ORION SERVICE DIGEST

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ORION

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FRONT AND BACK COVERS Patrol Squadron FIFTY exchanged their SP-5B Marlin seaplanes for P-3 Orions in the spring of 1967, marking the end of an era in the annals of Naval Aviation and beginning a new chapter in their professional career. Having carried out the last fully operational deployment of a seaplane squadron in the Pacific, PATRON FIFTY moved up to the most advanced — and their first land-based — ASW aircraft.

Commissioned originally at NAS Sand Point, Seattle as VP-892, the squadron first operated PBY-5A Catalinas as a reserve squadron, then, after being activated in the military build-up for the Korean conflict, in 1951 they were removed to San Diego, re-equipped with PBM Mariners, and deployed to Iwakuni, Japan.

By the time Korean hostilities ceased, VP-892 had been redesignated VP-50 and had become a regular member of the Seventh Fleet with its home base at NAS Alameda, participating in the regular WESTPAC deployment schedule. In 1956 PATRON FIFTY moved to NAS Whidbey Island and turned in their aging PBM's for new P5M-1 Marlins. For the remainder of the 1950's VP-50 ranged the Pacific from Whidbey Island to Japan to the Philippines. In May 1960, Iwakuni, Japan became the squadron's home port, and they began flying P5M-2's (SP-5B's), demonstrating such proficiency in the type as to win the Efficiency (E) Award for the 1961-62 competitive period, compiling over 30,000 accident-free hours during the 4-year tour.

PATRON FIFTY returned to San Diego in 1964 and was periodically deployed to Sangley Point, Republic of the Philippines during the next three years. Squadron assignments included participation in "Exercise Silver Lance" and performance of "Market Time" surveillance duty from Cam Ranh Bay.

Operating from NAS Moffett Field since transition to the P-3 in 1967, the squadron proved that they are no longer the "low and slow" class during a recent 6-month deployment to Sangley Point. Indeed, a PATRON FIFTY crew set a non-stop record from Atsugi, Japan to NAS Moffett Field, coursing the 4700 nautical miles in 11 hrs. 36 min., almost half an hour faster than any previous flight.

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P-3 ICE CONTROL SYSTEMS

FREQUENTLY AN aircraft's ice control systems provide the margin of performance that enables the aircraft to execute its mission and return home. A properly functioning ice control system is like an insurance policy, particularly to maritime patrol aircraft such as the P-3 Orion that routinely encounter icing conditions. This article describes the wing, empennage, propeller and power plant ice control systems, the bomb bay heating system, the instrument probe heating systems, the ice detector system, and some aspects of the aircraft bleed air system.

The P-3 is equipped with a composite ice control system that uses engine bleed hot air for some applications and electric heating elements for others. An ice detector system warns the flight crew when icing conditions are encountered. Bleed air is used to de-ice *or* anti-ice the wing leading edges, and to *anti-ice* the engine airscoops and engine inlet housing areas. Electrical heating elements are used to anti-ice and de-ice the propellers, empennage leading edges, and instrument probes. Recently the aircraft's bleed air system was modified so that engine bleed air could be used to heat the bomb bay. Several aircraft with this feature have already been delivered, and the remainder of those in service will be retrofitted to this configuration through incorporation of P-3 Airframe Change No. 142.

Aside from this, our subject systems vary only slightly throughout the Orion fleet. Some additional minor variations are discussed in related text throughout the article, but we cannot present the massive volume of information contained in official manuals, nor can we foresee future changes. For this reason, personnel must always consult the official manuals for new and more complete information, particularly when dealing with later (P-3C model) aircraft.

ICE DETECTION

P-3 aircraft are fitted with an ice detector that signals the presence and severity of icing conditions. The detector unit is located on the lower right side of the aircraft, forward of the nose wheel well, where its airfoil-shaped probe projects through the fuselage skin to sample the moisture content of the airstream. When ice accumulates on the probe, a pressure switch is actuated which energizes the "ICING" signal light on the flight station center instrument panel and the heating element that de-ices the probe. When the ice is melted, the pressure switch returns to its original position, extinguishing the signal light and de-energizing the probe heating element. The duration of the ice detector's de-icing period is about 3 seconds. If ice again accumulates on the probe, the cycle will repeat, and continue to do so until icing conditions are no longer encountered. The frequency of "ICING" light flashes will increase as the severity of icing conditions increases.

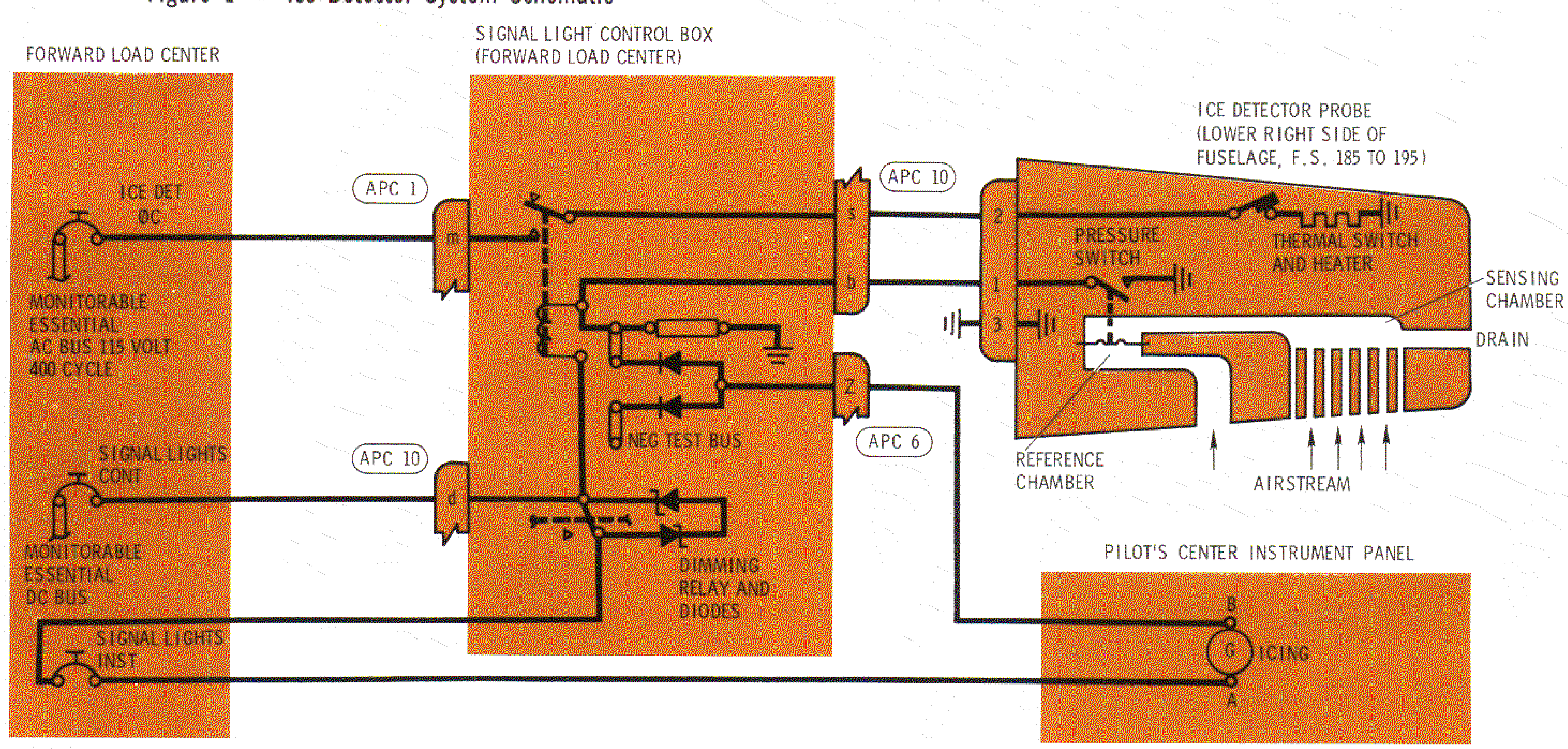
Figure 1 shows that the detector probe has two chambers separated by a diaphragm, one acting as a sensing chamber and the other acting as the reference chamber. Seven small orifices in the leading edge of the probe allow ram air to enter the sensing chamber, while one large orifice allows ram air to enter the reference chamber. The sensing chamber also has a drain hole to permit moisture to escape. Under normal (non-icing) operating conditions the air pressure in the sensing and reference chambers is statically balanced, but this balance is upset when ice crystals block the smaller inlet orifices that lead to the sensing chamber. The pressure differential produced during the icing moves the diaphragm, which mechanically actuates the detector's pressure switch, energizing the circuits to the "ICING" light and the probe heater. When the probe is de-iced, ram air again enters the sensing chamber and restores the pressure balance between the two chambers. The diaphragm returns to its original position and opens the circuit through the pressure switch, de-energizing the probe heater and the "ICING" light.

The ice detector electrical system employs both ac and dc power. The probe heater circuit is powered from the Monitorable Essential AC Bus ϕC, and the "ICING" light circuit is powered from the Monitorable Essential DC bus. Both circuits are routed through the Signal Lights Control Assembly which, with the system's circuit breakers, is located in the Forward Electrical Load Center.

INSTRUMENT PROBE ICE CONTROL

ANGLE OF ATTACK TRANSMITTER The angle of attack transmitter is installed on the right side of the aircraft at F.S. 300. Its slotted probe projects through the skin and senses the pitch angle between the aircraft centerline and the outside airstream. The probe is de-iced with a thermostatically-controlled electric heating element. The thermostat is located in the probe and adjusted to open the circuit at a temperature not higher than 121°C (250°F). 28-volt dc power is supplied to the heating element from the Monitorable Essential DC Bus via the ANGLE OF ATTACK INST PWR circuit breaker in the For-

Figure 1 Ice Detector System Schematic



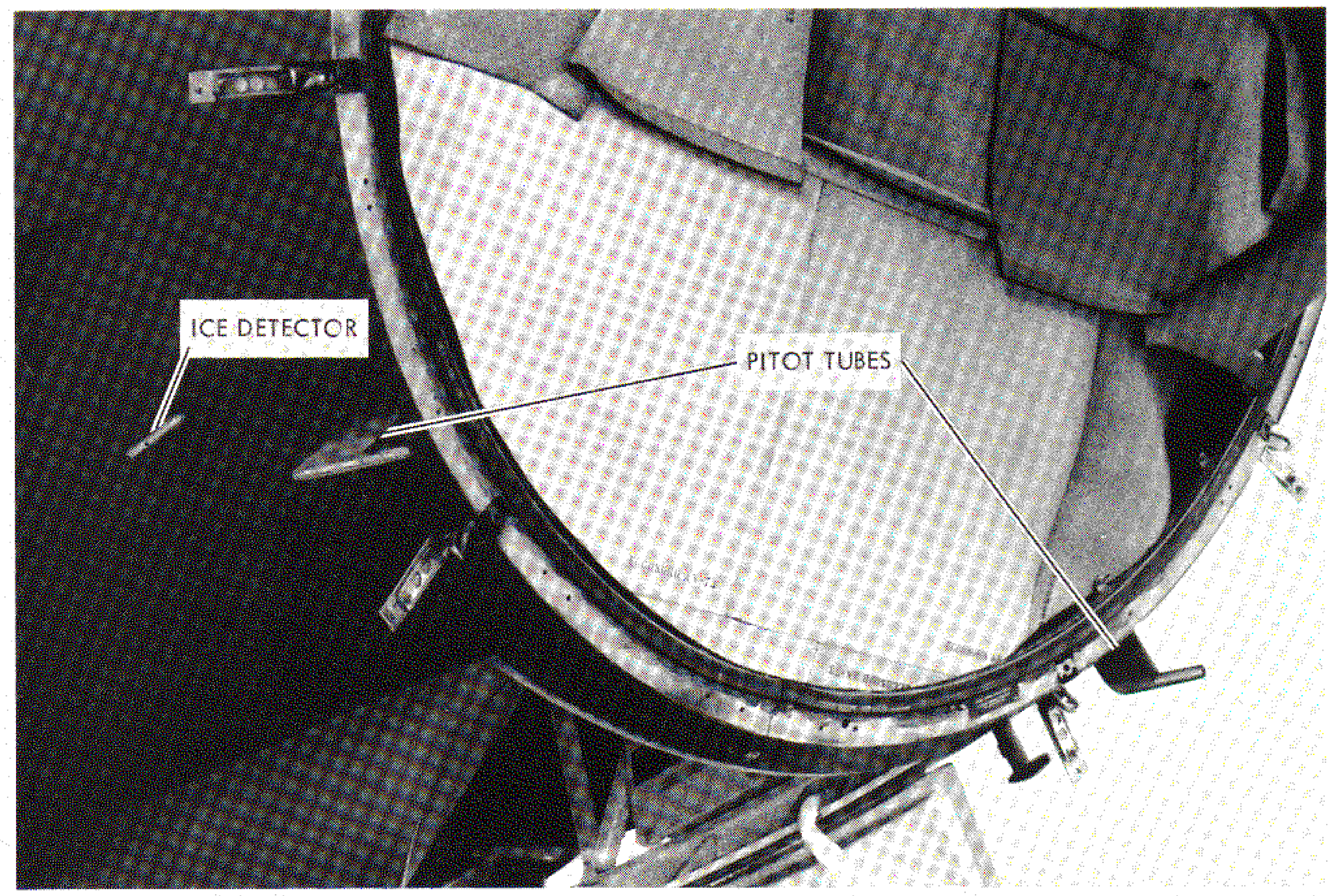
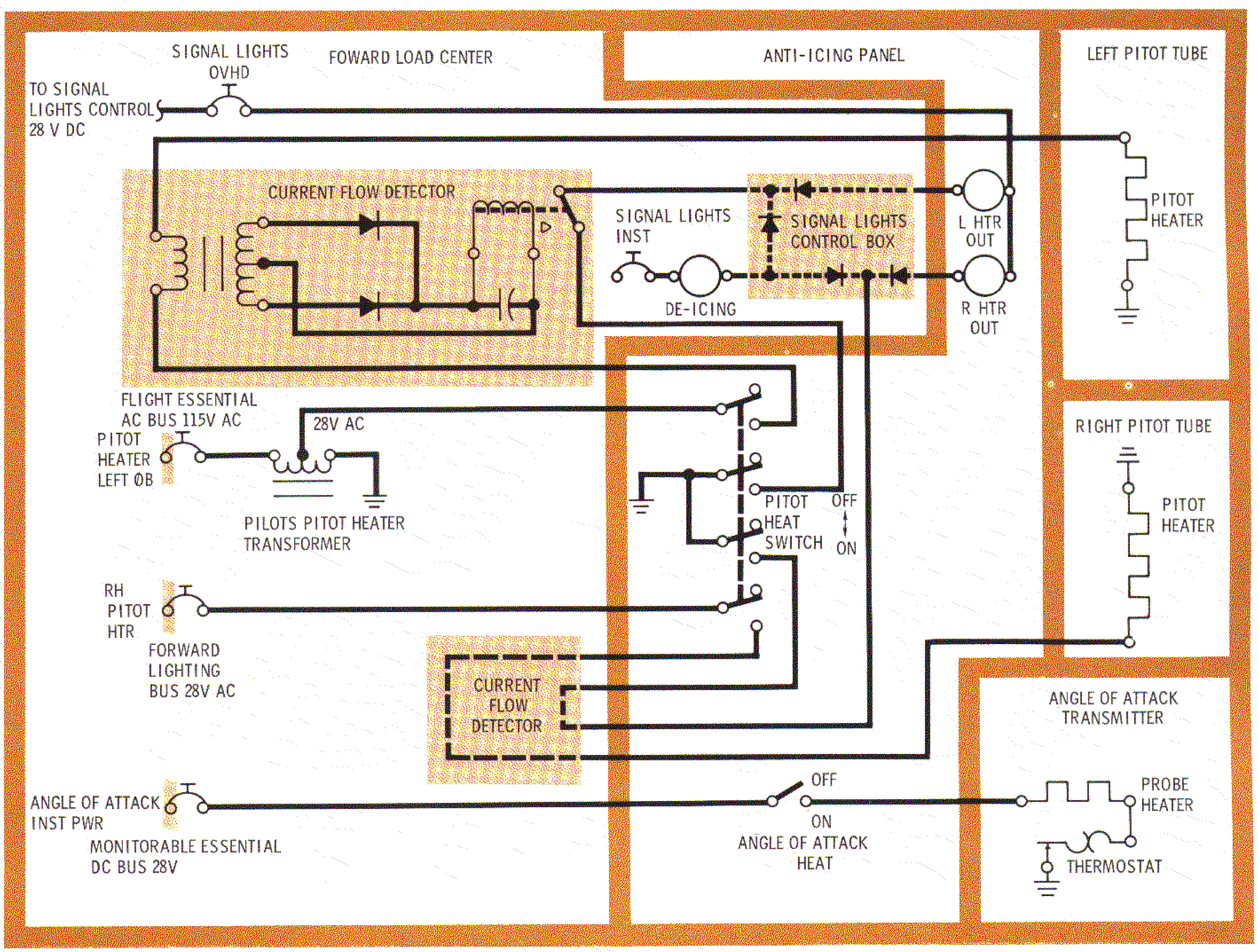
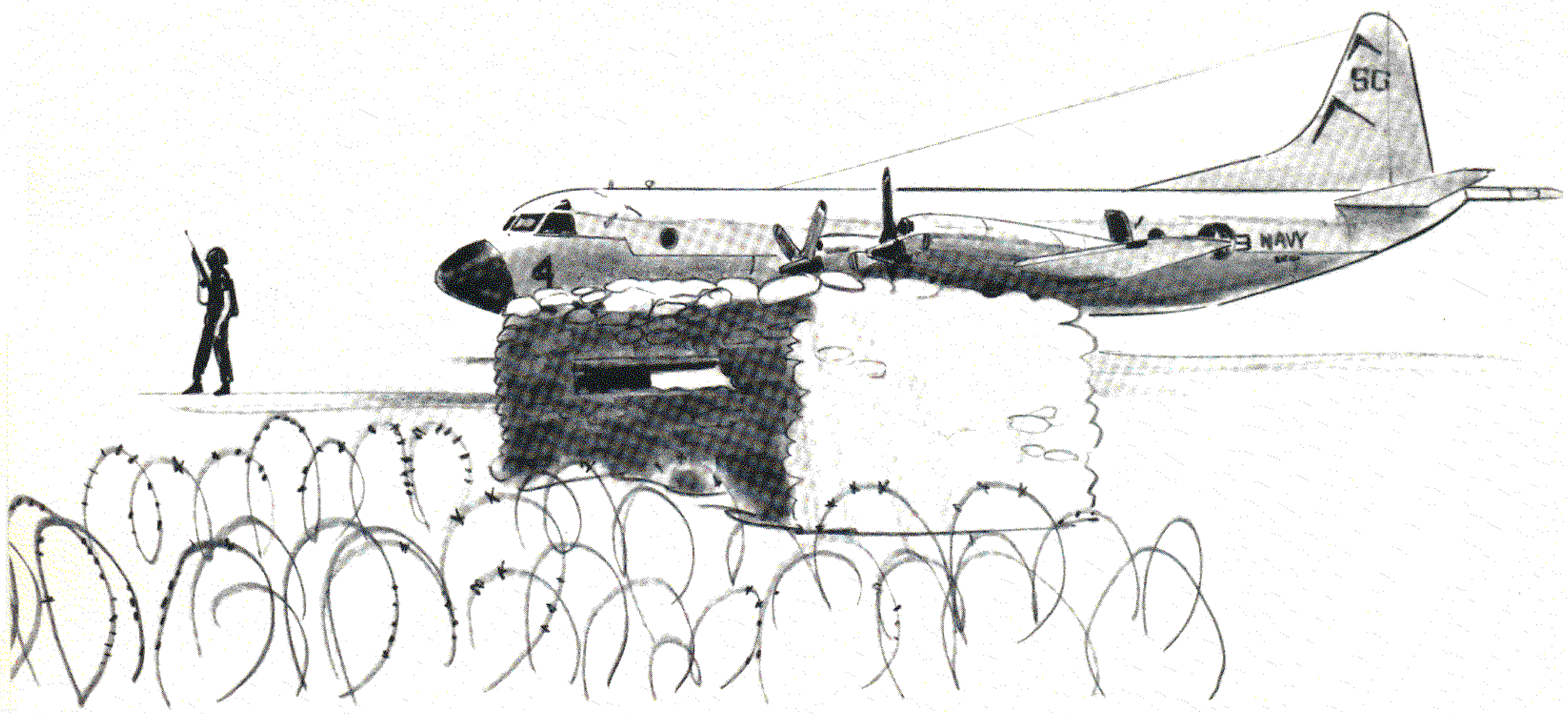


Figure 2
Ice Detector
and Pitot
Static Tubes
Installation

Figure 3 Instrument Probe Heating Systems Schematic





ward Load Center, through the Angle of Attack Heat control switch on the Anti-Icing Panel. According to the NATOPS Flight Manual, the heat control switch must be positioned "ON" prior to the engine start and left on until the engines are shut down upon completion of the flight. Figure 3 shows the angle of attack transmitter de-icing circuitry.

PITOT HEAT The pitot-static system's two tubes are mounted symmetrically on either side of the lower fuselage, just aft of the nose radome. Each tube is anti-iced by an electrical heating element that is integral with the tube and operates on 28 volts ac power. The tube heaters have separate power sources to ensure that failure of one tube's anti-icing circuitry would not render the aircraft's entire pitot-static system inoperative.

Power for the left (pilot's) pitot tube heater is supplied from the Flight Essential AC Bus via the PITOT HEATER LEFT ϕB circuit breaker in the Forward Load Center. Since power from this bus is 115 volts ac, it must be reduced to 28 volts ac by passing it through the Pilot's Pitot Heater Transformer. The low-voltage power is then routed through the caution light circuit's current flow detector to the tube's heating element. Power for the right (copilot's) pitot tube heater is routed from the 28-volt AC Forward Lighting Bus (also in the Forward Load Center), through the RH PITOT HEATER circuit breaker, through a caution light circuit current flow detector, to the tube's heating element.

Both pitot heat circuits are controlled by the Pitot Heat switch on the left overhead Anti-Icing Panel. Like the Angle of Attack Heat switch next to it, the Pitot Heat switch must be "ON" prior to engine start until engine shutdown at the conclusion of the mission. Circuitry for the pitot heat system is shown in Figure 3.

Each pitot tube has an individual "heater out" caution light circuit, but both circuits are also connected to the master "DE-ICING" caution light circuit. The pitot signal lights, "L HTR OUT" and "R HTR OUT", are on the anti-icing panel next to the Pitot Heat switch, and the master "DE-ICING" light is on the center instrument panel. Each circuit is powered from the SIGNAL LIGHTS OVHD circuit breaker on the Monitorable Essential DC Bus in the Forward Load Center. Power from the OVHD circuit breaker is routed through the signal light, through contacts on the pitot tube's current detector relay, through contacts on the Pitot Heat switch to ground. The signal light circuit will remain open as long as power passes through the detector transformer enroute to the pitot tube heating element, which keeps the detector relay energized. If either heating element's power supply is interrupted while the Pitot Heat switch is "ON," that current detector relay will de-energize and close the caution light circuit, illuminating the "HTR OUT" light and the master "DE-ICING" light.

WING ICE CONTROL SYSTEM

The wing ice control system is the branch of the aircraft's bleed air system that heats the wing leading edge surfaces by circulating hot compressed air through passages integral with the leading edge structure. This system can be operated manually to de-ice the wings or thermostatically to anti-ice them, however the NATOPS Flight Manual recommends that it be used primarily for de-icing for most efficient engine operation. Due to its high operating temperatures, full use of the wing ice control system is permitted only during flight.

Briefly, each engine can supply the wing ice control system with hot compressed air bled through dual ports at its final (14th) compression stage. This hot air is then distributed via the aircraft bleed air system manifold to one, two, or all three pairs of wing leading edge sections. Bleed airflow is controlled by a series of shutoff valves and modulating valves, and is applied to passages in the leading edge by jet pump assemblies called "piccolo tubes." After the air has circulated between the leading edge double skins, it is returned to the leading edge plenum chamber and eventually discharged overboard through louvers in the rear of each nacelle. A cross-ship duct connects the bleed air system of both wings so that hot compressed air from the engines on one wing can be supplied to the opposite wing duct for distribution as required to the wing ice control system, for engine starting, or for bomb bay heat. During flight the cross-ship duct is normally kept closed.

Bleed air system controls and instrumentation are located on the Engine Bleed Air and Anti-icing control panels, which are part of the flight station left overhead panel. The controls and signal lights are dc powered, the bleed air pressure and skin temperature indicators are ac powered. Other components of the system are located in the Forward and the Main Electrical Load Centers.

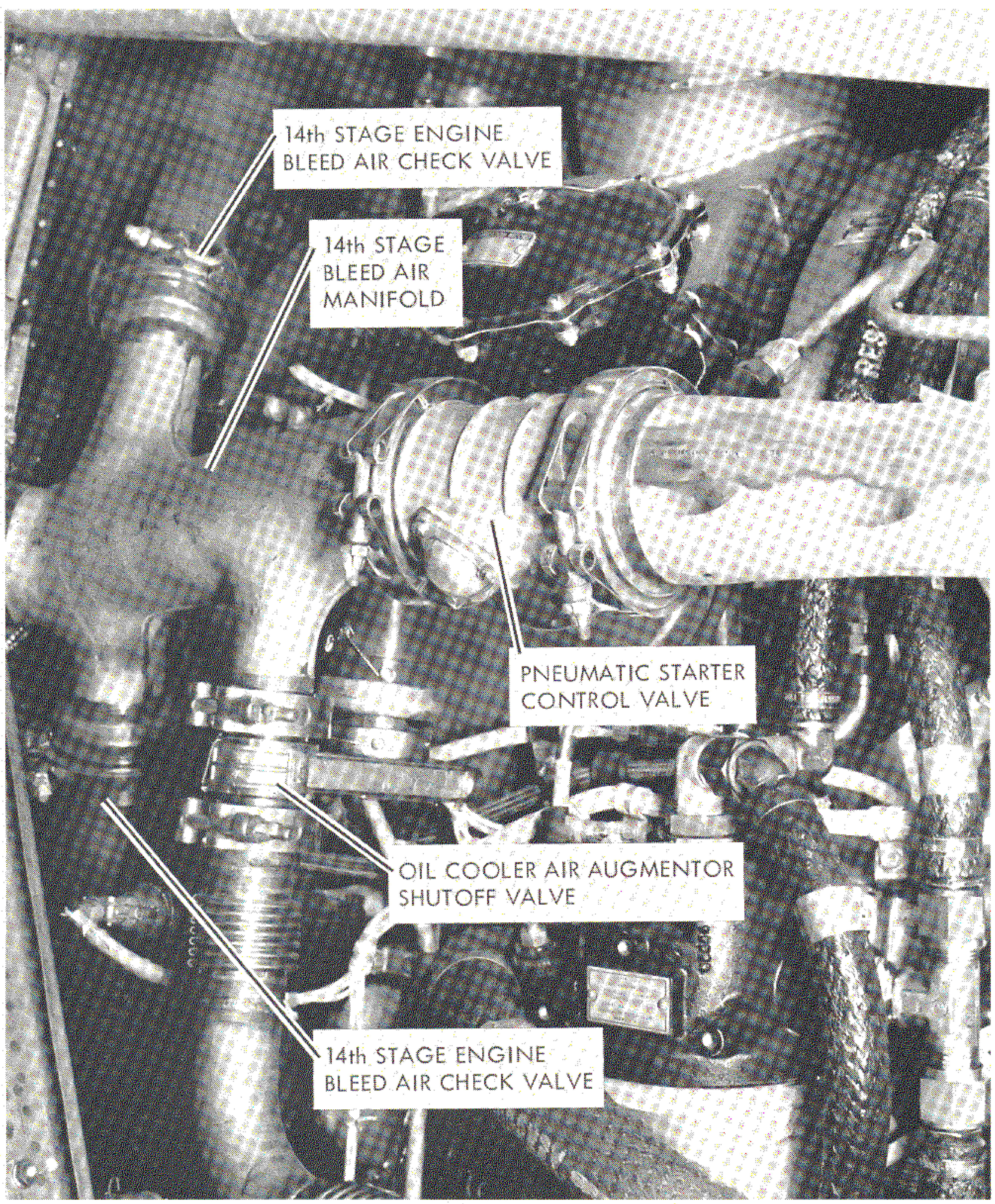


Figure 4
Engine
14th Stage
Bleed Air
Manifold

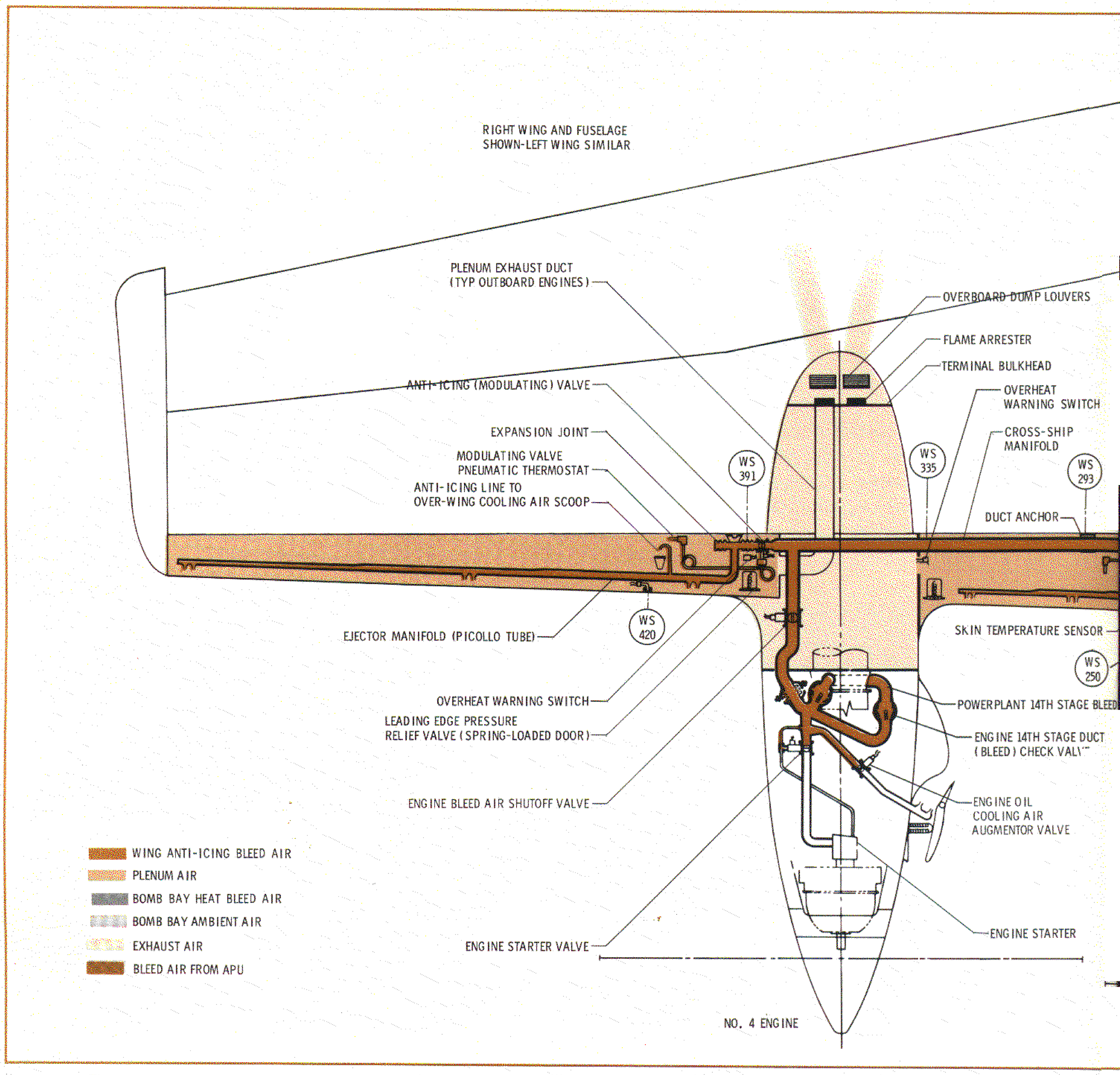
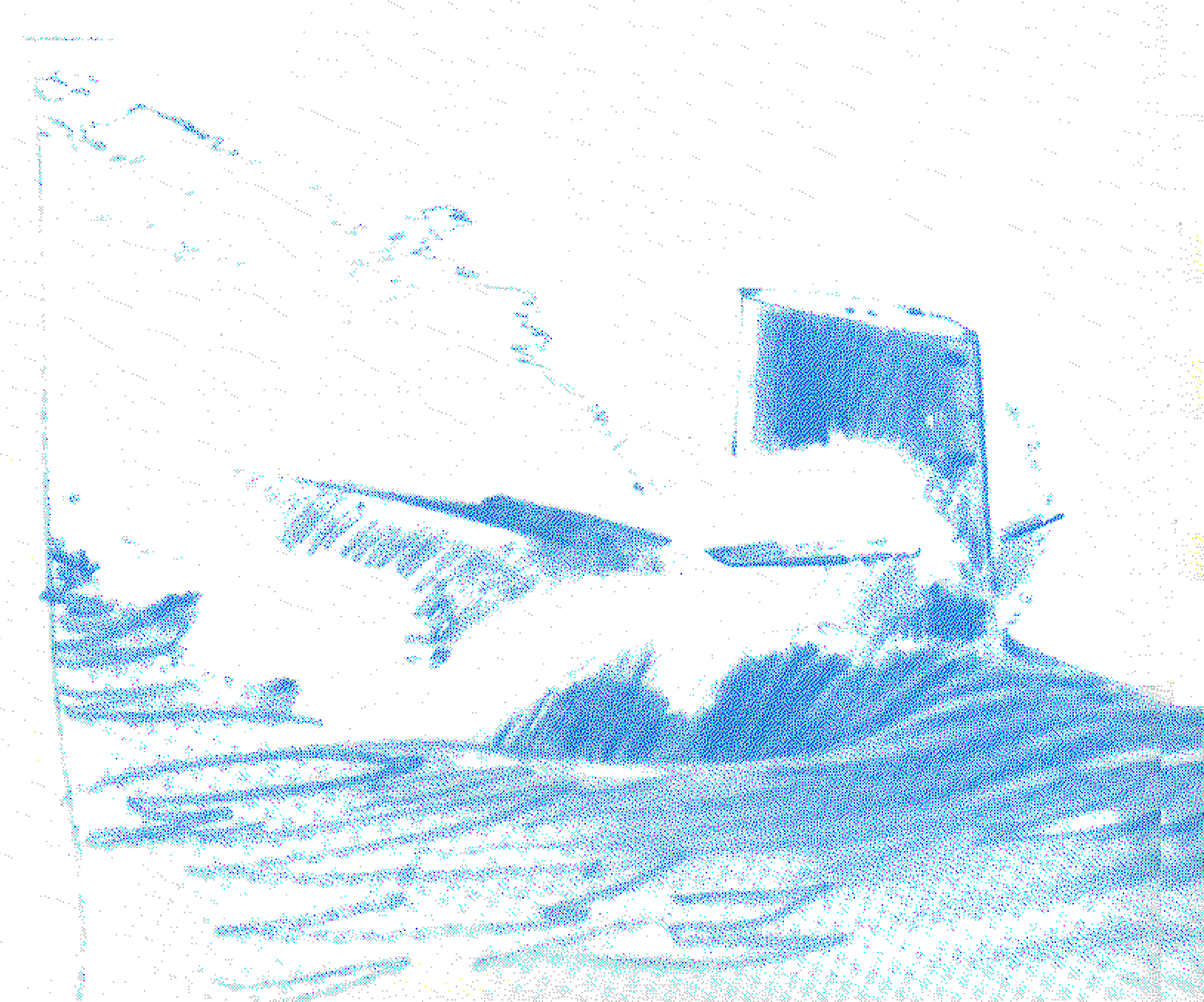
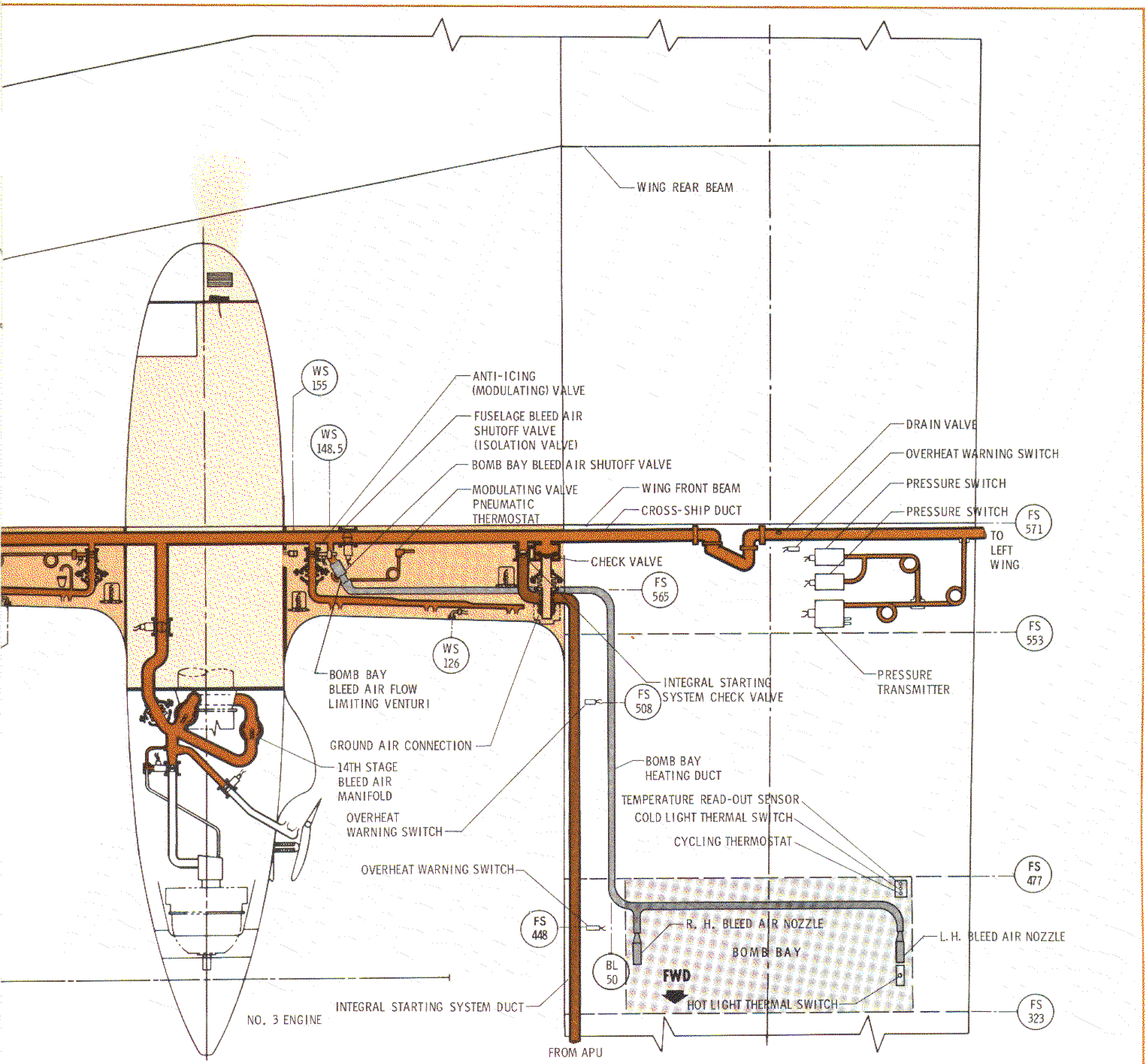


Figure 5 Bleed Air System Diagram

DESCRIPTION Discussion of the wing ice control system logically begins with a description of the aircraft's bleed air system, the source of the hot pressurized air used to anti-ice or de-ice the wing leading edge. The bleed air system has four sources of bleed air while the aircraft is airborne — each of the aircraft engines. The four engine bleed air systems are identical, thus we need to examine the system of only one engine in the following discussion and consider it as typical.

Engine bleed air is tapped from two ports located at the engine compressor's 14th stage and routed to the 14th stage bleed air manifold that distributes air to the wing ice control system and the bomb bay heat system, to the pneumatic starting system, and to the





engine oil cooling air augmentor (Figure 4). The mean bleed air temperature at the 14th stage manifold is about 282°C (540°F), although it may vary ±28°C (50°F) or more, depending upon the ambient air temperature and, to a lesser degree, upon the turbine inlet temperature. Check valves are installed between the manifold and the bleed air port to prevent bleed air from flowing back into the compressor section when the engine is being started.

Bleed air is ducted aft from the engine manifold, through the firewall, to the engine Bleed Air Shutoff Valve. This butterfly valve is operated by a reversible electric motor powered from the Start Essential DC Bus in the Forward Load Center. Mechanically actuated limit switches turn off the motor when valve travel reaches either full open or full closed. Valve operation is controlled by its individual switch on the Bleed Air Panel segment of the left overhead panel in the flight station. The "OPEN" signal light above the switch is illuminated by actuation of a limit switch within the valve assembly as the valve begins to open (switch actuation must occur by the time the valve has opened 5°). Wing ice control system and the bleed air system circuitry is shown in Figure 12.

After the air passes through the engine bleed air shutoff valve, it is discharged into the bleed air manifold that runs along the wing leading edge and combines with bleed air from the adjacent engine. Bleed air can also be routed from one wing to the other through the cross-ship duct that joins the left and right wing bleed air manifolds. Air flow through the cross-ship duct is controlled by two fuselage bleed air shutoff valves (also called fuselage isolation valves), one at each end of the duct. These valves, identical to the engine bleed air shutoff valves, are powered from the Start Essential DC Bus.

Bleed air is supplied from the bleed air manifold to each of the leading edge plenum chambers. Six modulating valves — one for the inboard, center and outboard section of each wing leading edge — regulate bleed airflow to the plenums. Each modulating valve (Figure 6) has a solenoid-operated pilot valve that must be opened before bleed air can pass through the modulating valve to the piccolo tube. The six pilot valve solenoids are controlled by three switches on the Bleed Air panel labeled LEFT & RIGHT WING ICE OUTBD, CTR, and INBD. These switches control symmetrical pairs of modulating valves, i.e. the inboard, center and outboard valves on each wing. When the switches are positioned "ON," the solenoids are energized by power from the Extension Main DC Bus in the Forward Load Center. Once the pilot valve is energized and bleed air has passed through the modulating valve into the leading edge section, the rate of airflow through the modulating valve is metered by the section's pneumatic thermostat.

Bleed air is ducted from the modulating valve to the leading edge section's piccolo tube (Figure 7)

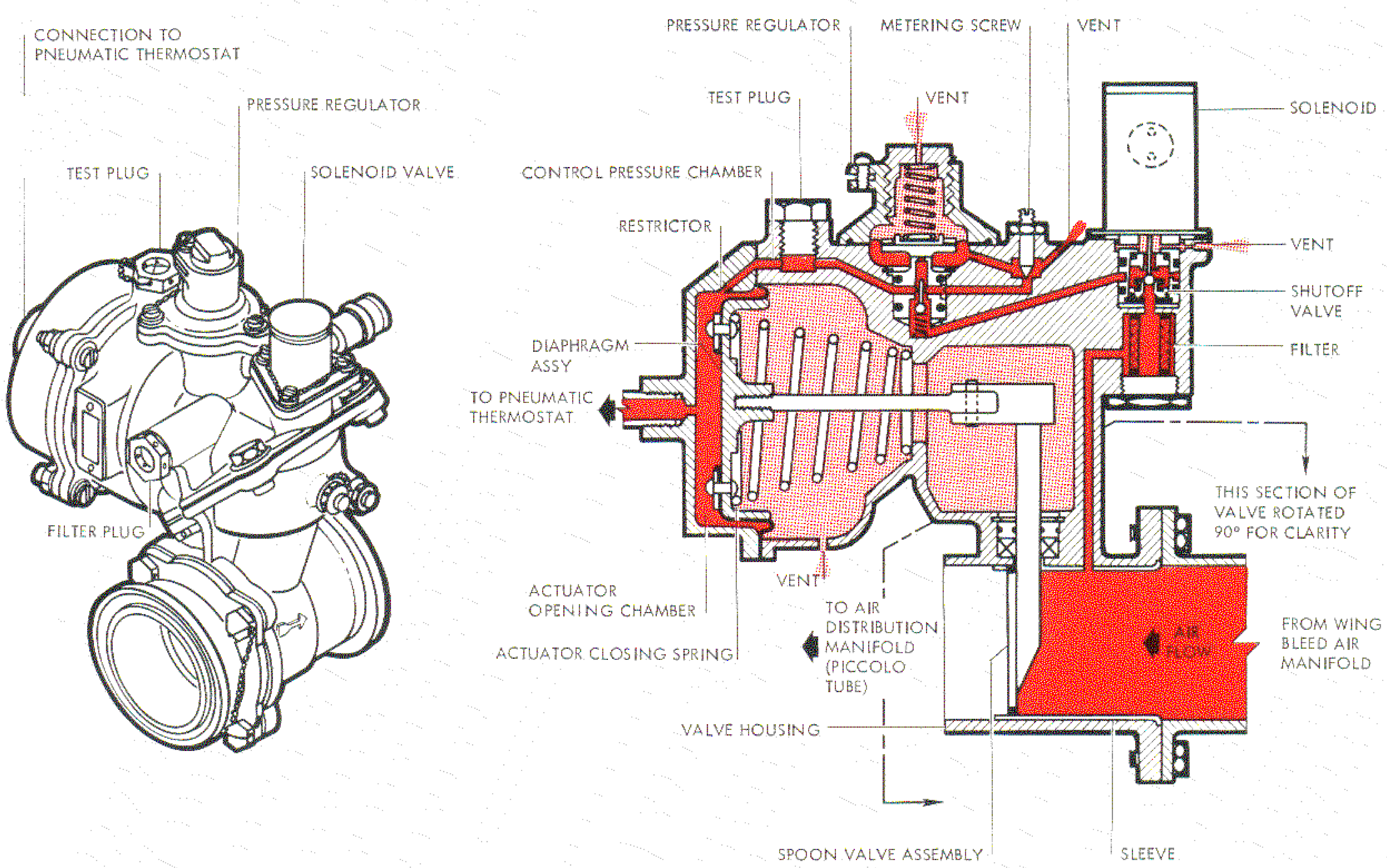
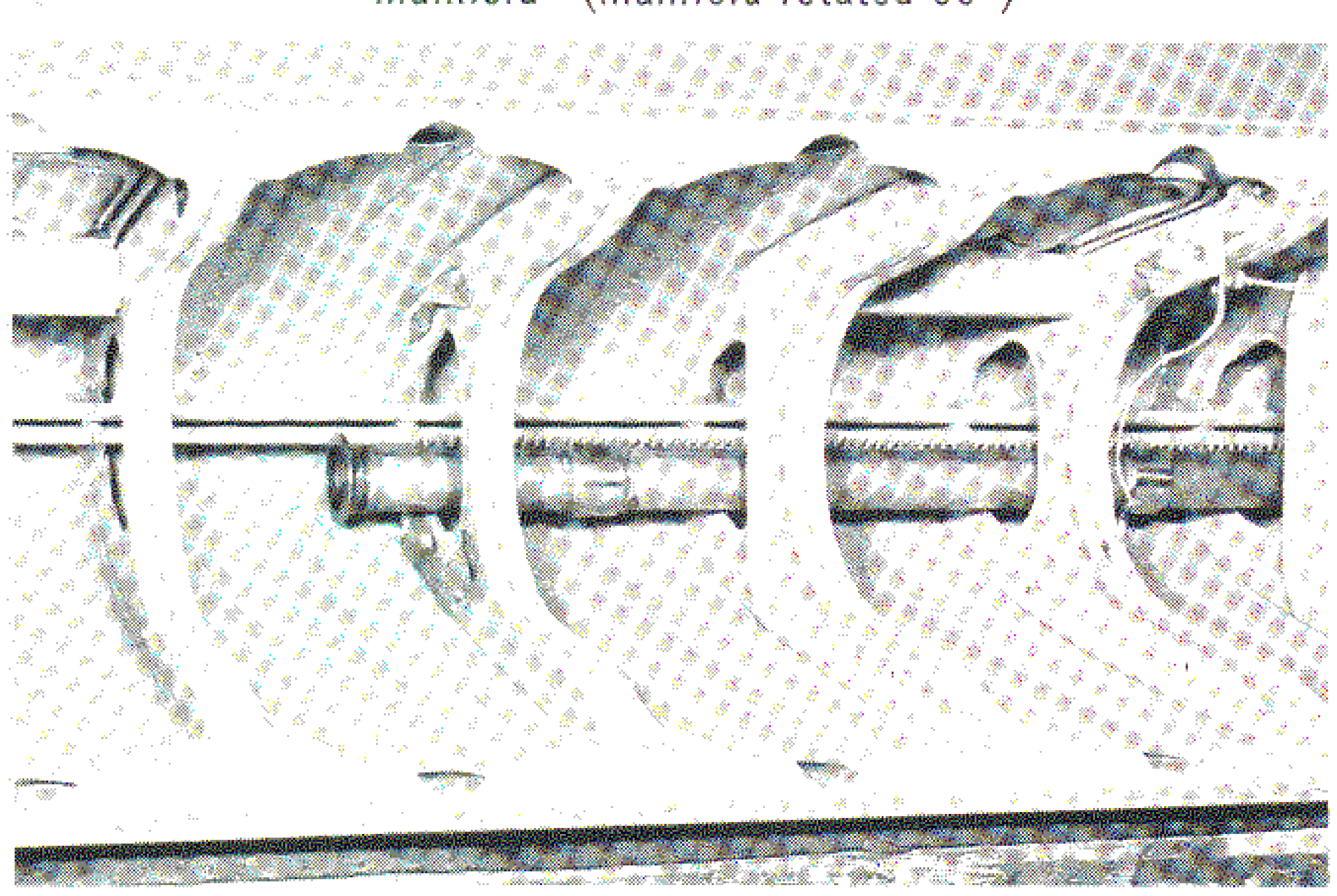


Figure 6 Wing Leading Edge Bleed Air Modulating Valve Diagram. (Section of valve rotated 90° for clarity.)

Figure 7 "Piccolo Tube" Air Distribution Manifold (Manifold rotated 90°)



which uniformly distributes the incoming air spanwise to the leading edge plenum. The jet pump action of the tube's nozzles dilutes the hot engine bleed air with recirculated air aspirated (drawn by suction) from the leading edge plenum area, reducing the air temperature significantly. This air mixture is directed through an inlet between the inner skins of the leading edge, it is circulated aft through the leading edge's upper and lower air passages, then it is discharged back into the plenum. Control temperature for the modulating valve is sensed by the pneumatic thermostat at the point air exits from the hollow-walled leading edge into the plenum chamber. The inboard and center plenum thermostats are set to restrict flow through their modulating valves at 49°C (120°F),

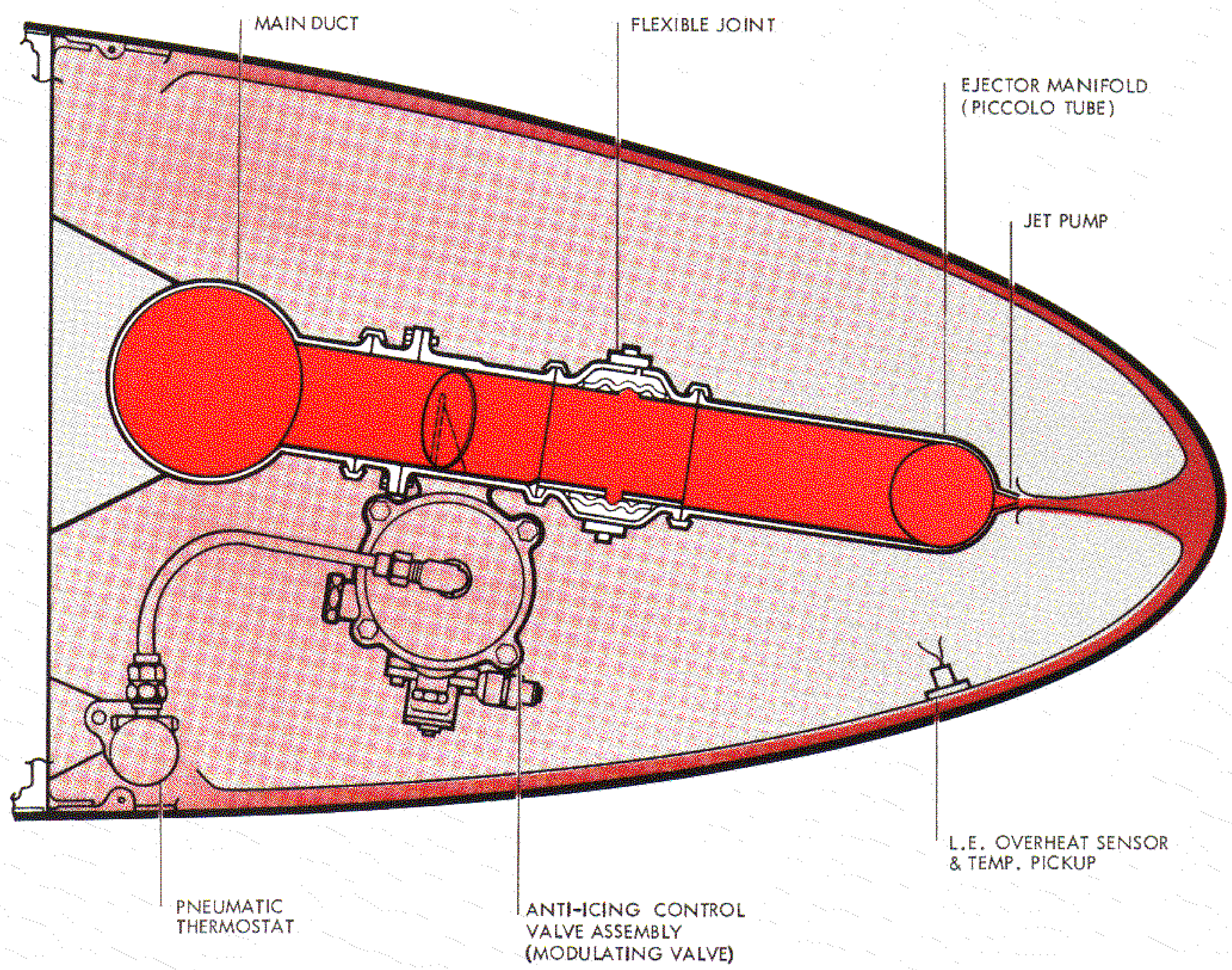


Figure 8
Leading Edge
Plenum Diagram

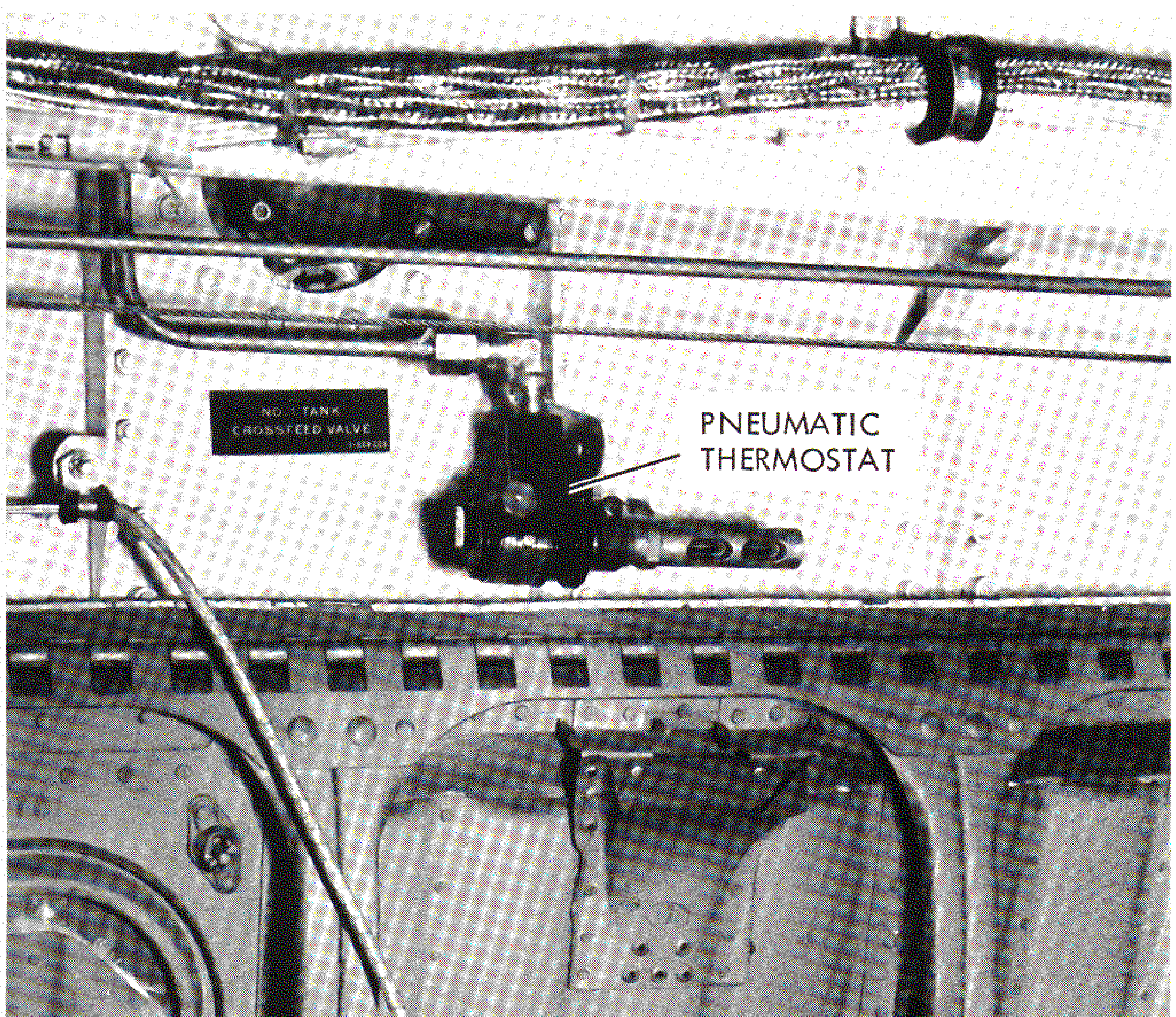


Figure 9
Pneumatic
Thermostat
Installation

the outboard plenum thermostats are set at 63°C (145°).*

The plenum air displaced by incoming air is exhausted overboard through louvers in the aft end of the nacelles. These provisions limit plenum air pressure to a maximum of 1 psi above ambient during normal aircraft operation. (During system design, 300 knots TAS at sea level was used as a guideline.) Spring-loaded pressure relief doors are installed in the leading edge sections to prevent pressure buildup in the event a break occurs in one of the bleed air distribution ducts. These doors are designed to open when the plenum pressure reaches approximately 5 psig, well below the 10 psig static pressure limit for which the structure was designed.

OPERATION The P-3 wing ice control system is an anti-icing system of proven design that dates back to the L-188 Electra commercial transport. This system is intended for anti-icing or de-icing only during flight. It should be operationally checked before takeoff, but *full operation of the wing ice control system on the ground is prohibited.* Even when icing conditions prevail, the wing ice control system should not be used during takeoff, for the power loss incurred by bleeding air from the engines is excessive in comparison to the value gained. Instead, employ timely ice control procedures as soon as possible after takeoff.

Although the NATOPS Flight Manual recommends that the wing ice control system be employed solely for de-icing, the fact remains that this is an anti-icing system, and the operator should be apprised of its operation. For anti-icing, simply position the engine bleed air switches "OPEN," then energize the modulating valves by positioning the Left & Right Wing Ice switches "ON." This permits engine bleed air to be metered by thermostatically-controlled modulating valves into the wing leading edge sections, warming the sections so that ice already present will be shed, and keeping them warm to prevent further ice buildup. Both fuselage bleed air shutoff valves are normally kept closed during flight.

The anti-icing system *could* be left energized for as long as the flight is exposed to icing conditions, but aircraft performance would be adversely affected by air being constantly bled from the engines. Consequently, the NATOPS Flight Manual recommends that the system be used for *de-icing* so that most efficient engine operation may be realized.

To use the system for de-icing, (a) first allow ice to accumulate on the wing leading edges (NATOPS suggests letting ice accumulate until the buildup is about ½-inch thick), (b) then position all four Bleed Air Valve switches "OPEN", and (c) position one of the Left & Right Wing Ice switches "ON". After the two symmetrical leading edge sections have been de-iced, (d) position the control switch "OFF" and (e) repeat the procedure with the other two Left & Right Wing Ice switches, one at a time. When the ice buildup is again significant, repeat the procedure. Thus, operation of the system in this manner causes the modulating valves to function as shutoff valves. Generally speaking, a "heat on" time of about 20 to 30 seconds is sufficient to remove ice that accumulates on the leading edges during the "heat off" time. The recommended de-icing procedure is included in the NATOPS Flight Manual, Section VI.

During de-icing the leading edge temperature indicator should be checked to ascertain if the leading edge sections are being heated. An indicated temperature rise of approximately 10°C during "heat on" is corroboration that the wing ice control system is functioning. Actually the leading edge temperature rise is considerably greater than 10°C, however the response of the temperature indicating system is slow.

The bleed air manifold pressure gage indicates the pressure sensed in the fuselage cross-ship duct. Although the fuselage bleed air shutoff valves are normally kept closed while the wings are being de-iced or anti-iced, some air will leak past the fuselage shutoff valves and gradually pressurize the cross-ship duct. This *gradual* pressure build-up is considered normal, regardless of how high the pressure goes. If the gage indicates a rapid pressure build-up or no pressure build-up with the fuselage bleed air shutoff valves closed, the condition should be investigated before the next flight.

When an engine is shut down with its Emergency Shutdown Handle while its bleed air shutoff valve is open, the valve will be automatically closed by

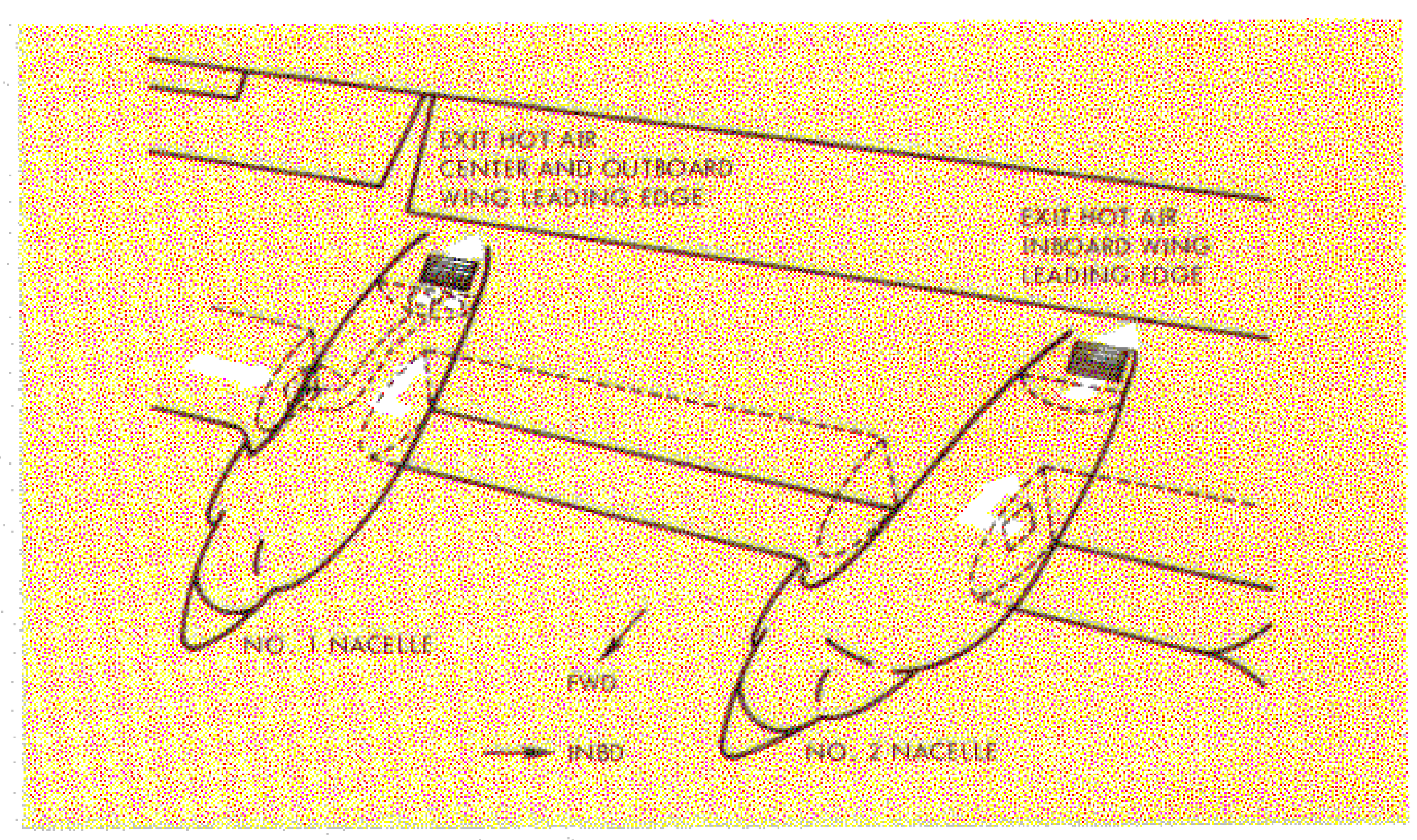


Figure 10 Wing Ice Control System Bleed Air Exhaust Diagram

* *When the wing ice control is used for de-icing (subsequently discussed under "OPERATION"), the system is operated so briefly that the thermostats cannot control their modulating valves. During such instances the modulating valves merely act as shutoff valves.*

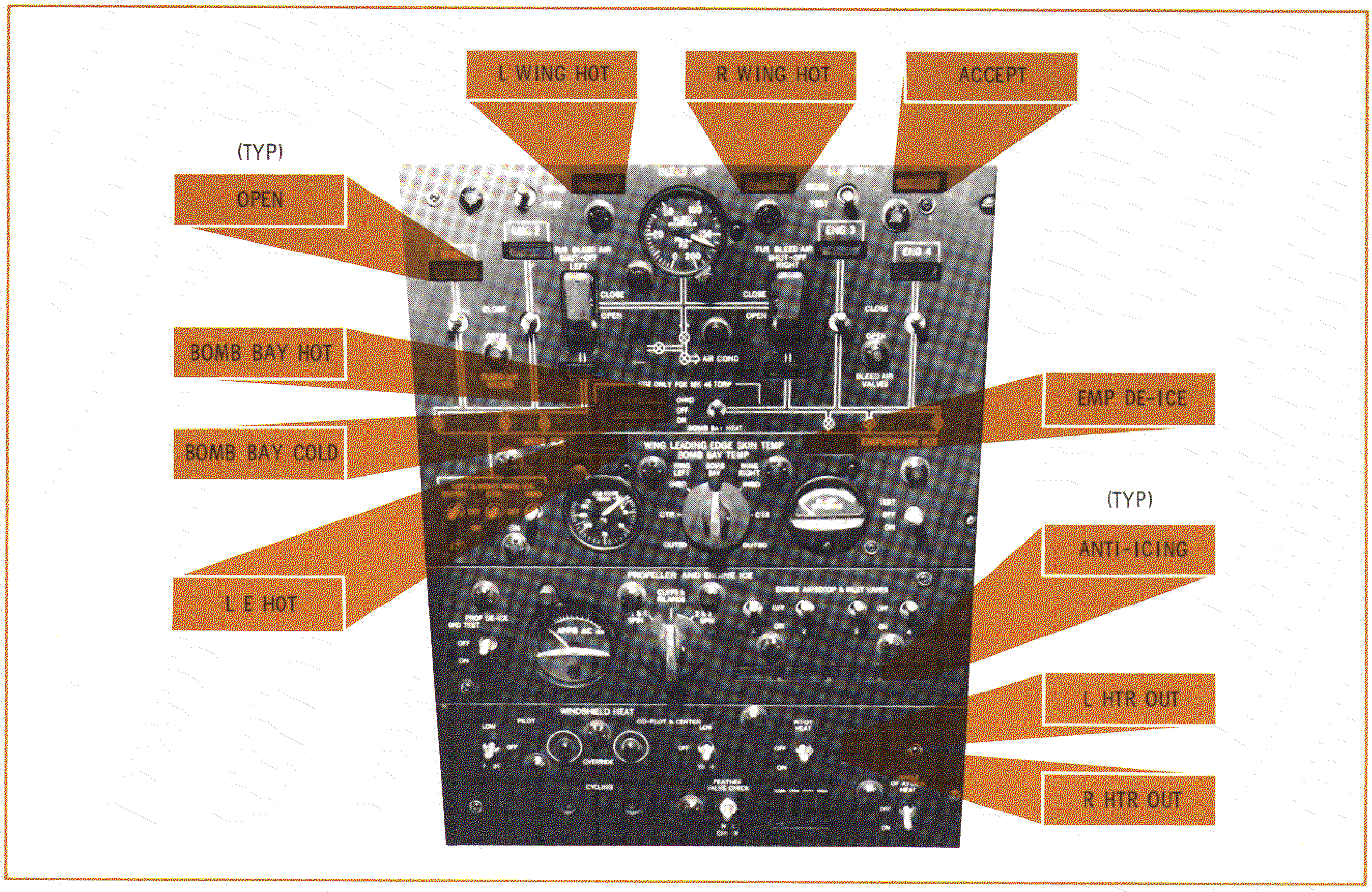
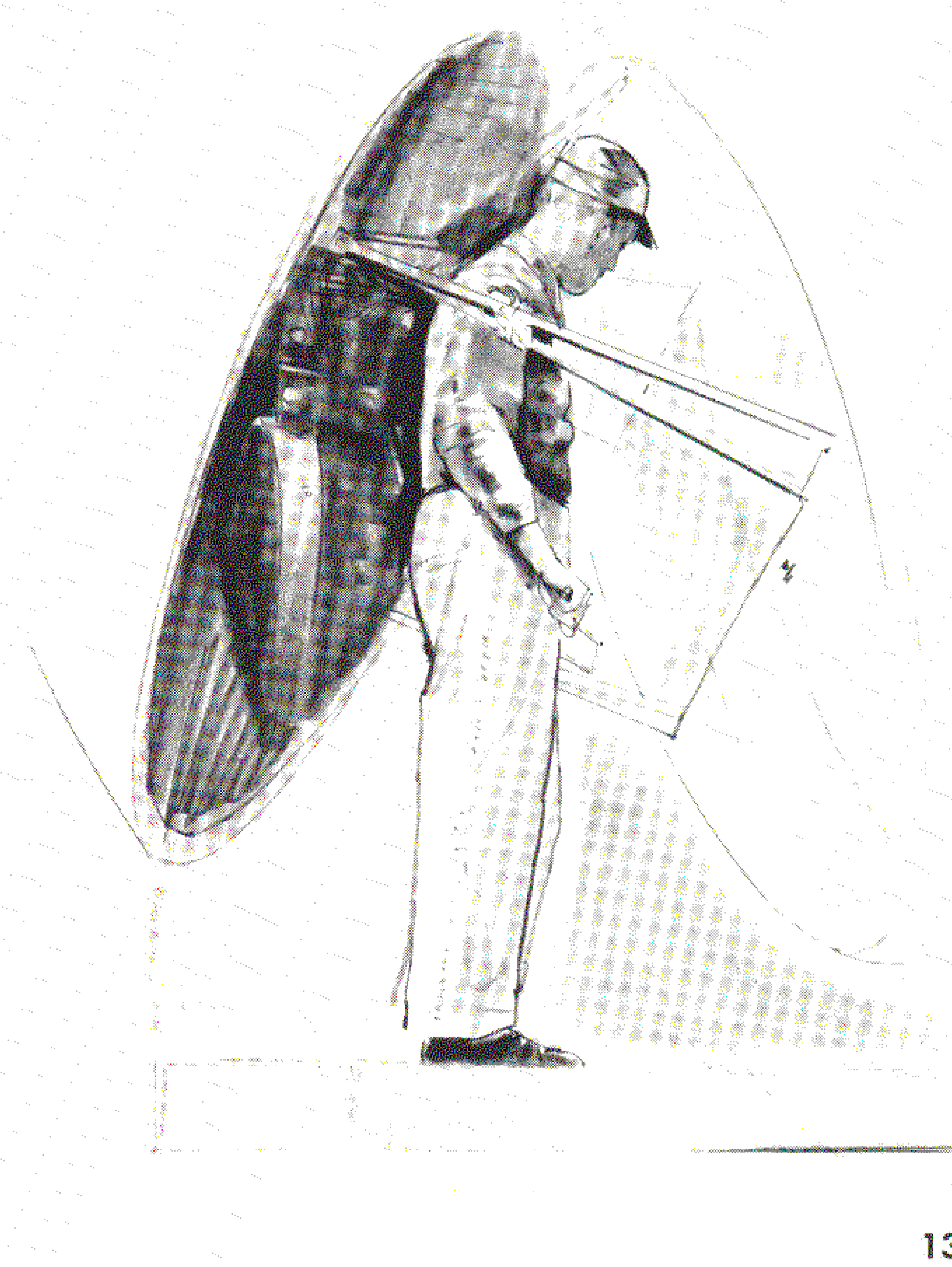


Figure 11 Engine Bleed Air and Anti-icing Control Panels

the aircraft's emergency shutdown circuitry. If the valve's control switch remains in the "OPEN" position, the valve will re-open automatically when the Emergency Shutdown Handle is pushed in.

OVERHEAT WARNING Bleed air is supplied to the bleed air distribution manifolds and ducts at a temperature in the neighborhood of 282°C (540°F). Even after the bleed air is diluted with leading edge plenum air, the temperature of the resultant mixture is still about 135°C (275°F) at the inlet to leading edge hot air passages. Slight leakage in the bleed air system is normal, but if the manifolds or ducts develop serious leaks or if the modulating valves fail to close when required, aircraft structure, equipment and wiring can be exposed to excessive temperatures. The aircraft is equipped with a bleed air distribution duct overheat warning system and a leading edge skin temperature indication and warning system to detect and signal overtemperature conditions so that immediate corrective action can be taken. Aircraft equipped with the bomb bay heating system have a separate temperature sensing and warning system for the bomb bay, but use the leading edge skin temperature indicator to show bomb bay temperature. This latter feature is discussed in the "Bomb Bay Heating System" section of this article.



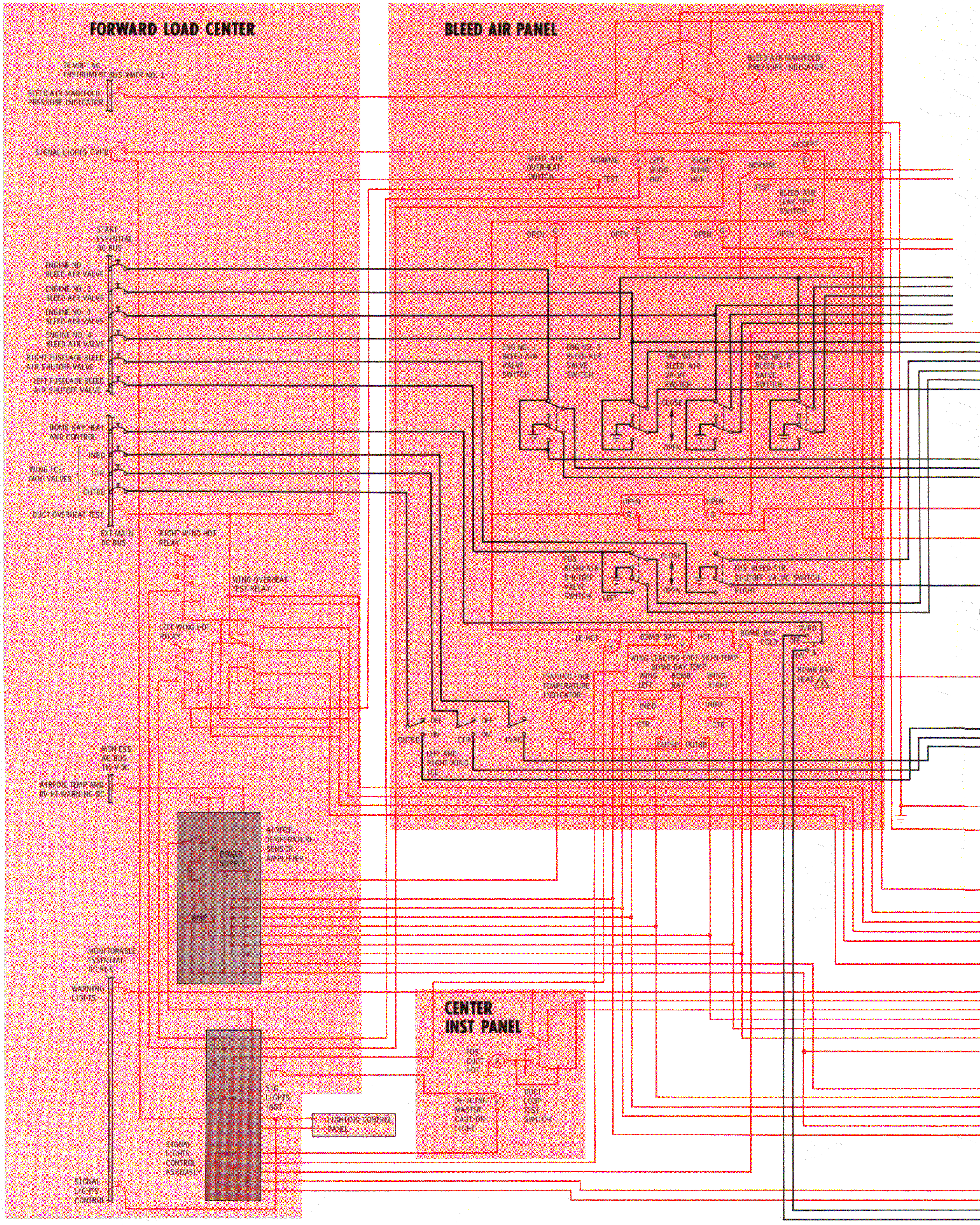

FORWARD LOAD CENTER
BLEED AIR PANEL
26 VOLT AC INSTRUMENT BUS XMFR NO. 1
BLEED AIR MANIFOLD PRESSURE INDICATOR
SIGNAL LIGHTS OVHD
START ESSENTIAL DC BUS
ENGINE NO. 1 BLEED AIR VALVE
ENGINE NO. 2 BLEED AIR VALVE
ENGINE NO. 3 BLEED AIR VALVE
ENGINE NO. 4 BLEED AIR VALVE
RIGHT FUSELAGE BLEED AIR SHUTOFF VALVE
LEFT FUSELAGE BLEED AIR SHUTOFF VALVE
BOMB BAY HEAT AND CONTROL
WING ICE MOD VALVES
INBD
CTR
OUTBD
DUCT OVERHEAT TEST
EXT MAIN DC BUS
RIGHT WING HOT RELAY
WING OVERHEAT TEST RELAY
LEFT WING HOT RELAY
MON ESS AC BUS 115 V AC
AIRFOIL TEMP AND OV HT WARNING OC
AIRFOIL TEMPERATURE SENSOR AMPLIFIER
POWER SUPPLY
AMP
MONITORABLE ESSENTIAL DC BUS
WARNING LIGHTS
SIG LIGHTS INST
LIGHTING CONTROL PANEL
SIGNAL LIGHTS CONTROL ASSEMBLY
SIGNAL LIGHTS CONTROL
BLEED AIR MANIFOLD PRESSURE INDICATOR
BLEED AIR OVERHEAT SWITCH
NORMAL
TEST
LEFT WING HOT
RIGHT WING HOT
ACCEPT
BLEED AIR LEAK TEST SWITCH
OPEN
ENG NO. 1 BLEED AIR VALVE SWITCH
ENG NO. 2 BLEED AIR VALVE SWITCH
ENG NO. 3 BLEED AIR VALVE SWITCH
ENG NO. 4 BLEED AIR VALVE SWITCH
CLOSE
FUS BLEED AIR SHUTOFF VALVE SWITCH
LEFT
RIGHT
FUS BLEED AIR SHUTOFF VALVE SWITCH
LE HOT
BOMB BAY HOT
BOMB BAY COLD
OVRD
OFF
ON
BOMB BAY HEAT
WING LEADING EDGE SKIN TEMP
BOMB BAY TEMP
WING LEFT
BOMB BAY
WING RIGHT
LEADING EDGE TEMPERATURE INDICATOR
LEFT AND RIGHT WING ICE
CENTER INST PANEL
FUS DUCT HOT
DE-ICING MASTER CAUTION LIGHT
DUCT LOOP TEST SWITCH

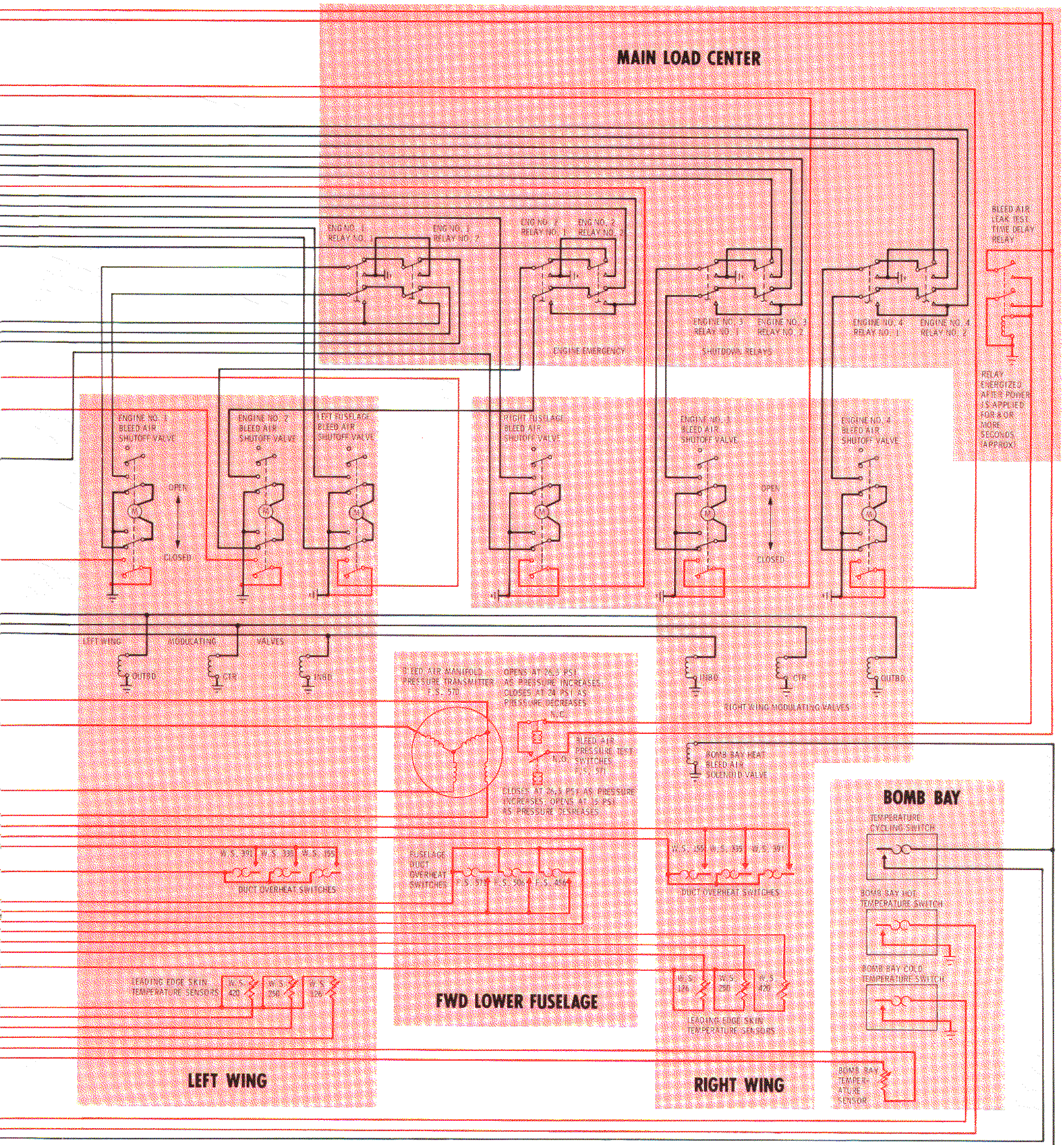
Figure 12 Wing Ice Control System and Bomb Bay Heating System Electrical Schematic

CODE —— CONTROL CIRCUITS
—— INDICATION CIRCUITS

NOTE

1. THIS DIAGRAM REFLECTS THE PRODUCTION CIRCUITRY OF LATER P-3B AIRCRAFT. THERE WILL BE MINOR DIFFERENCES IN THE CIRCUITRY OF EARLIER P-3 AIRCRAFT, DEPENDING UPON THE AIRCRAFT BLOCK NUMBER AND UPON THE INCORPORATION OF AIRFRAME CHANGES. REFER TO THE LATEST REVISION OF NAVWEPS 01-75PAA-2-13.2 TO RESOLVE CIRCUITRY VARIATIONS.
2. ALL SWITCHES AND RELAYS ARE SHOWN POSITIONED WITH THE SYSTEM DE-ENERGIZED WHILE THE AIRCRAFT IS ON THE GROUND.

⚠3 REFER TO ORION SERVICE DIGEST NO. 16, JUNE 1967, FIGURE 8 FOR CIRCUITRY REFLECTING USE OF APU TO PROVIDE BOMB BAY HEAT DURING GROUND OPERATIONS.



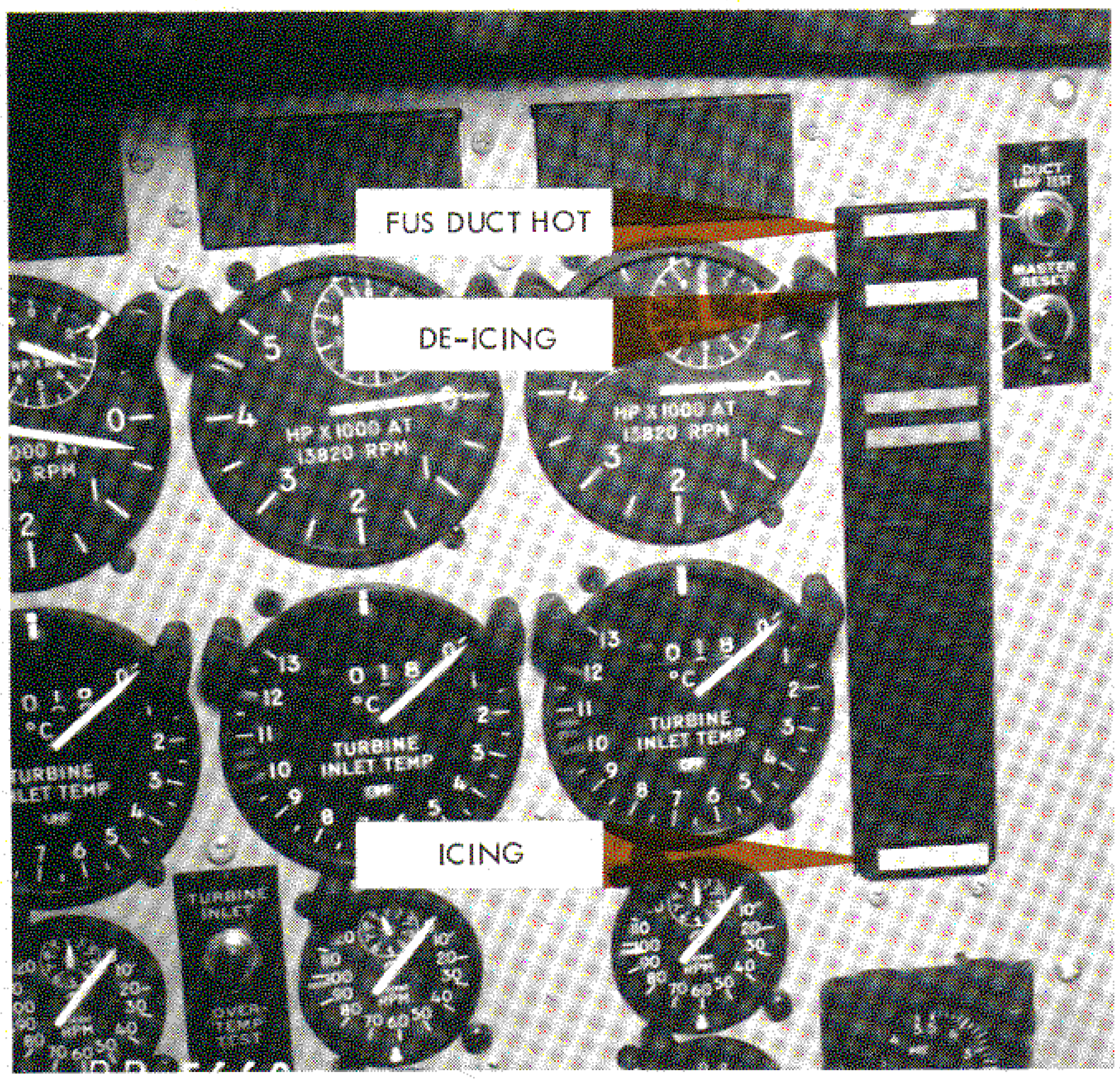


Figure 13 Pilot's Center Instrument Panel Annunciator Lights

The bleed air distribution duct overheat warning system senses the air temperature adjacent to the bleed air manifold and ducting. It is comprised of three independent branches, one protecting each wing and a third protecting the fuselage. The heart of each branch is a network of three thermal (leak detector) switches wired in parallel, and installed close to the ducting. A thermal switch is installed in each wing leading edge section near the opening through which the plenum air exhausts into the nacelle. The other three thermal switches are installed in the fuselage, one near the cross-ship duct and the other two on the right side of the fuselage near the APU bleed air duct and the bomb bay heat duct. Thermal switch locations are shown on Figure 5, while the duct overheat warning circuitry is shown on Figure 12.

Each wing's branch of the duct overheat warning system is composed of its thermal switch network, a signal light, and a signal light relay. The inboard and center leading edge section thermal switches close as the temperature increases past 88°C (190°F) and open as the temperature decreases past 79°C (175°F); the outboard section switches close as the temperature increases past 104°C (220°F) and open as the temperature decreases past 96°C (205°F). When any wing duct overheat switch closes, a circuit is completed from the Main DC Extension Bus, through the thermal switch to the coil of the system branch's wing hot relay, energizing the relay. When the relay is energized, power is routed from the Monitorable Essential DC Bus, through the Signal Lights Control Assembly, to the appropriate amber signal light (either the "L WING HOT" or the "R WING HOT" light) on the Bleed Air panel and to the "DE-ICING" Master Caution Light on the pilot's Center Instrument panel. Both branches of the wing overheat warning system share a test relay that bypasses the thermal switches. When the relay is energized by the Bleed Air Overheat Test Switch, both "WING HOT" signal lights and the "DE-ICING" Master Caution Light are illuminated.

Illumination of either the "L WING HOT" or "R WING HOT" light on the Bleed Air panel indicates that the air temperature in at least one of the leading edge plenums exceeds safe limits. The probable cause of this overheat indication is a leak in the bleed air system ducting, perhaps due to a loose clamp, a crack in the duct, etc. Since prolonged exposure of the leading edge structure to excessive heat could affect the wing's structural integrity, and since continued operation of the opposite wing's ice control would only result in asymmetrical icing conditions, the aircraft's entire wing ice control system must be secured immediately. From an operational standpoint, it is only common sense to get out of the icing conditions as soon as possible.

The fuselage branch of the duct temperature warning system is comprised of its three-sensor thermal switch network, a red "FUS DUCT HOT" light, and a Duct Loop Test switch, the latter two components being installed on the Center Instrument panel. Fuselage duct thermal switch response is the same as that of the wing inboard and center plenum switches (close at 88°C, re-open at 79°C). When any one of the switches is actuated, the warning circuit is energized with power from the Monitorable Essential DC Bus and the "FUS DUCT HOT" light is illuminated. Actuation of this branch of the system does *not* energize the "DE-ICING" Master Caution Light. Actuation of the Duct Loop Test switch bypasses the thermal switches and illuminates the warning light.

Illumination of the "FUS DUCT HOT" light on the Center Instrument panel indicates excessive air temperature in the lower portion of the forward fuselage, near the cross-ship duct or near the bomb bay heat and APU bleed air ducts. Like a "Wing Hot" indication, the probable cause is a leak in the ducting, so all engine bleed air valves and fuselage shutoff valves should be closed immediately. If the "FUS DUCT HOT" indication occurs during ground operations while the APU is being used as a bleed air source, the APU bleed air duct may be defective. In this latter situation, shut down the APU and engines and investigate.

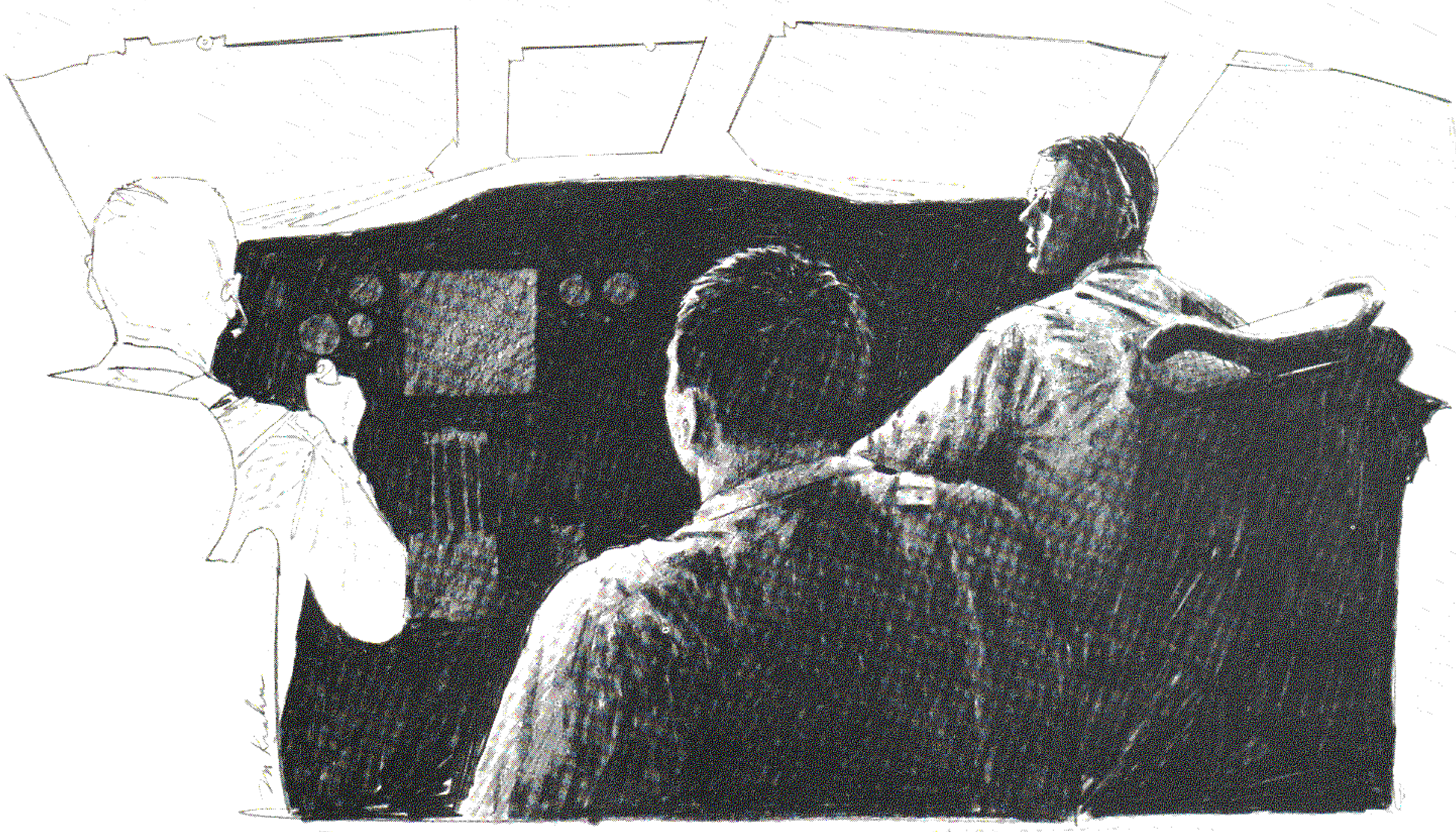
The wing leading edge skin temperature indication and warning system displays the skin temperature on an indicator and signals overheat conditions by illuminating the "LE HOT" signal light and the "DE-ICING" Master Caution Light. The system employs six thermistors as temperature sensors, one installed in the inner skin of each leading edge section. Signals from the thermistors are continually fed into the system's airfoil temperature sensor amplifier in the Forward Load Center. The amplifier is powered from ϕ C of the Monitorable Essential AC Bus, and the signal lights are powered from the Monitorable Essential DC Bus by way of the Signal Lights Control Assembly. The temperature of each leading edge section can be shown on the indicator by selecting the desired section with the system's rotary switch.

If any section's skin temperature exceeds 110°C (230°F), a relay within the amplifier is energized. This grounds the signal light circuit, which illuminates the "LE HOT" signal light and the "DE-ICING" Master Caution Light. If a leading edge skin overheat condition is signaled, the location can be determined by observing the indicator while placing the rotary switch in each of its positions. The temperature of all six leading edge sections should be checked, for there is always the possibility that more than one section is overheating.

Illumination of the "LE HOT" light after the recommended NATOPS de-icing procedure has been performed indicates that at least one of the modulating valves did not close when its control switch was positioned "OFF". The first response to this signal must be to determine, with the aid of the leading edge skin temperature indicator, which section or sections are overheating. Any section with an indicated temperature of 110°C (230F°) or greater must be considered excessively hot. Immediately thereafter, close all four engine bleed air valve switches to shut off the supply of bleed air to the bleed air manifolds of both wings. This will leave the leading edges on both wings temporarily without ice control protection, however it will prevent an asymmetrical icing condition from developing while determining what remedial action can be taken.

After the bleed air supply has been shut off, check the WING ICE MOD VALVE circuit breakers (located on the Extension Main DC Bus panel at the Forward Load Center). If any of the circuit breakers are open, reset them if possible. After the circuit breakers have been reset, cycle the modulating valve control switches to determine if the circuit breakers will remain closed. If the circuit breakers do remain closed, the engine bleed air valves for both wings may be opened and the NATOPS recommended de-icing procedure may be resumed.

If the overheat condition cannot be remedied by resetting the modulating valve circuit breakers or if the circuit breakers cannot be reset, the leading edges can still be de-iced by an alternate procedure: Position all modulating valve control switches "ON," then cycle the engine bleed air control valves open and closed to heat the leading edge sections. When this procedure is used, open the engine bleed air



valves only until the ice accumulations have been shed, then close them to prevent the defective section(s) from overheating. Be certain to monitor the leading edge skin temperature gage carefully, and do not let the skin temperature exceed 110°C (230°F).

BLEED AIR PRESSURE TEST Some leakage is inherent in the bleed air system's design, however the system can be ground-tested if excessive leakage is suspected. For example, the system should be tested if the bleed air pressure gage shows little or no pressure during wing de-icing, if the gage shows rapid pressure build-up during wing de-icing, or if APU bleed air is being used for an engine start and the gage shows less than 25 psi as the engine accelerates through 16% RPM (approx.). During this test the cross-ship manifold* is pressurized, then the actual manifold pressure decay time is compared with the minimum allowable pressure decay time. The system test "ACCEPT" light will illuminate if the manifold pressure decay time falls within acceptable limits.

The manifold pressure test system is comprised of two pressure switches installed in the cross-ship duct, a time-delay relay in the Main Load Center, and a test switch and "ACCEPT" light on the Bleed Air panel. Bleed air pressure can be read from the pressure gage at the top of the Bleed Air panel. System circuitry is shown on Figure 12. Bleed air for the test is supplied from one engine running at normal rpm. The other three engines and the APU must be shut down to prevent air pressure from these sources from affecting the test. If engine No. 2, 3, or 4 is used as the bleed air source the aircraft can provide electrical power for the test, but if engine No. 1 is used electrical power must be supplied from a ground power source since no generator is installed on this engine. A valid check cannot be performed if the APU is used as the bleed source because it cannot provide the 70 psig air pressure required for the test.

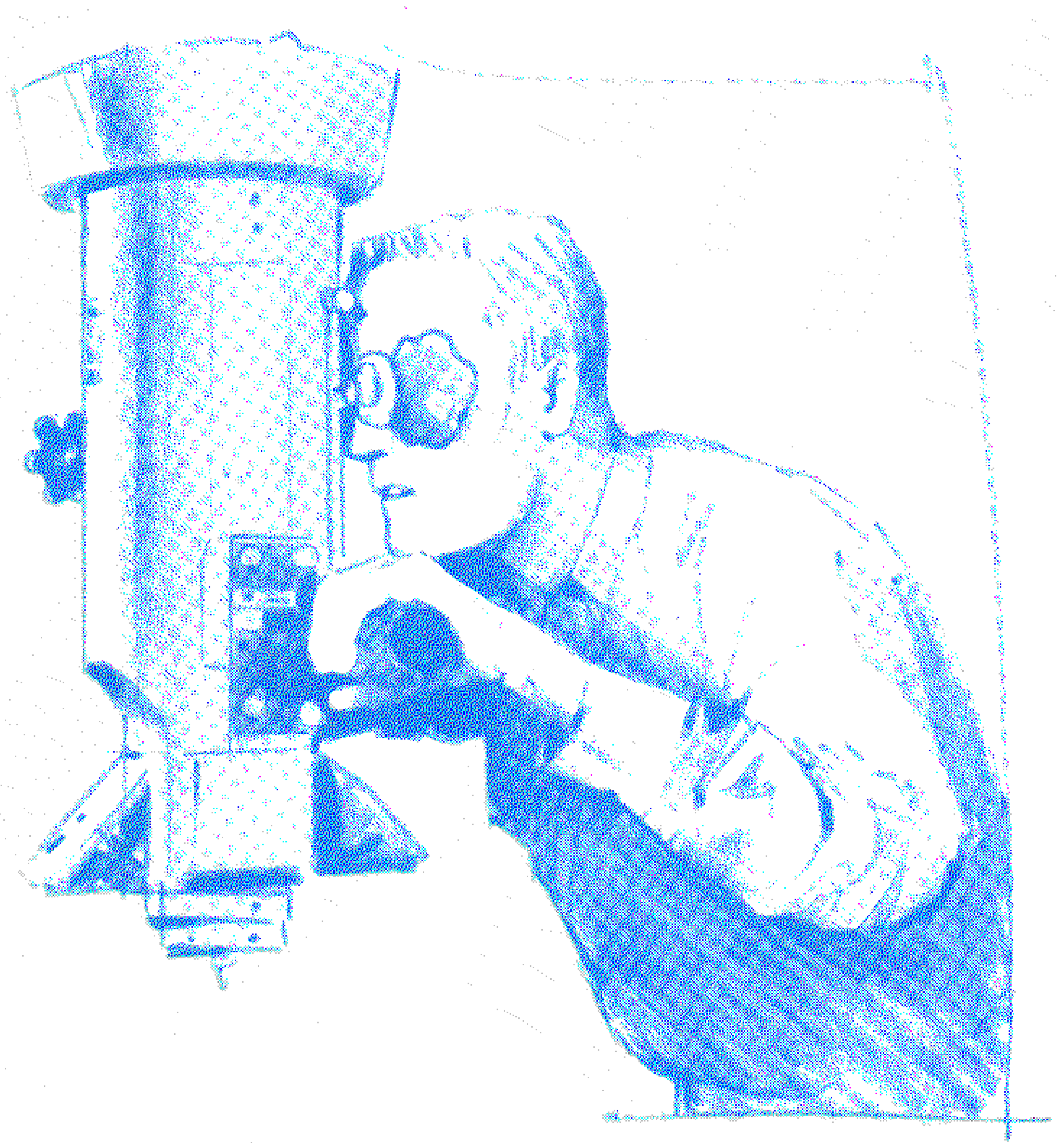
This pressure test system, the test procedure, and test values were adopted without change when the P-3 Orion was derived from its commercial transport antecedent, but the bleed-air distribution system on the L-188 Electra is considerably larger than the P-3 system, as hot air anti-icing is employed on the transport empennage. Consequently, the large-volume system test is quite stringent for the small-volume P-3 system, and a high order of system integrity is proven when the "ACCEPT" light illuminates. Field experience indicated that a slightly higher leak rate could be safely tolerated, and an alternate test procedure was formulated in which pressure decay is timed manually over an expanded range. In the following paragraphs, the regular test procedure is in normal type-face, the alternate procedure is in italics.

Prepare the bleed air system for the leak test by closing all six wing leading edge modulating valves, and (if the aircraft is equipped with bomb bay heat) by closing the bomb bay bleed air shut-off valve. Open the left and right fuselage bleed air shutoff valves and close the bleed air valves of the inoperative engines. Open the operating engine's bleed air valve and pressurize the cross-ship manifold to 70 psig (min.). As the pressure increases past 26.5 psig one of the pressure switches opens and the second closes.

Next position the Leak Test switch to "TEST" and hold, and close the operating engine's bleed air valve. When the manifold pressure decreases past 24 psig, the first pressure switch closes to complete the circuit to the time-delay relay. The manifold pressure will continue to decrease until at 15 psig the second pressure switch opens. If power has been applied through the pressure switches to the time-delay relay for at least 8 seconds, the relay will be energized and hold itself closed with a latching circuit until the spring-loaded test switch is released. A ground is provided for the "ACCEPT" light circuit through a set of the relay's contacts as long as the relay remains energized. Illumination of the "ACCEPT" light indicates that cross-ship manifold air leakage falls within acceptable limits.

If acceptable test results are not obtained, pres-

** The cross-ship manifold is comprised of the left and right wing bleed air manifolds and their interconnecting duct (the cross-ship duct) that runs through the fuselage.*



surize the cross-ship manifold again to 70 psig, close the operating engine's bleed air valve, then observe manifold pressure decay on the Bleed Air Manifold pressure gage. If the manifold air pressure remains between 37 and 15 psig for at least 8 seconds as the pressure decays, manifold leakage is considered to be within acceptable limits. If manifold leakage is not excessive, continue with the next part of the test. If bleed air leakage is excessive, corrective action must be initiated.

This second part of the test checks the fuselage duct and the fuselage bleed air shutoff valves for excessive leakage. With the test switch in the "NORMAL" position, pressurize the cross-ship manifold to 70 psig, then close both fuselage bleed air shutoff valves. After closing the fuselage bleed air shutoff valves, hold the test switch to "TEST", close the operating engine's bleed air shutoff valve, then open the left and right wing modulating valves. Operation of the pressure switches will be the same as during the first part of the test procedure. Illumination of the "ACCEPT" light approximately 8 seconds after the fuselage duct pressure has decreased to 24 psig indicates that the left and right fuselage bleed air shutoff valves are functioning properly and that fuselage duct leakage is within acceptable limits.

If the "ACCEPT" light does not illuminate, pressurize the cross-ship manifold again to 70 psig, close both fuselage bleed air shutoff valves, close the operating engine's bleed air shutoff valve, and open the modulating valves on both wings. Observe fuselage duct pressure decay on the Bleed Air Manifold gage. If the duct pressure remains between 37 and 15 psig for at least 20 seconds, fuselage bleed air shutoff valve operation is acceptable and fuselage duct leakage is within acceptable limits. If neither of these fuselage duct tests produce acceptable results, a hazardous bleed air leakage may exist in the fuselage area.

NACELLE COOLING SCOOPS Four small NACA-type flush scoops are installed on the lower surface of the wing leading edges, one slightly outboard of each nacelle. These scoops supply air to ventilate the areas between the wing and the engine tailpipe shrouds, primarily to prevent fuel vapor entrapment and secondarily to cool the skin.* Since the scoops are part of the leading edge structure, engine bleed air is also used to protect them from icing. Bleed air from the leading edge jet pumps flows aft between the leading edge's double skin, around the ramp and side walls of the scoop, and exits back into the leading edge plenum area around the elbow of the scoop duct. The lip of the cooling duct scoop, which is hollow, is supplied with the bleed air directly from the piccolo tube via a ½-inch line. After leaving the lip, this air mixes with the hot bleed air that has been used to heat the ramp and is routed to heat the duct elbow.

* *Whenever the upper wing surface is inspected (for example, during the special 28-day corrosion inspections), the condition of the wing skin paint beneath the tailpipes should be checked for discoloration due to overheating. If the wing did get hot enough to discolor the paint, the heat treatment of the wing planks could be nullified, and further investigation should be instituted.*

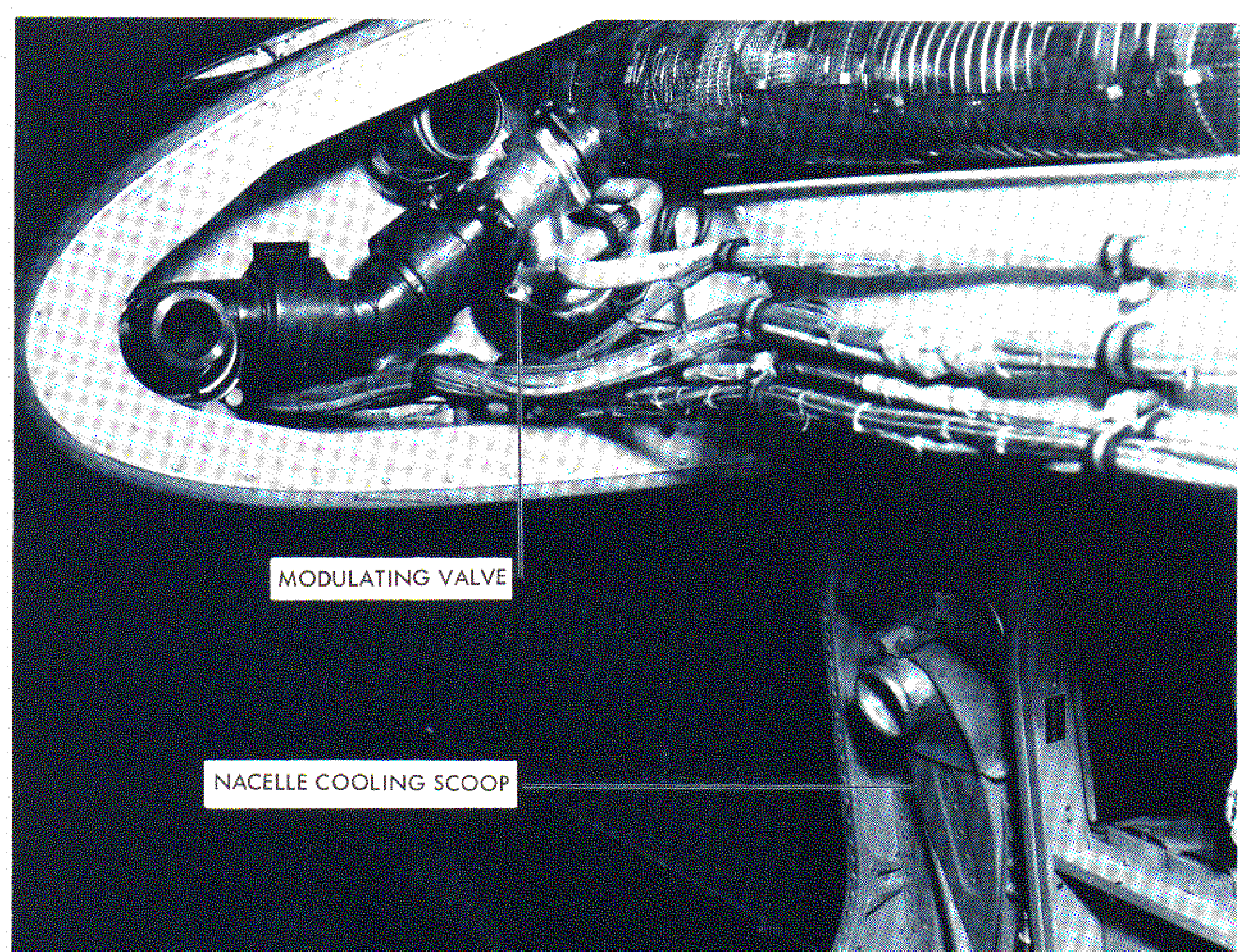


Figure 14
Nacelle Cooling
Scoop Installation

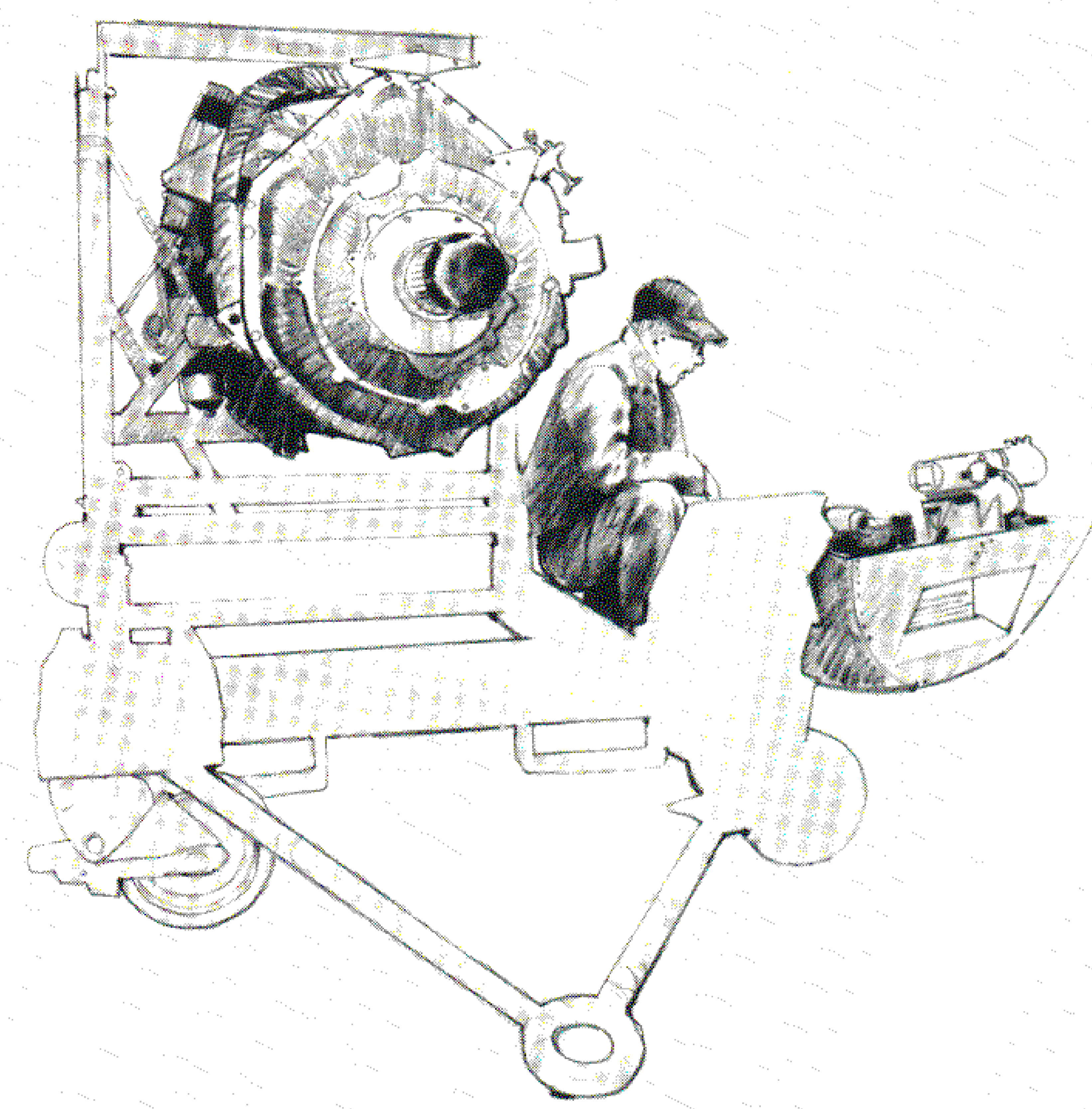
BOMB BAY HEATING SYSTEM

Later P-3B aircraft have been equipped with a bleed air heating system that can maintain the bomb bay temperature above −1°C (30°F) during both flight and ground operations. Bleed air is normally supplied by engines No. 3 and/or No. 4 for in-flight heating and for some ground operations. The APU can also serve as a source for bleed air, but only during ground operations. Operation of the bomb bay heat system will tax the bleed air source(s) a total of about 50 H.P. Earlier P-3 aircraft are being fitted with the bomb bay heating system during Progressive Aircraft Rework (PAR) through incorporation of P-3 Airframe Change No. 142.

Hot air for the bomb bay heating system is taken from the bleed air system's right wing manifold. The tee to the bomb bay system is located outboard of the right fuselage isolation valve as shown in Figure 15. Airflow to the bomb bay system is controlled with the solenoid-operated shutoff valve installed just downstream from the tee and by a flow-limiting orifice located at the valve outlet. The system's ducting runs inboard from the shutoff valve along the wing leading edge, enters the fuselage near fuselage station 565, runs forward in the right side of the aircraft, enters the bomb bay near fuselage station 469, and extends from right to left across the top aft end of the bomb bay (see Figure 16). The air ejectors, one on each side of the bomb bay, mix the hot bleed air with ambient bomb bay air by means of aspiration (similar to the piccolo tube arrangement described previously). The ejectors discharge the mixture forward and slightly downward along the bomb bay side panels.

The system is controlled by the three-position Bomb Bay Heat switch located on the Bleed Air panel, and by the cycling thermostat in the bomb bay. Positioning the switch "ON" energizes the system's control circuitry with power from the Extension Main DC Bus, enabling the cycling thermostat to open or close the system's air shutoff valve. The cycling thermostat is set to maintain the bomb bay's *minimum* temperature between approximately −1° and + 7°C (30° and 44°F). If the cycling thermostat fails to open the valve, the thermostat can be bypassed by positioning the Bomb Bay Heat switch to "OVRD." This applies power directly to the shutoff valve solenoid, opening the valve. Refer to Figure 12 for system circuitry.

Bomb Bay temperature and the heating system's performance can be monitored on the ice control panel with the temperature advisory lights and the Wing Leading Edge Temperature Indicator. A thermal switch will actuate the "BOMB BAY COLD" light if the bay's temperature falls below approximately −5°C (23°F). Similarly, if the bomb bay temperature exceeds approximately 54°C (130°F), another thermal switch will actuate the "BOMB BAY HOT" light. Actuation of either of these advisory lights will also energize the "DE-ICING" Master Caution Light on the center annunciator panel in the flight station. Power for the signal lights circuit is supplied from the Monitorable Essential DC Bus by way of the Signal Lights Control Assembly.



A thermal sensor in the bomb bay continuously signals the temperature to the Airfoil Temperature Sensor Amplifier just as the six airfoil thermal sensors do. The amplified signal is displayed on the Leading Edge Temp indicator when the selector switch is positioned to "BOMB BAY." If it is necessary to operate the bomb bay heating system by overriding the system's cycling thermostat, the bomb bay temperature should be continuously monitored on the indicator.

OPERATION When the bomb bay system is used during flight, bleed air must be provided by one or more of the aircraft's main engines. Normally, the bleed air is supplied from engines Nos. 3 and/or 4, but it is apparent that bleed air can also be supplied from engines Nos. 1 and 2. This latter practice is not recommended. The APU is not designed to supply bleed air to *any* aircraft systems during flight. During ground operations the bomb-bay heating system may be supplied bleed air either from engines Nos. 3 and/or 4 or from the APU.

Preparatory to heating the bomb bay during flight, ensure that the right fuselage bleed air shutoff valve is closed. Next, charge the right wing bleed air manifold with air by opening either or both No. 3 and No. 4 engine bleed air shutoff valves. Begin bomb bay heating by positioning the Bomb Bay Heat switch "ON," then monitor system operation with the advisory lights and with the temperature indicator on the Bleed Air panel. When the bomb bay heating system air shutoff valve is open, the engine(s) supplying the bleed air will experience a total power loss of about 50 H.P.

When the bomb bay heat system is operated on the ground with bleed air from the APU, the right hand fuselage bleed air shutoff valve must be open to allow bleed air to flow from the APU to the bomb bay air distribution line. The bomb bay heat system may now be operated in the conventional manner. Note that during an engine start from the APU, the bomb bay heat system is *not* monitored automatically. It will be necessary to turn off the bomb bay heat switch in order to maintain the proper manifold pressure for starting. However, the APU can supply bleed air to the bomb bay heating system and the air conditioning system simultaneously without overtaxing the APU. APU/Air Conditioning system circuitry relevant to ground operation of the bomb bay heat system is shown on Figure 8 of *Orion Service Digest, Issue 16.* Other pertinent comments are on pages 17, 18 and 20 of the same issue.

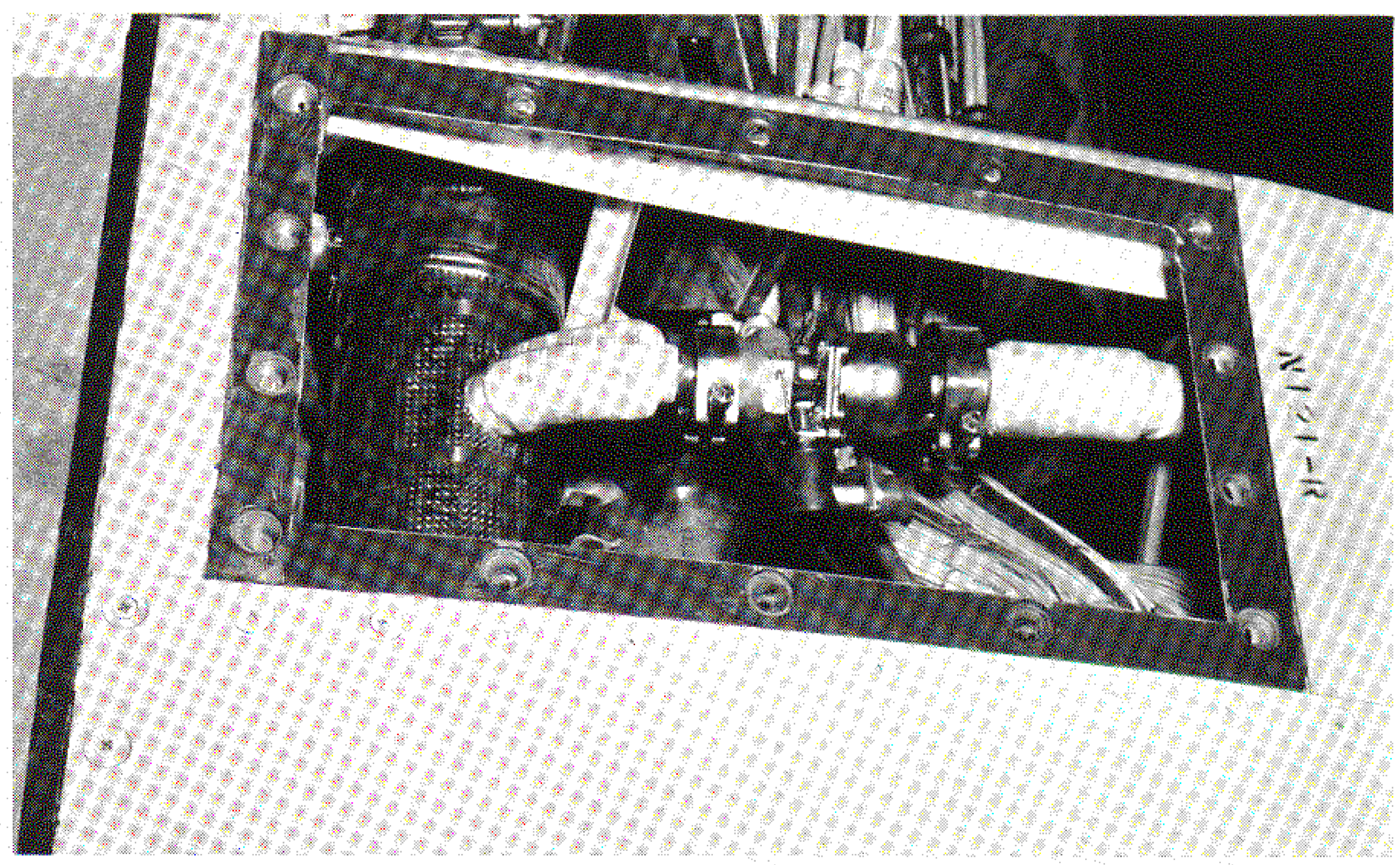


Figure 15
Bomb Bay Bleed Air
Shutoff Valve Installation

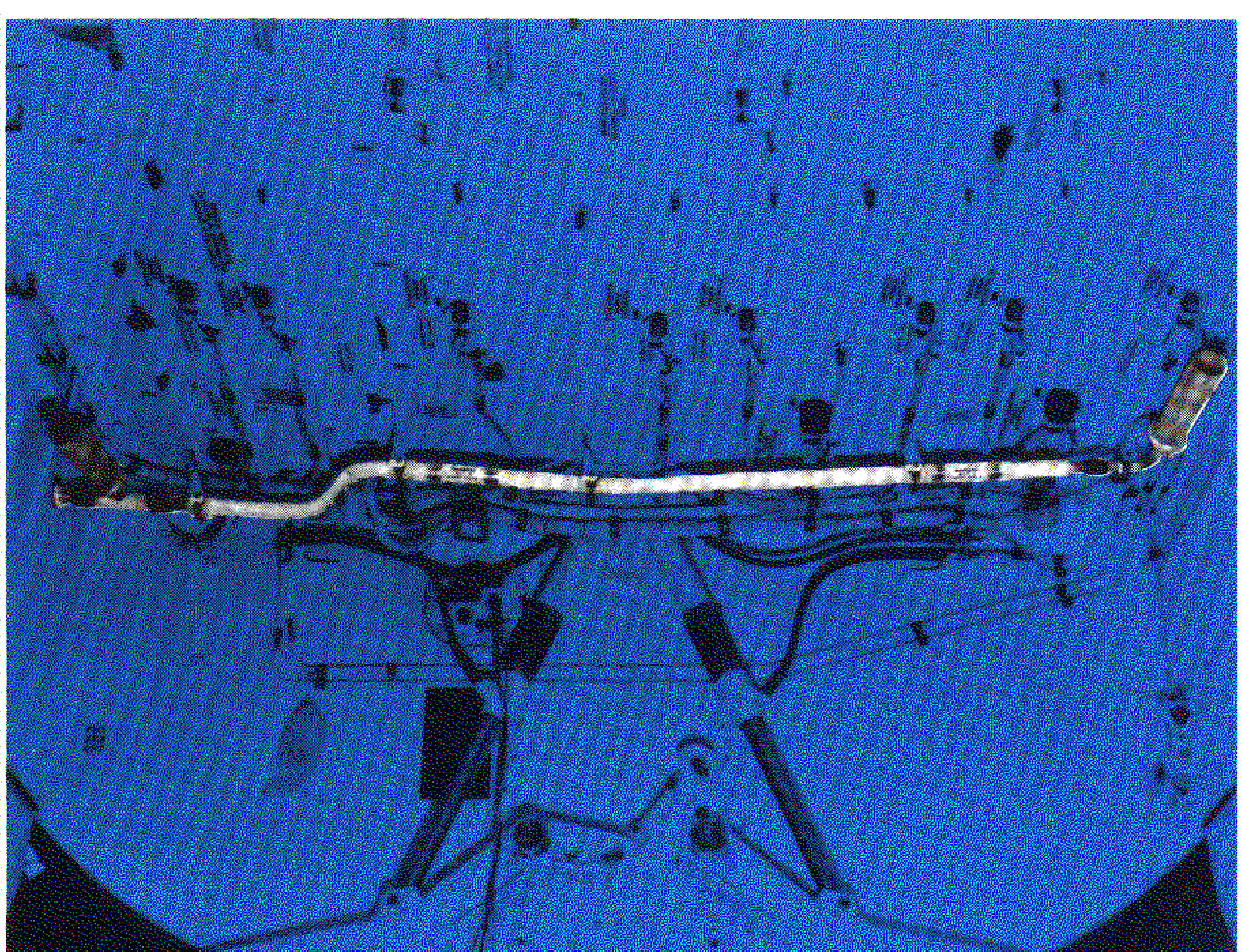


Figure 16
Bomb Bay Heating System
Air Ejectors and Ducting

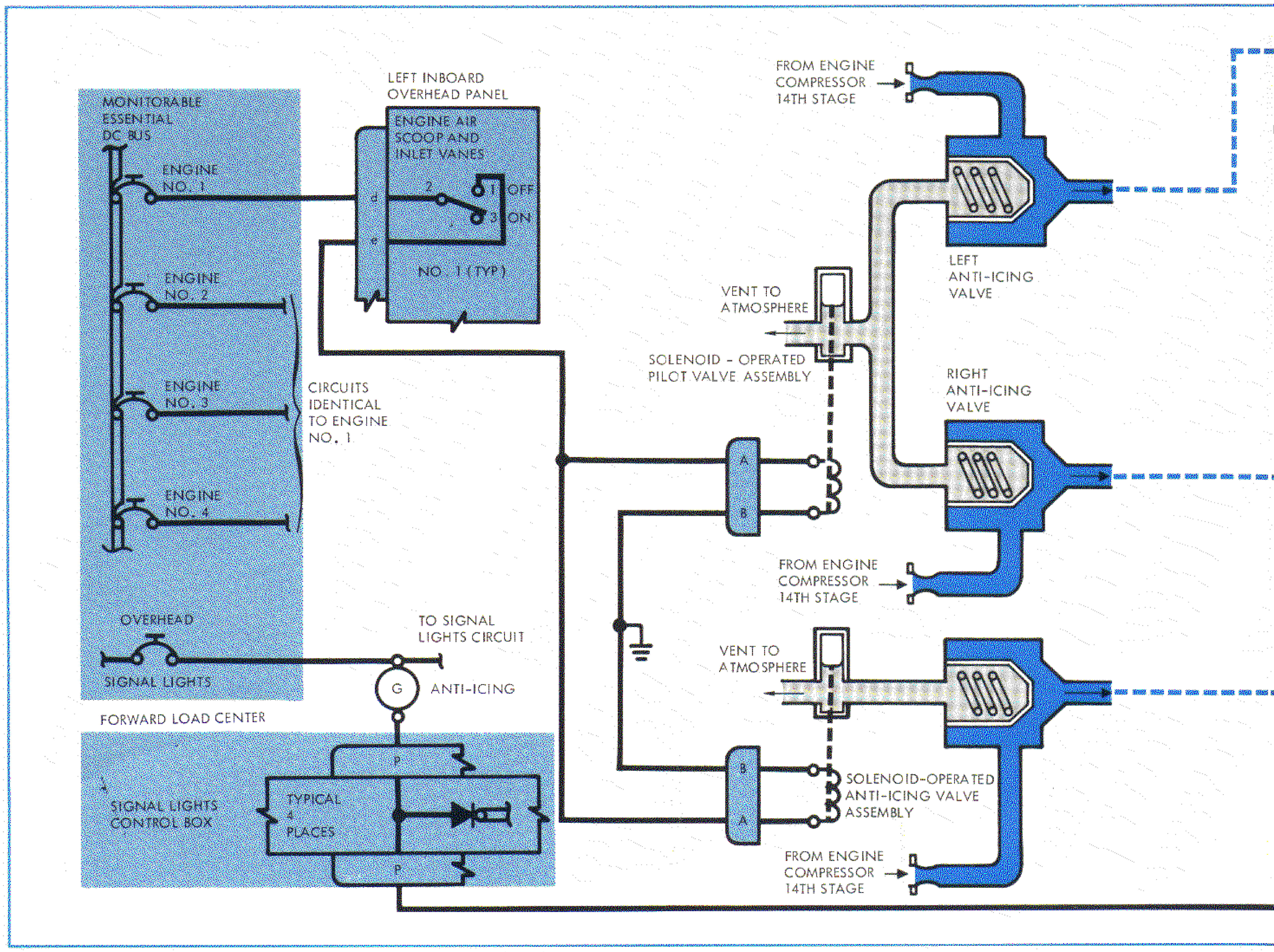


Figure 17 Engine Bleed Air Systems Schematic

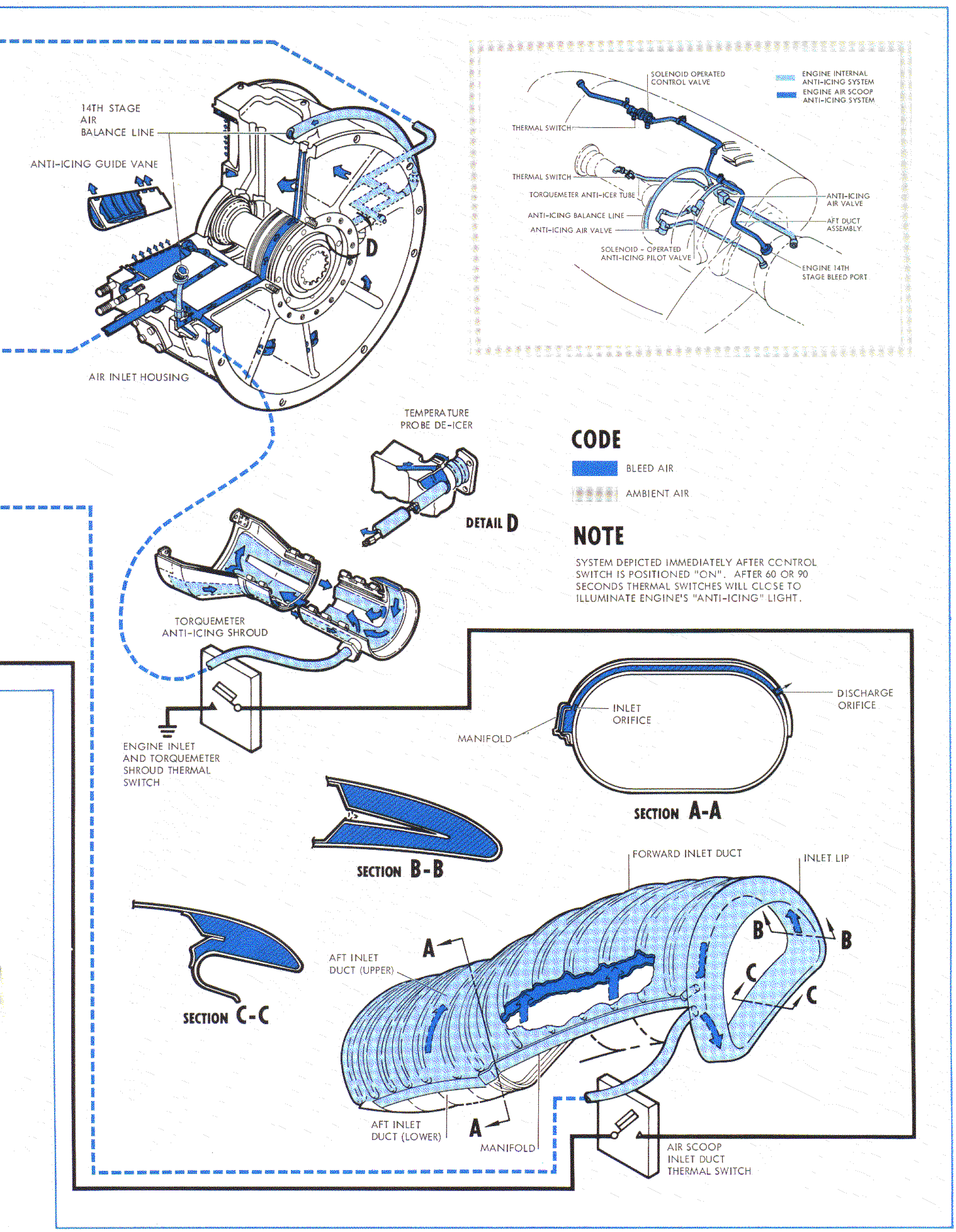
ENGINE ICE CONTROL SYSTEMS

Each power plant is equipped with two separate bleed air anti-icing systems, one system to heat the engine airscoop assembly and the other to heat forward engine components that are subject to icing. Four control switches, one for each engine's anti-icing system, are located on the pilot's overhead control panel. Once the systems have been turned "ON," their operation is automatic. Since engines can be damaged if ice is permitted to build up, the engine ice control systems must be activated whenever icing conditions are anticipated.

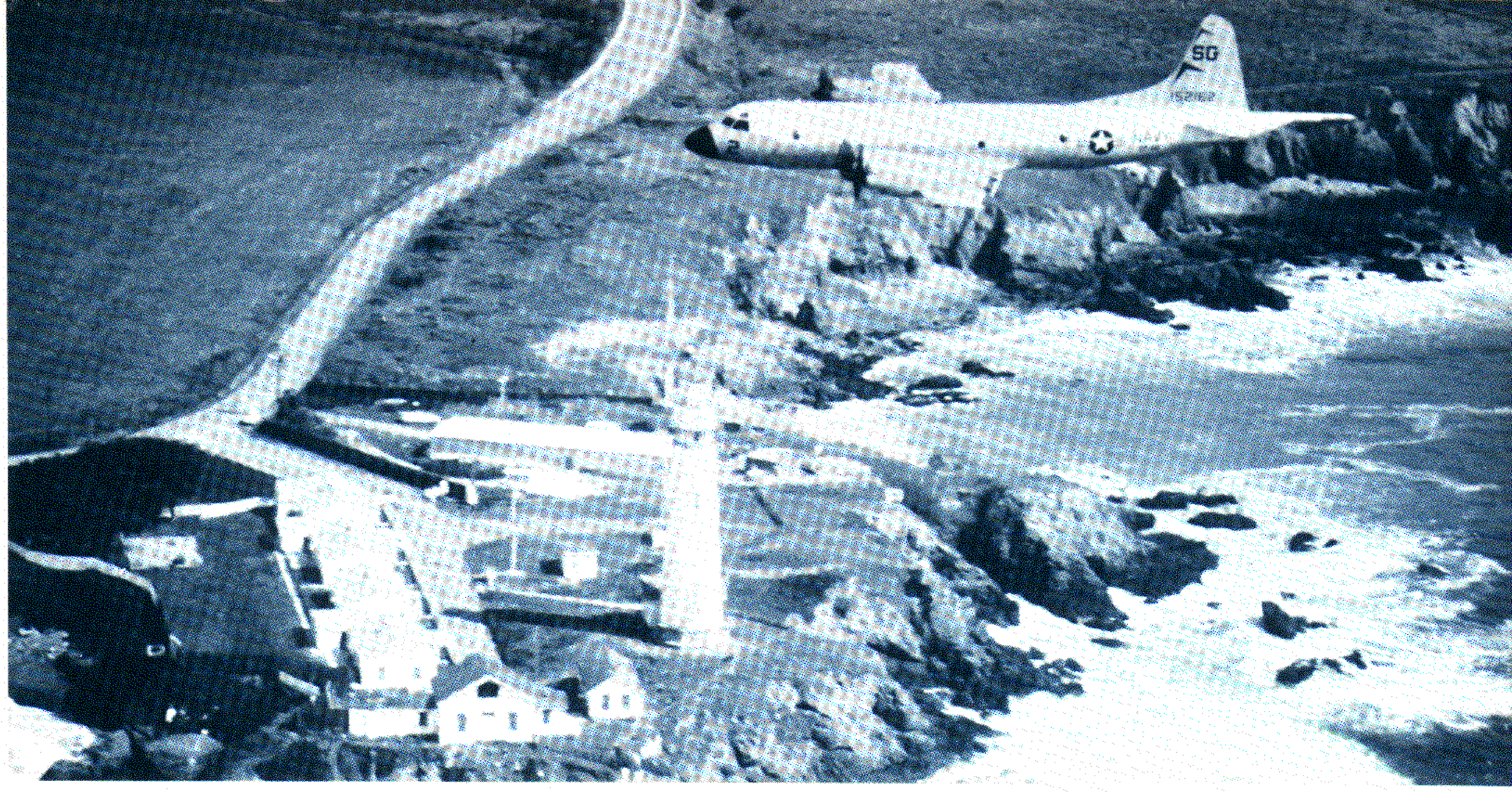
ENGINE AIRSCOOP ANTI-ICING The engine airscoop guides air aft and downward into the engine's air inlet. It can be readily deduced from the shallow "S" path the air follows through the intake duct that those surfaces most subject to icing are the airscoop lip and the upper inside portion of the intake duct. Ice is prevented from forming on these surfaces by heating them with engine bleed air. A schematic of the system is shown on Figure 17.

Bleed air is supplied to the system from a port at the engine compressor's 14th stage. A flow control orifice integral with the bleed air port protects the system from uncontrolled air leakage in the event of failure of the supply line that connects the port and the anti-icing system's shutoff valve. If the line fails, the orifice limits bleed airflow through the port to approximately 120 percent of normal flow.

The system's bleed air supply is controlled by a solenoid-operated, poppet-type shutoff valve installed in the tubing leading from the engine bleed air port to the airscoop lip. The valve is operated with the appropriate Engine Airscoop & Inlet Vanes switch on the Propeller and Engine Ice panel in the flight station. When the switch is positioned "ON," the solenoid is de-energized and *opens* the valve. Conversely, when the switch is positioned "OFF," the solenoid is energized to *close* the valve. In essence, this design is a fail-to-the-safe feature, for it ensures that bleed air will be supplied to the airscoop anti-icing system in the event of electrical power failure. The valve forms a sonic orifice that regulates the system bleed air supply to a relatively constant volume during all aircraft operating conditions.



14TH STAGE
AIR
BALANCE LINE
ANTI-ICING GUIDE VANE
D
AIR INLET HOUSING
SOLENOID OPERATED
CONTROL VALVE
ENGINE INTERNAL
ANTI-ICING SYSTEM
ENGINE AIR SCOOP
ANTI-ICING SYSTEM
THERMAL SWITCH
THERMAL SWITCH
TORQUEMETER ANTI-ICER TUBE
ANTI-ICING BALANCE LINE
ANTI-ICING AIR VALVE
SOLENOID - OPERATED
ANTI-ICING PILOT VALVE
ANTI-ICING
AIR VALVE
AFT DUCT
ASSEMBLY
ENGINE 14TH
STAGE BLEED PORT
TEMPERA TURE
PROBE DE-ICER
DETAIL D
CODE
BLEED AIR
AMBIENT AIR
NOTE
SYSTEM DEPICTED IMMEDIATELY AFTER CONTROL SWITCH IS POSITIONED "ON". AFTER 60 OR 90 SECONDS THERMAL SWITCHES WILL CLOSE TO ILLUMINATE ENGINE'S "ANTI-ICING" LIGHT.
TORQUEMETER
ANTI-ICING SHROUD
ENGINE INLET
AND TORQUEMETER
SHROUD THERMAL
SWITCH
INLET
ORIFICE
DISCHARGE
ORIFICE
MANIFOLD
SECTION A-A
SECTION B-B
SECTION C-C
FORWARD INLET DUCT
INLET LIP
AFT INLET
DUCT (UPPER)
A
B
B
C
C
A
AFT INLET
DUCT (LOWER)
MANIFOLD
AIR SCOOP
INLET DUCT
THERMAL SWITCH



Official Photograph, U.S. Navy

Regulated bleed air is routed from the shutoff valve, through a short length of tubing, to a fitting on the lower aft side of the airscoop lip. A thermal switch is attached to the tubing, just downstream from the shutoff valve. This switch is wired in series with a similar switch in the engine internal anti-icing system. When both switches close, a signal light on the Propeller and Engine Ice panel illuminates to indicate that the engine's ice control systems are functioning. Further discussion of thermal switch operation is included under Engine Anti-Icing Systems Operation.

The hot air enters the passage formed by the double-skin construction of the airscoop lip, flows around the circumference of the lip, then passes into a manifold on the right side of the airscoop duct. The manifold distributes the hot air to 12 tapered heat transfer passages that are integral with the double skin that makes up the upper half of the airscoop duct. Small air inlet and exit holes meter the airflow through these passages. System air, now at considerably reduced temperature, is exhausted through the exit holes in the left side of the airscoop to atmosphere in the nacelle.

ENGINE COMPRESSOR INLET HOUSING AND TORQUEMETER SHROUD ANTI-ICING SYSTEM This system prevents ice formation on the engine air inlet housing, its guide vanes and struts, the fuel pressure control probe, and the torquemeter shroud. Following the schematic in Figure 17, air is supplied to the system through two 14th-stage ports, one on either side of the engine compressor. These bleed air ports are separate from those which supply the P-3's engine starting and wing leading edge ice control systems. System air supply is regulated by two anti-icing air valves, one in the supply line from each port. These two valves, in turn, are controlled by a solenoid-operated pilot valve that is open when the solenoid is de-energized. The pilot valve is wired in parallel with the airscoop system control valve, and provides the same "valve open, fail-to-the-safe" feature as its companion system. The bleed air from both anti-icing air valves is routed to the engine air inlet housing for distribution.

Hot air is circulated through passages in the engine air inlet housing to heat its leading edge struts and guide vanes, and to the fuel control probe that projects through the housing into the airstream. A line connects the air passages on one side of the housing with those on the other side to balance the pressure within them. After the air has circulated through these passages, it vents into the engine air inlet.

Part of the anti-icing system branches forward from the engine air inlet housing and circulates bleed air through the torquemeter and drive shaft shroud. A thermal switch is installed on this line near the shroud, and is connected in series with a similar switch in the airscoop anti-icing system. Function of the thermal switches will be discussed in the next section. After the bleed air has traversed the passages in the shroud, it is exhausted into the engine inlet.

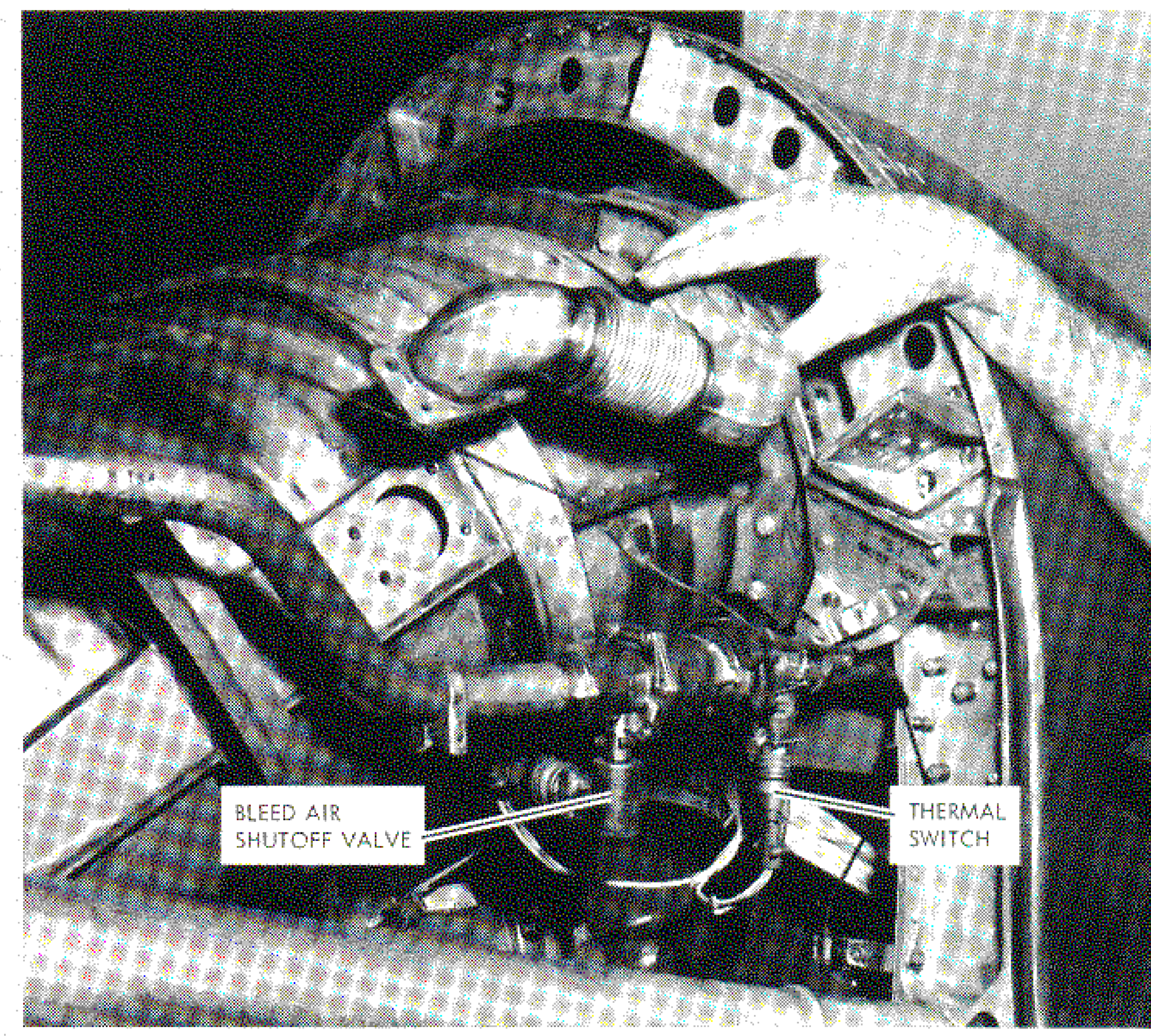


Figure 18 Engine Airscoop Anti-icing System Bleed Air Shutoff Valve and Thermal Switch Installation

ENGINE ANTI-ICING SYSTEMS OPERATION The airscoop and air inlet housing anti-icing systems for each engine are controlled by the engine's Airscoop & Inlet Vanes switch on the Propeller and Engine Ice panel, which is part of the Pilot's Overhead Control Panel. The bleed air control valves for both systems — the airscoop system's shutoff valve and the air inlet housing system's pilot valve — are powered from the Monitorable Essential DC Bus in the Forward Load Center. Due to the "fail-to-the-safe" feature mentioned previously, both solenoid-operated control valves are energized *closed* when electrical power is available and the Airscoop & Inlet Vanes switch is positioned "OFF". When the control switch is positioned "ON," power is removed from the valves' solenoids and the de-energized valves *open,* fully activating that engine's ice control facilities.

The first indication that the engine's anti-icing systems are functioning is registered as a drop on the engine horsepower gage, indicating that the bleed air control valves have opened. Further indication of systems operation is provided by the engine's "ANTI-ICING" advisory light that illuminates when both the airscoop and the engine inlet thermal switches (which are wired in series) close. The thermal switches are attached to the systems' supply tubing and *sense the temperature of the tubing* rather than the temperature of the engine bleed air. When the passage of bleed air has warmed the tubing sufficiently, the thermal switches close to illuminate the "ANTI-ICING" light.

The "ANTI-ICING" light will illuminate either 60 or 90 seconds (approx.) after the bleed air in both systems has reached an operating temperature of about 149°C (300°F), depending upon which of two types of thermal switches are installed on the system's tubing. The engine anti-icing systems on P-3 aircraft BuNo 152765 and earlier were production equipped with thermal switches (Lockheed P/N 613567-9 and -11) that close when they sense a temperature above 66°C (150°F), while later aircraft have been equipped with thermal switches (Lockheed P/N 613567-101 and -103) that close when they sense a temperature above 93°C (200°F). These latter switches are now stocked as spares for all aircraft. Note again that the switches sense the temperature of the tubing rather than the actual bleed air temperature. Within about 60 seconds after the system bleed air temperature has reached 149°C (300°F) on aircraft equipped with the earlier type of switch, the tubing will be sufficiently hot to cause the switches to close. For aircraft equipped with the later type of switch, about 90 seconds will pass before the switches close, assuming that all other operating conditions are the same.

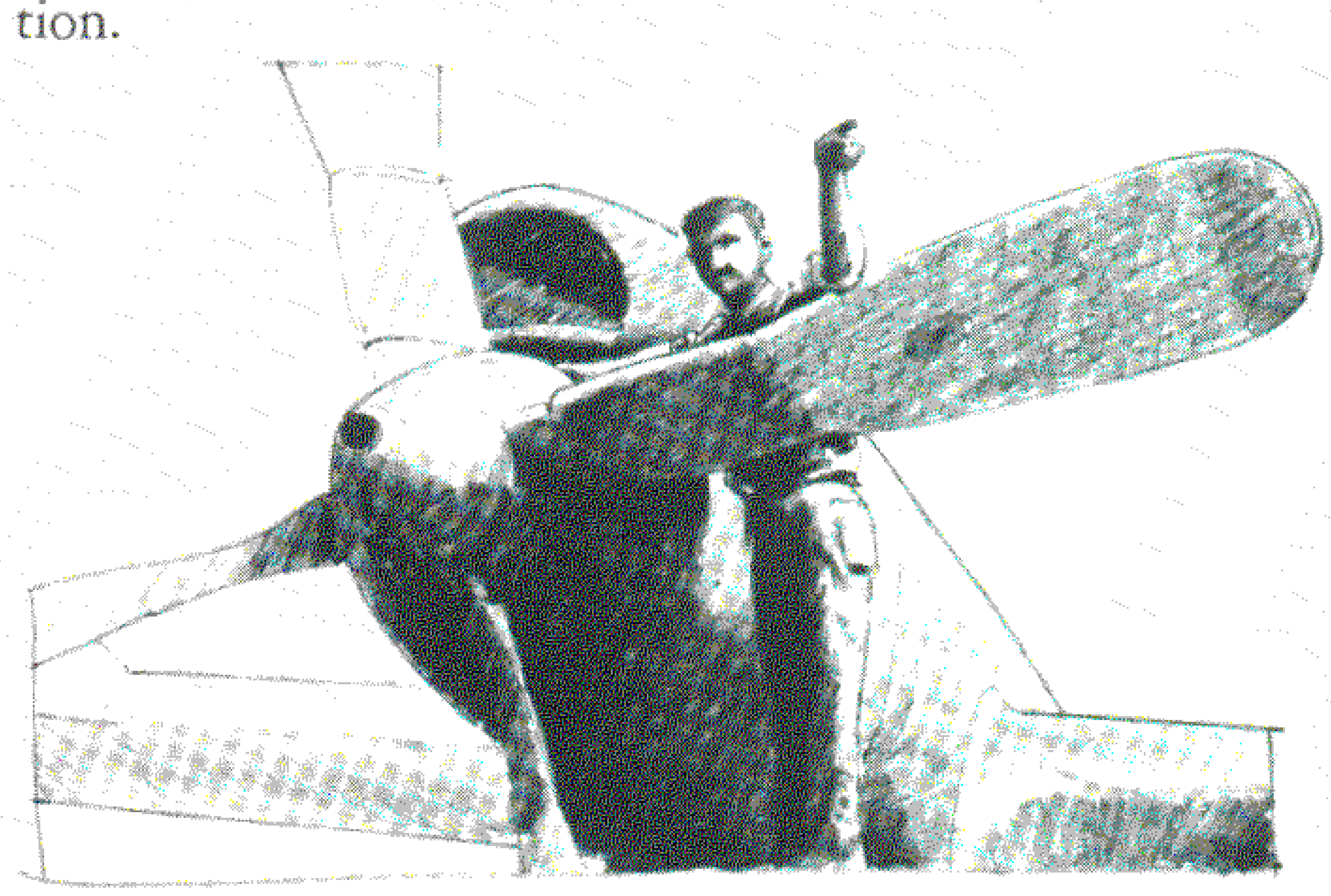
SYSTEM MALFUNCTION INDICATION Proper operation of the engine anti-ice systems is indicated by continuous illumination of the system's "ANTI-ICING" light for the duration of system operation, excluding, of course, the 60 or 90 second "warm up" period. However, if the "ANTI-ICING" light fails to illuminate after the control switch has been positioned "ON" and the warm-up period has elapsed, or if the light illuminates while the control switch is "OFF", a system malfunction is indicated. The shaft horsepower indication of the engine whose anti-icing systems are suspect may be compared with the horsepower indications of those engines whose anti-icing systems are functioning properly to assist in determining if hot air is being supplied to the systems. A similar horsepower drop does not guarantee that the engine's anti-icing systems *are* receiving hot air, but if a similar horsepower drop is not present it is probable that one or both anti-icing systems *are not* receiving adequate hot air.

For example, if the engine anti-ice system control switch is positioned "ON" and the "ANTI-ICING" light does not illuminate, and the engine horsepower drop is similar to that of other engines whose anti-icing switches are "ON" and whose "ANTI-ICING" lights are illuminated, one of that engine's thermal switches may have failed, or the signal light may be defective. However, if the engine horsepower drop is substantially less than that of other engines, it is possible that one system's solenoid-operated valve is malfunctioning.

To determine if engine operation can be continued, observe the suspect engine's airscoop leading edge. Ice accumulations would indicate that the airscoop anti-icing system is malfunctioning. Engine operation can be continued during light icing conditions, but if the ice accumulations become excessive shut down the engine and get out of the icing area.

If the engine anti-icing control switch is "ON," the "ANTI-ICING" light does not illuminate, and at least some engine horsepower drop is indicated, *but no ice is present on the airscoop,* it must be assumed that the engine air inlet and torquemeter shroud anti-icing system is not functioning properly. Since the flight crew cannot visually determine if ice accumulations are present on these surfaces, it must also be assumed that ice is present in quantities that represent a potential hazard to continued operation of the engine. Under these conditions, shut down the engine *immediately* and get out of the icing area.

A malfunction must also be suspected when the "ANTI-ICING" light illuminates while the systems' control switch is positioned "OFF". This could indicate either that (1) the two thermal switches have been actuated by abnormal heat from an unknown source or that (2) the system has lost electrical power, causing the solenoid valves to open and supply the anti-icing systems with hot air. If this condition is noted prior to take-off, it is recommended that the engine be shut down immediately and the aircraft returned to the flight line. If this condition is noted during flight, turn on the engine anti-ice control switch and note if there is a drop in engine horsepower. The NATOPS Flight Manual recommends that if a horsepower drop is observed, "secure the engine unless a greater emergency exists." If no horsepower loss is observed, continue engine operation.



Official Photograph, U.S. Navy

EMPENNAGE ICE CONTROL SYSTEM

The empennage ice control system is an all-electric hybrid de-icing/anti-icing system whose heating elements are integral with the airfoil leading edges. The system is designed solely for airborne operation, although it also has a test mode that can be used to test the circuitry either during flight or on the ground. When the system is energized in flight, its de-icing elements are turned on and off sequentially and cyclically while its anti-icing elements remain on continuously. Electrical power is distributed to the heating elements by a motor-driven timer that completes a cycle of all the sequentially operated circuits in slightly less than 3 minutes. The system is controlled and monitored from the ice control panel in the flight station. An overheat protection system automatically de-energizes all empennage heating elements if an overheat condition is detected.

Major system components include the heating elements in the empennage leading edges; control and test relays in the Main Electrical Load Center; a sensing relay forward of the Main Electrical Load Center and the timer, cycling relay, transformers and circuit breakers in the right rear fuselage at FS 1072. System circuit breakers are on the Main AC Bus A and B panels at the Main Electrical Load Center, and the Monitorable Essential AC Bus and the Main DC Extension Bus at the Forward Electrical Load Center. The system's signal light circuitry runs to the Signal Light Control Box in the Forward Electrical Load Center. The system controls and instrumentation are very simple, consisting of an "ON-OFF-TEST" switch, a fault indicator, and three signal lights. The switch, indicator and "EMP DE-ICE" light are located in the flight station on the left inboard overhead control panel. The master "DE-ICING" light is on the pilot's annunciator panel. The thermal sensor overheat light and relay assembly is in the electronics bay, forward of the Main Load Center. The overheat protection system thermal sensors are located in the left horizontal stabilizer leading edge, one in the parting strip area and the other adjacent to the No. 1 heating element.

The empennage ice control system has twenty de-icing elements, eight each on the left and right horizontal stabilizers and four on the vertical stabilizer. Anti-icing elements called parting strips divide the total de-iced area into ten discrete sections, each of which incorporates two de-icing elements. This arrangement, as shown on Figure 19, prevents ice from bridging the leading edges, thus permitting aerodynamic forces to remove the ice. It also limits ice accumulations to relatively small size so that failure to de-ice any one section will not produce a severe

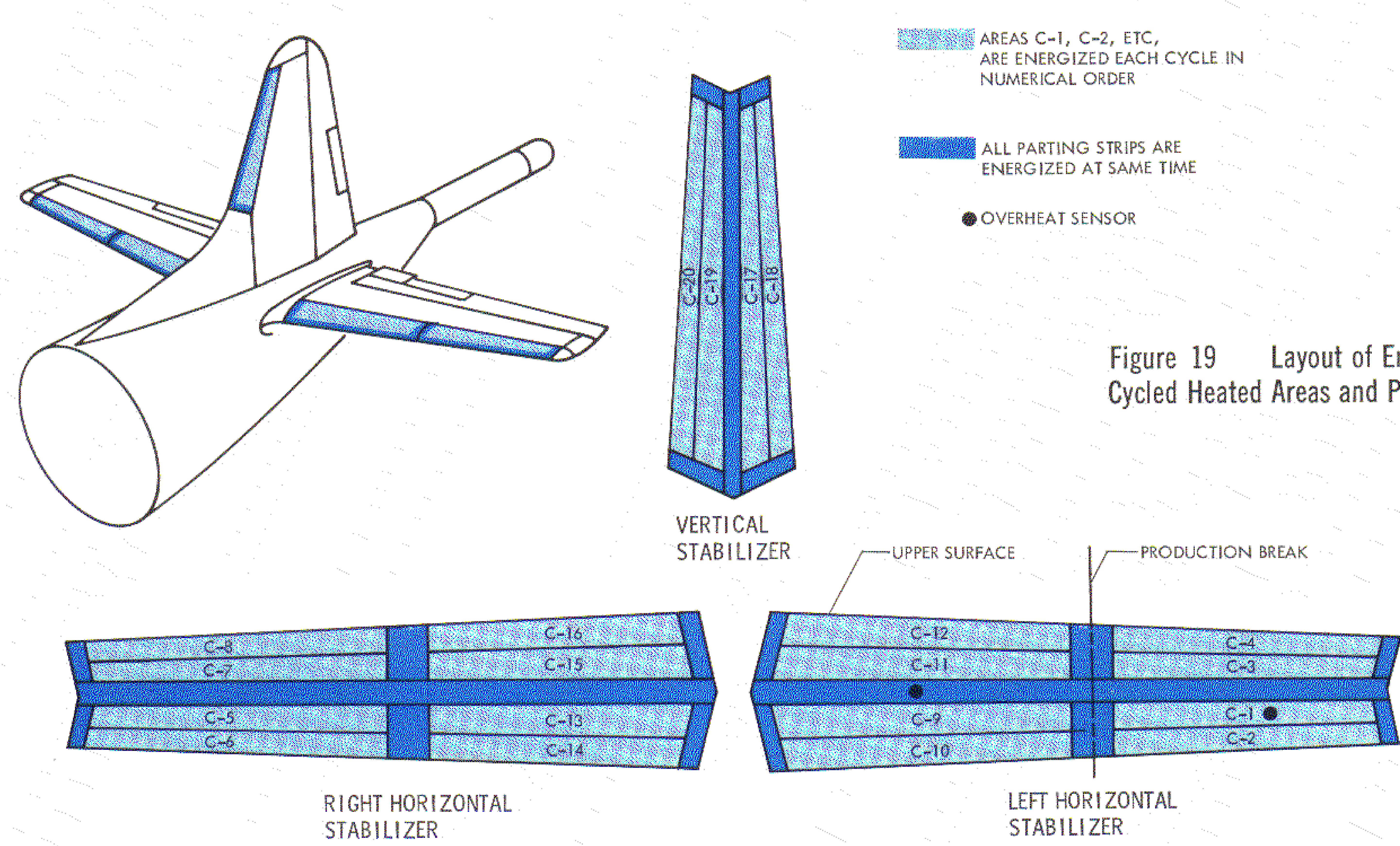


Figure 19 Layout of Empennage Cycled Heated Areas and Parting Strips

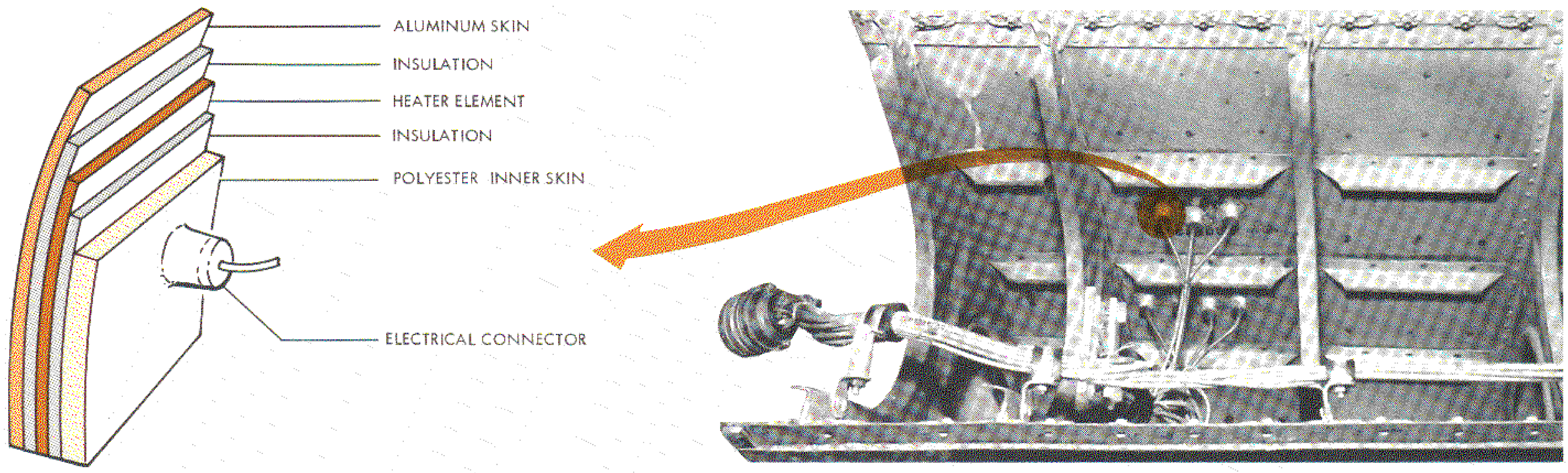


Figure 20 Construction of Empennage Leading Edge

control problem.

There are two modes of system operation: "ON" for normal low-speed operation, and "TEST" for one-cycle high speed operation. During low-speed operation each de-icing element is sequentially energized for 8 seconds, then de-energized for 168 seconds. Counting an 8-second auxiliary dead band at the beginning and an 8-second dead band at the end of each de-icing sequence (during continuous operation this merely becomes a 16-second interval between each cyclic heating sequence), the total time of the cycle is 176 seconds.* The parting strips are energized for the full 176-second cycle. Operation of the system in the low-speed mode is permitted only during flight because airflow and precipitation are required to dissipate the heat from the leading edges.

When the system is operated in the "TEST" mode, it completes one cycle in 44 seconds (four times normal speed), then stops. The first two seconds of the test determine if the low speed motor is operating, then the parting strip circuitry and each of the 20 de-icing element circuits are energized for 2-second intervals. During the test, the condition of the heating element circuitry is displayed on the fault indicator. Condition of the system power control circuitry is reflected by the "EMP DE-ICE" light on the pilot's left overhead control panel.

The system may be tested either during flight or while the aircraft is on the ground. If a ground test is performed, the parting strips are energized for only 2 seconds during the parting strip auxiliary band (to prevent overheating); during an in-flight test the parting strips are energized throughout the cycle. In either case, parting strip operation is displayed on the fault indicator only during the 2-second interval preceding the test of the twenty de-icing heater circuits. *When an empennage ice control system ground test is being conducted, be certain that the GRD AIR SENSING circuit breaker on the Main DC Bus is pushed in. Otherwise the parting strips will be energized throughout the test cycle, overheat, and may damage the leading edges.*

** These times may vary somewhat because the system's timer is only required to be ± 10% accurate, both in normal and test modes.*

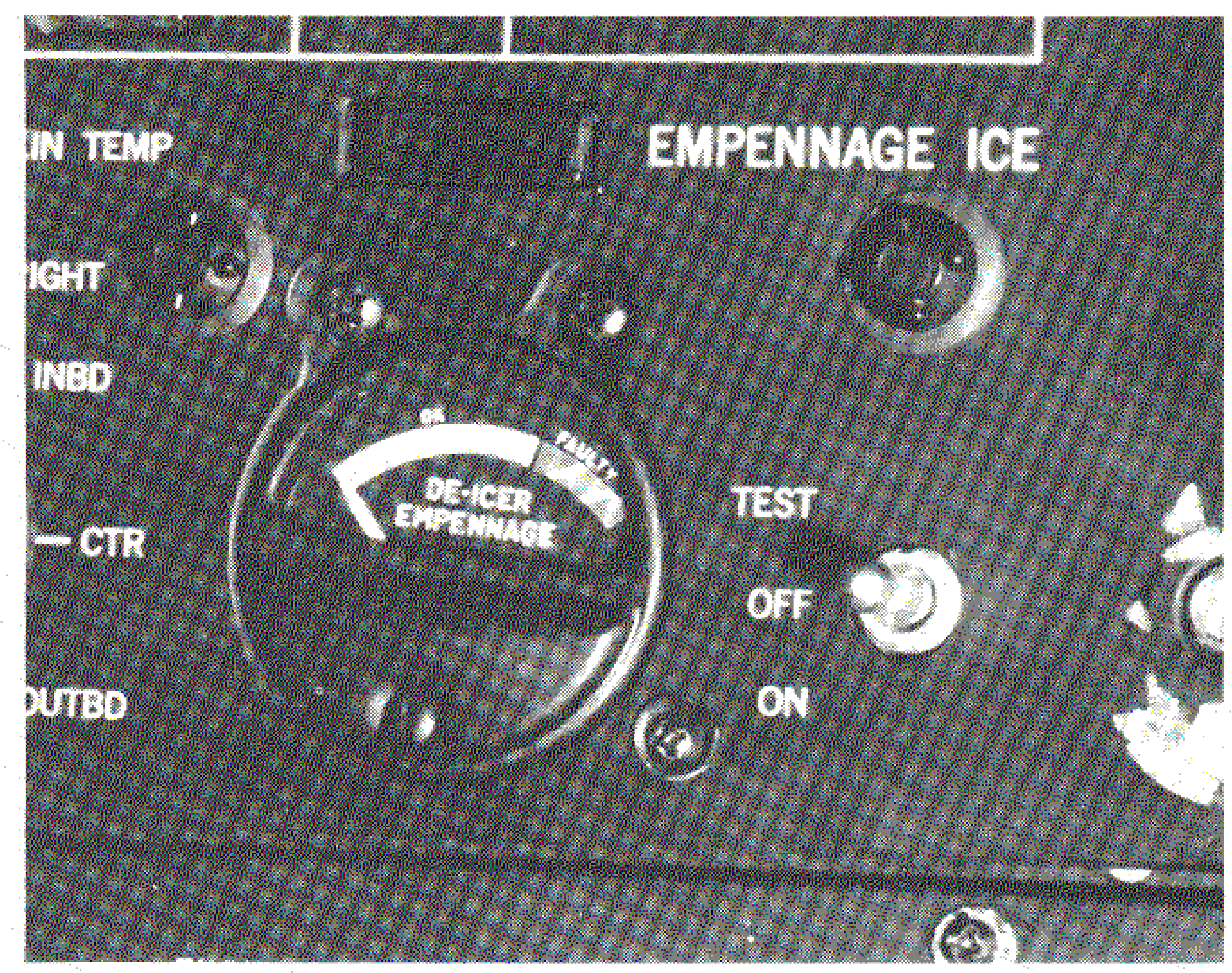
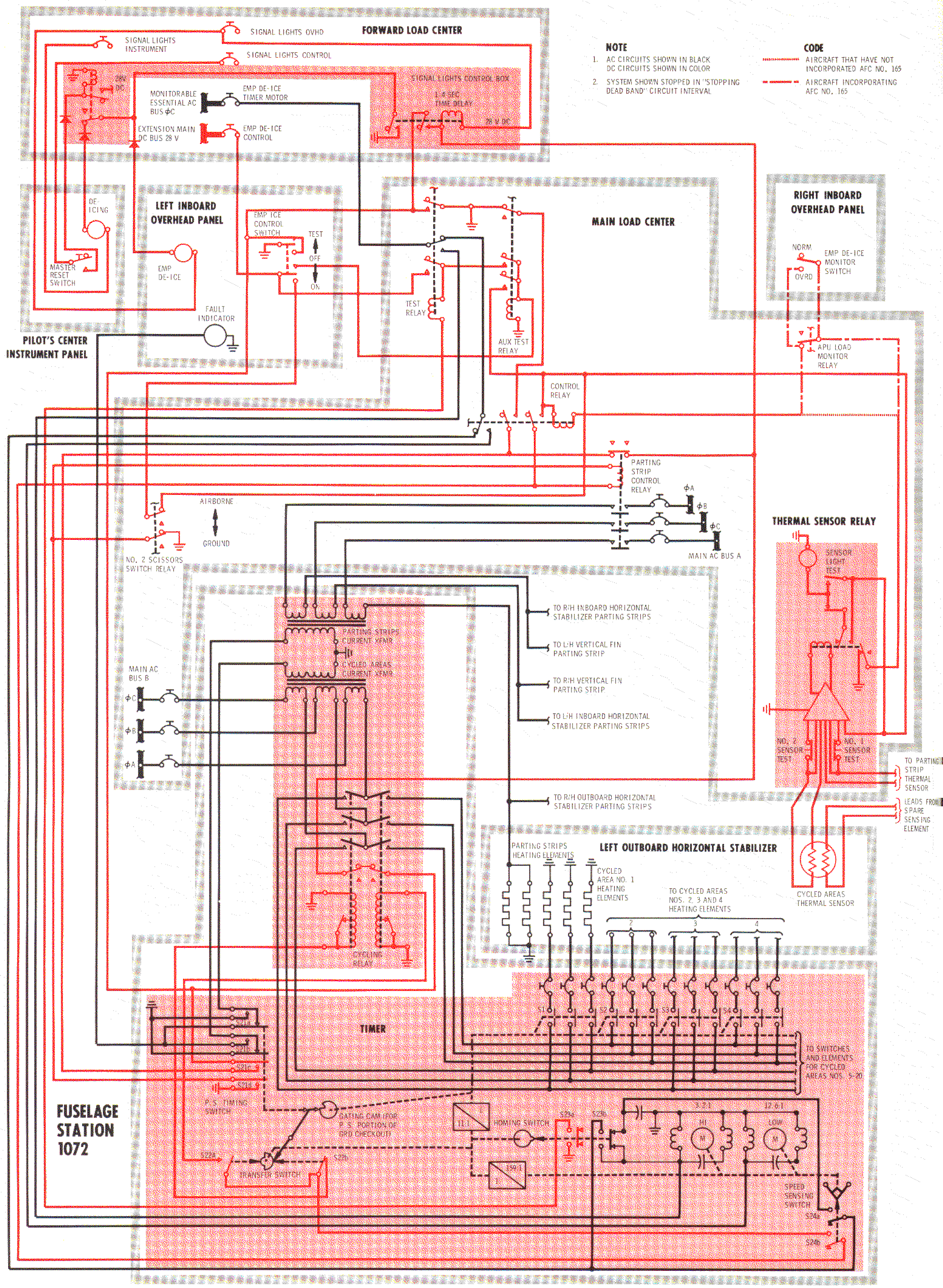


Figure 21 Empennage Ice Control Switch and Fault Indicator

OPERATION As was mentioned previously, the aircraft must be airborne before the empennage ice control system can operate in the normal mode. The system is energized by positioning the Empennage Ice switch "ON" (refer to Figure 22). This routes power from the 28-volt Main DC Extension Bus through the control switch, through the No. 2 scissors switch relay, through the control relay coil, and through contacts of the thermal sensor relay to ground, energizing the control relay. When the control relay is energized, dc power is routed from the control relay through the parting strip control relay coil, though contacts on the No. 2 scissors switch relay to ground, energizing the parting strip control relay.


FORWARD LOAD CENTER
SIGNAL LIGHTS OVHD
SIGNAL LIGHTS INSTRUMENT
SIGNAL LIGHTS CONTROL
SIGNAL LIGHTS CONTROL BOX
28V DC
MONITORABLE ESSENTIAL AC BUS ϕC
EMP DE-ICE TIMER MOTOR
EXTENSION MAIN DC BUS 28 V
EMP DE-ICE CONTROL
1/4 SEC TIME DELAY
28 V DC
NOTE
1. AC CIRCUITS SHOWN IN BLACK DC CIRCUITS SHOWN IN COLOR
2. SYSTEM SHOWN STOPPED IN "STOPPING DEAD BAND" CIRCUIT INTERVAL
CODE
AIRCRAFT THAT HAVE NOT INCORPORATED AFC NO. 165
AIRCRAFT INCORPORATING AFC NO. 165
DE-ICING
MASTER RESET SWITCH
LEFT INBOARD OVERHEAD PANEL
EMP DE-ICE
EMP ICE CONTROL SWITCH
TEST
OFF
ON
FAULT INDICATOR
PILOT'S CENTER INSTRUMENT PANEL
MAIN LOAD CENTER
TEST RELAY
AUX TEST RELAY
RIGHT INBOARD OVERHEAD PANEL
NORM
OVRD
EMP DE-ICE MONITOR SWITCH
APU LOAD MONITOR RELAY
CONTROL RELAY
PARTING STRIP CONTROL RELAY
ϕA
ϕB
ϕC
MAIN AC BUS A
AIRBORNE
GROUND
NO. 2 SCISSORS SWITCH RELAY
THERMAL SENSOR RELAY
SENSOR LIGHT TEST
NO. 2 SENSOR TEST
NO. 1 SENSOR TEST
TO PARTING STRIP THERMAL SENSOR
LEADS FROM SPARE SENSING ELEMENT
CYCLED AREAS THERMAL SENSOR
PARTING STRIPS CURRENT XFMR
CYCLED AREAS CURRENT XFMR
MAIN AC BUS B
ϕC
ϕB
ϕA
TO R/H INBOARD HORIZONTAL STABILIZER PARTING STRIPS
TO L/H VERTICAL FIN PARTING STRIP
TO R/H VERTICAL FIN PARTING STRIP
TO L/H INBOARD HORIZONTAL STABILIZER PARTING STRIPS
TO R/H OUTBOARD HORIZONTAL STABILIZER PARTING STRIPS
LEFT OUTBOARD HORIZONTAL STABILIZER
PARTING STRIPS HEATING ELEMENTS
CYCLED AREA NO. 1 HEATING ELEMENTS
TO CYCLED AREAS NOS. 2, 3 AND 4 HEATING ELEMENTS
2
3
4
CYCLING RELAY
TIMER
S1
S2
S3
S4
TO SWITCHES AND ELEMENTS FOR CYCLED AREAS NOS. 5-20
P.S. TIMING SWITCH
GATING CAM (FOR P.S. PORTION OF GRD CHECKOUT)
HOMING SWITCH
TRANSFER SWITCH
S22A
S22B
S23A
S23B
HI
LOW
SPEED SENSING SWITCH
S24A
S24B
FUSELAGE STATION 1072

When the parting strip control relay is energized, three-phase current from Main AC Bus A is routed through the relay's contacts through three primary coils on the parting strip current sensing transformer, through the parting strips to ground, energizing the parting strips' heating elements. The parting strips will remain energized so long as the control switch is "ON" and no overheat condition exists, when the No. 2 scissors switch relay is de-energized (airborne condition).

With the control relay energized, current is routed from EMP DI TIMER MOTOR ϕ C circuit breaker on the Monitorable Essential AC Bus, through normally closed contacts of the test relay, through the control relay, and through the timer's low speed motor to ground. This energizes the motor to drive the timer's gear train, which in turn operates the timer's speed sensing switch, transfer switch and roller cam.

DC power is picked up from the control relay and routed through the centrifugally driven speed sensing switch (fail-safe switch) to the transfer switch. The transfer switch's two contacts are operated by a double-lobed cam whose lobes are set 180 degrees apart. The cam, mechanically driven by the timer's gear train, closes the transfer switch's two contacts for alternate 8-second intervals. Power from the transfer switch contacts is routed to two complementary coils on the de-icer element cycling relay, energizing the coils alternately.

Three-phase current from Main AC Bus B is routed through the primary windings of the cycled areas current sensing transformer to the set of common contacts on the cycling relay. As the relay's two coils are energized alternately, three-phase current is shunted back and forth through two circuits to the timer's 20 three-phase switches.

Inside the timer, all the odd numbered de-icing

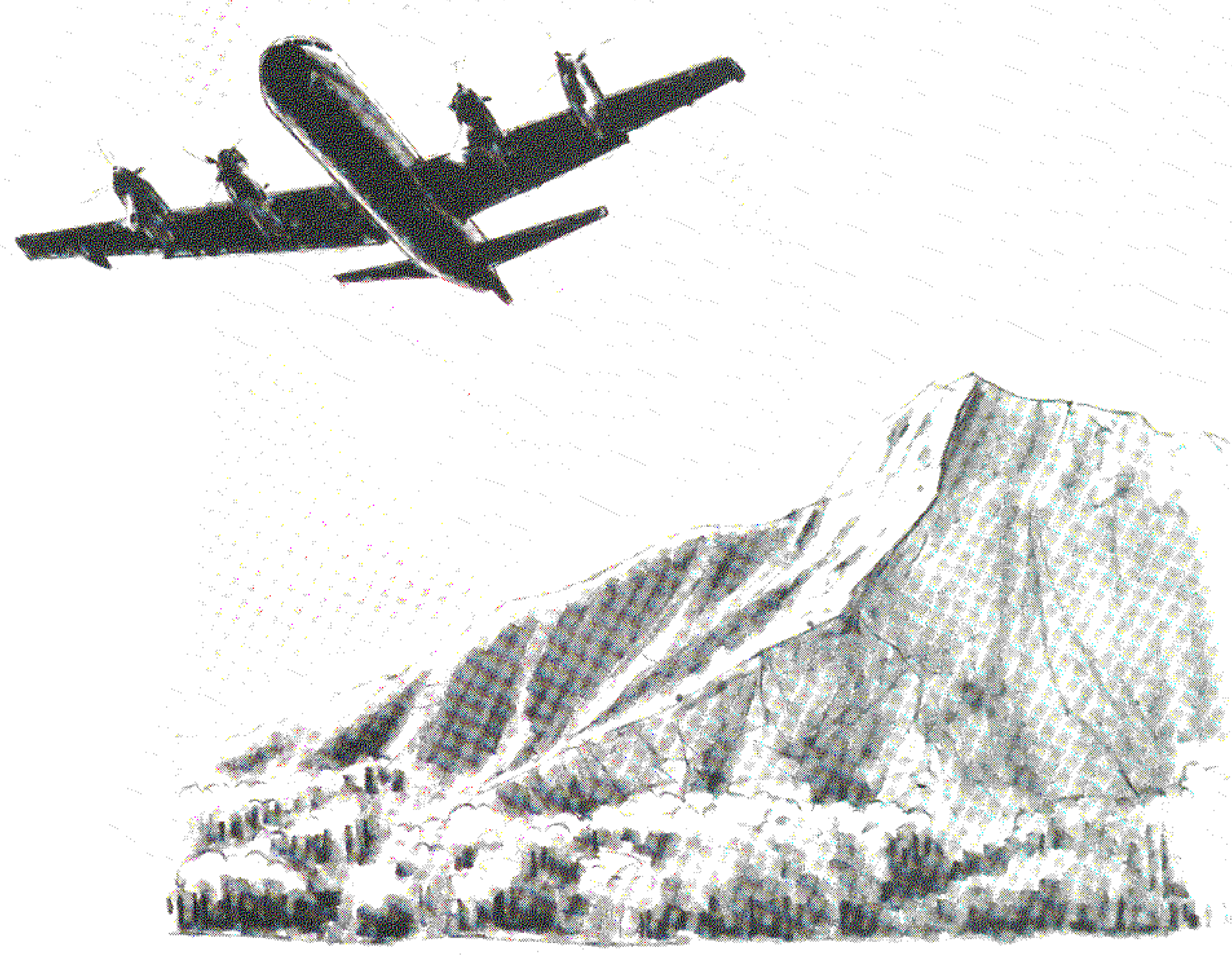


Figure 22
Empennage Ice Control
System Schematic

elements' three-phase switches are connected to one circuit, and the even numbered elements' switches are connected to the other. The de-icing elements are energized as the timer's roller cam actuates the 20 switches progressively throughout the timing sequence. Operation of the timer and the cycling relay is synchronized so that there is a slight overlap (less than 1 second during normal system operation) of switch actuation, unless the timer is at the beginning or end of the cycle. This ensures that the next switch will be closed when its circuits receive high-amperage power from the cycling relay, minimizing arcing between the switch contacts and the attendant erosion damage. The cycling will repeat as long as the Empennage Ice switch is "ON" and the No. 2 scissors switch relay remains de-energized (airborne condition).

When the system is operating satisfactorily, all three signal lights will be extinguished. The fault indicator pointer will be in the "OK" range while the heating elements are energized, and at the extreme left position while the system is operating in the dead band interval. The pointer will pulse every 8 seconds (or 2 seconds during a test), indicating that the parting strips circuitry is operational and that the cycled heating areas are being energized sequentially.

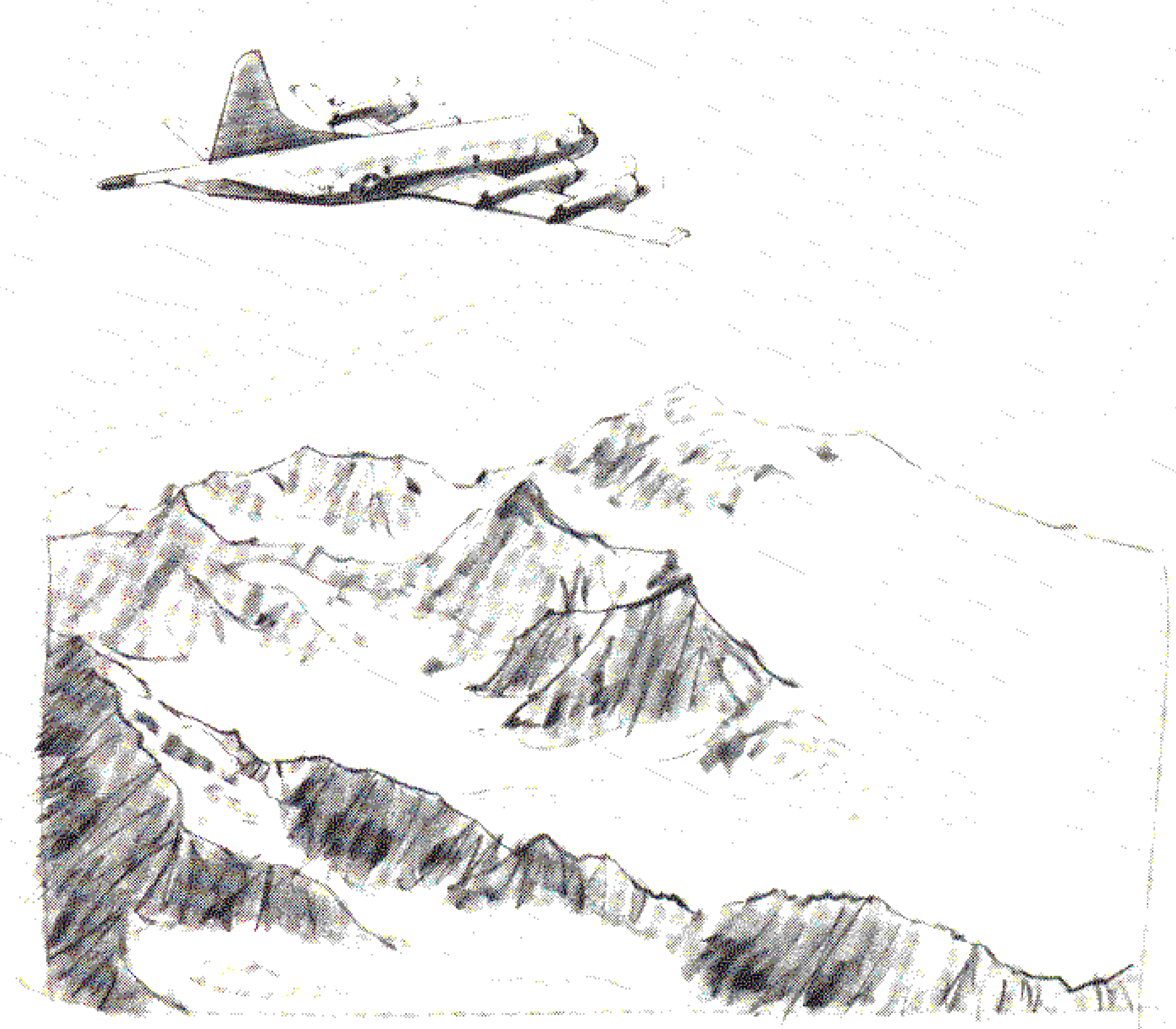
The system is de-activated by positioning the Empennage Ice switch "OFF" or automatically at touchdown by scissors switch actuation. If the control switch is left "ON" at touchdown, the "EMP DE-ICE" light and the master "DE-ICING" light will be illuminated. In either case, the control relay is de-energized, switching ac power from the low speed motor to the high speed motor by way of the timer's "homing" switch. The timer gear train is then driven at four times its normal speed to the end of the timing cycle. Upon reaching the end of the cycle, power is momentarily applied through the speed sensing switch to charge the timer's dynamic braking circuit capacitor. At this point a gear-train operated cam will open the homing switch and shut off the high speed motor. The dynamic braking circuit is applied to the motor through a second set of homing switch contacts, causing the timer to stop in the "stopping dead band" area. If the system is turned "OFF" while the timer is progressing through the "stopping dead band" area during operation in the normal mode, the low speed motor will be de-energized and braked to rest immediately.

Empennage De-icing with One AC Generator The loads on the electrical power system are automatically monitored when only one generator is operating to ensure that power will be supplied to vital aircraft systems. If the operating generator is No. 2, 3 or 4, use of either the empennage or propeller ice control system will cause the load monitoring relays to automatically disconnect the No. 1 and No. 2 ac electronics feeders,* the galley heat, and the radiant floor heaters. Non-essential electrical equipment also should be de-energized manually to reduce the load on the generator, and the empennage and propeller ice control systems should be used singly for the same reason unless the severity of icing requires their simultaneous operation.

If the single operating generator is the APU, additional factors must be considered before using the empennage or propeller ice control system. At altitudes below 10,000 feet the APU is able to handle the same reduced electrical loads as any one of the aircraft's other generators, but above this altitude the APU's engine efficiency declines until, at altitudes above 20,000 feet, use of the APU is not recommended. When empennage or propeller de-icing is to be used above 10,000 feet while the APU is the only operating generator, other equipment with equivalent loads must be manually shut down to avoid overloading the APU if the aircraft has not incorporated P-3 Airframe Change No. 165. The alternatives are listed in the NATOPS Flight Manual, Section V.

P-3 aircraft incorporating Airframe Change No. 165 have an automatic electrical load monitoring system that gives generator No. 4 priority over the APU, and automatically shuts down additional systems if the APU is the only operating generator

* *No. 2 ac electronic feeder only is disconnected on P-3 aircraft incorporating Airframe Change No. 165.*



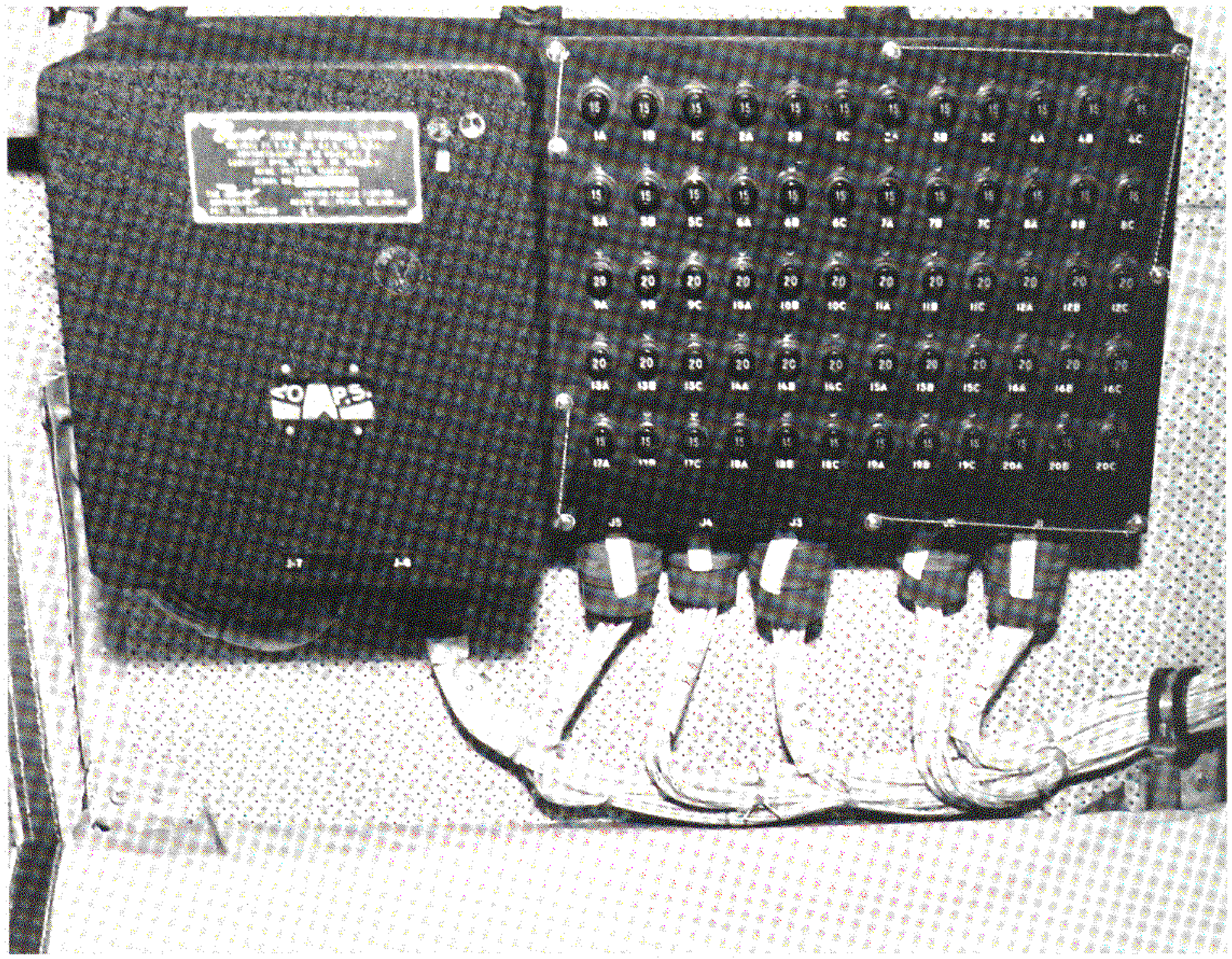


Figure 23a Empennage Ice Control System Timer

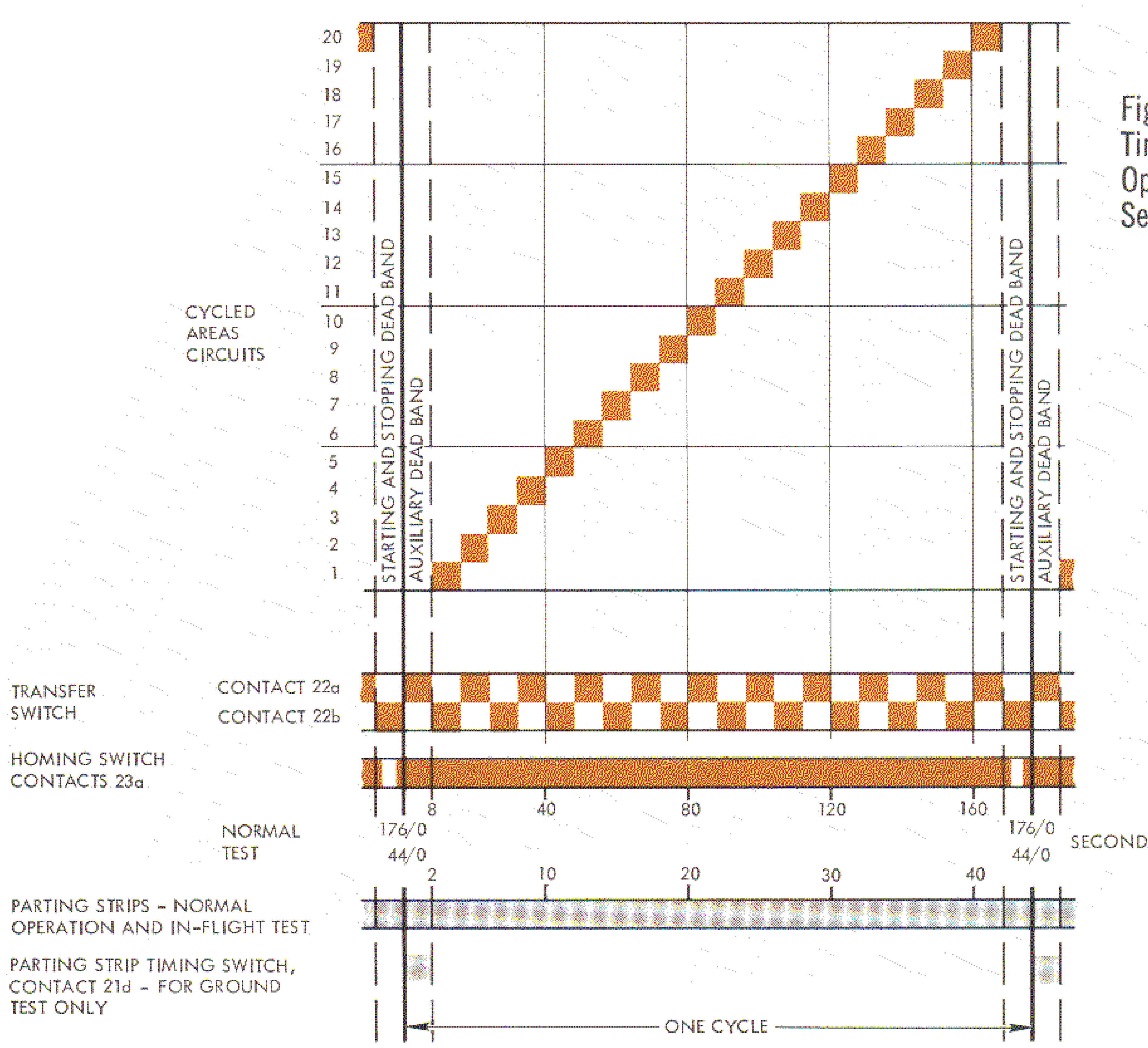


Figure 23b
Timer
Operation
Sequence

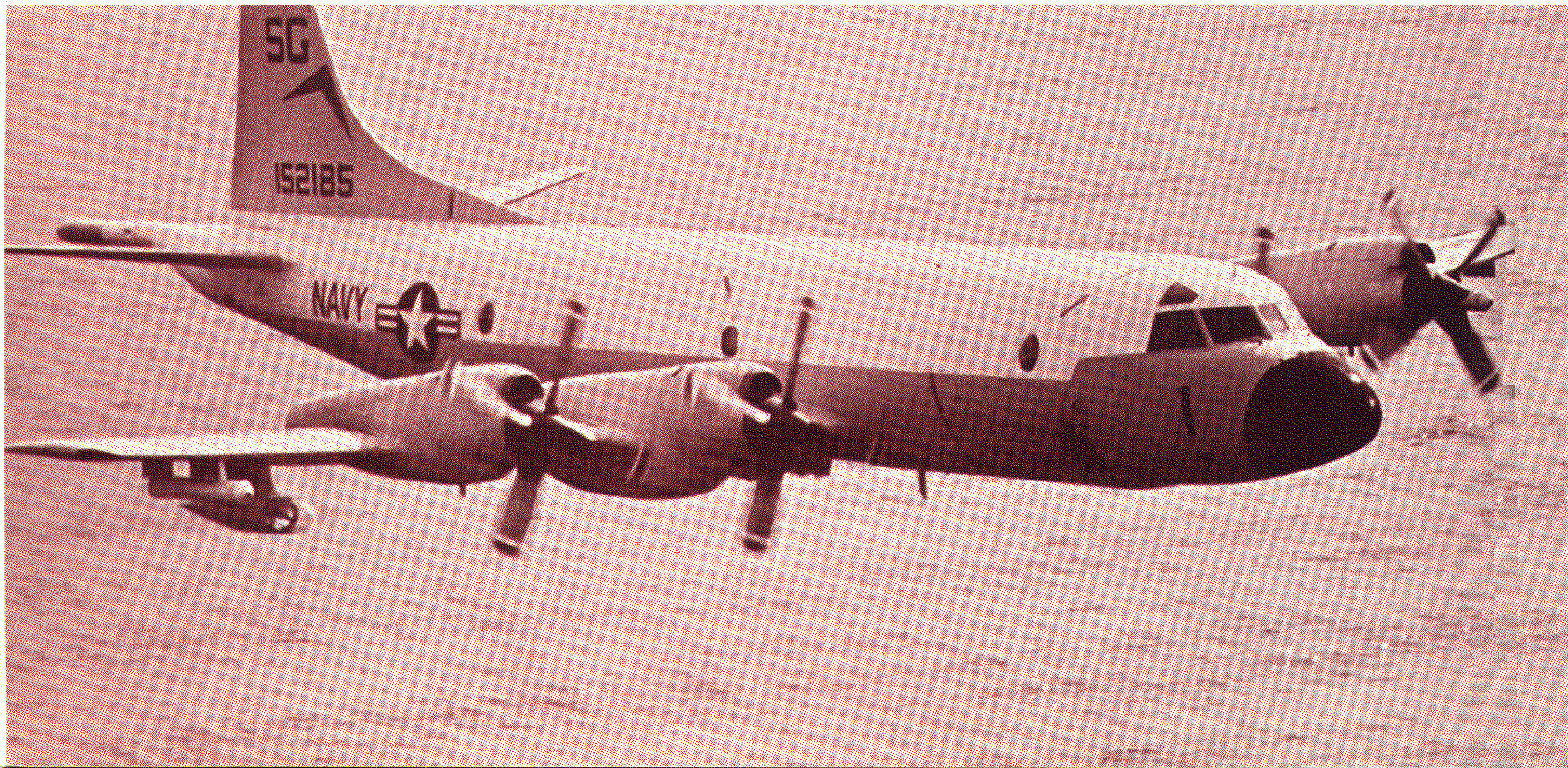
above 8,000 feet (± 500 ft.). Assuming that this latter condition exists, the empennage ice control system is among those additional systems that are shut down automatically. Since operation of the empennage ice control system may still be essential, it can be re-activated with the Empennage De-ice Monitor Override switch on the co-pilot's overhead electrical power system panel. However, electrical loads equivalent to the empennage ice control system must be shut down *before* actuating the monitor override switch to avoid overloading the APU. The exhaust fan, No. 1 and No. 2 hydraulic pumps (boost out), and all fuel boost pumps' loads, when combined, comprise a load equivalent to that of the empennage ice control system. If the monitor override switch has been actuated, a holding relay will keep those systems that were automatically shut down de-energized upon descent through 8,000 feet (± 500 ft.). Use of systems that have been shut down automatically can be restored during operations below 8,000 feet (± 500 ft.) by de-energizing the holding relay. The relay will be de-energized when both the empennage and the propeller de-ice switches are in the "OFF" position at the same time. Always consult the NATOPS Flight Manual, Section V, for guidance during one-generator operation.

FAULTS, MALFUNCTIONS, AND OVERHEATING Empennage ice control system operational problems can be classified as faults in the heating element power supply circuitry, malfunctions of control circuit components, or leading edge overheating. The nature of the problem is reflected by the fault indicator and the signal lights, and the data thus displayed can be useful when seeking the trouble's cause and its remedy. In general, in-flight corrective action is limited to resetting circuit breakers or to positioning the control switch "OFF," then returning it to "ON" in order to clear momentary operational problems. More extensive troubleshooting is described in Section II of the NAVAIR 01-75PAA-12 In-flight Maintenance Manual, but most of these remedial actions can be performed only after completion of the flight. If the trouble cannot be corrected during flight, turn the system "OFF," increase the airspeed to above 200 knots, and get out of the icing area. During empennage ice control system operation the fault indicator should be checked frequently to maintain current knowledge of the system's status.

System Faults Current flow through the empennage heating element power supply circuitry is sensed by the parting strips and the cycled areas current transformers. Signals from these transformers are routed through the parting strips timing switch to the De-Icer Empennage fault indicator. The fault indicator compares the signals from each heated area's power supply to determine if its three-phase current flow is acceptably balanced, or if there is an open circuit or a short. The results are presented on the indicator dial. Note however that a heating element power supply fault will not illuminate any of the system's signal lights nor will it cause the system to shut down automatically, therefore frequent observation of the fault indicator is vital.

The fault indicator compares the current flow of the three phases with one another to determine if their current flow is acceptably balanced. If there is

Official Photograph, U.S. Navy



no current flow, if there is a three-phase short, or if the current flow is perfectly equalized among the three phases, the indicator pointer will remain at the extreme left position. When one or two phases supply more or less current than the remaining phase(s), the indicator senses this phase unbalance and displays it as deflection of the pointer towards the right of the indicator face. If the phase unbalance falls within acceptable limits, the pointer will remain within the indicator's "OK" range. However, if the phase unbalance is excessive, the pointer will deflect past the "OK" range to the "FAULTY" range. It must be emphasized that the degree of pointer deflection does *not* indicate the combined intensity of the current flow in the three phases, but simply how well the current flow is balanced among the three phases.

Three distinct segments of system operation are reflected by the fault indicator — the "dead band" period, parting strip operation, and cycled areas operation. The dead band is an 8-second period during which the system is started and stopped, however during continuous normal system operation the dead band is merely an 8-second interval between cycles. No signals from the heating element circuits are sensed during the dead band period, therefore the fault indicator pointer is at the extreme left of the dial.

Following the dead band period, the parting strip timing switch routes signals from the parting strips current transformer to the fault indicator for 8 seconds. (Although the parting strips are energized throughout the 176-second cycle, only 8 seconds of parting strip operation are checked.) Next, the timing switch is again actuated, this time to route signals from the cycled areas current transformer to the fault indicator for the next 160 seconds. During this period, three-phase current flow balance to each of system's 20 de-iced areas is sensed and displayed by the fault indicator during 8-second intervals. The beginning of each successive 8-second interval is denoted by a pulse of the indicator pointer. After the last of the cycled areas has been energized, the system re-enters the dead band segment of system operation and the indicator pointer returns to the extreme left position on the dial.

The fault indicator will continue to monitor system operation as long as the control switch is positioned "ON." When the system is turned "OFF," the power supply to all system heating elements is removed, causing the indicator pointer to return to the extreme left of the dial. System fault indication is the same during a system test as during normal system operation except that the data is displayed for 2-second intervals.

Thus, power distribution to the heating elements

can be monitored whenever the system is operating, during either the "normal" or the test mode. If an open circuit fault is suspected, it may be possible to clear the fault by checking the EMP DEICE and EMP DEICING PARTING STRIPS circuit breakers at the Main Load Center and the circuit breakers on the timer circuit breaker panel, then resetting any open breakers. If the fault cannot be remedied, it may be necessary to shut down the system or to de-energize the faulty circuit and continue to use partial system heating, depending upon operational contingencies.

Malfunctions Control circuit malfunctions can interrupt power to several or all of the empennage heating elements. If a malfunction occurs, the fault indicator pointer will move to the extreme left of the dial for the length of time that the interruption of current flow is sensed. In addition, the following malfunctions will cause the empennage ice control system signal light circuit to be energized, illuminating the "EMP DE-ICE" signal light and the "DE-ICING" Master Caution Light: the parting strip power relay de-energizes; one or both cycling power relays will not energize; the control relay de-energizes; the EMP DEICER CONT circuit breaker and/or the EMP DI

TIMER MOTOR circuit breaker are open; the timer motor fails; an overheat condition* is sensed or simulated. In-flight remedial action is usually limited to re-setting the circuit breakers if possible, and to cycling the Emp De-ice switch "OFF", then "ON" on the possibility that the malfunction was only momentary.

The control malfunction signal light circuit is comprised of two lights wired in parallel — the "EMP DE-ICE" light and the master "DE-ICING" light — and a 1.4 second time delay relay with a latching

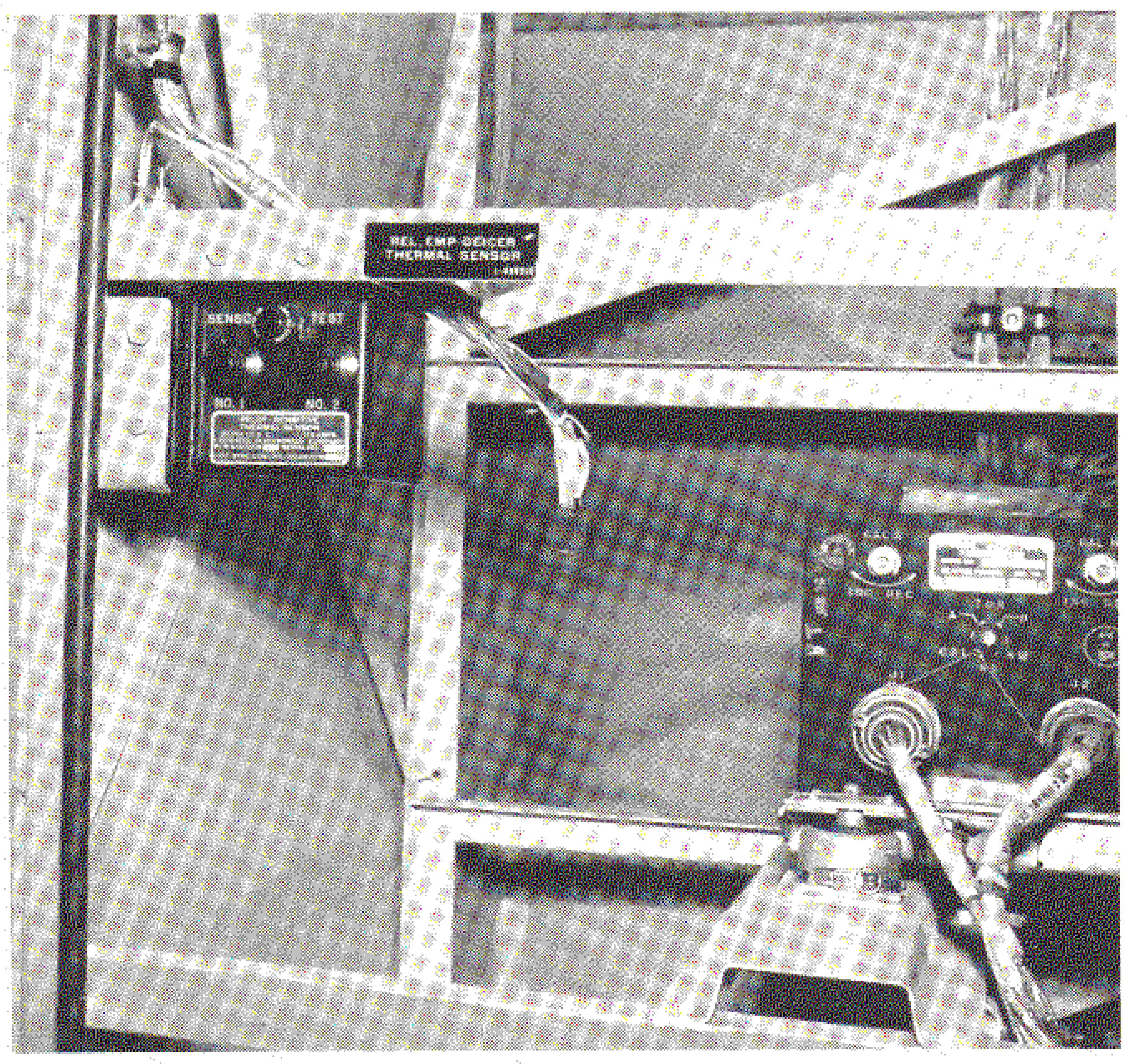


Figure 24 Empennage Thermal Sensor Relay Installation

circuit (see Figure 22). The "EMP DE-ICE" and master "DE-ICING" lights are powered from the SIGNAL LIGHTS OVHD and SIGNAL LIGHTS INST circuit breakers respectively, and the time delay relay is powered from the SIGNAL LIGHTS OVHD circuit breaker. If any of the previously mentioned malfunctions occur, the time delay relay is energized and is held in this position by its latching circuit. When this relay is energized, the signal lights circuit is grounded through a set of the relay's contacts, illuminating both signal lights. The relay will remain energized, as will the signal lights circuit, until the Emp De-ice switch is positioned "OFF." However, the master "DE-ICING" light can be extinguished without positioning the Emp De-ice switch "OFF" by pressing the Master Reset switch next to the "DE-ICING" light.

Leading Edge Overheating The empennage leading edges are protected from overheating whenever the empennage ice control system is energized. Power is routed from the Extension Main DC Bus to the thermal sensor relay assembly by way of the No. 2 scissors switch relay or the auxiliary test relay, depending upon whether the ice control system is being operated in the normal or test mode. If either thermal sensor (thermistor) detects a temperature of 104°C ± 6° (220°F ± 10°) or more, the sensor relay assembly is signaled. Upon receipt of the signal, the relay assembly's transistorized circuitry energizes the thermal sensor relay. This interrupts power to the system's control relay and illuminates the signal light on the thermal sensor relay assembly.

Since the control relay is de-energized, power is removed from all of the heating elements (causing the fault indicator pointer to move to the extreme left of the dial) and the "EMP DE-ICE" and "DE-ICING" lights are illuminated. If the overheat condition occurs while the system is operating in the normal mode, the timer will switch from low speed to high speed operation when the control relay is de-energized. When the empennage leading edge cools to 82°C ± 6° (180°F ± 10°), the thermal sensor relay is de-energized, the control relay circuit is completed again, the thermal sensor relay signal light is extinguished, and the empennage ice control system resumes operation. However, the "EMP DE-ICE" and "DE-ICING" lights will remain illuminated until the latching circuit on the time delay relay is de-energized by positioning the system's control switch "OFF."

To recapitulate, an overheat condition is indicated by illumination of the "EMP DE-ICE" light next to the fault indicator, the master "DE-ICING" light on the center annunciator panel, and illumination of the thermal sensor relay's signal light. And, since the power to the heating elements is shut off, the fault indicator pointer will be at the extreme left of the dial for the duration of the overheat condition. However, during operational conditions it may be only a few seconds until the leading edges cool sufficiently to negate the overheat signal, de-energize the thermal sensor relay, extinguish the relay's signal light, and permit the system to resume operation. This period may be so brief that the thermal sensor relay could be de-energized and its signal light extinguished before a crew member can gain access to the relay to observe its signal light. Thus, the crew could find itself with an empennage ice control system that has apparently

* *Actually, an overheat condition is not a bona fide control circuit malfunction, but it does affect the system the same way. The system's control relay is de-energized for the duration of the overheat condition (or the simulated overheat condition), energizing the signal light circuit and illuminating an additional signal light on the thermal sensor relay. Overheating is discussed in greater detail in the latter part of this section.*

resumed "normal" operation, but with the "EMP DE-ICE" and master "DE-ICING" signal lights still illuminated.

The following procedure is included in the NATOPS Flight Manual to enable the flight crew to determine if it is possible to use the empennage ice control system after the "EMP DE-ICE" light illuminates. Attempt to extinguish the "EMP DE-ICE" and "DE-ICING" signal lights by positioning the Emp De-ice switch "OFF" for 2 or 3 seconds, then return it to "ON." If the signal lights are extinguished and the system resumes normal operation, it is probable that a transient overheat condition occurred. Under these conditions system operation may be continued, although special attention should be given the system fault indicator and signal lights for the remainder of the operational period. If the signal lights are illuminated again when the control switch is returned to "ON," it is indicative of a system malfunction or of a persistent (or recurring) overheat condition. Under these conditions shut down the system, increase airspeed to above 200 knots, and get out of the icing area. After the system has been shut down, its circuit breakers can be checked to determine if operation can be restored. Upon completion of the mission, thoroughly check the system's electrical circuitry and the empennage leading edges to determine if the system has been damaged and to ascertain why the signal lights were illuminated.

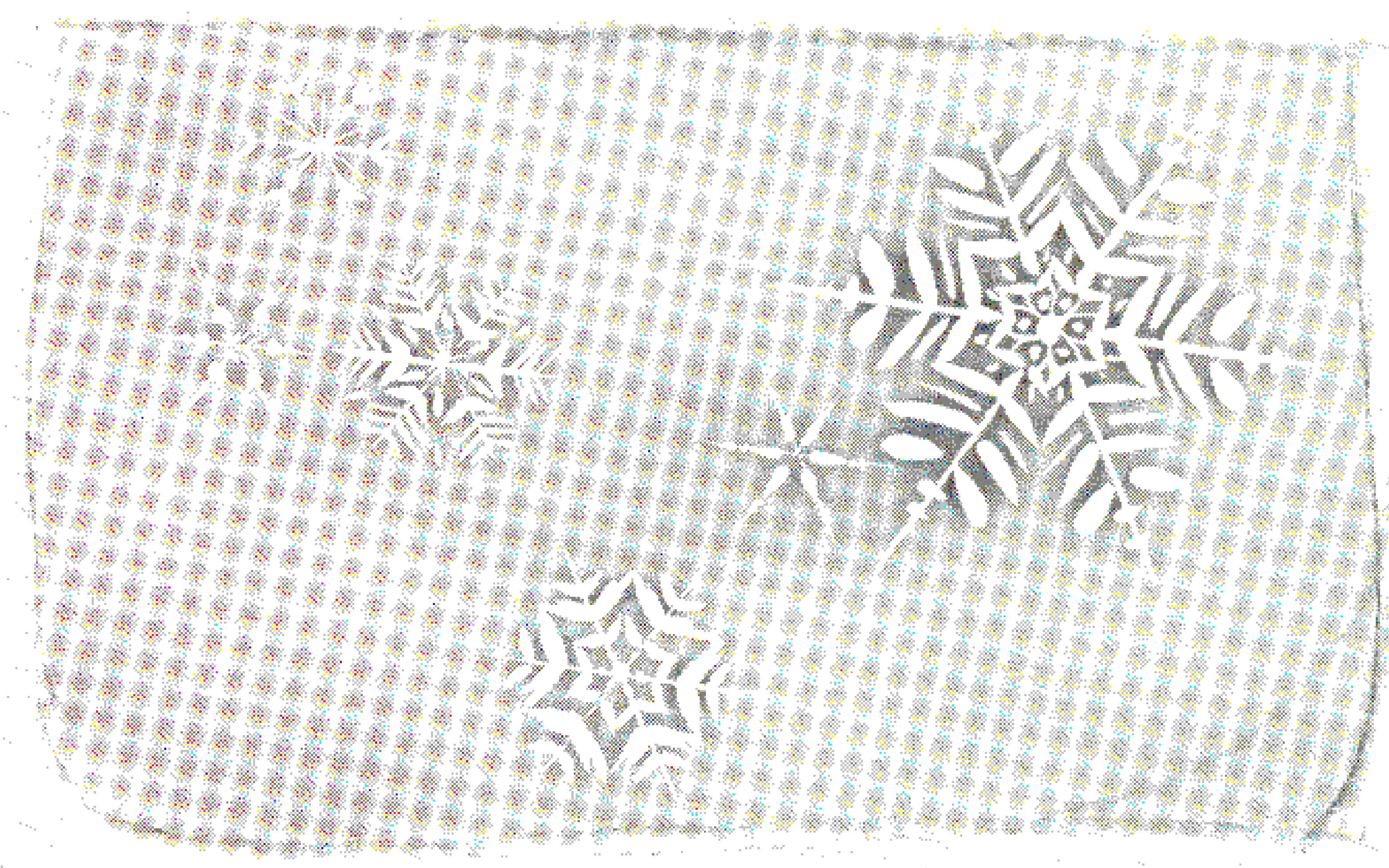
There are two makes of thermal sensor relay assemblies, the Lockheed Electronics relay assembly and the United Controls assembly. Both models are physically and functionally interchangeable, however the United Controls assembly is more likely to be encountered because it superseded the other model during P-3A production and has also become the standard spares relay assembly.

The thermal sensor relay assembly has two test buttons, one for each sensor circuit. Sensor circuit continuity can be checked by pressing each circuit's test button whenever power is available to the sensor relay's amplifier. This applies high resistance to the sensor circuit to simulate an overheat condition, causing the relay assembly amplifier to respond as if an actual overheat condition is sensed. This continuity check is described further under "SYSTEM TEST". If icing conditions are anticipated during the mission, the thermal sensor circuitry should be tested prior to takeoff.

Each thermal sensor has two resistors, one connected to the overheat sensing system and the other serving as a spare. In case of burnout, the sensor's defective resistor can be disconnected and the spare resistor substituted into the system. Detailed repair instructions for the empennage leading edge are contained in the NAVWEPS 03-25EK-11 Empennage Leading Edge Assemblies overhaul manual. Subjects covered include skin repairs, heating element repairs, and thermal sensor replacement.

SYSTEM TEST The empennage de-icing system is equipped with a test circuit that will energize the system for one complete cycle, operating the system at high speed. This system test was designed primarily with ground testing in mind so that the heating elements would not overheat during conditions when heat dissipation is at a minimum. The condition of the system may be observed on the fault indicator and the "EMP DE-ICE" signal light either during the test period or during normal system operation. During ground maintenance operations, allow at least 5 minutes to elapse between consecutive tests to provide ample time for the heating elements to cool.

The test is initiated by holding the Empennage Ice switch in the "TEST" position, causing several events to occur almost simultaneously. First, the aux-



iliary test relay is energized and supplies power from the Extension Main DC Bus to the control relay coil. If the thermal sensors are not signaling an overheat condition, power will proceed from the control relay coil to ground by way of contacts on the thermal sensor relay. Now that the control relay is energized by way of the test relay, ac power is routed through one set of control relay contacts to energize the low speed motor, and dc power through another set of control relay contacts to the parting strip relay coil. Concurrently, dc power is routed from the auxiliary test relay, through the test relay coil, to a set of open contacts on the timer's homing switch.

During an in-flight test the parting strip relay is energized throughout the test cycle by grounding the relay's coil through closed contacts of the No. 2 scissors switch relay. Power is then routed from Main AC Bus A through the parting strip relay contacts, through the parting strip transformer's primary coils, to the parting strip heating elements for the duration of the test cycle. During a ground test the parting strips are energized only briefly, as will be explained subsequently.

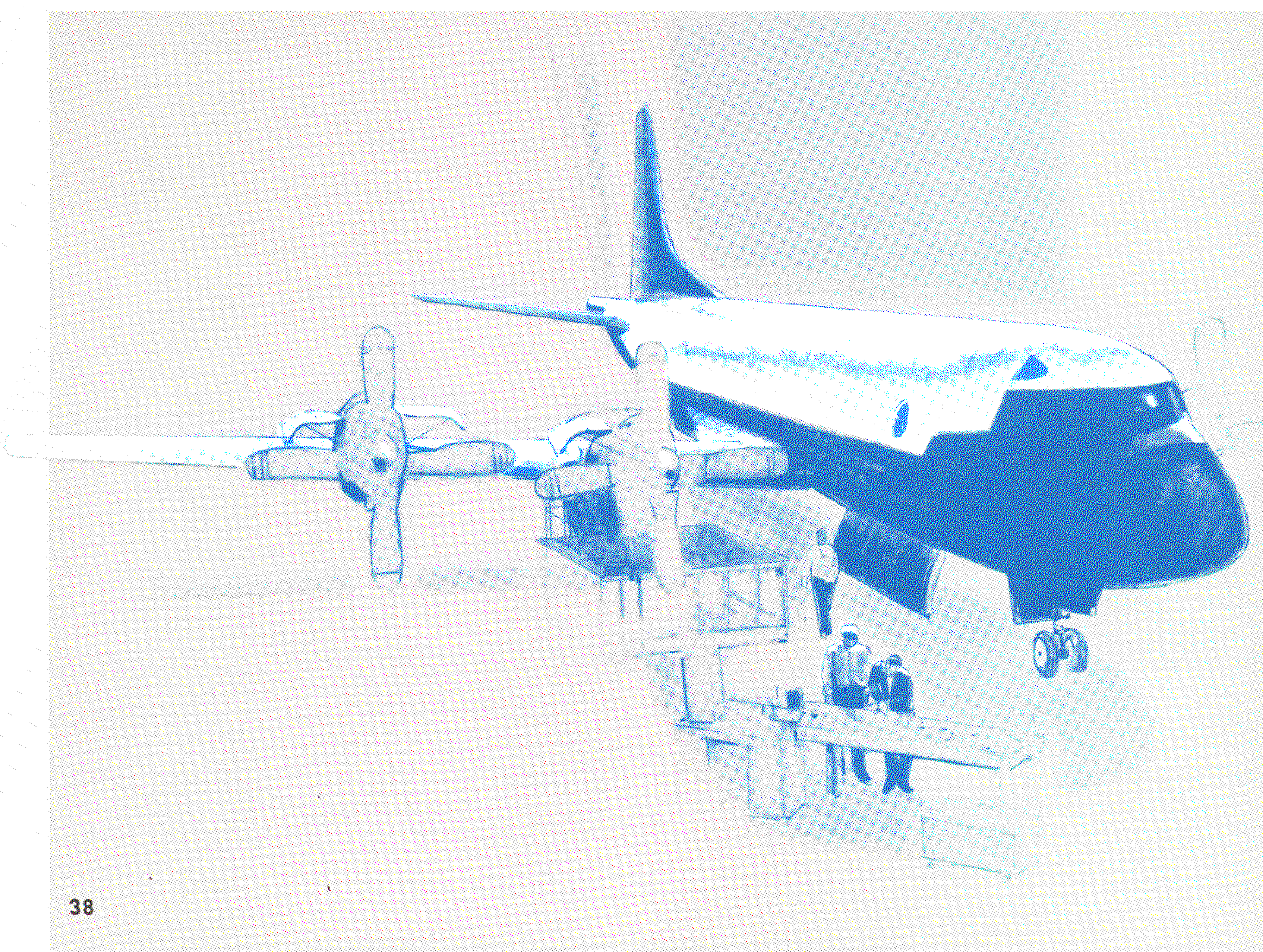
After low speed motor operation has been demonstrated for approximately 2 seconds, the cam-actuated homing switch closes and energizes the test relay. This establishes a holding circuit for both the test relay and the auxiliary test relay, and switches ac power from the low speed motor to the high speed motor. The Empennage Ice switch may now be released to the "OFF" position.

The next 2-second segment of the test checks the operation of the parting strip heating elements. If the test is conducted during flight, the heating elements have already been energized, as described previously. If this is a ground test, the parting strip relay is now energized through the S21d contacts of the cam-operated parting strip timing switch. Power is routed to the heating elements through the relay and transformer, but only for the duration of this 2-second operational check. At the same time, contacts on the S21b part of the timing switch close to complete a circuit from the parting strip transformer secondary coil to the fault indicator.

Performance of the parting strip heating elements is now reflected on the fault indicator. After 2 seconds the parting strip timing switch is again actuated, switching the fault indicator circuit from the parting strips transformer to the de-icing elements transformer, and de-energizing the parting strip relay ground operation circuit.

The next segment of the test checks the operation of each of the de-icing element circuits for 2 seconds each. As the timer's roller cam actuates each set of heating element switches, the cam-operated transfer switch alternately energizes the cycling relay coils to direct Main AC Bus B power to the odd and even numbered de-icing circuits. System performance is displayed by the fault indicator and the "EMP DE-ICE" light. Upon completion of the cycle — about 44 seconds — the timer's homing cam will open the homing switch and de-energize the system.

The final segment of the test checks the operation of the thermal sensor relay assembly. First, position the Emp De-ice switch to "TEST". This not only sets the empennage ice control system in operation, it also energizes the thermal sensor relay assembly circuitry. If the system is being checked during ground maintenance operations, remember to ob-

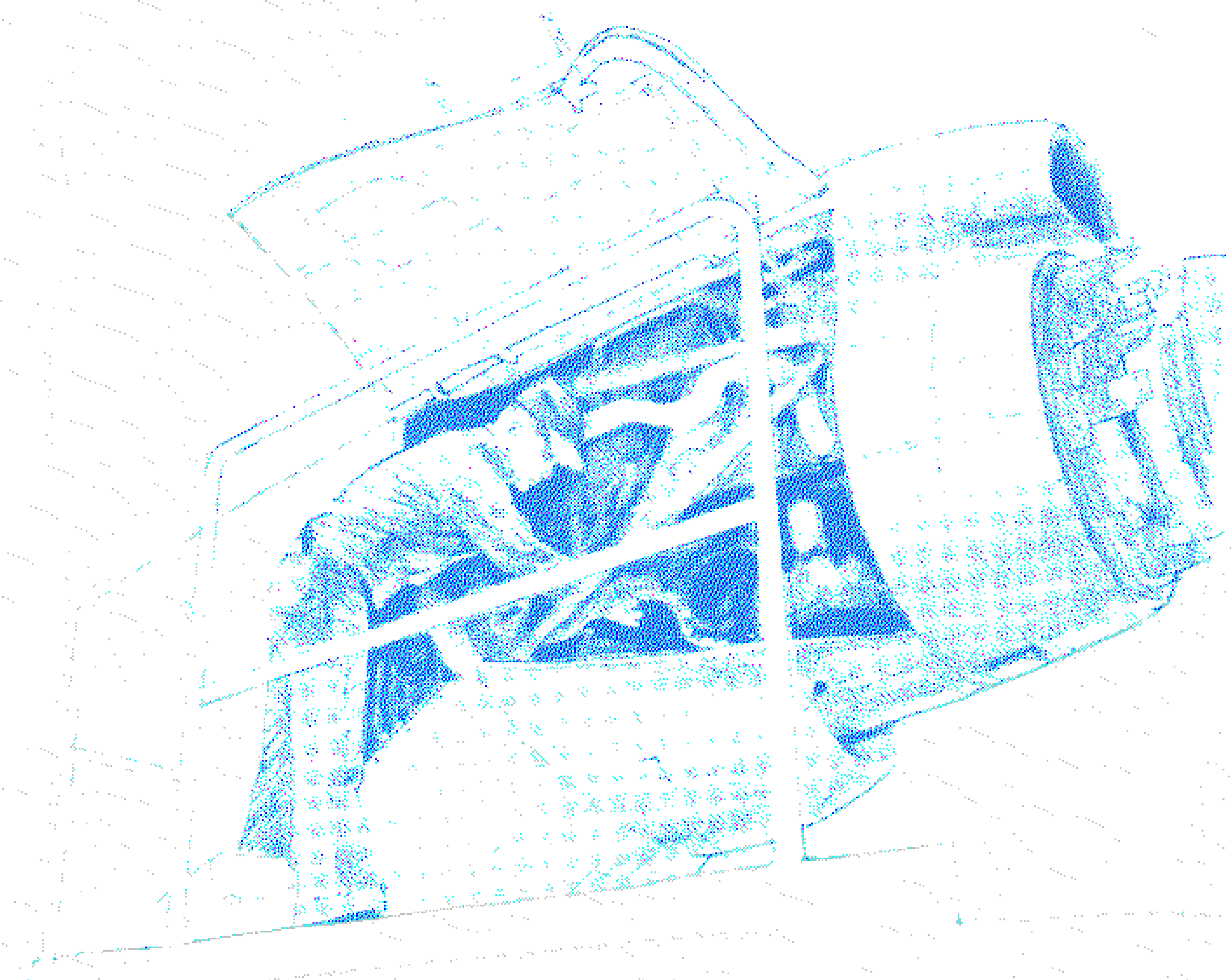


serve the 5-minute interval between consecutive tests. Determine if the relay assembly's signal light is operable by pressing on the light. If the relay assembly is receiving power and if the light is operable, it will illuminate. Next, test the parting strips thermal sensor circuit by pressing the "No. 1" test button on the relay assembly and hold for approximately 2 seconds to simulate an overheat condition. If the circuit is functioning properly and the thermal sensor relay is energized, the relay assembly signal light will illuminate. Operation of the thermal sensor relay is also indicated by illumination of the "EMP DE-ICE" and "DE-ICING" signal lights which will remain illuminated until the test cycle is completed. The thermal sensor relay will be de-energized and its signal light extinguished when the "No. 1" test button is released. Press the "No. 2" button on the relay assembly to simulate an overheat condition in the cycled areas temperature sensor circuitry. If the circuitry is functioning properly the thermal sensor relay will again be energized and its signal light will illuminate. The relay will be de-energized and its signal light extinguished when the "No. 2" test button is released.

PROPELLER ICE CONTROL SYSTEM

Propeller icing is controlled by electrically heating the spinners and blade cuffs. When the system is energized, the nose portion of each spinner is continuously anti-iced while the remainder is cyclically de-iced by elements in the aft spinner (skirt), in the spinner's four blade-root fairings (islands), and in the cuff covering the shank of each blade. The system uses 115-volt power from Main AC Bus B for heating these elements — B phase for de-icing, A and C phases for anti-icing — and 28-volt dc power for control and for distributing Phase B power to the cyclically heated elements. A low-voltage test circuit is incorporated in the system, permitting continuity checks to be made on the ac power circuitry without raising the temperature of the heated components.

The system's major components are the heating elements, the controls and instruments, and the timer and power distribution relays. The spinner heating elements are integral with the spinner structure, but the cuff elements are separate from the blade shank fairings — a replaceable external heater assembly is secured to the leading edge of each fairing. As shown in Figure 26, the controls and instruments are comprised of a three-position control switch (ON-OFF-TEST), an ammeter and a three-position rotary selector for the ammeter. These are located on the Propeller and Engine Ice Panel in the flight station. The system timer and the power distribution relays are located in the Main Electrical Load Center.



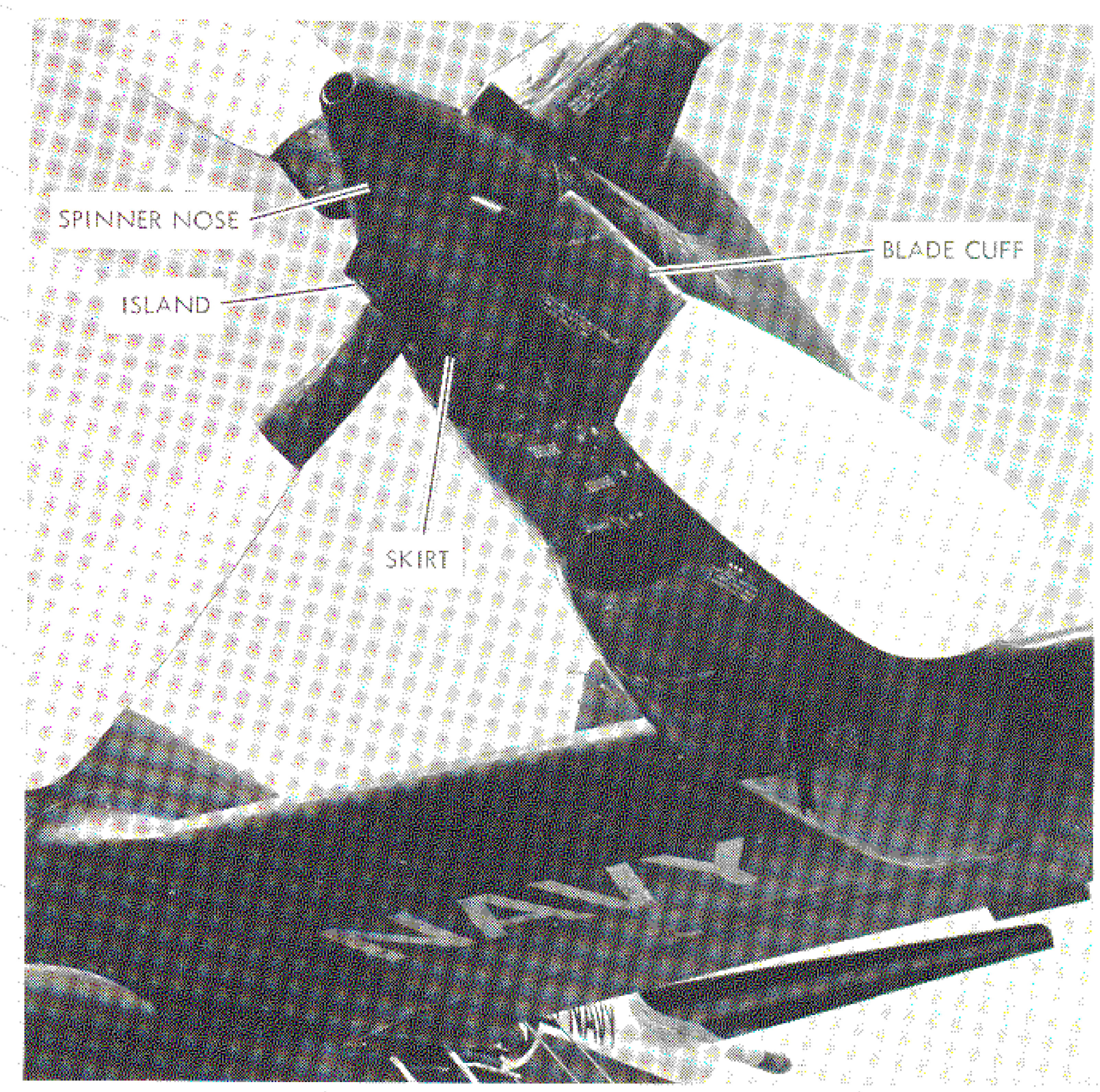


Figure 25 Propeller Heating Element Locations

Figure 26 Propeller Ice Protection System Controls

OPERATION Positioning the control switch "ON" while the aircraft is airborne* closes two control circuits originating from the Extension Main DC Bus (see Figure 27). One circuit, from the PROP DE-ICE CONT circuit breaker, is routed through contacts on the No. 1 Scissors Switch Relay and the De-ice Test Relay to energize the Prop Anti-ice and De-ice Power Relay; the other circuit, from the PROP DE-ICE TIMER circuit breaker, is routed through a different contact of the No. 1 Scissors Switch Relay to the system's timer.

When the Prop Anti-ice and De-ice Power Relay is energized, 3-phase Main AC Bus B power (from the 90-amp PROP ANTI-ICE AND DE-ICE POWER circuit breakers) is supplied to three single-phase power distribution busses. A transformer-type pick-up senses current flow to each bus. From these busses, power is taken from 50-amp Anti-ice or 60-amp De-ice circuit breakers, also mounted on Main AC Bus B panel, through the propeller slip rings to the heating elements. Phase A and Phase C power is supplied continuously to the forward spinner anti-icing elements — Phase A to the outboard props, Phase C to the inboard props. Phase B power is supplied sequentially to the two groups of cycled de-icing elements on all four propellers.

The energized timer motor operates 7 cam-actuated switches to sequentially energize and de-energize the 8 Prop De-ice power relays which distribute Phase B power to the heating elements in the order shown on Figure 27. The timing cycle traverses the four propeller stations from left to right, first energizing the elements in a propeller's blade cuffs for 20 seconds, then energizing the skirt-and-island elements for an equal interval before proceeding to the next propeller. With 40 seconds total dwell on each propeller, 160 seconds are required to complete the cycle. Since the timer has no homing device, there is no fixed point of origin for the prop de-icing cycle as there is with the empennage de-icing cycle. When the system is turned on, the timing sequence will resume at the point it was interrupted at last use.

The ammeter and its adjacent selector on the Propeller and Engine Ice Control panel permit individual read-out of prop ice control loads on A, B, and C phases of Main AC Bus B. With the selector in its normal position ("CUFFS and ISLANDS") the cycling de-icing loads on Phase B are reported, pulsing normally at 20-second intervals between 58-to-72 amps (during cuff de-icing) and 68-to-82 amps (during skirt-and-islands de-icing). When the selector is held to the "1 & 4 SPIN" position, the ammeter reports the steady anti-icing loads on Phase A due to continual heating of the outboard prop spinners; the loads of the inboard spinners on Phase C are reported when the selector is held to the "2 & 3 SPIN" position. Normally these loads are 84 to 100 amps on each phase.

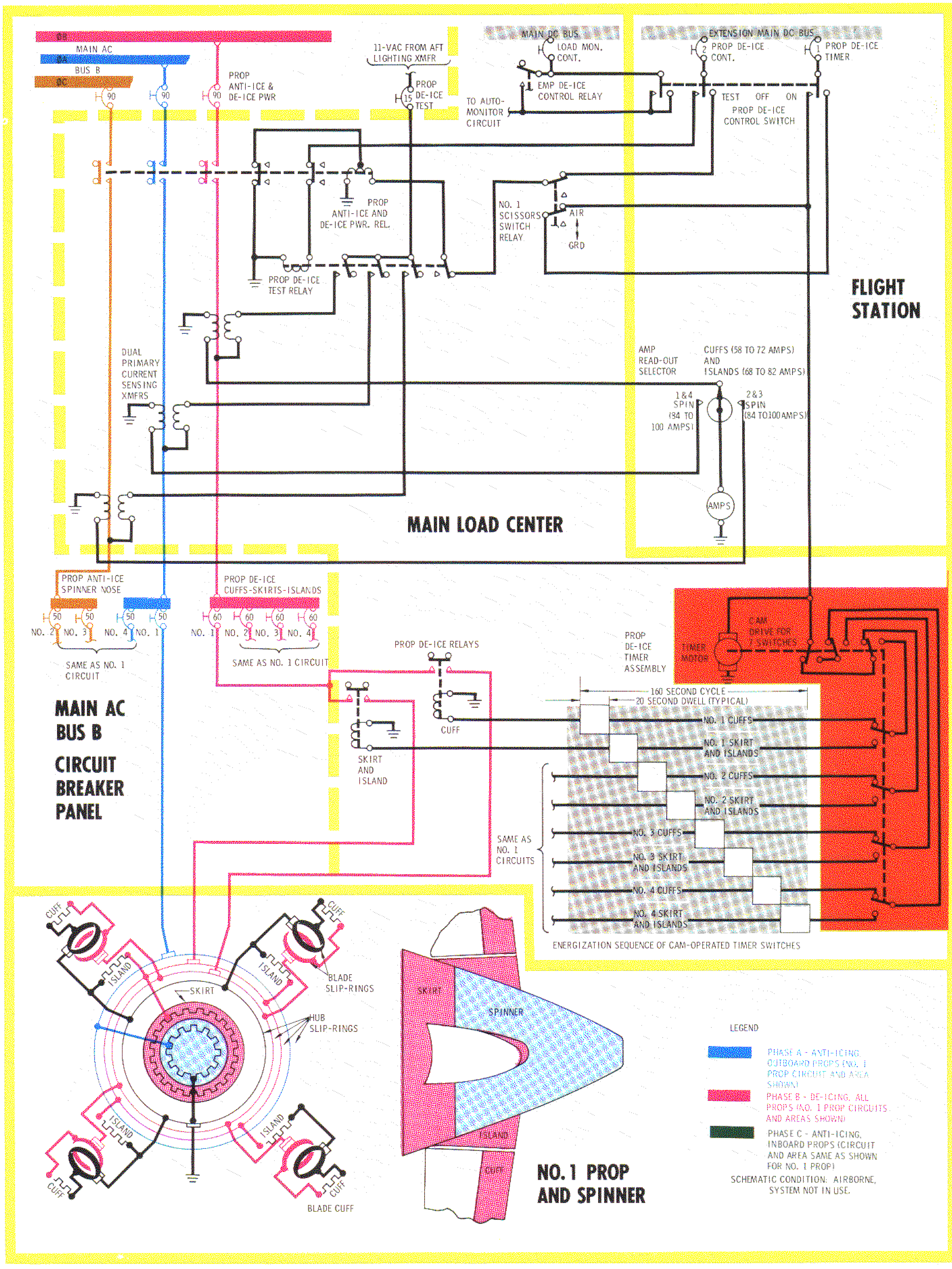
SYSTEM TEST A special 11-volt tap on the Aft Lighting Transformer provides system test power through the 15-amp Prop De-ice Test circuit breaker mounted on the Main AC Bus B panel. When "TEST" is selected at the Prop De-ice Control switch, 11-volt power is introduced into the three leads that normally supply A, B, and C-phase 115-V power to the prop ice control circuits. This low voltage actually induces a very low current flow (protecting the anti-icing and de-icing elements against overheating, much as the accelerated test cycle protects the empennage ice control elements) but the amperage reported on the system ammeter will be approximately the same as is reported during normal operation. This is due to the special dual primary coupling at the ammeter pick-ups shown in Figure 27.

Control power in the Test mode by-passes Scissor Switch Relay No. 1; thus a check can be run either in-flight or on the ground. Inasmuch as dynamic components (slip rings) are involved, a check made with engines operating is a truer test of operation.

As mentioned previously, both the propeller and the empennage ice control systems are interlocked with the automatic electrical load monitoring system. Selecting either "ON" or "TEST" at the Prop Ice Control switch will arm the auto-monitor system, and a number of non-essential loads, including ASW search systems, will be de-energized if a single generator is supporting both Main AC busses. Since this is usually the case during ground operation, personnel making maintenance checks of the Prop or Empennage ice control systems should take care that they do not inadvertently interfere with work on the many systems supported by the Main Bus A Electronic Feeders. If work cannot be coordinated to avoid conflict, the Auto-Monitoring system can be de-activated temporarily by opening the LOAD MON CONT circuit breaker on the Main DC Bus while running a test cycle of the prop ice control system.

* *The "ON" position of the control switch has no effect, either in supplying 115-V AC for heating or 28-V DC for the de-ice timer operation, unless the No. 1 Scissors Switch Relay is in its de-energized (airborne) position. However, a third contact of the switch always arms the Auto Monitor circuit when the switch is positioned either to "ON" or "TEST" whether or not the aircraft is airborne.*

Figure 27
Propeller Ice Control
System Schematic



MAIN AC
BUS B
11-VAC FROM AFT LIGHTING XMFR
MAIN DC BUS
LOAD MON. CONT.
EXTENSION MAIN DC BUS
PROP DE-ICE CONT.
PROP DE-ICE TIMER
PROP ANTI-ICE & DE-ICE PWR
PROP DE-ICE TEST
TO AUTO-MONITOR CIRCUIT
EMP DE-ICE CONTROL RELAY
TEST OFF ON
PROP DE-ICE CONTROL SWITCH
PROP ANTI-ICE AND DE-ICE PWR. REL.
NO. 1 SCISSORS SWITCH RELAY
AIR
GRD
PROP DE-ICE TEST RELAY
FLIGHT STATION
DUAL PRIMARY CURRENT SENSING XMFRS
AMP READ-OUT SELECTOR
CUFFS (58 TO 72 AMPS) AND ISLANDS (68 TO 82 AMPS)
1&4 SPIN (84 TO 100 AMPS)
2&3 SPIN (84 TO100AMPS)
AMPS
MAIN LOAD CENTER
PROP ANTI-ICE SPINNER NOSE
PROP DE-ICE CUFFS-SKIRTS-ISLANDS
NO. 2 NO. 3 NO. 4 NO. 1
NO. 1 NO. 2 NO. 3 NO. 4
SAME AS NO. 1 CIRCUIT
SAME AS NO. 1 CIRCUIT
PROP DE-ICE RELAYS
CUFF
SKIRT AND ISLAND
PROP DE-ICE TIMER ASSEMBLY
TIMER MOTOR
CAM DRIVE FOR 7 SWITCHES
160 SECOND CYCLE
20 SECOND DWELL (TYPICAL)
NO. 1 CUFFS
NO. 1 SKIRT AND ISLANDS
NO. 2 CUFFS
NO. 2 SKIRT AND ISLANDS
NO. 3 CUFFS
NO. 3 SKIRT AND ISLANDS
NO. 4 CUFFS
NO. 4 SKIRT AND ISLANDS
SAME AS NO. 1 CIRCUITS
ENERGIZATION SEQUENCE OF CAM-OPERATED TIMER SWITCHES
MAIN AC BUS B CIRCUIT BREAKER PANEL
CUFF
ISLAND
SKIRT
BLADE SLIP-RINGS
HUB SLIP-RINGS
BLADE CUFF
SPINNER
NO. 1 PROP AND SPINNER
LEGEND
PHASE A - ANTI-ICING OUTBOARD PROPS (NO. 1 PROP CIRCUIT AND AREA SHOWN)
PHASE B - DE-ICING, ALL PROPS (NO. 1 PROP CIRCUITS AND AREAS SHOWN)
PHASE C - ANTI-ICING, INBOARD PROPS (CIRCUIT AND AREA SAME AS SHOWN FOR NO. 1 PROP)
SCHEMATIC CONDITION: AIRBORNE, SYSTEM NOT IN USE.

TROUBLE INVESTIGATION When an out-of-tolerance ammeter reading is noted in flight, it is advantageous to subsequent maintenance if the problem is explored sufficiently before landing to provide a complete report of symptoms and to designate the fault area exactly. The low-voltage test-cycle run after engine shutdown may not always disclose faults caused by engine vibration and slip-ring discontinuities.

For example, if outboard spinners read low, in-flight ammeter readings taken operating in normal and test modes with first the No. 1 and then the No. 4 Spinner anti-ice circuit breaker pulled will disclose which engine is affected, and provide valuable supplemental data to direct maintenance action.

A malfunction of the Cuffs-Skirts-Islands de-icing presents a greater possibility of trouble due to eccentric ice build-up, and if operating in icing conditions the flight crew may wish to expeditiously determine which engine is affected. If the fault produces a significant ammeter reading (above zero), this can be accomplished as follows:

1. Select Prop Ice Control switch "OFF" immediately after out-of-tolerance reading is registered on ammeter.
2. Pull all four 60-amp Prop De-ice circuit breakers on Main AC Bus B.
3. Select "ON" at Prop Ice Control switch, and reset the Prop De-ice circuit breakers in rapid sequence until amperage is indicated (test must be finished before the timer completes the 20-second dwell period).

A malfunction which produces a zero ammeter reading cannot be directly located by the foregoing procedure, but by continuing operation past the zero-amperage interval, the position of the next interval in the timing cycle can be established in the same manner. Then, by noting whether "Cuffs" amperage (62 amps) or "Skirt and Islands" amperage (72 amps) has been in progress during the test interval, the fault can be assigned either to the tested engine position (if "Skirts and Islands" de-icing was in progress) or to the preceeding engine in the test cycle (if "Cuffs" de-icing was in progress). To confirm that the correct engine has been located, the Prop De-ice system should be run through a full 160-second cycle with the suspected engine's PROP DE-ICE circuit breaker pulled, observing that the ammeter registers zero for 40 seconds and pulses in the normal range for 120 seconds.

The foregoing technique has proven to be effective in flight testing at Lockheed, and it provides a simple and direct means of identifying a problem area in flight. However, since the amperage tolerance for cuffs de-icing (58 to 72 amps) overlaps the skirts & islands tolerance (68 to 82 amps), a mistake could be made in identifying the first significant reading following a zero-amperage interval, and *it is essential to always carry out the full-cycle check as described to assure that the troublesome engine position has been correctly identified.*

AVOID OVERHEAT DAMAGE The number of reported incidents in which prop spinner integral heating elements have been ruined by overheat suggests laxity in observing approved procedures among both operating and maintenance personnel.

Although the Ground/Air sensing system will automatically de-activate the propeller and empennage ice control systems at touchdown, Step 6 of the NATOPS "After Landing" check-list specifically directs the Flight Engineer to select "OFF" at the Ice Control Panel switches. If the switches are left "ON," any subsequent action which de-energizes the Ground/Air sensing system will energize spinner heating elements in still air, and some or all of them will be ruined in a very short time.

Instructions for jacking the aircraft specify, as a preliminary step, opening and tagging a list of circuit breakers that ensures against incidental damage to systems, such as the electrical ice control systems, which would otherwise become operable when weight is removed from the landing gear. Nevertheless, most incidents of overheat occur while the aircraft is on jacks.

Personnel may feel there is little reason to delay maintenance checks of the prop and empennage ice control systems until the aircraft is removed from the jacks, reasoning that since Test-mode operation is permissible when airborne or on the ground, both systems may be operated in Test mode at will.

An airplane on jacks is neither airborne nor on the ground.

Operation of the prop system in Test mode while the aircraft is jacked is not harmful, but working on any system while its automatic safety feature is disabled can easily lead to mishaps. If the control switch is inadvertently selected "ON," full 115-V bus power will be connected to the prop heating elements, and damage will result if operation is not terminated within a few minutes. ▲▲

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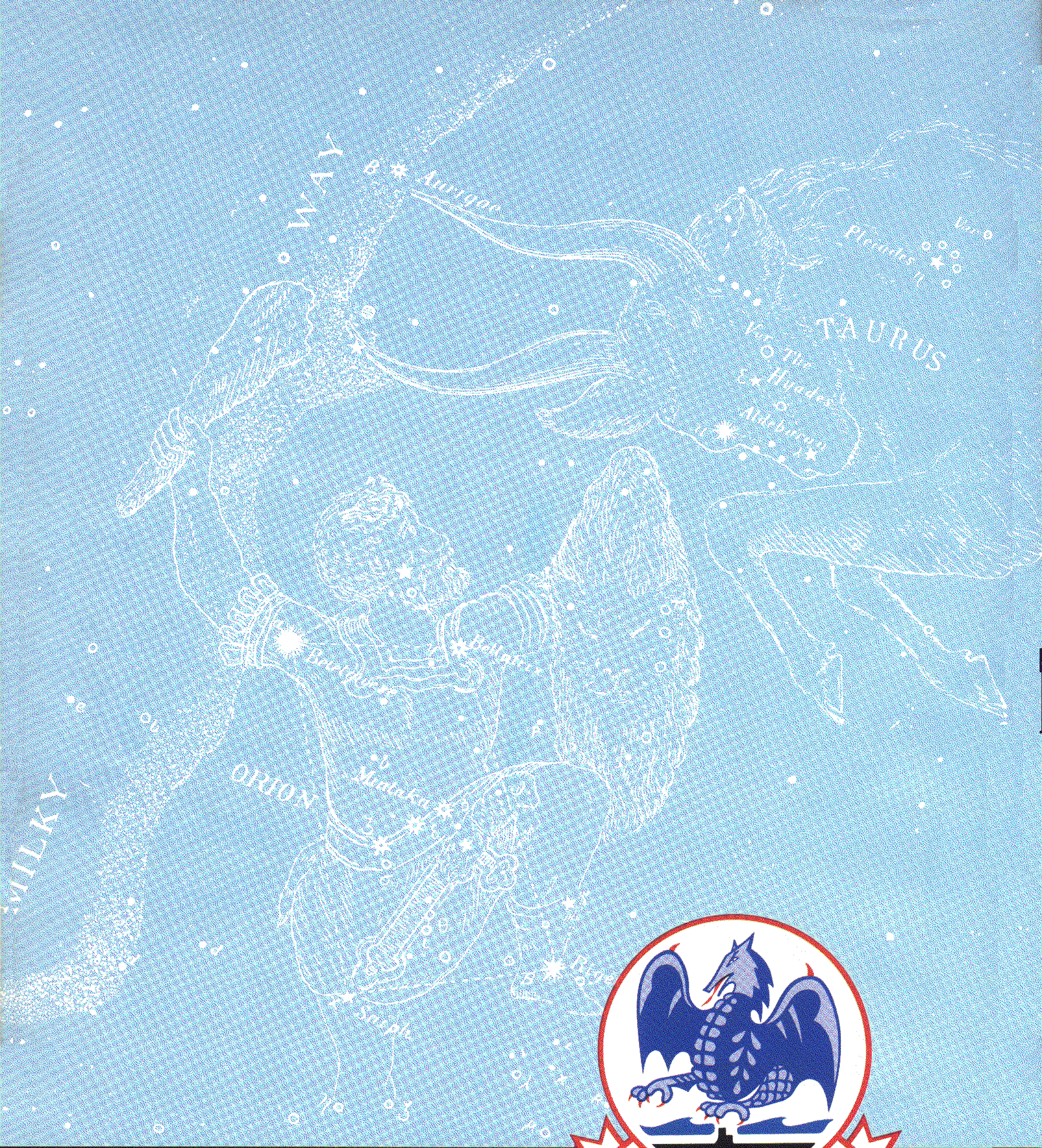

PATRON FIFTY

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