CONFIDENTIAL

ORENDA

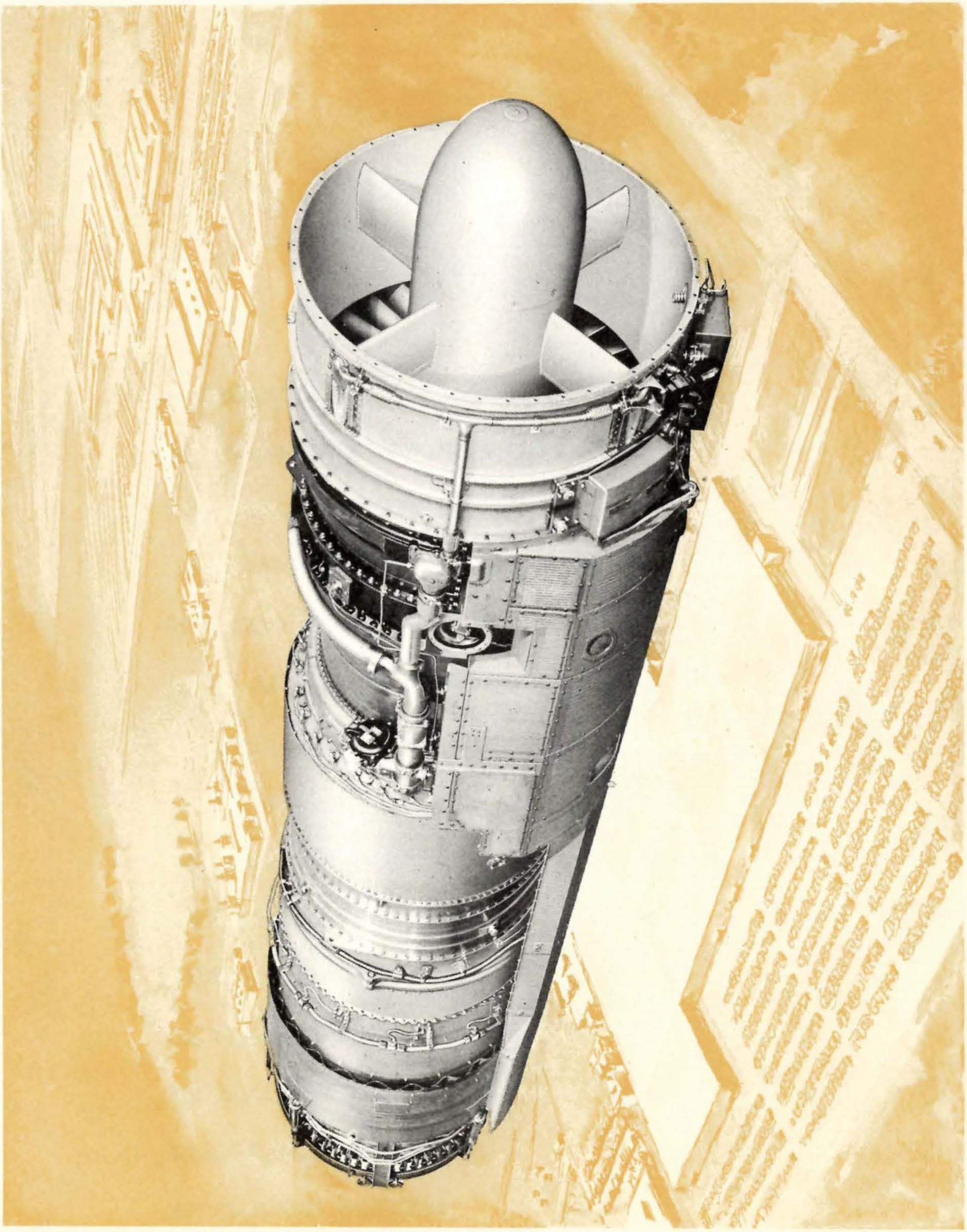
IROQUOIS 2 MOCK-UP ENGINEERING EVALUATION CONFERENCE

July 24th and 25th

B17-58

ORENDA ENGINES LIMITED

MALTON ONTARIO CANADA



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OXYGEN SYSTEM

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INTRODUCTION

The Iroquois 2 Mock-Up Engineering Evaluation Conference has been arranged in accordance with the requirements of the Iroquois Engine Development Agreement, dated September 7, 1956.

This brochure has been prepared to assist members in understanding the conference schedule and the nature of the topics slated for discussion. To assist in this regard, descriptions of the systems have been included in the rear sections of the brochure.

Participating members are responsible for arranging their own accommodation. The Secretary will assist with accommodation arrangements if requested in sufficient time.

Members requiring Company transportation are requested to meet in the lobby of the Royal York Hotel at 8:00 A.M. daily.

Board Members and Technical Advisors are requested to enter and leave the Conference and Mock-Up areas via the Main Lobby of the Production Plant. The location of this entrance and the parking area and the relevant routes to be taken through the Plant are shown in Figure 1.

Personnel attending the Conference must register in the Main Lobby where suitable identification will be issued.

Copies of the following specifications will be available in the Mock-Up area:

a) Model Specification
b) Advance Data Sheets
c) Installation Drawings
d) Installation Manual
e) Arrow 2 Engine Installation Manual
f) ARDCM 80-1
g) MIL-E-5007
h) MIL-E-5008
j) MIL-E-5009

Other specifications can be made available on request to the Project Engineers or the Conference Secretary.

Change Request Forms will be available from the Secretary and should be completed as follows:

1. The originator should check with the ORENDA specialist for accuracy and correct nomenclature.

2. The completed Change Request Form must be passed to the Conference Secretary who will record it and have four copies typed.

3. The Secretary will return these copies to the originator for his signature.

4. One copy will be retained by the originator the remaining three copies must be returned to the Secretary.

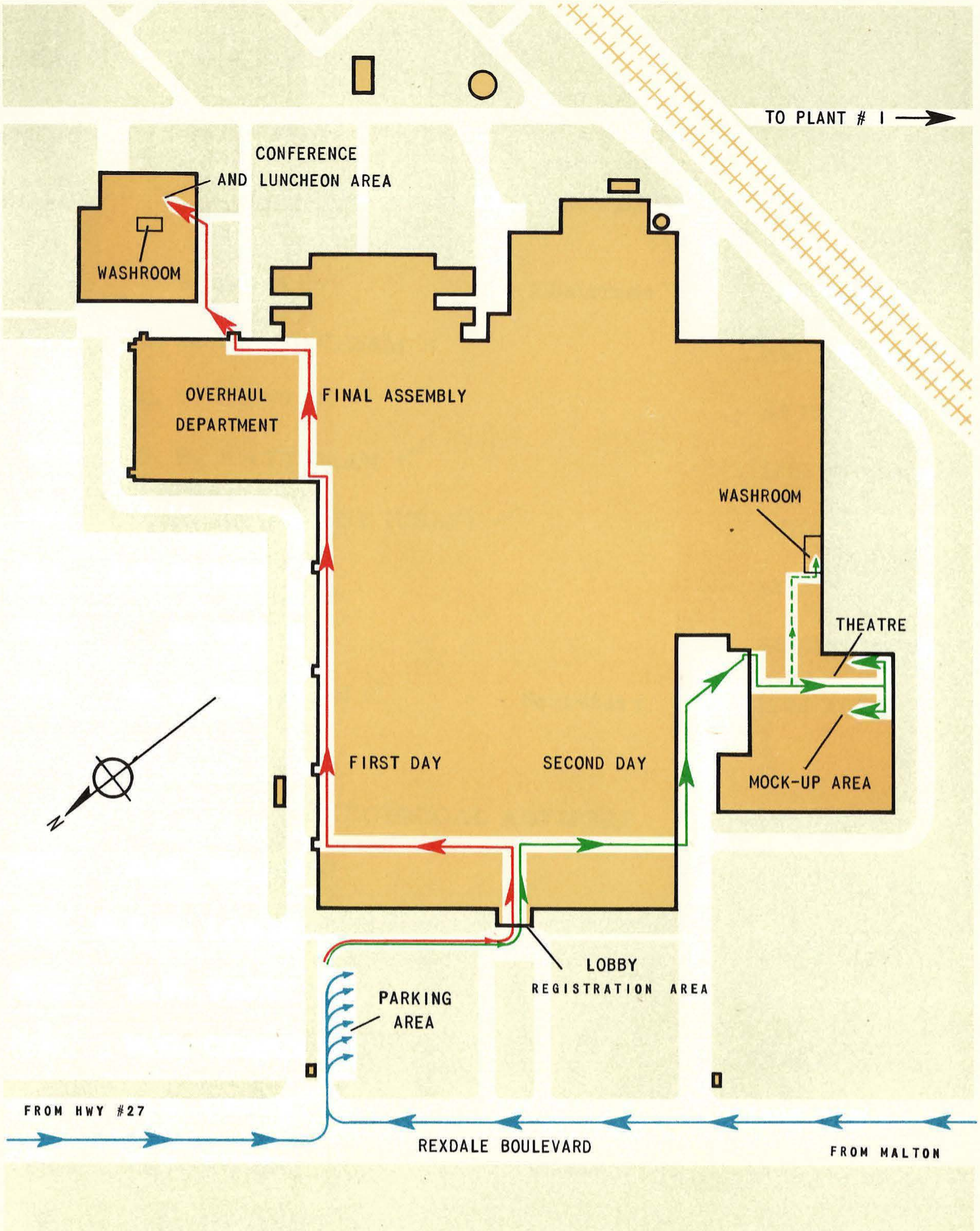


FIG. 1 GENERAL LAYOUT OF PLANT
SHOWING CONFERENCE AND MOCK-UP AREAS

MOCK-UP BOARD MEMBERS

W/C	D. W. GOSS	Chairman	AFHQ/AAWS
G/C	D. D. CUNNINGHAM		AMCHQ/SACO
W/C	C. L. V. GERVAIS		AFHQ/DADR
W/C	G. B. WATERMAN		1202 TSD Avro
Mjr	J. M. AMBRECHT (USAF)		ADCHQ
S/L	J. O. H. NEFF		1202 TSD MAT
S/L	D. WHITE		AFHQ/DIE Eng
S/L	L. J. SULLIVAN	Secretary	1202 TSD

TECHNICAL ADVISORS

G/C	D. M. HOLMAN	AMCHQ/SACO
W/C	H. S. M. LONDEAU	AFHQ/AAWS
W/C	R. B. WAITT	AMCHQ/SACO
S/L	W. G. CHANDLER (2)	TCHQ
S/L	R. S. CROSBY	1202 TSD
S/L	T. R. FUTER	AFHQ/DADR
S/L	J. E. NEELIN	AMCHQ/SACO
S/L	O. B. PHILP	AFHQ/AAWS

S/L	J. R. ROMANOW		AMCHQ/AAWS
S/L	A. J. S. WRIGHT		AFHQ/DIE Eng
F/L	R. A. DOIRON		AMCHQ/SACO
F/L	W. G. GALLOP		ADCHQ
F/L	J. B. MURRAY (2)		AFHQ/AAWS
F/L	J. D. YOUNG		1202 TSD
F/O	J. A. WALLINGTON ✓		1202 TSD
WO_1	E. H. ROSSELL		1202 TSD
WO_2	G. STEEL		1202 TSD
Sgt	P. A. BELL		1202 TSD
Sgt	R. SHERRARD		AMCHQ/SACO

GOVERNMENT AGENCY REPRESENTATIVES

DDP	J. C. FINLAYSON		Ottawa
	C. A. HORE	Senior Representative	Malton
	J. B. SILVESTER	Production Officer	Malton
DRB	H. C. OATWAY		Ottawa

SUB-CONTRACTOR PERSONNEL

D. F. MOWBRAY ✓
S R Scott

AVRO AIRCRAFT PERSONNEL

Arrow Weapon System Co-Ordinator - W. R. STEPHENS

Engineering	A.	BINDING
	S.	WHITELEY
	D.	THOMAS
	C.	BARKER
	C. V.	LINDOW
	A.	BULEY
Service	D.	REYNOLDS

ORENDA ENGINES PERSONNEL

Engineering	C. A.	GRINYER
	F. H.	KEAST
	B. A.	AVERY
	S. L.	BRITTON
	J. T.	PURVIS
	R. M.	SACHS
	R. G.	DENNYS
	D.	QUAN
	C. D.	RUPPEL
	F. D. M.	WILLIAMS
	J. C.	WHITE
	J. R.	JOY

	J. L. BRISLEY
	A. W. SMALLWOOD
Inspection	F. M. STAINES
Sales & Service	A. L. SUTTON
	R. STANYAR
	G. WILSON
	P. Y. DAVOUD
	D. J. CAPLE
	H. D. CULHAM
Test Pilots	M. COOPER-SLIPPER
	L. HOBBS

IROQUOIS 2 MOCK-UP ENGINEERING EVALUATION CONFERENCE

PROGRAM

DATE	TIME	FUNCTION
Thursday		
July 24	9:00 - 9:30 A. M.	Registration
	9:30 - 10:00	Welcome and Introduction by Mr. C. A. Grinyer
	10:00 - 10:30	General description of IROQUOIS 2 engine
	10:30 - 10:45	FREE PERIOD
	10:45 - 12:30 P. M.	Lecture Period - Systems Design
	12:30 - 1:45	LUNCH
	1:45 - 3:30	Lecture Period - Systems Design
	3:30 - 4:30	Question period

Friday		
July 25	9:00 - 9:30 A. M.	Short talk on physical aspects of IROQUOIS 2 engine
	9:30 - 10:30	Visual inspection of mock-up engine
	10:30 - 10:45	FREE PERIOD
	10:45 - 12:30 P. M.	Submission of comments during maintenance demonstration
	12:30 - 1:45	LUNCH
	1:45 - 4:30	Meeting of Chairman, Members and Advisors to discuss the change request forms.

GENERAL DESCRIPTION

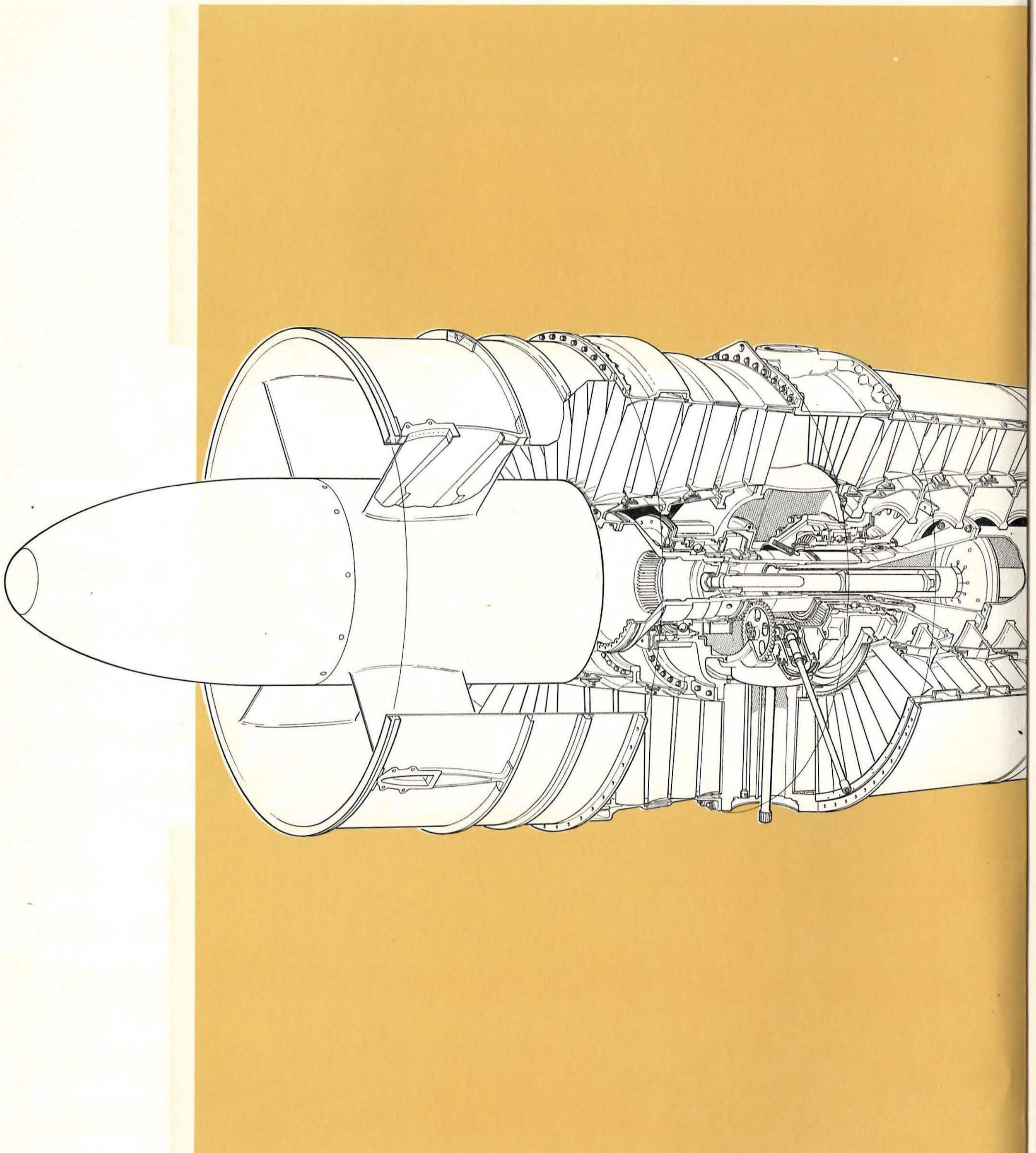
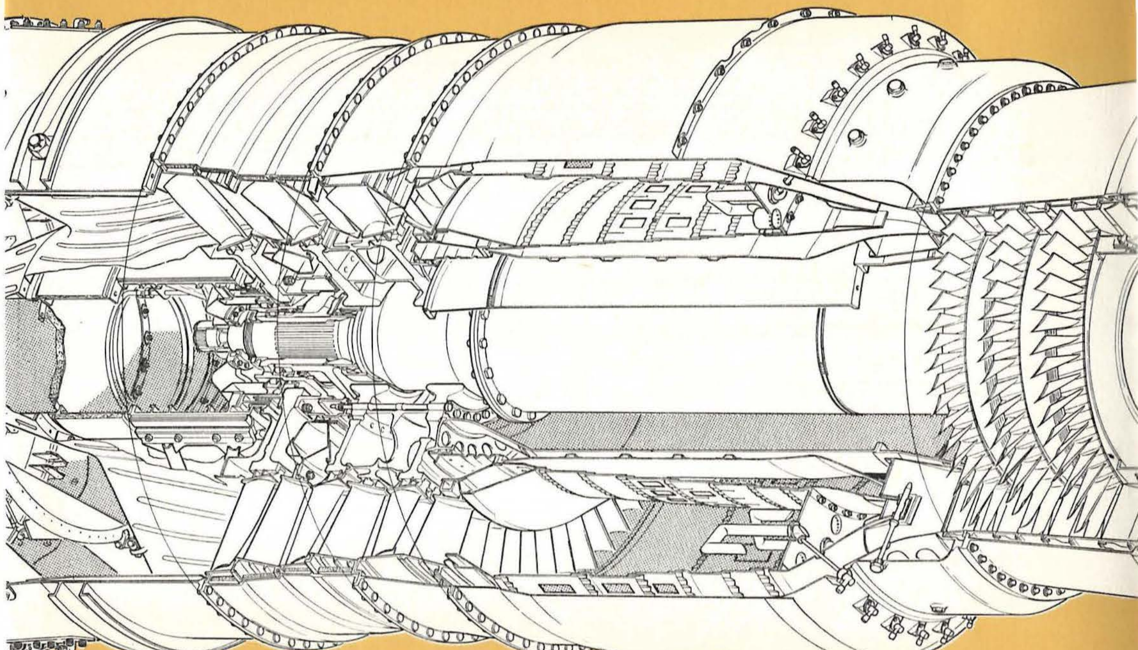
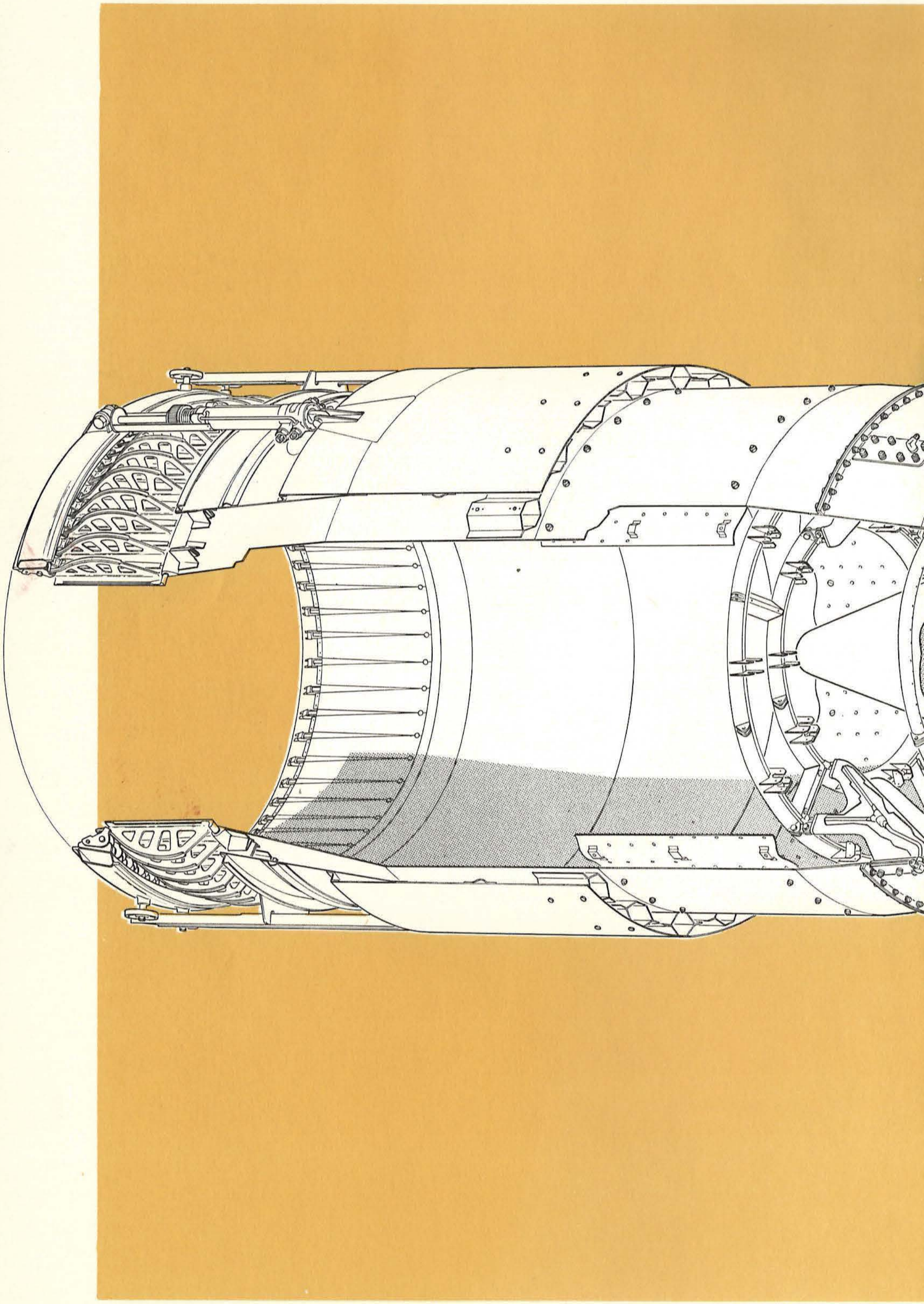


FIG. 2 SECTIONED VIEW OF ENGINE





GENERAL DESCRIPTION

The Iroquois is a high performance, axial flow, two-spool turbojet engine with integral afterburner, and is specifically designed for operation under supersonic flight conditions. The engine derives its thrust from the reaction to the increase in momentum of the air mass passing through it. This increase in momentum is produced by burning fuel in the air mass to increase its temperature and volume and by accelerating the exhaust gases to a high velocity by means of a final nozzle at the rear of the engine.

The engine comprises essentially a two-spool compressor, an annular type combustion system, a turbine section, an afterburner-equipped exhaust system and the necessary support frames and casings.

An inlet frame assembly directs the air flow into the three-stage low pressure compressor, from which it is forced into the seven-stage high pressure compressor. Each compressor is individually coupled to its respective turbine by direct shafting, the HP compressor being driven by a two-stage turbine and the LP compressor by a single stage turbine.

The LP and HP compressor rotor assemblies are mounted in two separate ball thrust bearings supported in the front frame assembly, interposed between the LP and HP compressors. Additional support is provided by a steadying roller bearing at the front of the HP rotor assembly. The turbine rotor assemblies are mounted in two separate roller bearings housed in the rear frame assembly located at the rear of the turbine section. In addition to supporting the LP and HP rotors, the front frame and rear frame assemblies distribute the high thrust and tangential flight loadings from the engine casings to the airframe.

The mid-frame assembly is located between the compressor and turbine sections and accommodates an annular combustion chamber in which the main fuel supply is injected and burned by means of 32 walking stick vaporizer type burners.

The afterburner assembly, being an integral part of the engine, is mounted aft of the rear frame assembly. It comprises a casing which houses flame stabilizing equipment and fuel spray rings. A variable area final nozzle is attached to the rear of the afterburner assembly.

Internal gearing located in the hub of the front frame assembly, provides for power take-off points to operate the various accessories necessary to sustain engine operation and meet airframe service requirements. All accessories are mounted around the bottom of the engine forward of the mid-frame. These are enclosed in a sheet metal shroud. Lubrication of internal bearings and gearing and cooling of the heat producing areas of the engine are provided by self-contained oil and air systems. Protection against the formation of ice at the engine intake under certain operating conditions is by means of a hot air anti-icing system.

The engine has five mounting points. The front mount located on the top of the front frame transmits the net axial thrust load and side loads, while a side mount takes vertical loads. Two mounts, one on each side of the rear frame, take vertical loads and a rear mount at the top of the rear frame takes side loads.

POWER CONTROL SYSTEM

POWER CONTROL SYSTEM

HIGH PRESSURE FUEL SUPPLY SYSTEM

The supply of pressurized fuel required for engine and afterburner operation is provided by two air turbine driven centrifugal type pumps operating in parallel. Bleed air from the high pressure compressor is used to drive the pumps. Pump exhaust is collected in the tertiary air duct and discharged with the main gas stream at the engine final nozzle. The pump shaft rotates on fuel lubricated sleeve bearings. Use of fuel provides a completely self-contained lubrication system and reduces the number of rubbing type shaft seals to one per pump.

Two pumps in parallel are used for high pressure fuel supply for increased reliability and to obviate the necessity for a separate afterburner pump and control. Each pump is capable of supplying the main engine fuel requirements so that in the event of failure of one pump maximum dry engine thrust can still be achieved under all conditions. Check valves fitted in the main and servo outlets, provide complete isolation of an inoperative pump.

In addition, a relief valve is incorporated in each pump to prevent excessive fuel pressures in the main and afterburner circuits.

Centrifugal type pumps, being less sensitive to contamination of the fuel supply than either gear or piston pumps due to the absence of critical clearances, do not require micronic filtration of the fuel supply. Since a micronic filter with capacity adequate to handle maximum system flows would be large, a considerable saving of space and weight is realized through the use of centrifugal pumps. Micronic filtration for critical circuits is provided by two small elements; one is used to filter the fuel for pump bearing and seal lubrication, the other to filter servo fuel for the control.

The output of the fuel pumps is controlled by regulating the air supply to the pumps by means of a throttling valve in the supply duct. The pump pressure control which regulates pump output senses the pressure rise across the compressor, this being a measure of the air mass flow through the engine, and hence varies output according to changes in engine speed, aircraft forward speed and altitude.

The pump output is controlled by the control unit at a value above the discharge pressure required at the burners. The excess pressure delivered by the pumps provides for initial acceleration of the engine, compensates for the pressure losses in the system components, and supplies a small controlling pressure drop across the metering orifices located elsewhere in the system. The principal advantage of this type of control system over a by-pass system is that the demand on the pumps is decreased, resulting in an extension of their effective life. Under most operating conditions, a by-pass type of system is continuously over-pressurized, resulting in a large proportion of pump delivery pressure being wasted during the return of excess fuel to the inlet side of the pumps. In addition, extensive overheating of the fuel is commonly associated with the by-pass system.

Fuel from the aircraft tanks is supplied at boost pressure to the engine fuel inlet where it is pressurized to the required operating pressure by the engine fuel pumps. From the pumps, the fuel is delivered at high pressure to the main and afterburner control systems.

Regulation of the air supply is effected by a hydraulic actuator which is mechanically linked to the throttling valve. The position of the hydraulic actuator is controlled by the pump pressure control unit which varies the servo pressure loading on the actuator piston. At low engine speeds corresponding to low pressure ratios across the air turbines the throttling valve is open to provide a minimum of restriction to the air; as engine speed increases the valve tends to close to maintain the pressure schedule established by the pump pressure control. When delivery pressure differs from the scheduled value, the control will cause the hydraulic actuator to increase or decrease the pump air supply until the required output is achieved, at which time fuel pressure which is also sensed by the control will cause the control to stabilize at its new equilibrium position.

In addition to its primary function of positioning the air valve, the hydraulic actuator incorporates two secondary features to safeguard pump operation. A valve actuated by the power lever is incorporated to override the pump pressure control and cause the air valve to close when the power lever is moved to the cut-off position. This prevents pump rotation in the stalled engine condition with resultant heating of the fuel during engine windmilling and shut-down from high speeds. Overspeed protection for the pumps is also provided by the actuator which is spring loaded to the closed position. High pressure fuel is required to overcome this spring and cause the valve to open. In the event of interruption of the fuel supply which would result in a sudden excess of torque, tending to overspeed the pumps, the accompanying reduction in fuel pressure causes the actuator to move to close the air valve.

An auxiliary starting pump is provided in the system to produce sufficient delivery pressure during the initial stages of the starting cycle until the value of compressor delivery pressure is sufficient to drive the air turbine pumps. The starting pump is hydraulically driven. It is energized by the 'Start' or 'Relight' switch in the aircraft, and is automatically cut out as the main pumps become self-sustaining during completion of the starting cycle.

MAIN CONTROL SYSTEM

The main control system is comprised of the units which under normal circumstances provide automatic regulation of the fuel flow to the burners in the combustion chamber.

High pressure rotor speed is used as the basic parameter to control main engine thrust. A schedule of rotor speed versus throttle angle is established to provide a linear thrust versus throttle angle relationship. The relationship between rpm and thrust is generally consistent throughout the entire operating range of the engine, and is relatively unaffected by the changing set of conditions encountered during engine operation. If, for example 85 percent rpm produces 50 percent available thrust output at sea level, the same engine rpm will produce approximately 50 percent available thrust at altitude. Thus, the setting of the power control lever establishes a demand for a certain percentage of available thrust. The fuel control units interpret this setting as a request for the engine rpm which will deliver the desired thrust output and, in turn, the exact fuel flow required to produce this rpm.

The proportionality between thrust and throttle angle is altered when limits incorporated in the control to protect the engine from excessive stresses are encountered. The effect of these limits is to create flat spots in throttle movement such that increasing throttle angle does not produce an increase in thrust. These limits are, however, encountered only in certain extreme operating conditions and, therefore, do not normally affect response of engine thrust to throttle movement. These limits are as follows:

LP rotor overspeed.

Maximum compressor pressure rise.

Maximum engine fuel flow.

During acceleration fuel flow is trimmed by the electronic control amplifier according to a schedule of jet pipe temperature versus high pressure rotor speed. This schedule is corrected for compressor inlet temperature to allow for changes in the engine surge line with temperature in order to obtain minimum safe acceleration time under all conditions. During steady state operation jet pipe temperature trim is transferred to the final nozzle area control.

A further measure of stall control is provided by the variable incidence inlet guide vanes on the HP compressor. At low engine speeds these are positioned at low incidence to assist acceleration. At high engine speeds the vanes are moved to their high incidence position to improve compressor efficiency.

The flow through the engine fuel system is established by the orifice sizes of the main metering valve and the servo throttle valve which controls the pressure drop across the metering unit. The orifice size of the main metering unit is established by inter-compressor pressure (P_3). The orifice size in the servo throttle valve is made variable by servo pressure signals originating in the proportional flow circuit. The underlying principle of the proportional flow type of control is to regulate a small by-pass flow in exact proportion to the fuel flow requirements of the engine, and to establish the main fuel flow at some multiple of the by-pass flow by proper calibration of the control units involved. The proportional by-pass flow is tapped from the inlet to the main metering unit and passes through a hydraulic potentiometric circuit, consisting primarily of a variable orifice (the potentiometric throttle valve operated by the pilot's power lever) and a fixed orifice in series. A pressure tapping between these two orifices is taken to the spring side of the pressure drop unit. This pressure establishes the pressure in the opposite side of the pressure drop unit and this resulting pressure establishes the flow rate into the servo throttle valve chamber. For example if the area of the potentiometric throttle valve is increased, the pressure on the spring side of the pressure drop unit increases. This increase is balanced by the diaphragm plate partly closing over the orifice in the pressure drop unit, increasing the pressure on that side. This smaller orifice reduces the flow to the servo throttle valve, and if the make-up rate is less than the bleed rate, the valve moves in the opening direction, increasing fuel flow.

Various trim orifices act to vary this basic potentiometer. An orifice opening in parallel with the potentiometric throttle valve will increase the fuel flow without movement of the potentiometric throttle valve. This is the operation of the Minimum Idle Flow valve.

An orifice opening in parallel with the fixed orifice downstream of the potentiometric throttle valve reduces the inter-orifice pressure supplied to the pressure drop unit without changing the potentiometric throttle valve. These orifices reduce fuel, and consist of the HP and LP governor proportional and floating orifices.

The setting of the HP governor is varied as a function of power lever position by the governor reset unit. Rotation of the cam in the governor reset unit varies the loading pressure on the governor spindle. The spindle is held in equilibrium by the combined force of loading pressure and trim spring in one direction and governor flyweight force in the opposite direction. When high pressure compressor rotor speed exceeds the selected speed the flyweights cause the spindle to move, thereby creating a bleed in parallel with the fixed orifice in the proportional flow circuit causing the reference pressure in the pressure drop unit to decrease. As a result the servo throttle valve tends to close to reduce the pressure drop; hence the flow through the metering unit. Reduction in engine fuel flow causes engine speed to decrease thereby restoring equilibrium. Operation of the governor is made isochronous (i. e. maintains constant speed independent of operating conditions) through the governor floating valve. Whereas a slight excess of engine speed over selected speed can produce only a small opening of the spindle valve (governor proportional valve) it can produce a large opening of the floating valve. Hence, the floating valve is capable of producing a relatively large reduction in the reference pressure applied to the pressure drop unit for small movement of the governor spindle. In this manner large reductions in scheduled engine fuel flow can be created by small speed changes, hence isochronous control is achieved.

The operation of the low pressure rotor governor is identical to that of the HP governor except that it is set to operate at only one speed to prevent overspeed of the low pressure rotor.

An emergency HP rotor overspeed governor is also included in the system. Although this unit remains in the control circuit at all times, it is normally inoperative when running on the main control, since its setting is above the maximum speed of the high pressure governors. The operation of this unit is covered in the description of the emergency system.

The minimum idle flow valve also operates on the proportional flow circuit to provide a variable orifice in parallel with the power lever operated throttle valve to increase the reference pressure applied to the pressure drop unit under altitude idling conditions. The effect of this valve is to override the HP rotor governor control when the power lever is in the idle position to provide an increasing idling speed with altitude to satisfy minimum aircraft and combustion requirements. This valve is progressively closed by a cam as the throttle is advanced in order to prevent it overriding the governor at high throttle angles.

As mentioned previously the opening of the metering unit orifice is controlled by inter-compressor pressure. In order to protect the turbine and combustion chamber from excessive pressures at high compressor inlet pressures, a pressure limiter is incorporated in the control. When the compressor pressure rise exceeds 160 psig, inter-compressor pressure (P_3) applied to the control, is vented to ambient to reduce the effective P_3 sensed by the control thereby reducing the opening of the metering unit valve.

During acceleration the servo control pressure applied to the servo throttle valve is biased by the proportional fuel valve to vary the orifice size in the servo throttle valve. The opening of the proportional fuel valve is controlled by a signal from the electronic control amplifier. Voltages proportional to jet pipe temperature and HP rotor speed are sensed by the control and are compared with reference voltages in the amplifier. The error between actual and required values is amplified and used to actuate the proportional fuel valve.

The torque required to operate the power lever is minimized through the use of a power assist arrangement. This is a simple follower servo which utilizes fuel pressure to amplify the input signal from the aircraft throttle. This unit incorporates a mechanical back-up which maintains positive control of power lever operation in the event of malfunction of the power assist. However, in this case the operating torque will increase.

EMERGENCY CONTROL SYSTEM

The emergency control system provides an alternative route for the engine fuel flow should malfunctioning of the automatic control units develop. When the pilot selects 'Emergency', the selector valve in the main control system closes and simultaneously a valve in the emergency system opens. This allows pump delivery fuel to by-pass the main control units and pass directly to a manually operated emergency throttle valve. Transfer to the emergency system also isolates the electronic amplifier rendering inoperative all trims operated by this control.

The emergency throttle valve, being mechanically linked to the power control lever, provides the pilot with full manual control of the fuel flow during emergency conditions. On emergency the afterburner remains operative if there is no interruption in DC supply. With the automatic control inactive, the pilot must exercise extreme care during the throttle handling and when necessary, must modulate the throttle setting to protect the engine from adverse operating conditions since the all-automatic trims are inoperative. However, a degree of altitude compensation is provided by the nature of the pump pressure control unit which, as previously described, reduces or increases pump delivery pressure as a function of the engine mass air flow. In addition, overspeed protection for the HP rotor is provided by an emergency governor which applies a trim on the pump pressure control.

AFTERBURNER CONTROL

The fuel supply to the afterburner is automatically regulated to the exacting flow values required for safe and efficient operation of the afterburner under the wide range of operating conditions encountered by the engine. During operation of the afterburner, a flow of HP pump delivery fuel passes through a series of afterburner control units, including a metering unit and a servo throttle valve, which are located in parallel to the main control system. The mechanical configuration and operation of the afterburner metering unit and servo throttle valve are similar to those in the main control system with the exception that the valve in the afterburner metering unit is positioned as a function of the HP compressor delivery pressure (P_4), which, like P_3, is a measure of the mass airflow through the engine. Thus, the afterburner fuel flow is also altitude compensated.

Afterburner thrust is varied between 93 degrees and 110 degrees to provide smooth modulation of thrust with power lever movement. Light-up is accomplished at a fuel/air ratio slightly greater than the minimum required for combustion; as the power lever is advanced, the fuel/air ratio is increased to provide increasing thrust. When the power lever is retarded below 93 degrees the afterburner remains on, and thrust is modulated by varying main engine fuel flow down to 78 degrees power lever position. In this manner the thrust discontinuity which occurs at light-up can be overcome, and engines can be operated at any thrust between idle and maximum.

The operation of the afterburner control is similar to the main engine control in that two variable orifices in series are used to control fuel flow. One orifice in the afterburner metering unit is positioned as a function of P_4. The second orifice, the servo throttle valve, is positioned to maintain a constant pressure drop in the proportional flow by-pass circuit. The pressure drop is varied by a throttle valve in the proportional flow circuit. This valve, being mechanically linked to the power control lever, is opened and closed by power lever movement in the afterburning range, thus providing full thrust modulation of the afterburner. In addition, a small scheduling valve in the proportional by-pass circuit, actuated by compressor delivery pressure (P_4), establishes the fuel flow schedule and maintains the minimum fuel flow required to obtain afterburner light-up and sustain combustion.

A normally closed, solenoid operated shut-off valve located downstream of the servo throttle valve, opens to permit passage of the fuel to the afterburner manifolds when the power lever is advanced into the afterburning range. Afterburner ignition is accomplished by a hot streak igniter system which injects fuel into the main engine combustion chamber to ignite the afterburner fuel. The hot streak igniter is energized by the same circuit which energizes the shut-off valve but is de-energized by its own timing circuit at the end of a pre-determined interval. Premature operation of the shut-off valve and hot streak ignition system, when the power lever is advanced into the afterburning range during engine acceleration is prevented by a speed lock-out switch which prevents energization of the electrical circuits involved until the engine rotor speed is beyond the stall range.

During afterburning, final nozzle area is automatically increased to allow for increased expansion of the exhaust gases caused by afterburner combustion. The nozzle area control causes the final nozzle to open to maintain correct turbine pressure ratio. This action is described more fully in the operation of the nozzle area control.

Closing of the shut-off valve, and hence shut-down of the afterburner, is accomplished by retarding the power control lever below 78 degrees. Between 93 degrees and 78 degrees the afterburner control schedules the minimum flow to sustain combustion, and thrust is modulated by varying engine speed, hence main engine thrust.

FINAL NOZZLE AREA CONTROL

The area of the final (propelling) nozzle is varied by controlling the flow of hydraulic oil to the actuators which operate the nozzle. Turbine pressure ratio (P_4/P_7), sensed by the pressure ratio actuator, is used to position the valve which controls the flow of oil to the actuators. In this manner nozzle area is varied to maintain a constant turbine pressure ratio in the upper speed range. The area is trimmed by jet pipe temperature (T_7), to prevent engine damage due to excessive turbine blade temperatures, and to extend engine life without loss of performance through accurate temperature control both during cruising and maximum speed operation.

Turbine pressure ratio is sensed by the pressure ratio actuator. This is a hydro-pneumatic servomotor consisting of a variable area metallic diaphragm which operates a pilot valve in the area control valve. Movement of the area control valve modulates the hydraulic pressure on the head end of the pistons of the area control actuators. The resultant change in final nozzle area causes the turbine pressure ratio to change, thereby providing feedback to provide closed loop control of final nozzle area. The pressure ratio actuator is set to maintain a turbine pressure ratio of 3.15:1. If the pressure ratio exceeds this value the nozzle will be caused to close until the required value is re-established. If the pressure ratio is below the required value, the nozzle area will open. Thus at low engine speeds, before the required pressure ratio is established, the nozzle will be fully open thereby assisting acceleration. In the upper power range the nozzle area will close to maintain required pressure ratio and in so doing increase thrust through increased jet pipe temperature.

Pressure ratio is biased by an electronic trim to prevent excess jet pipe temperature as follows. Exhaust gas temperature is measured by means of thermocouples; the output of the thermocouples is compared to a reference voltage in the electronic amplifier. The reference voltage is varied by power lever position to provide safe limits during cruise and maximum engine speed conditions. An excess of measured jet pipe tem-

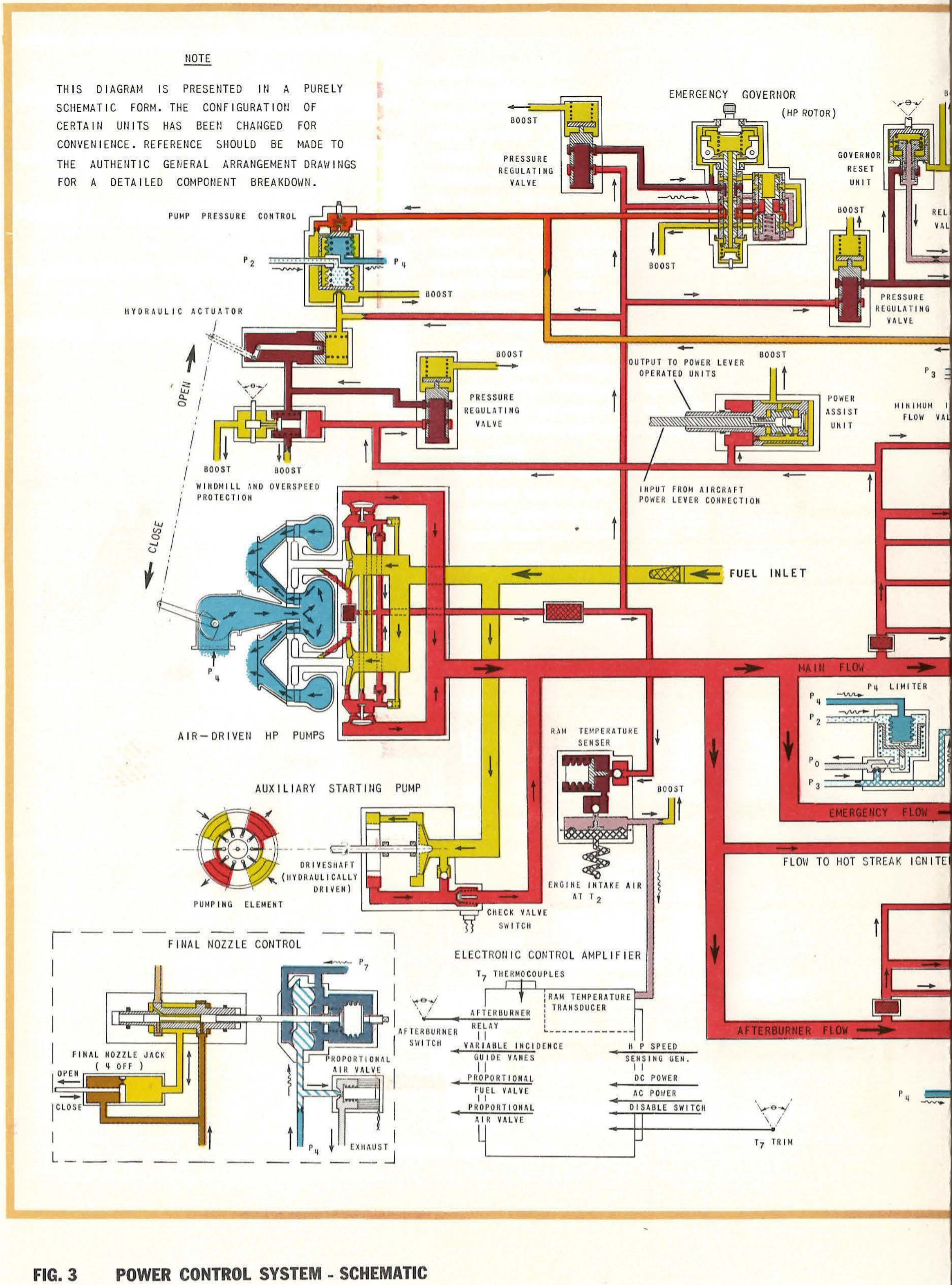
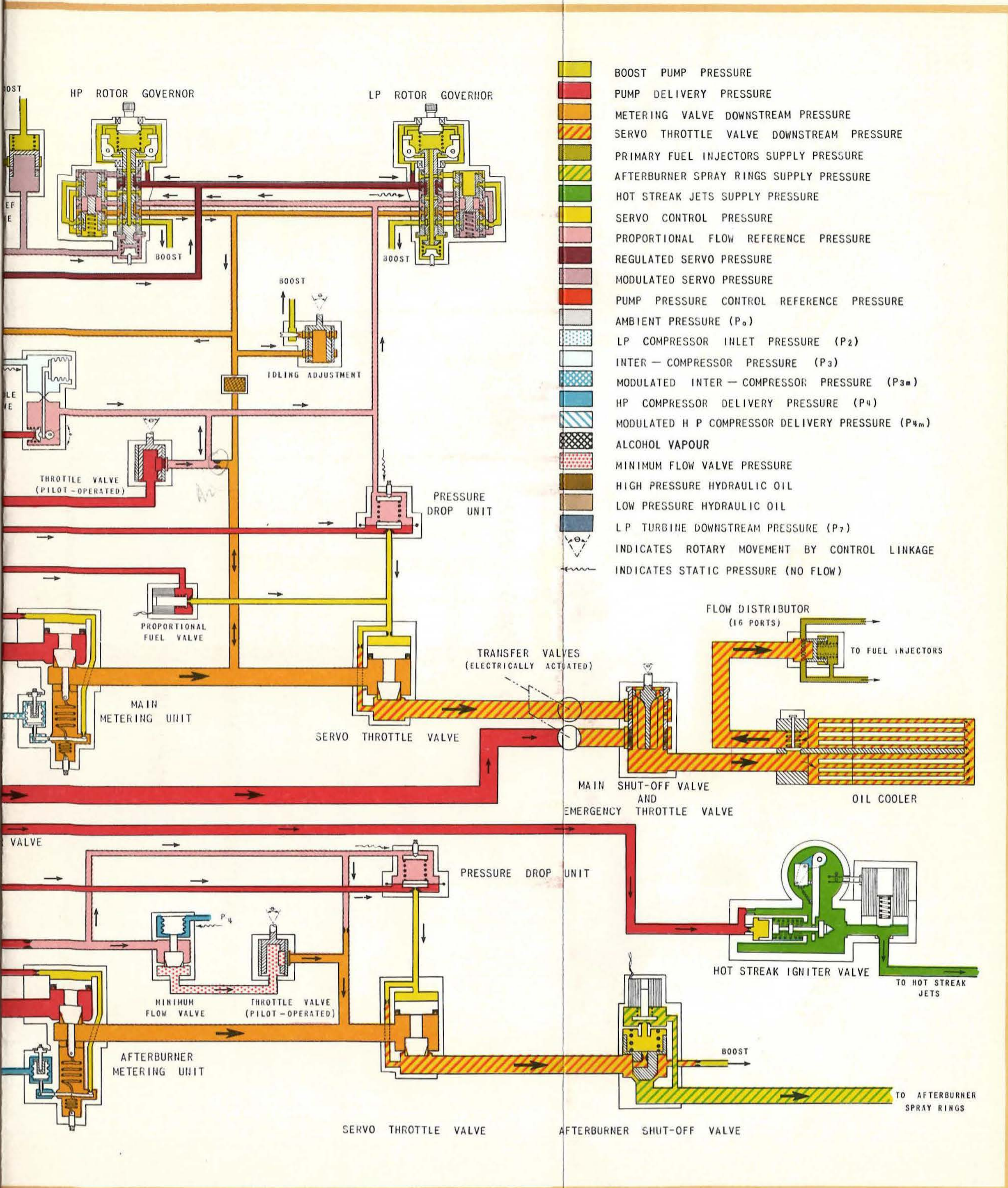


FIG. 3 POWER CONTROL SYSTEM - SCHEMATIC



BOOST PUMP PRESSURE
PUMP DELIVERY PRESSURE
METERING VALVE DOWNSTREAM PRESSURE
SERVO THROTTLE VALVE DOWNSTREAM PRESSURE
PRIMARY FUEL INJECTORS SUPPLY PRESSURE
AFTERBURNER SPRAY RINGS SUPPLY PRESSURE
HOT STREAK JETS SUPPLY PRESSURE
SERVO CONTROL PRESSURE
PROPORTIONAL FLOW REFERENCE PRESSURE
REGULATED SERVO PRESSURE
MODULATED SERVO PRESSURE
PUMP PRESSURE CONTROL REFERENCE PRESSURE
AMBIENT PRESSURE (P_0)
LP COMPRESSOR INLET PRESSURE (P_2)
INTER — COMPRESSOR PRESSURE (P_3)
MODULATED INTER — COMPRESSOR PRESSURE (P_{3m})
HP COMPRESSOR DELIVERY PRESSURE (P_4)
MODULATED H P COMPRESSOR DELIVERY PRESSURE (P_{4m})
ALCOHOL VAPOUR
MINIMUM FLOW VALVE PRESSURE
HIGH PRESSURE HYDRAULIC OIL
LOW PRESSURE HYDRAULIC OIL
L P TURBINE DOWNSTREAM PRESSURE (P_7)
INDICATES ROTARY MOVEMENT BY CONTROL LINKAGE
INDICATES STATIC PRESSURE (NO FLOW)
HP ROTOR GOVERNOR
LP ROTOR GOVERNOR
BOOST
BOOST
BOOST
IDLING ADJUSTMENT
THROTTLE VALVE (PILOT-OPERATED)
PRESSURE DROP UNIT
PROPORTIONAL FUEL VALVE
MAIN METERING UNIT
SERVO THROTTLE VALVE
TRANSFER VALVES (ELECTRICALLY ACTUATED)
FLOW DISTRIBUTOR (16 PORTS)
TO FUEL INJECTORS
MAIN SHUT-OFF VALVE AND EMERGENCY THROTTLE VALVE
OIL COOLER
VALVE
PRESSURE DROP UNIT
P_4
MINIMUM FLOW VALVE
THROTTLE VALVE (PILOT-OPERATED)
HOT STREAK IGNITER VALVE
TO HOT STREAK JETS
AFTERBURNER METERING UNIT
BOOST
TO AFTERBURNER SPRAY RINGS
SERVO THROTTLE VALVE
AFTERBURNER SHUT-OFF VALVE

perature over required value creates an error voltage which is amplified and applied to the proportional air solenoid valve causing P_4 to be vented to nacelle pressure. The movement of the air solenoid is proportional to the applied voltage. The reduction in sensed pressure creates an artificially low turbine pressure ratio which causes the nozzle to open to compensate, thereby reducing temperature as a result of a larger expansion ratio across the turbine.

LUBRICATING OIL SYSTEM

LUBRICATING OIL SYSTEM

GENERAL

The engine lubricating oil system is self-contained and operates basically on the dry sump principle. The system consists mainly of an internal tank, ~~two~~ an oil pressure supply pump~~s~~ and ~~a~~ ~~two~~ a scavenge pumps, an oil temperature regulator, and the necessary components for oil distribution, see Figure 4.

The quantity of oil supplied to the various components considerably exceeds that normally required for lubrication; this is to ensure that such items as bearings and gears are effectively cooled. The oil flow is controlled by jet orifice size and line resistance to the various outlets.

A feature of the lubricating system is the use of steel-backed carbon ring seals with air at high pressure applied to the seals to prevent oil leakage; the pressurizing effect is a major factor in the efficient scavenging of the oil system. Inverted flight for one minute is a normal military requirement but no special provision is made for anti-g conditions. The engine will function inverted without detriment for periods up to one minute should the oil supply to the bearings and gears be interrupted.

OIL TANK

An annular shaped oil tank is provided, using the inner hub of the front frame as the tank outer wall. The internal gearbox casting forms the inner circular wall of the tank, the front and rear walls being provided by the LP

ABS ~~and HP~~ bearing housing~~s~~ and front frame respectively. The oil tank is vented to the front sump through a double ball vent valve, in order to exhaust air transferred by the scavenge return system to the tank. The double ball vent valve also prevents oil flow from the tank to the front sump in any flight attitude. Refilling is carried out by attaching a pressure oil supply to a quick disconnect fitting located on the underside of the engine. An adjacent quick disconnect fitting connects to an overflow pipe from the tank and indicates when the correct oil level has been reached. The tank is drained by means of a cock at the bottom of the front frame No. 5 strut. The capacity of the tank is five Imperial gallons (six US gallons), four-fifths of which is occupied by oil.

PUMPS

The oil pumps are of the positive displacement type employing a special form of internal-external gear system known as the Gerotor mechanism. The two pumps employed in the engine lubricating system are mounted on the HP external gearbox. The single element pressure pump supplies the engine requirements. The scavenge pump unit, composed of two elements in a single casing, returns oil to the tank from the engine main circulating system. An oil ~~flow~~ low pressure warning indicator is provided in the pilot's cockpit. The indicator is actuated by a switch which senses the pressure differential across the main pressure pump.

FILTER AND BY-PASS VALVE

The filter has a re-usable 33 micron element of the stacked disc type, formed from calendered wire cloth. A by-pass valve incorporated in the filter assembly opens at 30 psi pressure difference and prevents undue restriction of the oil flow should the element become clogged by foreign matter. The filter body is attached to its housing by a single self-locking bolt, and access for servicing is provided by a panel on the underside of the engine shroud. A check valve located on the filter adaptor prevents oil draining from the tank into the circulating system through the pressure pump element when the engine is inoperative.

OIL TEMPERATURE REGULATOR

The engine oil is cooled by circulation through a conventional heat exchanger through which fuel is passed as the cooling medium. A pressure relief valve operating at 150 psi protects the oil temperature regulator from excessive pressure. A combined thermal and by-pass valve opens at temperatures below 15 degrees C (59 degrees F) or pressure differences above 40 psi allowing oil to by-pass the cooling element to achieve minimum warm up time.

INTERNAL GEARBOX LUBRICATION

Pressure oil is supplied through an oil jet to the meshing point of the spiral bevel gears which transmit the drive from the HP rotor shaft to the internal gearbox; pressure oil is also fed to the driven gear support bearings. The gear train from the LP rotor, being lightly loaded, is lubricated by oil mist. The lower portion of the internal gearbox casting forms the front sump. Oil from the separator and internal gearbox components collects in the front sump, and gravitates to the external gearbox sump through No. 5 strut of the front frame.

EXTERNAL LP AND HP GEARBOX LUBRICATION

Pressure oil is supplied by external connections both to the LP and HP gearboxes for jet lubrication of the gear trains. The oil seals are of the double floating ring type, the space between the seals being supplied with high pressure air tapped from the HP compressor. Drainage from the LP to the HP gearbox is by an external gravity line. The HP gearbox sump is scavenged by the main scavenge pump element.

OIL SEALS

The main oil seals used in the Iroquois are each composed of a steel-backed carbon ring located in a housing. The housing locates the ring axially and permits it to float with minimum clearance on the surface of the adjacent rotating component. As the pressure oil supply to the bearings and gears is by jets, no appreciable internal pressure head need be contained by the seals, thus a supply of high pressure air tapped from the HP compressor and applied to the seal, is sufficient to prevent oil leakage. It should be noted here, that a limited flow of air escapes through the seal to mix with the oil, as detailed later.

ENGINE MAIN BEARING LUBRICATION

The HP and LP thrust ball bearings, the LP mainshaft steady roller bearing, and the HP and LP turbine roller bearings, comprise the main engine bearings. Each of these is supplied with oil through the main pressure line. In the instance of the LP thrust bearing, oil is fed from a tapping in the front frame through screens to six jets, three equally spaced on each side of the bearing to ensure, in addition to lubrication, adequate cooling and heat distribution. A radially vaned scavenge impeller is mounted on each side of the bearing and these discharge the oil through passageways leading directly into the tank. The air on the front of the LP bearing housing is at third-stage LP compressor pressure. At this point a double carbon-ring oil seal is fitted, and to maintain an efficient sealing effect, the space between the two seals is pressurized by seventh-stage HP compressor air.

The feed arrangements, jets and impellers used on the HP thrust bearings are similar to those described for the LP bearing. Scavenge oil in the passages from the impeller to the tank can, however, drain to the front sump by means of a drain valve operated by pressure oil. As the main supply pressure falls during engine rundown, the valve opens to drain away oil accumulating in the HP thrust and steady bearings, which might otherwise leak through the seals into the compressor casing. A single carbon-ring oil seal is fitted at the rear of the HP bearing, the air at the exterior of the bearing housing being at seventh-stage HP compressor pressure.

The oil supply for the LP mainshaft steady roller bearing is conveyed by drillings in the internal gearbox casting to a single screened jet discharging into the annular space between the LP compressor shaft and the HP compressor front shaft. After flowing through the bearing, the oil passes through drillings in the HP compressor front shaft to the HP bearing impeller and is scavenged to the tank. A single carbon-ring oil seal is fitted at the rear of the bearing, the ambient air being at seventh-stage HP compressor pressure.

The LP and HP turbine roller bearings are each supplied with pressure oil fed into an annular ring on the inner face of the rear sump. Three equally spaced holes in the forward face discharge oil into the annular space between the oil return tube and the LP shaft. This oil, through the action of centrifugal force, makes its way through drillings and spline clearances to the forwardface of the HP bearing. Oil for the LP bearing is piped from the annular ring to the concentric lubrication groove immediately forward of the bearing. In both cases the oil flows rearward through the bearings and drains into the rear sump. The front inner, front intermediate and front outer oil seals on the HP and LP turbine assemblies are all pressurized by seventh-stage compressor air, and single carbon ring oil seals are fitted at these locations.

OIL CIRCULATION

The circulation of pressure oil from the internal tank to the various engine components is indicated in the oil system schematic. A large portion of the oil collected in the rear sump is in the form of an oil air mist due to air leakage through the seals and the churning action of the bearings. This vapour flows through the rear sump vent into the rotating oil return tube, as the rear sump air pressure is always slightly higher than that at the separator in the front sump. This flow is assisted by the forward air flow from a double floating-ring seal pressurized by seventh-stage compressor air, located at the rear of the ~~oil return~~ air vent tube. At extreme altitude when the air flow is reduced, the oil may tend to separate from the air in the oil return tube. To accommodate this condition the diameter of the oil return tube outer member increases in two increments, from rear to front. These

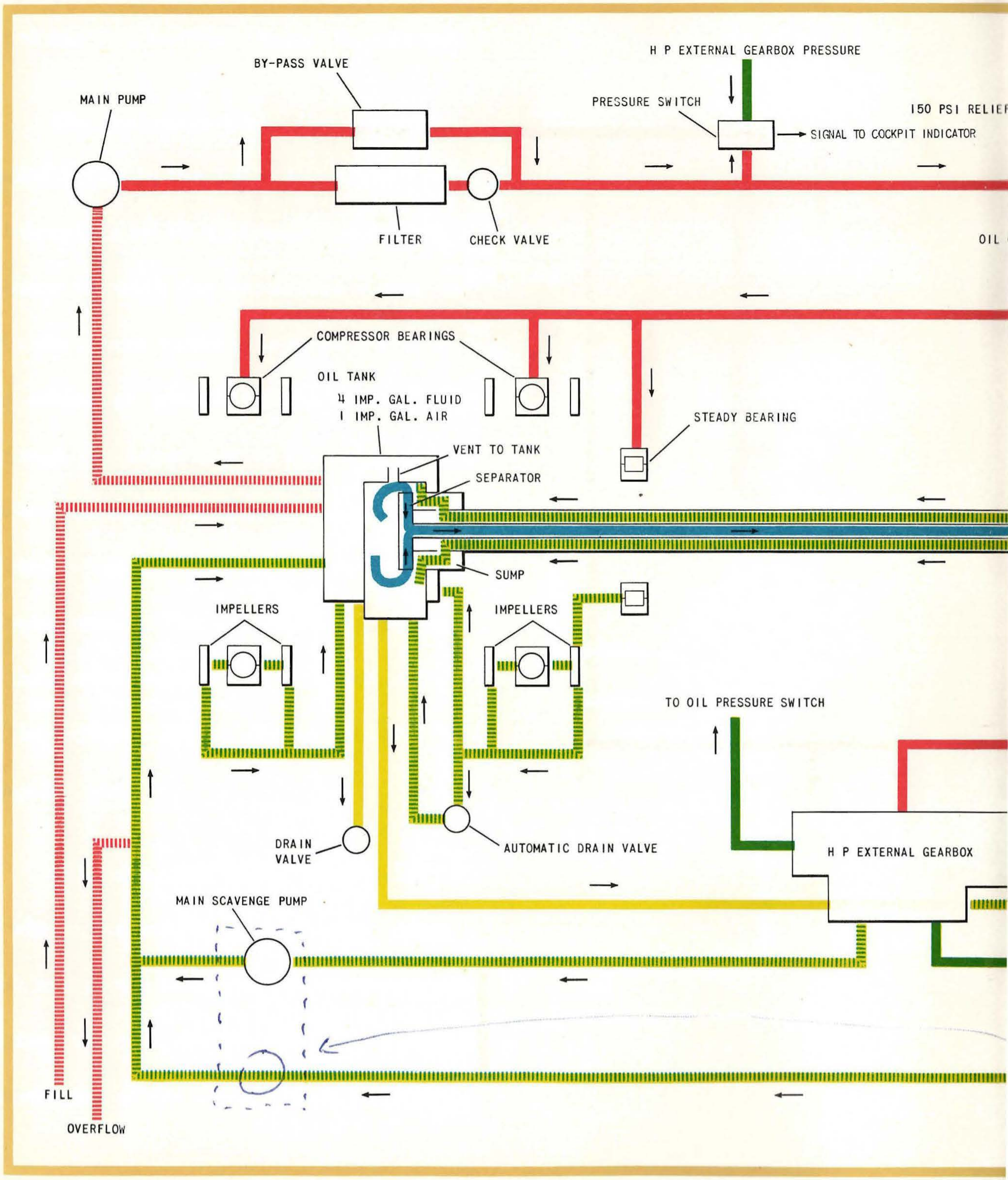
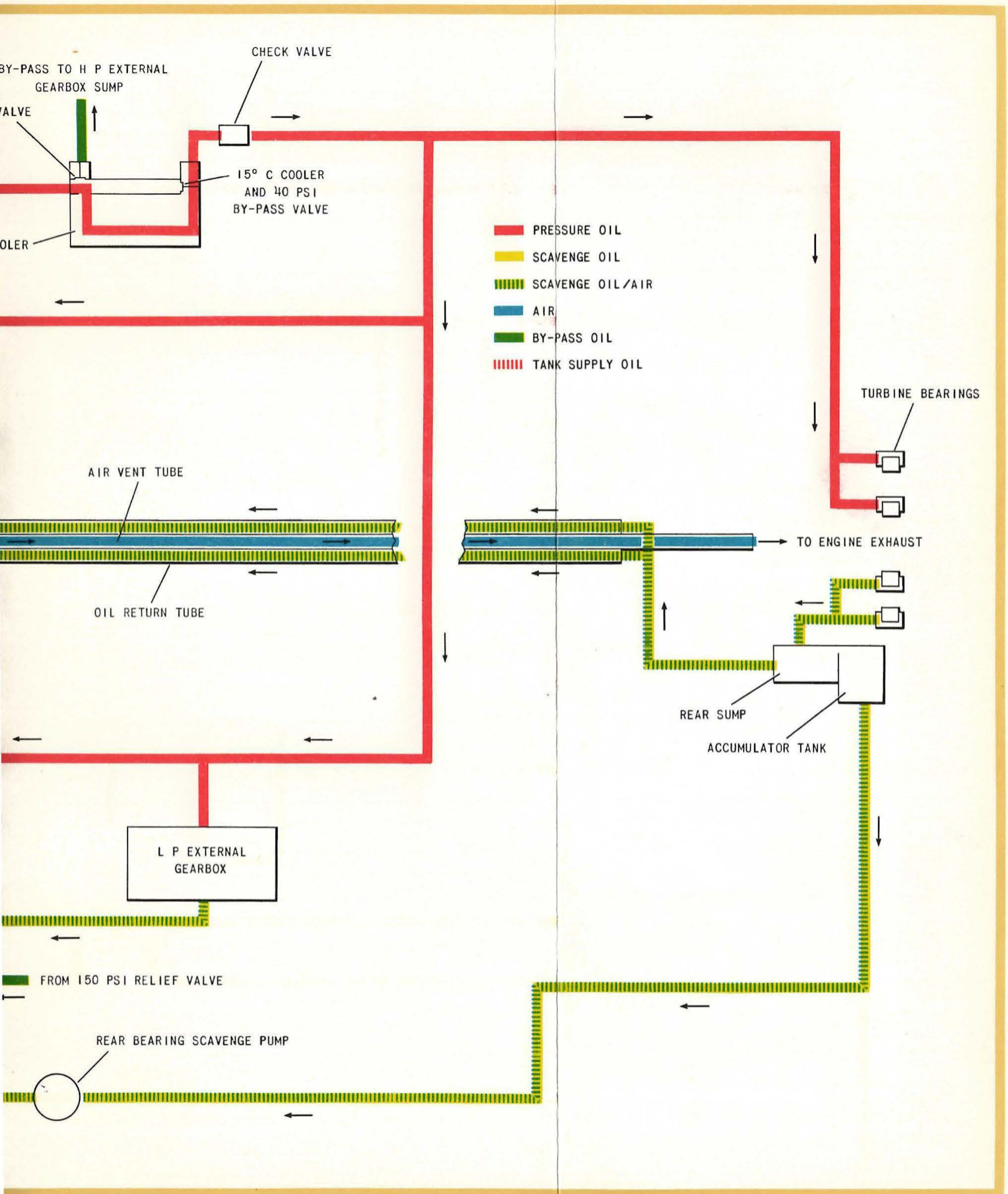


FIG. 4 LUBRICATING OIL SYSTEM - SCHEMATIC


BY-PASS TO H P EXTERNAL GEARBOX SUMP
CHECK VALVE
VALVE
15° C COOLER AND 40 PSI BY-PASS VALVE
OLER
PRESSURE OIL
SCAVENGE OIL
SCAVENGE OIL/AIR
AIR
BY-PASS OIL
TANK SUPPLY OIL
TURBINE BEARINGS
AIR VENT TUBE
TO ENGINE EXHAUST
OIL RETURN TUBE
REAR SUMP
ACCUMULATOR TANK
L P EXTERNAL GEARBOX
FROM 150 PSI RELIEF VALVE
REAR BEARING SCAVENGE PUMP

local conical sections impel the oil forward due to centrifugal effect.

On emerging from the oil return tube the air and oil flows through radial holes in the LP rotor shaft into the separator, which centrifuges all oil droplets of 10 microns and larger into the front sump. The air flows inward through the separator vanes into the air vent tube, then rearward into the space behind the rear sump in the exhaust bullet. Air returned by the scavenge pumps to the tank is vented to the front sump, and passes through the separator to the exhaust bullet in the same manner. An orifice plate located at the rear of the air vent tube restricts the flow, and therefore controls the overall air pressures throughout the system to the required proportions. The air discharges through a collector box on the underside of the ~~exhaust bullet~~ rear frame housing into the main exhaust gas stream. Any oil accumulating in the ~~exhaust bullet~~ rear frame housing drains into the collector box and is expelled with the outgoing air.

Oil collecting in the rear sump, drains into a small accumulator tank to which the auxiliary scavenge pump is connected. The purpose of this tank and pump is to ensure continuous scavenging of the rear sump during engine rundown, during which the air pressure may be inadequate for efficient scavenging of the sump through the oil return tube.

OXYGEN SYSTEM

OXYGEN SYSTEM

Oxygen is employed as a means of extending the normal flight relighting envelope of the engine and is not used during ground starting. Should any malfunction of the system occur or should the oxygen supply become exhausted, air relights are not prevented, but the ignition capabilities of the engine are merely reduced to the normal envelope.

The oxygen is stored in gaseous form in a pressure cylinder and a coupling between the cylinder and the system is self-sealing to ensure a minimum pressure loss to the system during coupling or uncoupling operations. A blow-out disc in the coupling releases the oxygen below the bursting pressure of the bottle. This pressure is more than twice the normal working pressure of the system, and more than 1.5 times the maximum working pressure. From this point the oxygen is led to the pressure regulating and shut-off valve. A pressure gauge is mounted on the regulating valve. The regulating valve consists of a throttling valve actuated by a set of pressure sensitive bellows on its downstream side. A filter is incorporated to protect the regulating valve orifice. A solenoid operated plunger on the regulating chamber outlet provides the necessary means for isolating the system. When this valve is in the shut position a spring arrangement provides for a relief valve mechanism which protects the system against excessive pressures in the event that the pressure regulator fails. Here the system splits and passes by way of two non-return valves with integral metering orifices. These orifices and the regulating valve determine the flow rate through the system. The non-return valves prevent combustion chamber pressure from entering the regulator; the shut-off valve also acts in this capacity. The regulator valve will withstand combustion chamber pressure, but with a line full of combustion products ignition is somewhat delayed.

The oxygen is injected into the hemispherical pots in the combustion base plate, near the igniter plugs through the oxygen ejector plugs. Thus the oxygen rich atmosphere locally created provides an area in which flame propogation is rapid and from which the flame may spread throughout the combustion chamber.

HYDRAULIC OIL SYSTEM

HYDRAULIC OIL SYSTEM

GENERAL

The engine hydraulic system provides a supply of oil at high pressure which is used primarily to actuate the variable area final nozzle, and the variable incidence HP compressor inlet guide vanes. The high pressure oil is also used, during engine starting, to drive a hydraulic motor to which the auxiliary fuel pump is coupled. The system operates on a closed circuit and consists basically of a reservoir to maintain a reserve of oil at low pressure, an accumulator, filters and a variable output hydraulic pump. Provision is made to bleed the HP side of the system.

A section of the lubricating oil temperature regulator is used to regulate the temperature of the hydraulic oil as it flows through the LP return line to the hydraulic pump; fuel passing through the regulator is used as the cooling medium. A thermal valve in this section of the regulator opens at oil temperatures of 85 degrees C (185 degrees F) and below allowing LP oil to by-pass the cooling element. This is to ensure rapid warm-up to normal operating temperatures on initial engine start.

HYDRAULIC PUMP

The supply of pressurized oil required for the operation of the system is provided by a variable stroke, axial piston type pump which is mounted on the HP external gearbox. The pump embodies an integral control which is made sensitive to pump output pressure, and which maintains a pressure of 3000 psi in the high pressure side of the system by adjusting the effective stroke of the pump. If, for example, the output pressure tends to drop due to the opening of the actuator control valve or a decrease in engine rpm, the pump control increases the stroke of the pump to maintain the pressure. Conversely, when a valve in the system closes, or when an increase in engine rpm occurs, the pump stroke is reduced to prevent an excessive build-up of pressure beyond the required value. When the full HP line pressure is attained, the pump delivers only sufficient oil to compensate for loss through the bleed holes in the final nozzle actuator pistons. This bleed flow provides a continuous minimum flow through the system to prevent overheating of the fluid due to stagnation in the actuators and also provides cooling flow for the pump.

FILTERS

Two 10 micron filters are provided in the system; one is located on the inlet and the other on the outlet side of the hydraulic pump. The filter elements are of the re-usable type and are formed of sintered wire cloth. Each filter is fitted with a pressure drop indicator which operates at 50 psi pressure difference to indicate that excessive pressure drop has occurred and that a filter change is necessary.

ACCUMULATOR

A 25 cu in. capacity piston type accumulator is fitted in the HP side of the system. The accumulator provides a reserve supply of HP oil and tends to stabilize the system when, for example, a sudden change occurs in the demand for HP oil as a result of a control valve opening or closing. The accumulator is pre-charged with nitrogen gas to a pressure of 1500 psi. A connection is provided for pressure checks, and for recharging the accumulator should this be found necessary during ground servicing.

RESERVOIR

A 90 cu in. capacity self-energizing oil reservoir is provided in the system. The reservoir consists of a piston and rod assembly which is free to move in a cylinder. HP oil acting on the small area of the piston rod is balanced by LP oil acting on the larger area of the piston outer face. The ratio of these areas is arranged so that the 3000 psi HP oil at the rod end exerts a pressure of 45 psi on the LP oil in the system. Unlike the conventional spring-loaded reservoir, the self-energizing type maintains an almost constant pressure at the pump inlet, irrespective of the volume of oil in the reservoir, or atmospheric pressure variations. Provision is made on the reservoir for the installation of a magnetic type switch which operates a warning light in the pilot's cockpit should the volume of oil fall below a pre-determined minimum amount.

An LP relief valve is incorporated in the reservoir to prevent excessive pressure build-up of the LP oil due to thermal expansion effects and over-filling. This valve is located in the reservoir piston and is operated mechanically when the valve stem contacts the inner face of the cylinder end wall. Oil passing through the valve to the space behind the piston is drained overboard.

For servicing purposes the volume of oil in the reservoir is indicated by a pointer and suitably graduated scale which is visible when the aircraft access panel is opened. The pointer is connected to the reservoir piston by a flexible cable.

HP RELIEF VALVE

An HP relief valve protects the HP side of the system from excessive pressure which might result from malfunctioning of the system components. The valve operates at pressures in excess of 3300 psi to by-pass HP oil to the LP side of the system.

PRESSURE FILLING CONNECTION

A quick-disconnect fitting for the attachment of pressure filling equipment is located on the underside of the engine. This fitting is positioned in close proximity to the oil reservoir volume indicator for ease of servicing, and is designed to prevent the entry of air during the filling operation.

AUXILIARY FUEL PUMP DRIVE MOTOR

The auxiliary fuel pump which provides the initial fuel supply to the engine during starting, is driven by a hydraulic motor of the fixed displacement, axial piston type. The hydraulic motor is operated by a supply of HP oil which is conveyed to the motor inlet through a solenoid-operated valve; the motor outlet is connected to the LP oil return to the hydraulic pump. The speed of the motor is controlled by a maximum flow valve which senses the volume of oil flowing through the motor.

When the engine starting button is pressed, a relay simultaneously energizes the solenoid which opens the valve and hence the HP oil line to the hydraulic motor. As the engine begins to rotate, pressure builds up rapidly in the hydraulic system, thus driving the hydraulic motor. The speed of the hydraulic motor increases until the oil flow through the motor approaches the maximum value established by the maximum flow valve. At this point the flow valve restricts the HP oil supply to the motor and stabilizes the motor speed.

As the engine speed increases, the main fuel pumps build up sufficient pressure in the fuel control supply line to close a check valve in the outlet of the auxiliary fuel pump. A switch on the check valve opens and de-energizes the hydraulic motor solenoid, thus cutting off the supply of HP oil to the hydraulic motor, stopping the motor and auxiliary fuel pump.

FINAL NOZZLE ACTUATOR AND CONTROL

Four hydraulic actuators, equally spaced around the engine final nozzle, are used to vary the final nozzle area to suit changing engine operating requirements. Each actuator consists basically of a piston and cylinder mechanism, the linear piston movement being transmitted to the final nozzle unison ring by direct mechanical linkage.

Movement of the piston in the actuator cylinder is regulated by an actuator control which is positioned as a function of the ratio of HP compressor delivery pressure (P_4) to turbine downstream pressure (P_7), by means of a pressure ratio actuator. The pressure ratio actuator maintains a constant trim of the final nozzle area at all flight conditions, to suit the the varying engine requirements. Variations in the P_4 and P_7 pressure ratio regulate the position of a valve in the actuator control, to direct HP oil or LP oil to the space on one side of the final nozzle actuator piston. HP oil is fed to the space on the other side of the final nozzle actuator piston at all times. Due to the difference in piston areas subjected to pressure, the piston will move to the right and increase the area of the engine final nozzle.

When an increase in the P_4 and P_7 pressure ratio occurs the actuator control valve moves and reduces the HP oil supply to the control. Simultaneously the LP port is uncovered and the pressure acting on the other side of the final nozzle actuator piston is relieved, allowing the actuator piston to move in and reduce the engine final nozzle area.

Under steady state operating conditions the pressure ratio actuator maintains a pressure on the left side of the piston which balances the existing nozzle load. A bleed through each of the actuator pistons maintains a continuous circulation of oil through the actuators for cooling purposes. A hydraulically operated roller type brake which locks the actuator in the event of failure of the engine hydraulic system is also embodied in each of the final nozzle actuators.

HP COMPRESSOR INLET GUIDE VANES CONTROL

The angle of incidence of the HP compressor inlet guide vanes is changed from minus 15 degrees to plus 25 degrees by means of three hydraulic actuators equally spaced around the periphery of the LP thrust bearing housing. (Later model engines will have one actuator located externally). The control and actuator consist of a solenoid-operated valve which controls the supply of HP and LP oil from the engine hydraulic system to the guide vane actuators. Adjustable stops in the actuators limit the range of movement. The supply of electric current to the solenoid is controlled by a switch in the engine amplifier unit as a function of HP compressor speed.

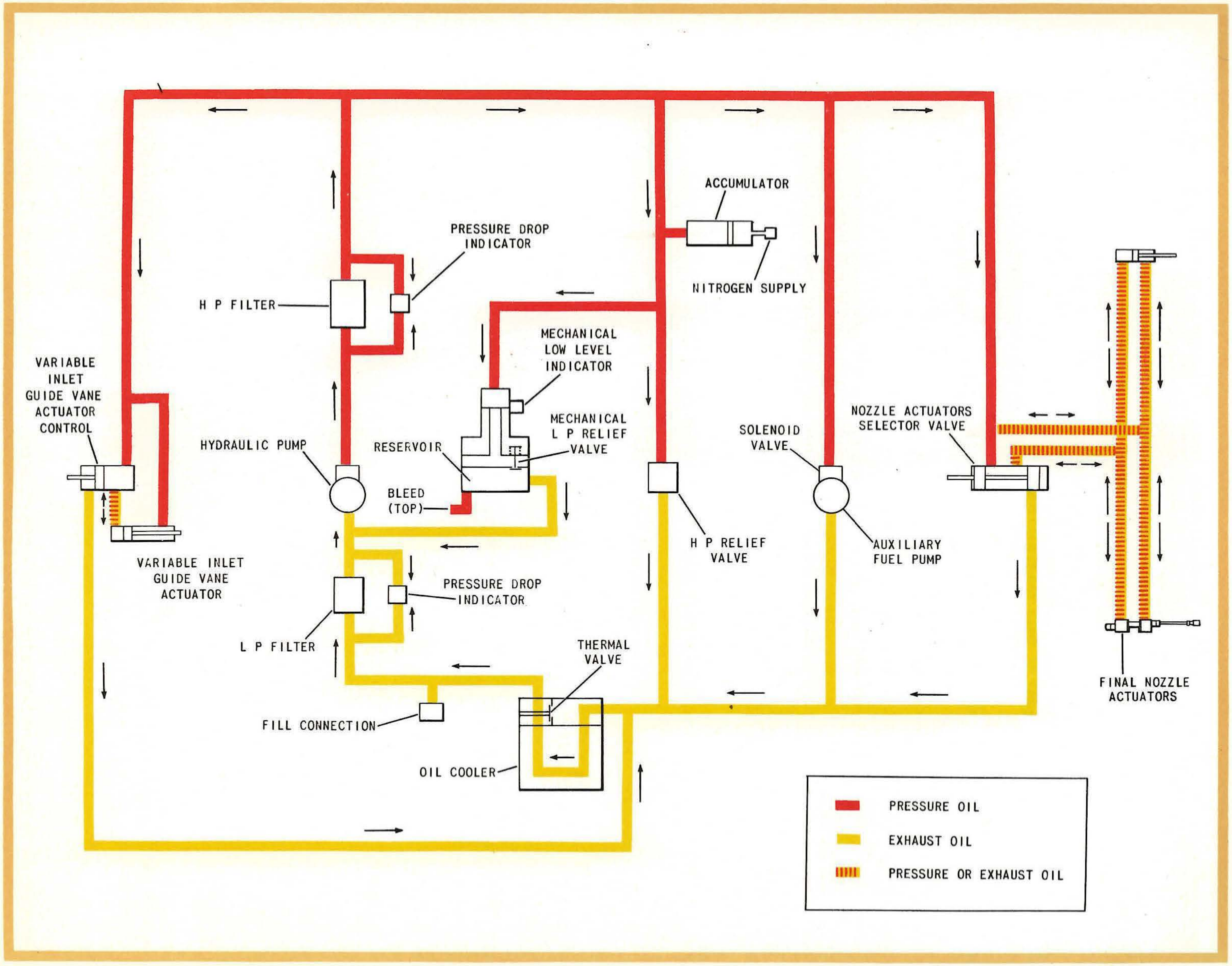


FIG. 5 HYDRAULIC OIL SYSTEM - SCHEMATIC

A spring mechanism incorporated in the linkage for each actuator, acts in opposition to the actuator travel from 0 degrees to plus 25 degrees guide vane angle. In the event of failure of the engine hydraulic system, the springs ensure that the guide vanes return to the 0 degree position, at which setting the engine will continue to operate satisfactorily under all conditions.

ELECTRICAL SYSTEM

12

ELECTRICAL SYSTEM

12.1 GENERAL

All the power required for operation of the engine electrical system is obtained from the aircraft electrical system. The engine system operates on 28 volts DC and 110 volts 400 cycle single phase AC, the latter is required for the electronic control amplifier and the anti-icing magnetic amplifier.

The main aircraft-to-engine electrical connector is of the quick release type locked by a latch mechanism mounted on the electrical connector panel on the lower right-hand side of the engine firewall assembly. All connections between the engine and airframe circuits are routed through this connector except for the fire detector and thermocouple circuits which are individually harnessed. The fire detector and thermocouple connectors are mounted on the panel adjacent to the main connector.

The operation of the engine electrical system is almost entirely automatic; the only pilot-operated controls are the Start-Motor switch, the relight switch and the Normal-Emergency switch. Additional switches in the power lever switch box are actuated by movement of the power control lever.

A schematic wiring diagram of the engine electrical system and associated details of the aircraft electrical system is shown in Figure 6.

The following engine controls, mounted in the cockpit, are used in conjunction with the engine electrical system and constitute all the controls necessary for the operation of one engine.

(a) Throttle lever

(b) Emergency-Normal switch

(c) Start-Motor switch

(d) Relight switch.

A summary of the circuits shown in Figure 6 is given in the paragraphs which follow:

STARTING SYSTEM

Ground Starting: With the power lever advanced past eight degrees the Start-Motor switch is moved to the start position for approximately one second and then released. A circuit will be made via the 2500 rpm speed switch and the zero to ~~four~~ eight degrees throttle angle switch to energize relay K-1. This relay will then lock-in through one of its own contacts.

The starter solenoid valve on the ground starting trolley is now energized through the contacts on K-1 and when the engine has reached a speed of approximately 200 rpm relay K-2 is also energized. The contacts on relay K-2 then provide the circuits for energizing the ignition exciter, ~~fuel valve~~ and starting fuel pump solenoid valve. Engine light-up will now take place.

At an engine speed of approximately 2500 rpm the speed switch will open causing relay K-1 to be de-energized. This will in turn de-energize relay K-2 and the ignition exciter. The starting pump solenoid valve will remain energized via the check valve switch. This switch is closed by the starting fuel flow and provides a hold-on for the starting pump until the main pumps take over. If light-up does not occur and the engine does not reach a speed of 2500 rpm the system will be reset by moving the throttle lever ~~below four degrees~~ to the off position, i.e. de-energizing relays K-1, K-2 and the starting pump solenoid.

Engine Motoring: The Start-Motor switch when moved to the 'Motor' position will energize the starter solenoid on the ground starting trolley for as long as the switch is depressed. As none of the other starting circuits is energized, motoring of the engine can be carried out with the throttle lever in any position beyond eight degrees.

Flight Re-starting: The relight switch is a push button mounted on the throttle lever. While depressed it completes a circuit for energizing relay K-2 directly. Relay K-2 energizes the same components as on ground starting with the addition of the oxygen valve solenoid which is energized through the aircraft undercarriage scissor switch relay. If the engine fails to relight, the starting pump solenoid will be de-energized by moving the throttle lever to the 'Off' position.

IGNITION SYSTEM

The engine utilizes a high energy (12 joule) ignition system. Two igniter plugs per engine are used to initiate combustion in the annular combustion chamber. Each plug is separately operated through a two circuit system and energized through its own section of a dual exciter box. The igniter plugs are of the low voltage, surface gap type.

In order to provide high energy discharge at the plugs, DC current from the 28 volt aircraft supply is passed through the exciter box which operates on the 'make and break' principle, to charge a storage capacitor with a series of high voltage surges from a step-up transformer. A sealed spark gap in the exciter box discharges the stored energy to the two igniter plugs to give one spark per second per plug.

The exciter box is mounted on the side of the inlet frame and the two igniter plugs are fitted one on each side of the engine at the forward end of the combustion zone, below the engine horizontal axis. Leads with flexible metal shielding connect the plugs to the exciter box. The shielded leads, in conjunction with a filter in the exciter box, ensure that radio interference from the ignition system is within the limits laid down in Specification MIL-I-6181.

124 CONTROL SYSTEM

Electronic Amplifier: Inputs to amplifier are as follows:

28 volts DC

110 volts 400 cycles AC

HP rotor speed signal

Inlet temperature signal

Thermocouples (jet pipe temperature reference - PL Potentiometers)

Functions controlled by the amplifier are:

1) Fuel flow control during accelerations to permit acceleration to be achieved at the maximum rate consistent with jet pipe temperature and engine surge characteristics.

2) Steady state jet pipe temperature control.

3) Automatic speed switching of:

a) Reference temperatures for (1) and (2).

b) Changeover in amplifier output from (1) to (2) in the high speed range.

c) Variable inlet guide vanes.

d) Afterburner lock-out below a certain minimum engine speed.

Function (1) is provided by the fuel proportional valve and function (2) by the proportional air solenoid valve.

Afterburner Relay Control System: This controls the operation of the afterburner shut-off solenoid valve and hot-streak solenoid. The system is energized by the afterburner throttle switch in series with the speed lock-out switch in the control amplifier.

Emergency Changeover Valve: This is a two-position motorized valve which changes the fuel system to 'Normal' or 'Emergency' on selection of the switch.

12.5 PHYSICAL CONFIGURATION OF ELECTRICAL SYSTEM

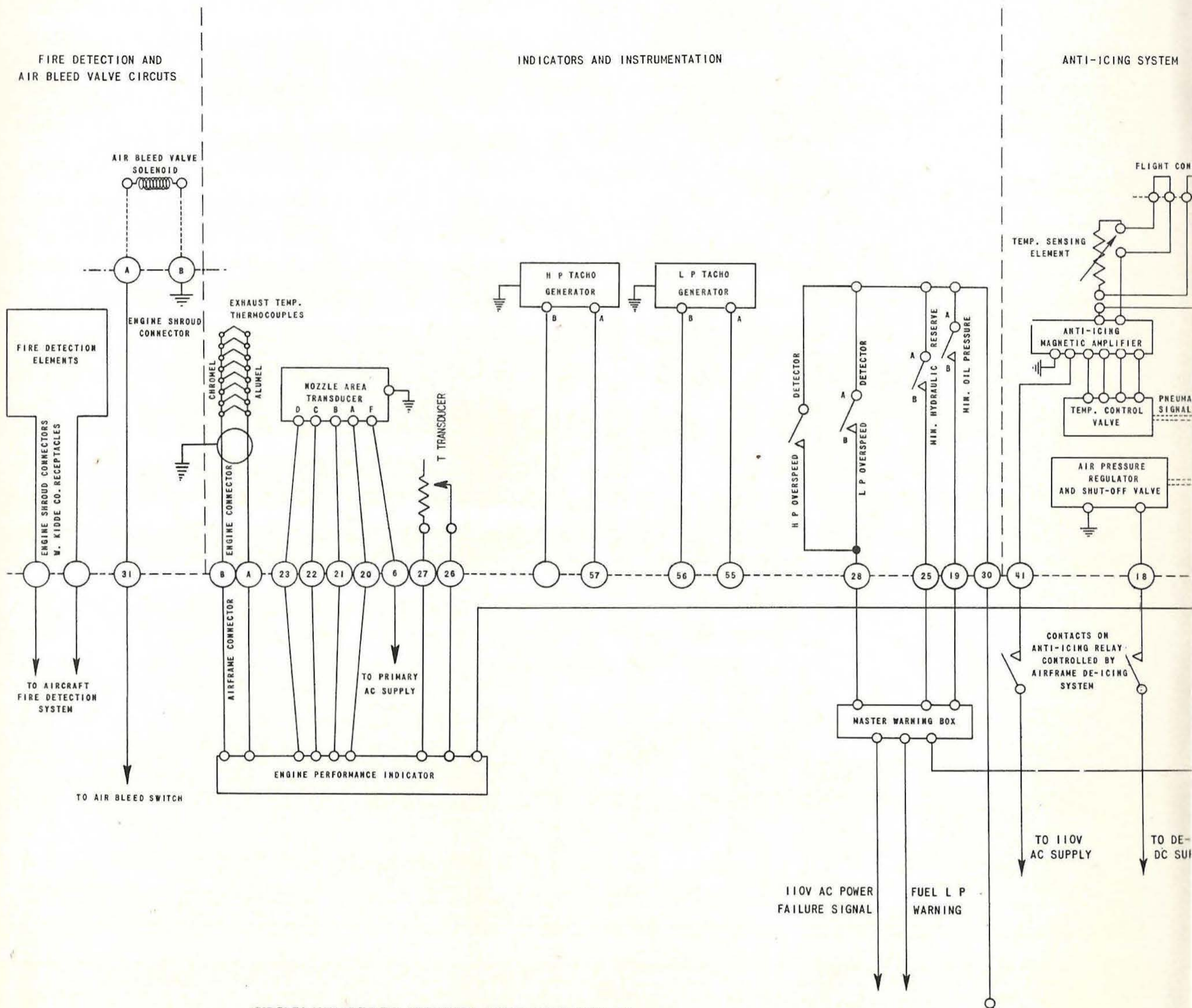
The engine circuit is designed to operate at altitudes up to 75,000 feet within the ambient temperature range of -54 degrees C (-65 degrees F) to 177 degrees C (350 degrees F). All connectors are 'potted' to provide sealing and support, and are of the type rated at 177 degrees C (350 degrees F) continuous operation. With the exception of the thermocouple circuit, all wires conform to the requirements of Specification MIL-W-7139A.

The engine to aircraft electrical connector is a quick-release type controlled by a latching device attached to the engine shroud.

ANTI-ICING SYSTEM

12.6 General

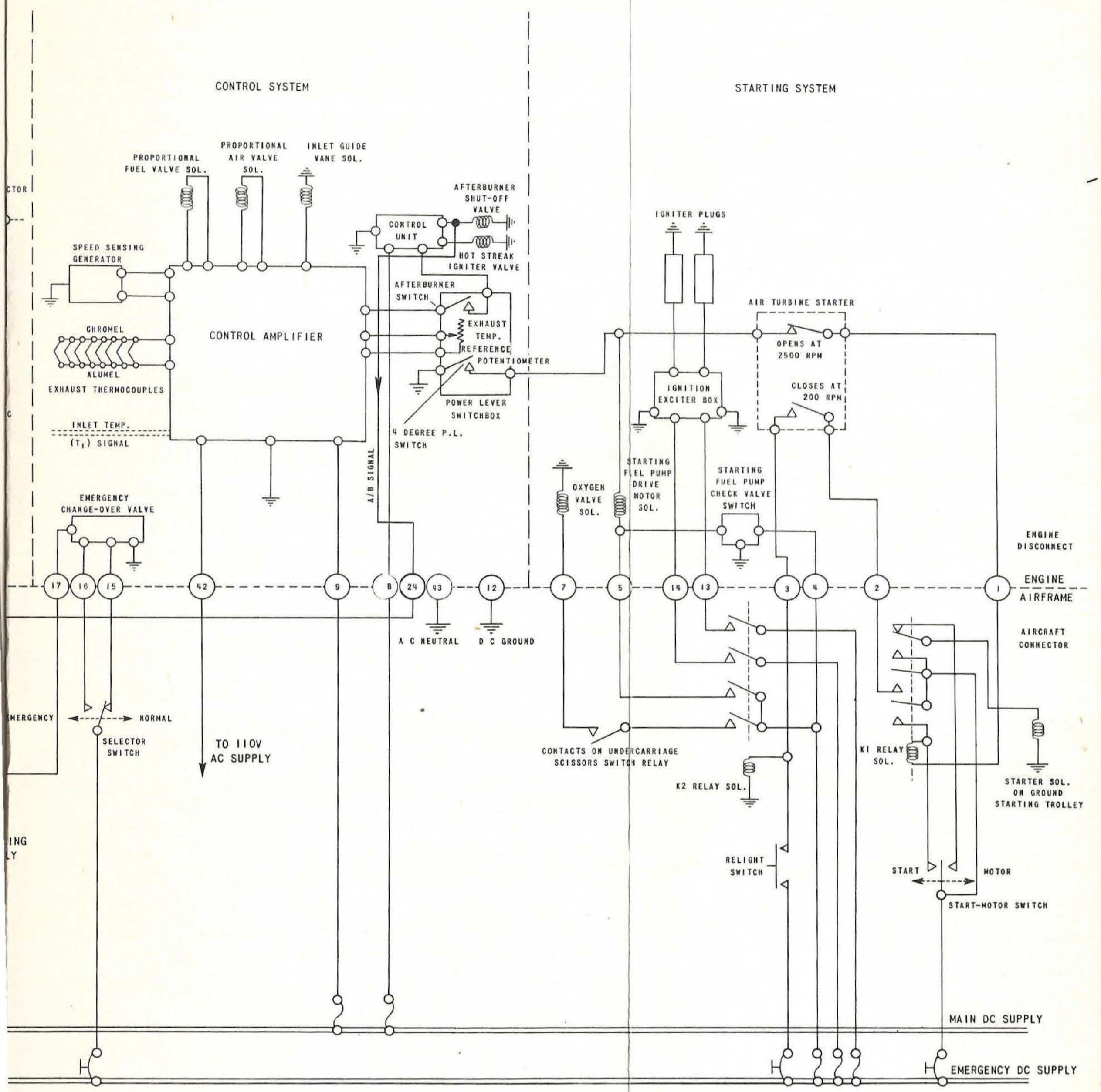
The inlet frame when fitted, provides support for the constant speed unit and alternator. It must be de-iced in order to provide all-weather



CIRCLES NUMBERED BETWEEN 1 AND 58 REPRESENT PIN NUMBERS ON THE ENGINE TO AIRFRAME MAIN CONNECTOR.

SEPARATE CONNECTORS ARE PROVIDED FOR THE FIRE DETECTOR AND THERMOCOUPLE CIRCUITS, AND ALSO IN THE ANTI-ICING AND AIR BLEED VALVE CIRCUITS.

FIG. 6 ELECTRICAL SYSTEM WIRING DIAGRAM



CONTROL SYSTEM
STARTING SYSTEM
PROPORTIONAL FUEL VALVE SOL.
PROPORTIONAL AIR VALVE SOL.
INLET GUIDE VANE SOL.
AFTERBURNER SHUT-OFF VALVE
CONTROL UNIT
HOT STREAK IGNITER VALVE
SPEED SENSING GENERATOR
AFTERBURNER SWITCH
CHROMEL
CONTROL AMPLIFIER
EXHAUST TEMP. REFERENCE POTENTIOMETER
ALUMEL
EXHAUST THERMOCOUPLES
POWER LEVER SWITCHBOX
INLET TEMP. (T1) SIGNAL
4 DEGREE P.L. SWITCH
A/B SIGNAL
EMERGENCY CHANGE-OVER VALVE
IGNITER PLUGS
AIR TURBINE STARTER
OPENS AT 2500 RPM
CLOSES AT 200 RPM
IGNITION EXCITER BOX
OXYGEN VALVE SOL.
STARTING FUEL PUMP DRIVE MOTOR SOL.
STARTING FUEL PUMP CHECK VALVE SWITCH
ENGINE DISCONNECT
ENGINE
AIRFRAME
A C NEUTRAL
D C GROUND
AIRCRAFT CONNECTOR
EMERGENCY
NORMAL
SELECTOR SWITCH
TO 110V AC SUPPLY
CONTACTS ON UNDERCARRIAGE SCISSORS SWITCH RELAY
K2 RELAY SOL.
K1 RELAY SOL.
STARTER SOL. ON GROUND STARTING TROLLEY
RELIGHT SWITCH
START
MOTOR
START-MOTOR SWITCH
MAIN DC SUPPLY
EMERGENCY DC SUPPLY

protection for the engine. For the same purpose provision is also made for de-icing the first stage stator blades. A diagram of the system is shown in Figure 7 and the electrical connections are shown in Figure 6.

The engine portion of the control system consists of a thermistor, magnetic amplifier, electro-magnetic pilot valve, regulator valve and automatic shut-off valve. The aircraft portion consists of an ice detector, timer (or equivalent), power supplies, control switch and cockpit indicator.

FIRE DETECTION SYSTEM

Two connectors on the engine firewall electrical connector panel are fitted to the end of the fire detector loop and connect into the aircraft detector system.

ORENDA ENGINES LIMITED

ANTI-ICING SYSTEM

ANTI-ICING SYSTEM

GENERAL

During engine operation in weather conditions conducive to icing, the formation of ice at the engine air intake is prevented by a hot air, surface heating, anti-icing system. The system is fully automatic in operation and is designed to operate at air pressures up to approximately 45 psig. The system is illustrated in Figure 7.

A supply of air, bled from the delivery side of the HP compressor, is supplied to the LP compressor hollow first stage stator blades, and to the inlet frame assembly where the air heats the outer skins of the nose bullet and frame struts.

EXTERNAL PIPING AND DUCTING

The supply of air is piped externally from the upper right-hand side of an air take-off manifold immediately downstream of the HP compressor to an air manifold formed around the LP compressor stator casing, thence to the outer ends of the four inlet frame struts.

COMPRESSOR STATOR BLADE HEATING

The compressor first stage stator blades are heated by a flow of air from the LP compressor stator casing air manifold. As each blade is hollow and open-ended, the air flows through the blades and is expelled inward to the engine air stream.

NOSE BULLET AND INLET FRAME STRUT HEATING

The flow of air delivered to the air ducts of the inlet frame struts passes into the struts through a cavity at the front of each. Holes in the strut inner skin direct the air flow against the leading edge inner surface and into the gap formed between the inner and outer skins. The air flows rearward through the gap and is then led through a cavity at the rear of each strut to the inlet frame internal air manifold.

The air is then piped from the internal air manifold to the front of the nose bullet and flows into the gap formed by the double skin of the bullet. The air flows rearward through the gap to a corresponding gap in the inlet frame inner casing and is exhausted to the engine air stream immediately upstream of the compressor first stage rotor blades.

AIR PRESSURE REGULATOR AND SHUT-OFF VALVE

The quantity of air required for adequate anti-icing protection is controlled by the air pressure regulator and shut-off valve. The valve is comprised mainly of a spring-loaded, pressure sensitive diaphragm which opens and closes a butterfly type air modulator valve located in the main air supply pipe, together with a solenoid-operated shut-off valve which is electrically connected to an aircraft mounted ice detection system. Opening and closing of the modulator valve regulates the amount of air passing into the anti-icing system.

When the system is inoperative, pressure derived from a tapping upstream of the air modulator valve is applied to the spring-loaded side of the actuator diaphragm to hold the valve closed. When icing conditions are detected, the normally closed, solenoid-operated shut-off valve is energized by the aircraft mounted ice detection system, and the pressure on the under side of the actuator diaphragm is transferred to the upper side of the diaphragm. The air modulator valve opens and the air flow to the system begins. The shut-off valve remains energized, and hence the air flow continues, until the end of a pre-determined cycle which is timed by the ice detection system. If the icing condition is still sensed by the ice detection system at the end of the cycle, the solenoid valve remains in the same position.

A maximum pressure limiter, which is integral with the air pressure regulator, is sensitive to the air pressure at the LP compressor stator casing air manifold. A rise in manifold pressure above calibrated limits unseats a valve in the pressure limiter and permits this pressure to be applied to the spring-loaded side of the actuator diaphragm in the air pressure regulator, thus moving the air modulator valve toward the closed position to reduce the air ~~flow~~ pressure.

TEMPERATURE CONTROL VALVE

As high engine speeds result in a higher bleed air temperature, less air is required to maintain adequate anti-icing protection. To prevent excessive usage of air, a trim is applied to the degree of modulator valve opening, as a function of the actual skin temperature of the engine intake components, by an electro-magnetic temperature control valve.

A skin temperature sensing element, comprising a thermistor mounted at the front of the inlet frame inner casing, varies the current output from the magnetic amplifier as a function of its change in resistance; the thermistor resistance decreases with a rise in temperature. The amplifier in turn transmits an amplified signal in the form of current flow to the temperature control valve in proportion to the skin temperature. The control valve regulates a bleed-off of the pressure acting on the air pressure regulator valve actuator diaphragm. If a low temperature is sensed by the thermistor, the control valve reduces the bleed-off, and the pressure acting on the actuator diaphragm increases. This results in an increased opening of the air modulator valve and a corresponding increase in the air flow. As the skin temperature of the intake components rises, the temperature

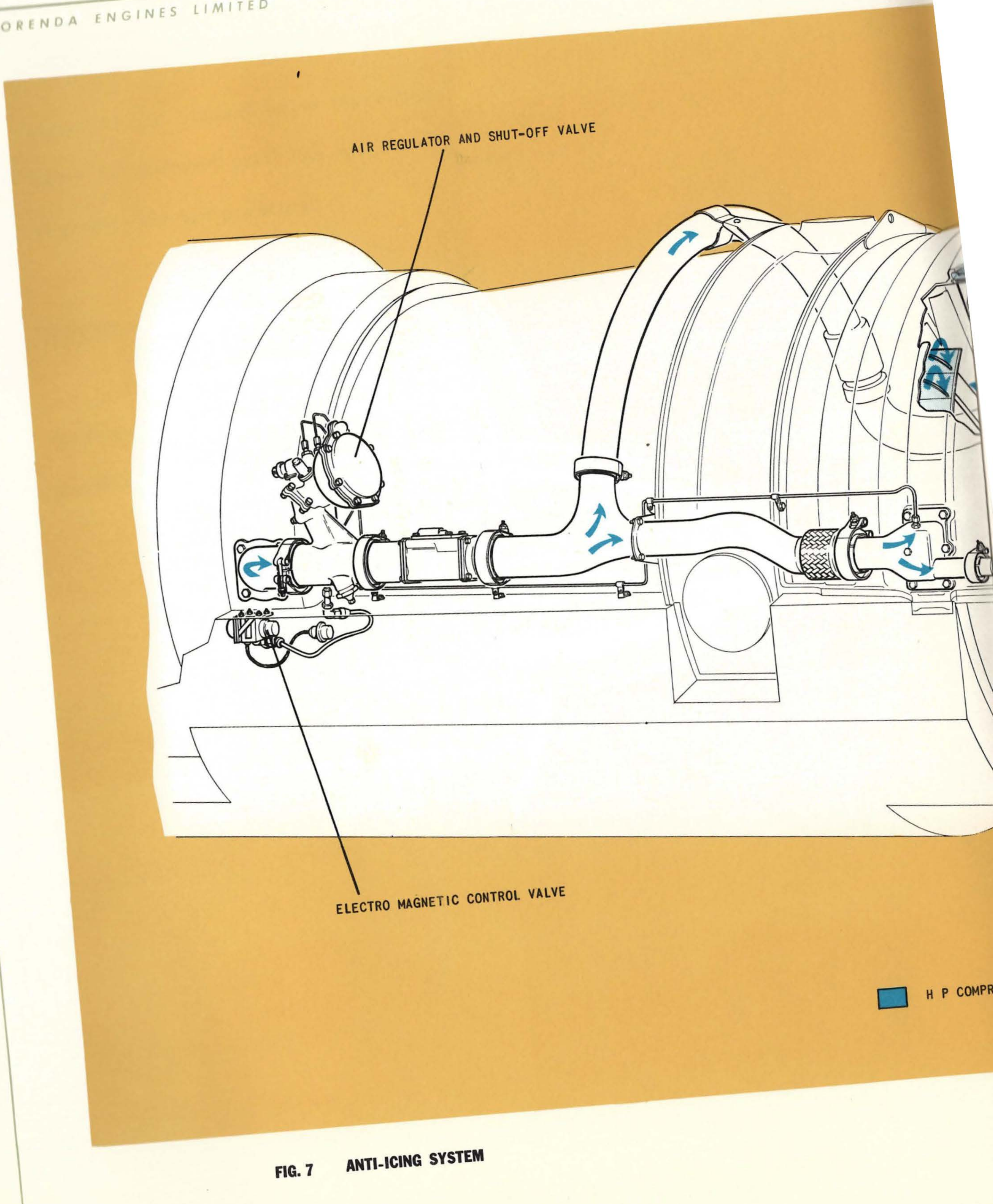
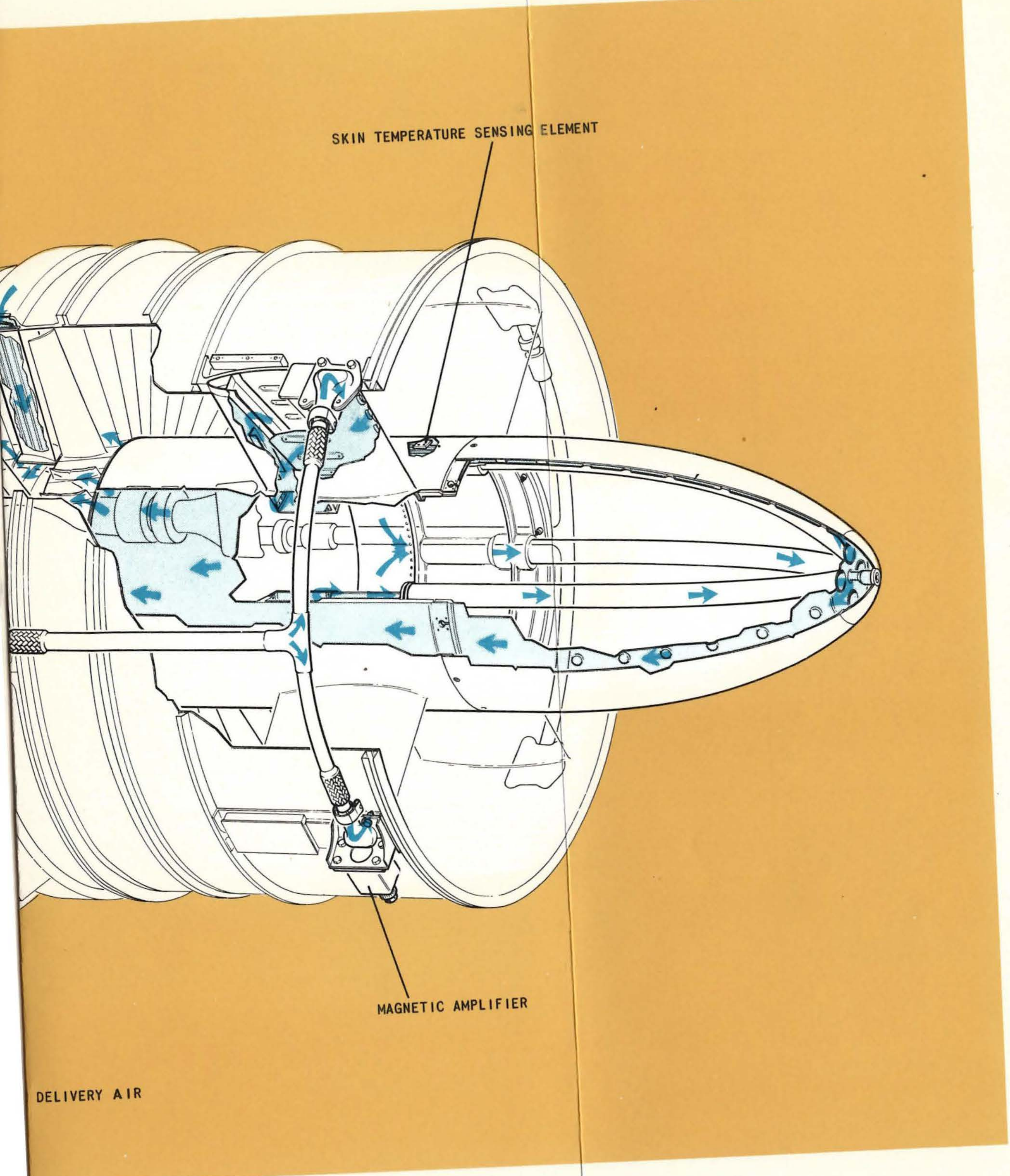


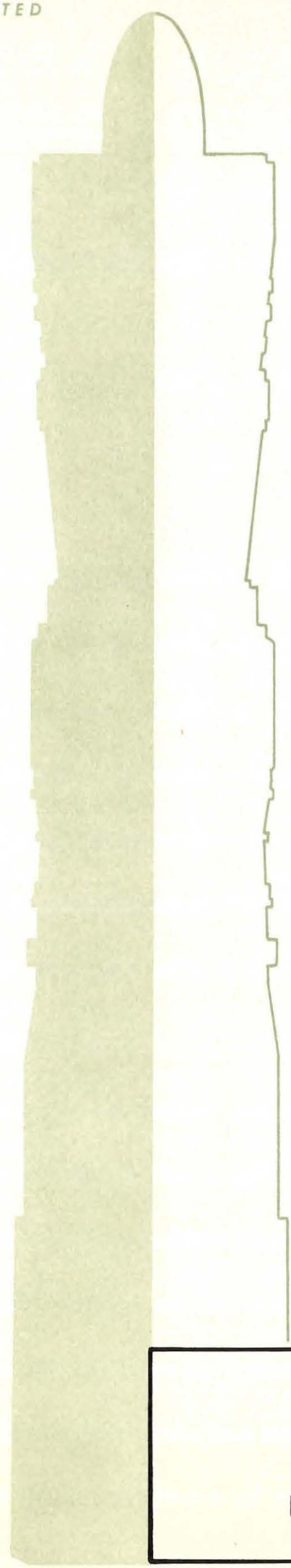
FIG. 7 ANTI-ICING SYSTEM


SKIN TEMPERATURE SENSING ELEMENT
MAGNETIC AMPLIFIER
DELIVERY AIR

signal to the control valve increases and the opening of the air modulator valve is reduced until the air flow is at the value which maintains the desired skin temperature.

PRESSURE CHECK VALVE

A pressure check valve, which is installed in the main supply pipe downstream of the air modulator valve, cuts off the hot air supply in the event of a failure in the air pressure regulator and shut-off valve.



FIRE DETECTOR AND EXTINGUISHER SYSTEMS

FIRE DETECTOR AND EXTINGUISHER SYSTEMS

GENERAL

The fire detector system for the Iroquois is of the continuous-wire, heat and fire detection type consisting of heat sensing elements, a control unit and two coloured cockpit warning lights. The fire extinguisher system comprises a network of lines which carry extinguisher fluid to the forward region of the engine accessories compartment. A threaded connector on the engine firewall assembly connects the lines to a supply line from the aircraft mounted fire extinguisher bottles, see Figure 8

FIRE DETECTOR SYSTEM

Three heat sensing elements together with a fire resistant cable, form a continuous loop inside the engine firewall assembly, passing near the top of the engine firewall assembly on the right-hand side of the engine, down toward the bottom centre at the mid-frame, and then rearward along the bottom of the firewall to the rear frame. The element assembly returns along the left-hand side of the engine in a route opposite to the right-hand route, with the exception that at the front of the firewall assembly it is routed under the engine to the right-hand side. Each end of the element assembly is connected to a socket receptacle on the engine firewall electrical connector panel located near No. 4 strut of the front frame; the receptacles are connected to mating plugs in the aircraft fire detector system. Two elements are also positioned around the engine nacelle. These, together with the fire detector control unit, are included in the aircraft part of the system.

The three heat sensing elements are identical and consist of a semi-flexible Inconel tube enclosing a ceramic thermistor core and ~~one~~ two internal wires. Each element is fitted with a plug at one end and a receptacle at the other. A short length of fire resistant cable, with a socket at each end, connects the elements at the rear section of the engine firewall. As the temperature coefficient of resistivity of the thermistor material is negative, the resistance between the wires and the Inconel tube varies inversely with the temperature of the element. The system monitors the resistance variations of the sensing element to provide an amber light overheat signal at 205 degrees C (400 degrees F) and a red light fire signal at 288 degrees C (550 degrees F). No signal is given for normal temperature changes up to the average maximum ambient temperature of 177 degrees C (350 degrees F).

The system also incorporates an averaging temperature sensing control which ensures that the total length of the detector element must be subjected to the preceding temperatures before an alarm is given. Since a fire would probably not affect the complete element at one time, the temperature necessary to cause an alarm is inversely proportional to the length of the element in the fire. An excessive rate of heat rise, above the normal ambient temperature, illuminates the red light instead of the amber light to indicate an abnormal temperature increase due to a fire hot spot. Upon elimination of the hazard, the high resistance of the thermistor re-establishes itself, and the system is ready to detect any further hazards. The existence of a break in the sensing element can be established by a continuity check. However, the system continues to function as a detector in spite of a break.

FIRE EXTINGUISHER SYSTEM

The engine portion of the fire extinguisher system, when operating, is supplied with extinguisher fluid through a connection located on the underside of the engine firewall assembly, adjacent to the combustion drains fitting. Inside the engine firewall a short length of pipe extends forward from the connector to a cross fitting from which three open-ended lines carry the fire extinguisher fluid to the region between the outer ends of the front frame No. 5 and 6 struts, and to the region fore and aft of the HP external gearbox.

In the event of fire during an engine ground run, provision is made for a manual fire extinguishing procedure. Two fire traps, each consisting of

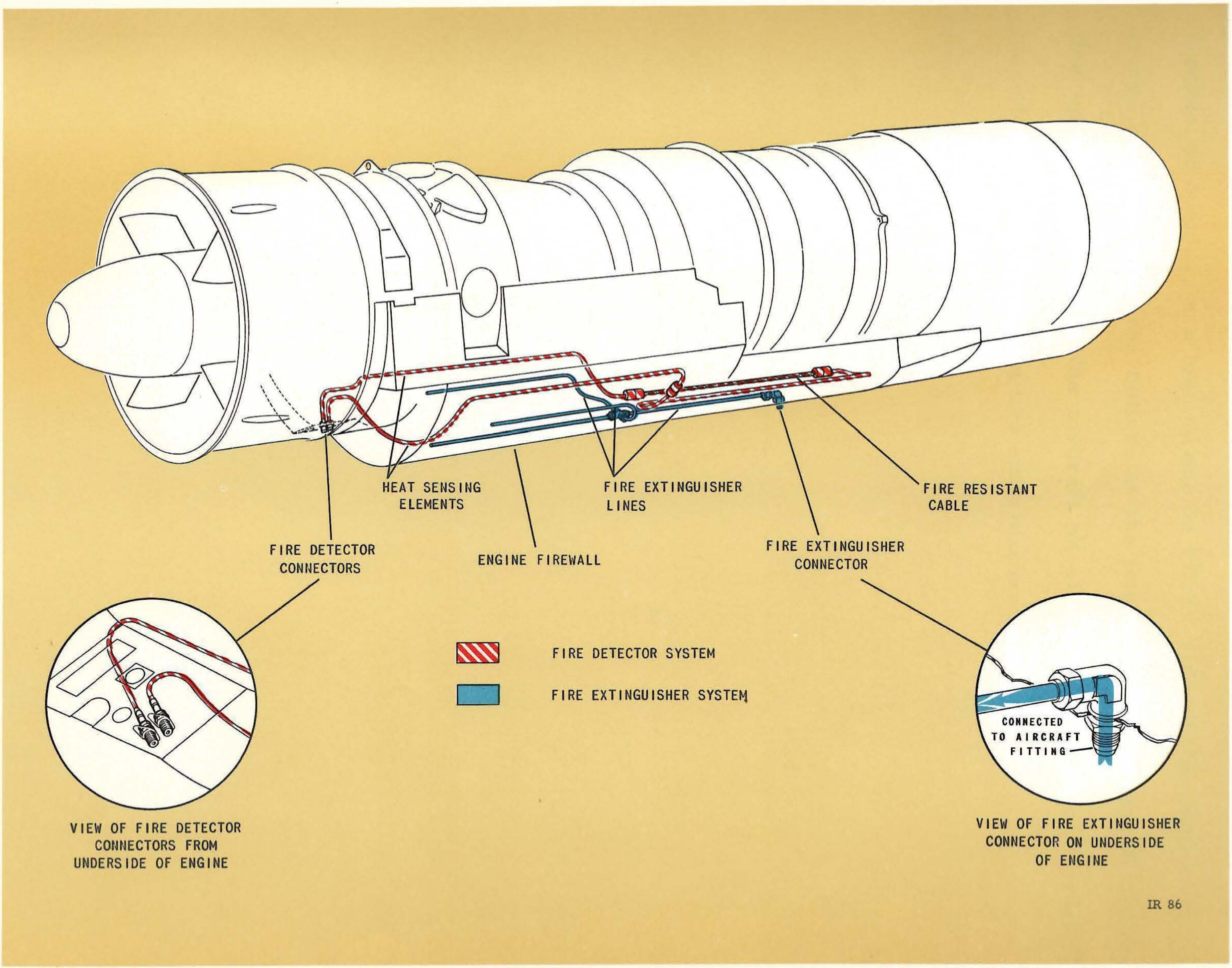


FIG. 8 FIRE DETECTOR AND EXTINGUISHER SYSTEMS

a screened opening, are located one on either side of the engine firewall centre section, near the bottom of the engine. The screened openings line up with openings in the aircraft nacelle structure which permit the insertion of hand or mobile fire extinguisher nozzles.

NOTES

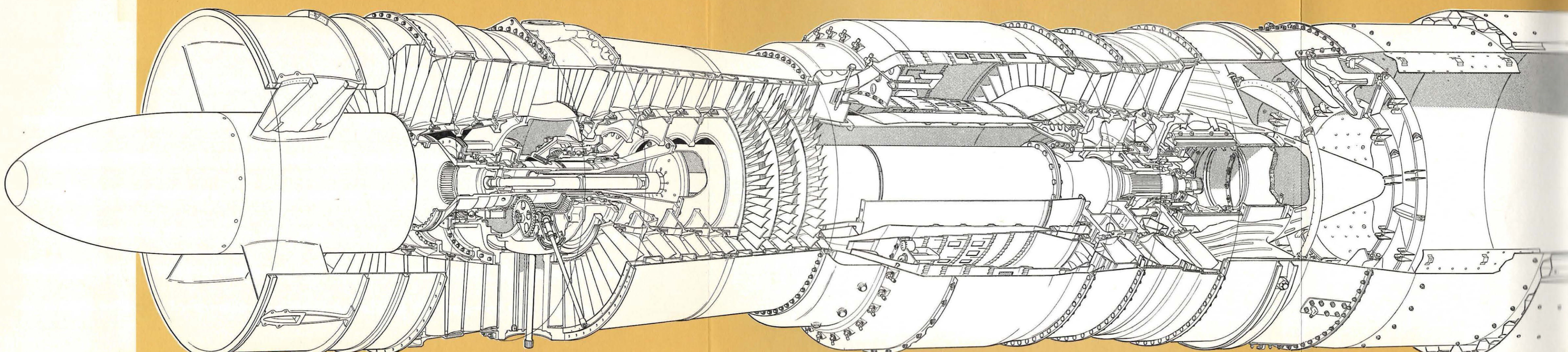


FIG. 2 SECTIONED VIEW OF ENGINE

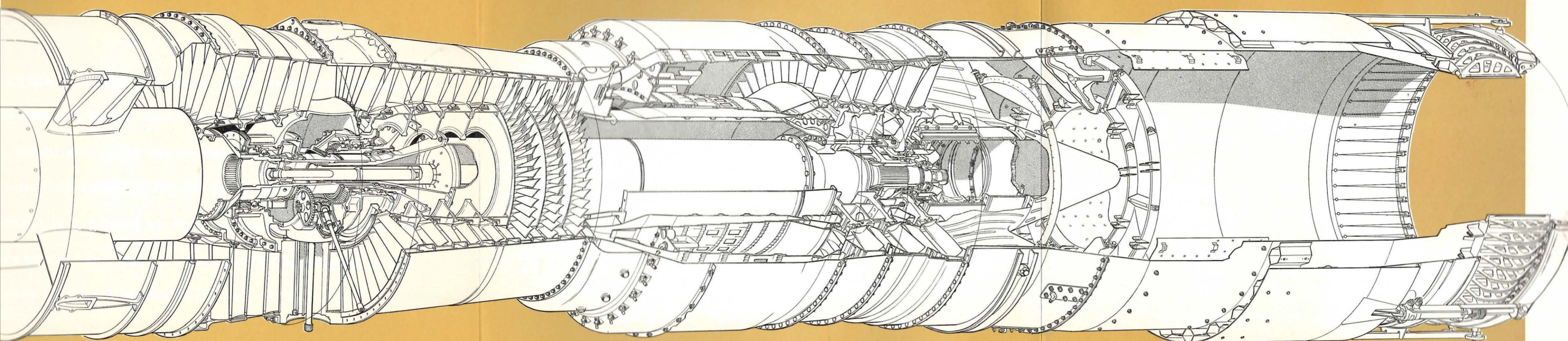


FIG. 2 SECTIONED VIEW OF ENGINE

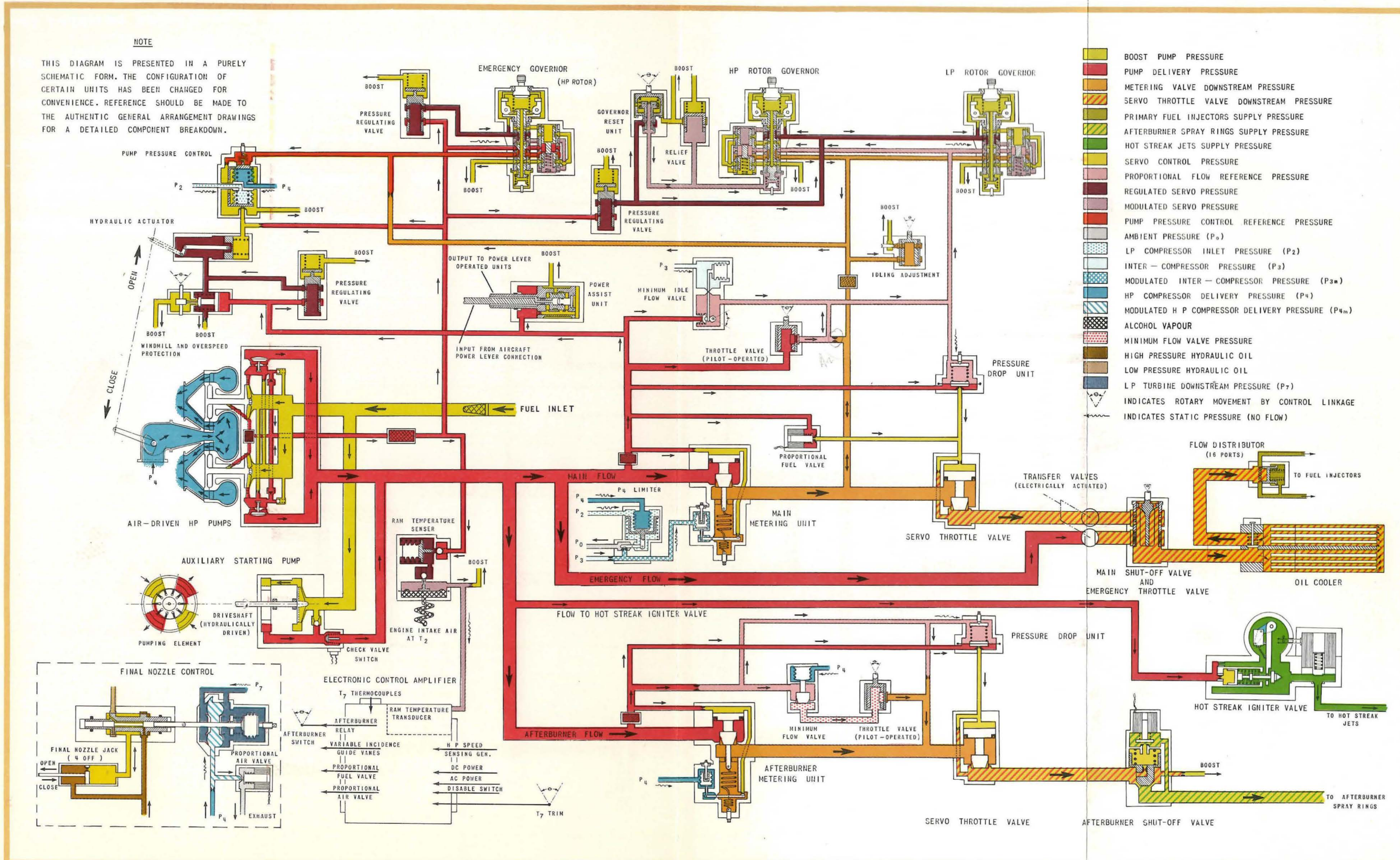


FIG. 3 POWER CONTROL SYSTEM - SCHEMATIC

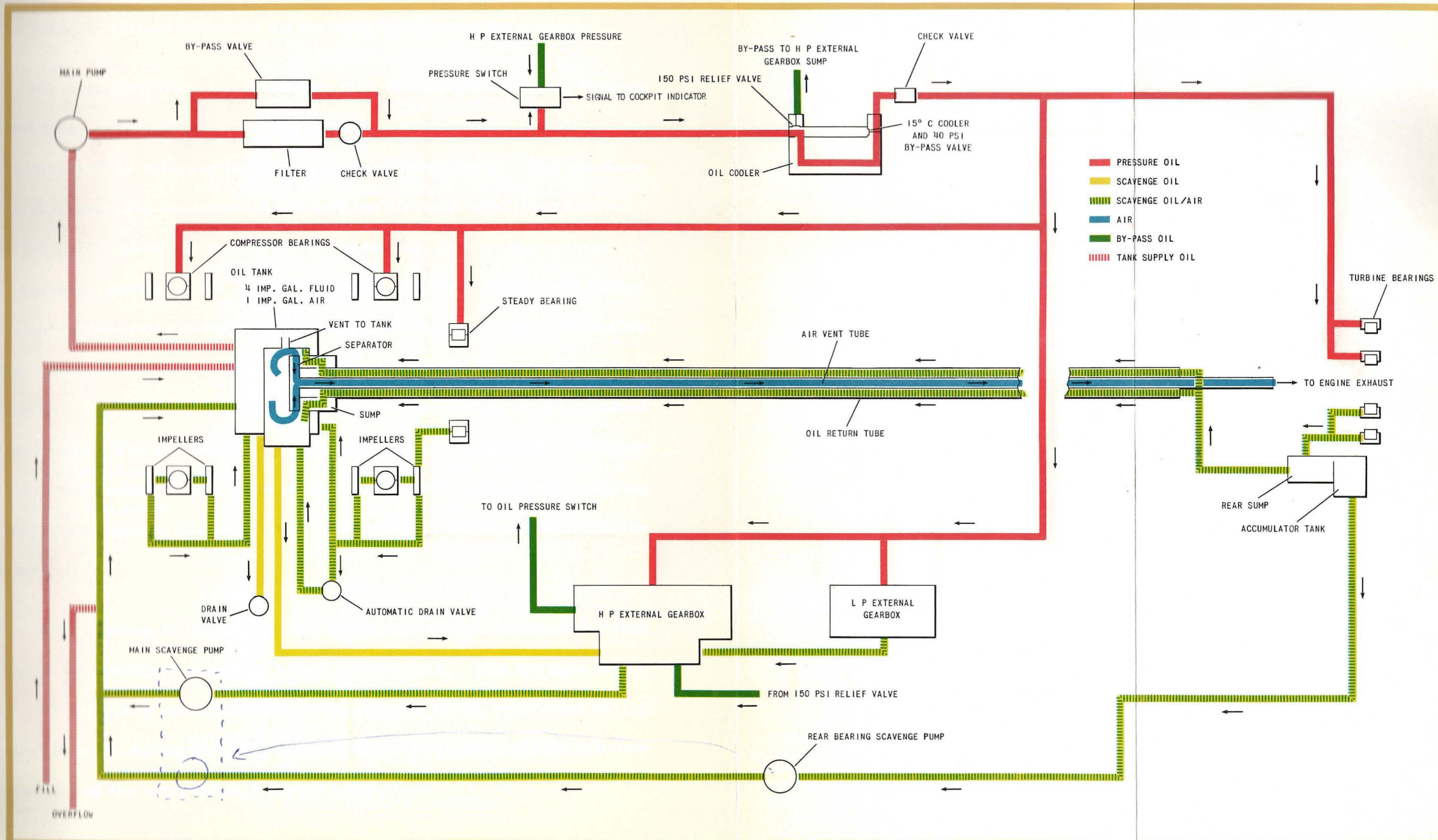
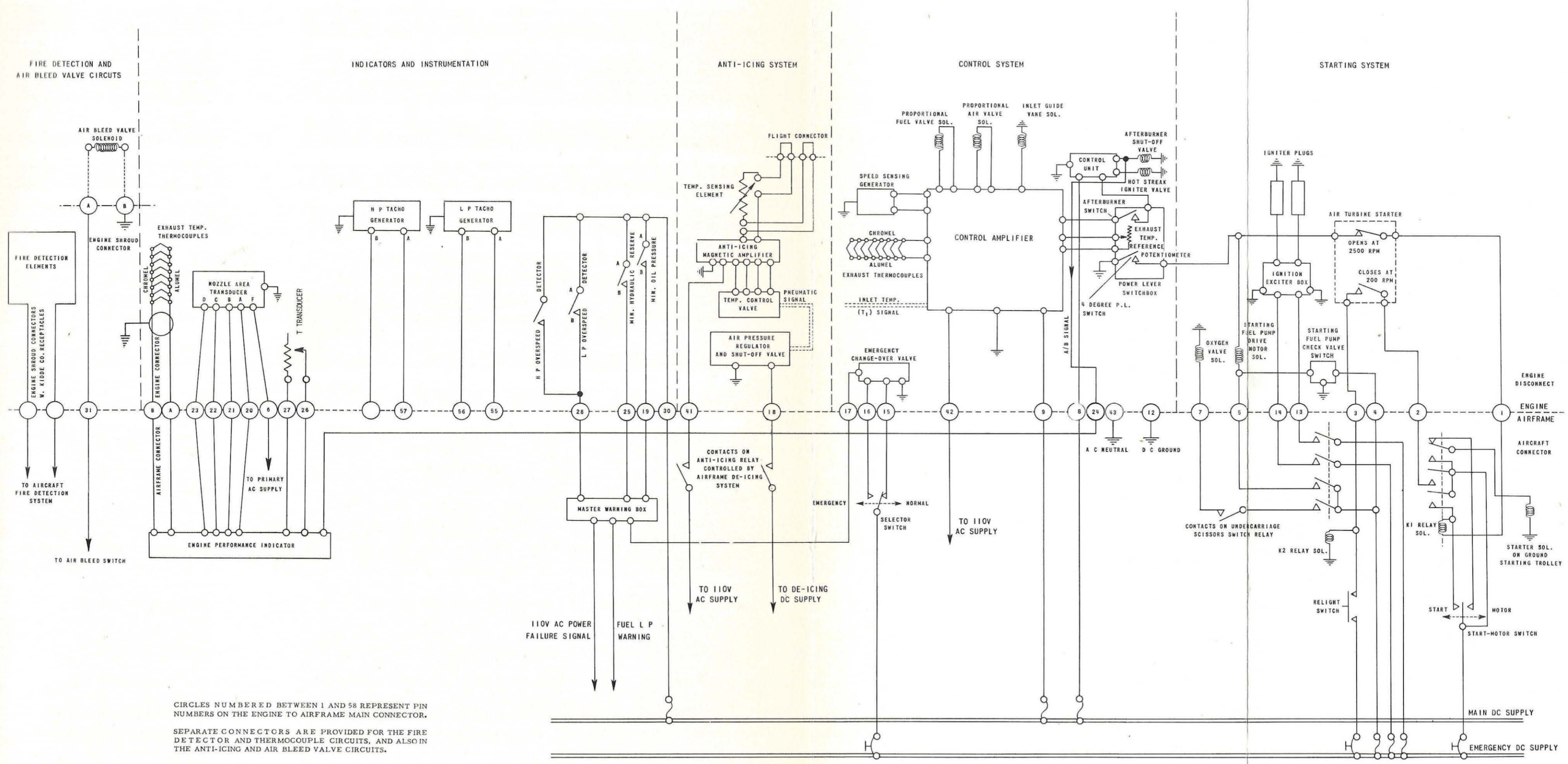


FIG. 4 **LUBRICATING OIL SYSTEM - SCHEMATIC**



CIRCLES NUMBERED BETWEEN 1 AND 58 REPRESENT PIN NUMBERS ON THE ENGINE TO AIRFRAME MAIN CONNECTOR.

SEPARATE CONNECTORS ARE PROVIDED FOR THE FIRE DETECTOR AND THERMOCOUPLE CIRCUITS, AND ALSO IN THE ANTI-ICING AND AIR BLEED VALVE CIRCUITS.

FIG. 6 ELECTRICAL SYSTEM WIRING DIAGRAM

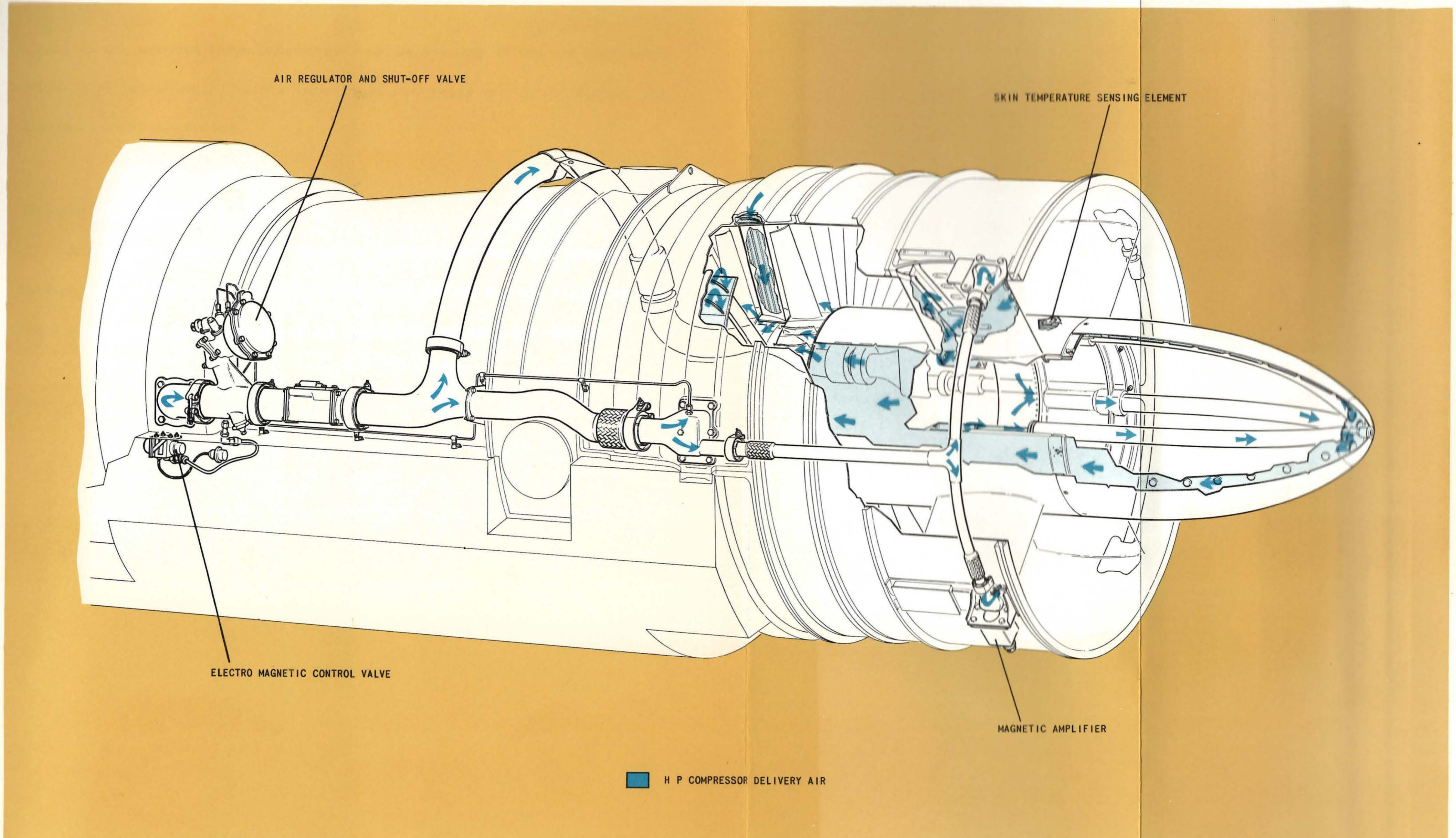


FIG. 7 ANTI-ICING SYSTEM