to provide fuel pressure for starting and to ensure uninterrupted fuel supply to the engines at any other time (takeoff, landing, climb above 10,000 feet), and serves as backup for the engine-driven fuel pump. The control switch for each pump is mounted on the instrument panel in each engine switch grouping (above). Boost pumps are to be turned ON at first indication of fluctuating or low (29 PSI) fuel pressure/fuel flow, or climb above 10,000 feet. Engine operation, at other than idle conditions, in the 12 to 29 PSI fuel pressure range can result in an unacceptable fuel/air ratio and could result in shorter engine life, engine damage, or engine failure.

**CROSSFEED SYSTEM**

The fuel system incorporates a X-FEED (Crossfeed) supply system to correct fuel imbalance and lateral trim, and to utilize fuel from an inoperative engine wing tank for prolonged single engine operation. Crossfeed is selected by rotating either fuel selector switch (above) to the X-FEED (Crossfeed) position. Crossfeed selection allows an engine to draw its total fuel directly from the opposite wing tank bypassing the sump check valves.

Double X-FEED (Crossfeed) must not be selected except in emergency when the LOW FUEL Warning Light is illuminated. The fuel selector X-FEED (Crossfeed) position is to be used in level coordinated flight only. Each operating engine fuel selector must be in the ON position for takeoff, climb, descent, approach and landing.

Should it be determined or suspected that the fuselage tank fuel quantity has been depleted, maximum usable wing fuel is obtained in coordinated flight at a level or slightly nose up attitude. Avoid nose down descents (up to 22 gallons unusable in each wing).

The X-FEED (Crossfeed) warning light will illuminate if the landing gear is extended when one or both of the fuel selector switches are inadvertently left in the X-FEED.
FUEL QUANTITY INDICATOR
A capacitance type fuel indicating system is installed. The system is comprised of a probe in each fuel tank, a signal conditioner and an indicator incorporating an individual display for each of the three tanks, left wing, fuselage and right wing.

The indicator, located in the center of the panel, displays fuel quantity in U.S. gallons. Wing tank fuel quantities above 50 gallons are un-gaugeable. The fuselage fuel quantity gauge scale has a yellow arc from 0 to 12 gallons. Takeoff is prohibited with the fuselage tank indicated quantity within the yellow arc.

The fuel quantity system is designed and calibrated for highest accuracy when the airplane is in a normal cruise attitude. For this reason, when reading fuel quantity, the airplane should be in a laterally and longitudinally level coordinated attitude (approximately 2° nose up) for the fuel indicator to read accurately. The wing fuel tank quantity readings are extremely sensitive to airplane attitude and turbulence. During uncoordinated flight the low wing may read a lesser quantity than the high wing even if the actual fuel quantity in the low wing is greater.

The aircraft is equipped with an optional digital display fuel flow totalizer system which can be manually selected to indicate total fuel expended or fuel remaining. For the system to display fuel remaining the pilot must first input his estimated total fuel load and then the totalizer electronically subtracts fuel consumed from the initial quantity entered. Regardless of the system accuracy as a fuel consumed totalizer, it should never be used as a primary fuel quantity indicator because of the following:

1. The displayed remaining fuel quantity on the totalizer is based on the initial estimated fuel load.
2. The totalizer only shows estimated total remaining fuel load and not how the fuel load is distributed between the different fuel tanks.
3. The totalizer only subtracts fuel consumed by the engines and consequently cannot indicate fuel that was mistakenly not added or fuel lost due to possible fuel system leaks.

LOW FUEL WARNING LIGHT
A panel mounted LOW FUEL warning light, activated by a float switch located in the fuselage tank is provided as standard equipment. Takeoff is prohibited with the LOW FUEL warning light illuminated or inoperative.

The LOW FUEL warning light, located on the annunciator panel, first illuminates continuously when 12 gallons of fuel remain in the fuselage fuel tank and the light will remain on as fuel is depleted until fuselage fuel tank is again serviced above 12 gallons. In turbulence or during climbs and descents intermittent illumination may
occur when the fuselage fuel tank has slightly more than 12 gallons of fuel remaining.

The fuel system is designed so that during normal operation the wing fuel tanks should be empty when the fuselage fuel tank quantity is approximately 12-16 gallons. The LOW FUEL warning system operates independent of the primary fuel quantity indicating system. Power for the LOW FUEL warning light is provided through the annunciator panel circuit breaker.

ELECTRICAL SYSTEM

ALTERNATOR AND BATTERY SYSTEM
The Aerostar is equipped with a 28V DC electrical system. Power to operate the various circuits and components is provided by two self exciting 70 amp alternators; one on each engine. Maximum continuous output from each is limited to 60 amps, total dual alternator capacity is 120 amps. The voltage from each alternator is controlled by its voltage regulator and over-voltage relay. The voltage regulator senses the output of the alternator and automatically adjusts the alternator field current to increase or decrease the output voltage. Should one regulator become inoperative, the over-voltage relay will prevent any over-voltage condition of that alternator by interrupting the field current, shutting off the alternator. This alternator may be reactivated by turning it OFF (this resets the relay) and then back ON. Each alternator system is separated from the battery and the other alternator system by isolation diodes. These diodes prevent reverse flow of current from an operating alternator or the battery into an inoperative alternator.

An alternator warning light will illuminate when the respective alternator fails or is turned off. There are other system malfunctions that will illuminate the light, such as:
- Engine RPM too low-alternator not on the line.
- Failure of alternator output fuse and/or filter.
- Over-voltage relay open.
- Voltage regulator failure.

NOTE
Multiple failures may not illuminate the warning lights, however, these failures are readily determined by monitoring the voltameter. If system load exceeds 60 amps per alternator a circuit breaker should also pop, but the warning light will not illuminate. Additionally, there will be no annunciation if there is a total failure of the electrical system.

]individual alternator switches are provided to turn their respective system ON or OFF and to check alternator circuit operation. Should an alternator fail, or an engine not be running, the respective alternator should be turned OFF. Since the alternators are self exciting, battery power is not required for excitation.

One 24-volt lead acid battery, located just aft of the rear baggage compartment, provides power for engine starting and a reserve source of power in the event of dual alternator failure. Alternator and battery switches must be in the OFF position prior
to connecting external power. Since this will provide a direct connection to the main aircraft bus, care should be taken to ensure that all switches and equipment are turned OFF before connecting the external power. This will protect the individual systems from voltage transients from a poorly regulated external power source which can damage the airplane's electrical equipment. Prior to turning on any electrical equipment, check bus voltage does not exceed 30 volts. Leave all nonessential switches OFF until external power is disconnected.

VOLT-AMMETER

The volt-ammeter is located on the copilot's lower panel and reads either voltage or amperage. A three position amperage selector switch, mounted adjacent to the volt-ammeter, is used to select the circuit to be read on the meter. Total OPERATING EQUIPMENT amperage load is displayed when the' switch is in the BUS position. The appropriate ALTERNATOR amperage load is displayed by moving the switch either left or right.

If left and right alternator loads are added, the slim may be greater or less than displayed in BUS position. If greater, the battery is being charged; if less, battery is being discharged. The left and right positions are also useful in determining alternator load sharing, or paralleling. At lower load values (not to exceed 60 amps) one alternator may carry much more load than the other. This condition is acceptable if the lower load value alternator has any load indication and will carry the load when the higher load value alternator is turned off. This test is valid only when both engines are operating above 1800 RPM to assure 28V output for each alternator.

CIRCUIT BREAKERS

An equipment circuit breaker panel is located on this same lower right instrument panel. Each of the push-to-reset type circuit breakers will pop out when excessive current flows through the circuit it protects. After allowing the circuit breaker to cool for 1-3 minutes, it may be reset by pushing it in until it clicks.

NOTE

The circuit breakers are not only push to reset but also are of the pull to disable type, except for alternator and battery breakers. The battery and both alternators are protected by panel mounted 60 amp circuit breakers. After a 1 to 3 minute cooling period, these panel mounted breakers may be reset.
Exterior Lighting
Appropriate switches and circuit breakers on the instrument panel control all exterior lighting. Combination strobe/navigation lights are located in the wing tips and on the tail cone. Dual taxi/landing lights are recessed in the bottom of the nose section. Simultaneous operation of both taxi/landing lights for extended periods while on the ground should be avoided, as this may cause overheating and possible deformation of the acrylic lens.

**WARNING**
Strobe lights should not be operating when flying through cloud, fog or haze, since the reflected light can produce spatial disorientation.

Interior Lighting
The instrument panel is lighted by means of red/white lights located under the glare shield and side panels. This flood lighting is controlled by the Flood Lights switch on the pilot's lower panel. The avionics equipment, engine/navigation instruments and the magnetic compass are internally lighted are controlled by the Panel Lights switch located on the pilot's lower panel.

Instrument Panel
When properly equipped the instrument panel contains instruments and controls necessary for day or night IFR flight. The function and operation of instrument panel features can be found in the Panel section of the POH.

**WARNING LIGHTS**
The warning light system is composed of an annunciator light panel and test switch mounted in the glareshield. Various warning or caution signals from the aircraft are routed to the annunciator controller which in turn triggers the appropriate annunciator light. The TEST switch may be depressed any time electrical power is supplied to the aircraft to check bulb operation.

**NOTE**
The TEST switch is used to check all of the annunciator light bulbs simultaneously. It only tests the operation of the bulbs and not the respective system or related circuits.

If an annunciation illuminates, take appropriate corrective action.

**FLIGHT INSTRUMENTS**
The standard flight instrument group consists of a pneumatic directional gyro and artificial horizon powered by air pressure, a 28V D.C. turn coordinator and an altimeter, airspeed indicator, vertical velocity indicator, Horizontal Situation Indicator (HSI) and a clock/stopwatch.
**Pitot-Static Pressure System**

A pitot head is located on the left side of the fuselage forward of the instrument panel location. A static opening is located on each side of the forward fuselage. Both pitot and static pressures are transmitted to the airspeed, altimeter, and rate of climb instruments, as required, through the pitot-static system lines. The pitot head incorporates an electric heating element, which is activated by a pitot heat switch located in the lower pilot's instrument panel switch grouping.

**Pneumatic System**

The pneumatic system consists of two engine-driven dry air pumps, two solenoid operated/spring-loaded pressure regulating valves located in the inboard wing bays, a manifold with check valves, and an in-line filter. The system can operate fully with one pump inoperative.

**Inflatable Cabin Door Seal**

The air supply is tapped downstream of the in-line filter and fed into a pilot operated door seal control valve and subsequently into the inflatable seals. The door seal control valve is comprised of a solenoid shutoff valve, a check valve (for airplanes equipped with optional surface de-ice, the check valve is replaced with a solenoid shutoff valve) and a pressure switch set at 12 ± 1 psig. When the seal inflation mode is selected by the pilot, the pneumatic system pressure regulating valve is energized by the pressure switch from the door seal control valve and air is pumped into the seals until the pressure regulating valve is de-energized allowing excess pumped air to discharge overboard and system pressure to return to normal. The door seal switch located on the pilot's console will show DEFLATE regardless of modes selected until the emergency exit handle has been properly secured.

**Gyro Pressure System**

The air from the pneumatic system in-line filter is routed under the cabin floor to an in-line pressure regulator located behind the instrument panel then to the gyros and overboard. A pressure gauge mounted on the copilot’s instrument panel shows the gyro system pressure in inches of mercury. The gauge also has L and R red indicators that come into view when the air supply from either the left or right engine driven pump fails or drops below approximately 2.5 PSIG.
Heating, Ventilating And Defrosting Air Distribution System

During ground operation, outside air may be drawn into the cabin by activating the cabin ventilation fan switch.

The air is drawn through the dorsal fin air inlet and mixed with air drawn from the cabin through the plenum static air door. The air is then distributed through the overhead air outlets. If desired, the heater fan switch may be activated, thereby circulating outside air through the floor outlets into the cabin. During unpressurized flight, ram air is forced through the inlet in the base of the dorsal fin, mixed with bleed air from each engine and directed through both the overhead and the floor outlets, regardless of cabin ventilation fan and heater fan selection. As pressurized flight begins, the ram air check valve closes and bleed air becomes the only source of cabin air. Distribution of the bleed air is discussed elsewhere in this section. The cabin ventilation fan and the heater fan may be utilized on the ground or in flight to augment ventilation air flow if desired, for either pressurized or unpressurized mode of Operation.
Aircraft Systems
Description And Operation

Heater System
A 35,000 BTU, gasoline powered Janitrol heater supplies heat as required for cabin comfort. The heater is located in the fuselage, aft of the baggage compartment and obtains its fuel supply from the airplane fuel system fuselage tank. During continuous operation the heater fuel consumption is approximately 0.6 GPH. When pressurized, the heater air supply comes from the bleed air only.

Operation of heater system


Digital cabin temperature indication can be found above, and to the left of the heater controls.

For ground operation the battery switch must be ON. Operating the heater or cabin fan on battery only is not recommended because of high current drain. The cabin blower switch must be ON for heater operation whether on the ground or in flight.

Bleed Air System
The bleed air system provides the air required for ventilation and cabin pressurization. Compressor discharge air from each engine turbocharger is routed from the intercooler mounted on the engine compartment aft fire wall through a sonic nozzle and then through a two position electrically operated diverter valve assembly (cabin/overboard) mounted within the inboard wing leading edge. From the valve, the bleed air passes through a second heat exchanger and a check valve prior to entering the cabin.

The air entering the cabin is ducted to a common plenum located on the aft cabin bulkhead. From the plenum, air is distributed forward to the cabin overhead ventilation outlets and aft through the heater. From the heater, air returns to the cabin and is distributed through the floor level outlets (see Environmental Systems Schematic).
Propeller Synchrophaser
The synchrophaser system is designed to provide automatic propeller synchronization and to maintain a selected phase angle. The system will hold synchrophasing throughout all normal aircraft maneuvers if the manual RPM and throttle control settings are not altered.

The system consists of two (2) electric pulse generators which sense engine RPM and compares this information through a computer and signals a solenoid control on the right engine governor, thereby, "slaving" the right engine propeller speed to the left engine.

The system is activated by a two position (MANUAL and AUTO) toggle switch mounted on the console just above the propeller controls. In the MANUAL position the synchrophaser computer is not energized, but a small current is continuously applied to the right engine governor solenoid to hold it in a mean travel position. When AUTO is selected the full synchrophaser system becomes functional. Regardless of switch position, pilot can manually override the synchrophaser system by using the propeller controls.

A panel mounted pull type circuit breaker, marked SYNC, provides electrical system protection, and when out, it will completely de-activate the system.

NOTE
If the synchrophaser system becomes inoperative, the aircraft can be operated following basic Airplane Flight Manual Procedures providing the right engine propeller governor is mechanically readjusted to prevent RPM above 2500, and SYNC circuit breaker is pulled to the out position.

Correct operation:
- For Starting Engines set the synchrophaser switch to MANUAL.
- For Takeoff and Landing check that synchrophaser switch is in MANUAL position.
- For Climb/Cruise set the throttle controls as desired. Manually synchronize propeller controls. Select AUTO position on synchrophaser switch.

NOTE
Engine RPM must be within 30 RPM before switching from MANUAL to AUTO or system will not synchronize propellers. Prior to making power changes, select MANUAL position, select desired MAP and RPM, manually synchronize propellers and reselect AUTO position.
NOTE

If the synchrophaser system loses synchronization during the flight, select the MANUAL position and manually synchronize propellers. Wait for 45 seconds minimum then reselect AUTO position. For all single engine operations set the synchrophaser switch to the MANUAL position.

Engine Fire Detect Lights
The Engine Fire Detect option provides the pilot with the means to detect engine fires. It consists of a temperature sensing circuit in each engine accessory compartment which triggers a warning horn and indicator light inside the cockpit, in the event of excessive temperature level.

The engine fire detect test switch on the copilot’s upper panel is used to test the annunciator lights and warning horn. When this switch is depressed the warning horn and appropriate annunciator light should activate. This indicates the fire sensors are working.