YL. 126-00/00

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SYLLABUS

IROQUOIS ENGINE

GENERAL MAINTENANCE TRAINING

GENERAL:

Course duration: 4 weeks (120 hours)
Class capacity: 12 students

This course of instruction is designed to enable the Gas Turbine Technician to operate, service and troubleshoot the Iroquois engine efficiently and economically under normal field operating conditions. The student is introduced to the basic elements of turbojet theory and performance. The construction and operation of the engine and its systems are thoroughly covered while servicing and troubleshooting are emphasized throughout.

More specialized training in the fuel control system is offered as a four week extension to this course for students of good standing.

SUBJECTS:		Time Allotted (hrs)
1	Orientation.	1
2	Elements of Turbojet Theory.	
2.1.	Thrust derivation: mass; inertia; momentum; reaction; diffusion; expansion.	3
2.2.	Gas flows: variation of pressures, temperatures and velocities throughout the engine.	2
2.3.	Performance: engine component and overall efficiencies; variations in fuel consumption, thrust and exhaust temperature with rpm and inlet conditions.	3
3	Engine Construction.	18
3.1.	Description of engine sub-assemblies; methods of assembly; limits of strip and assembly in the field; field checks of basic engine components.	

		Time Allotted (hrs)
4	Auxiliary Systems.	
	Construction, operation, servicing and trouble shooting as applicable to the following:	
4.1.	Air System: main flow; combustion; cooling; oil seal and sump pressurizing; aircraft services.	3
4.2.	Engine Anti-icing System.	3
4.3.	Engine Lubricating System.	3
4.4.	Engine Hydraulic System.	3
4.5.	Engine Electrics.	
4.5.1.	Electrical System, General.	1
4.5.2.	Ignition System.	1
4.5.3.	Starting Circuit.	2
4.5.4.	Electronic Control, General.	1
4.7.	Oxygen System.	1
5	Fuel Control System.	
5.1.	Introduction: basic engine control requirements; elementary principles of hydromechanical systems; general layout of Iroquois system.	3
5.2.	Turbine Fuel Pumps.	3
5.3.	Pump Pressure Control and Hydraulic Actuator.	3
5.4.	Primary Fuel System.	
5.4.1.	Main Metering Unit and P_3 Limiter.	2
5.4.2.	Potentiometer and Pressure Drop Control Circuits.	2
5.4.3.	Minimum Idle Flow Valve.	1
5.4.4.	Governors; Governor Reset Valve; Constant Pressure Valves.	4
5.4.5.	Idling Control.	1
5.4.6.	Shut-off Cock and Emergency Throttle Valve; 16 point distributor.	2
5.4.7.	Proportional Fuel Valve (Acceleration Control); P.C.L. Power Assist; Filters.	2
5.5.	Afterburner Fuel Control.	
5.5.1.	Introduction: general requirements, sequence of operation; performance.	2

			Time Allotted (hrs)
		5.5.2. Metering Unit; Potentiometer and Pressure Drop Control Circuits; A/B S.O.C.	4
	5.6.	Emergency Fuel Control.	
		5.6.1. Emergency throttle valve; governor; pump pressure control.	4
	5.7.	Ram Temperature Senser.	1
	5.8.	Engine Starting Fuel System.	3
6	Engine Operating Limitations.		3
		Sea level static ratings; fuel and oil specs, consumptions, temperatures and pressures; auxiliary equipment limitations; operation on emergency fuel system; transient and steady run exhaust temperatures; engine stability; max. rotor speeds.	
7	Field Servicing.		
	7.1.	Installation and removal; inhibiting and de-inhibiting.	4
	7.2.	Inspections: between flight, primary and periodic, supplementary and "hot end" inspections.	2
	7.3.	Ground Running: preparation for ground running; motoring and start procedures; ground running checks; start characteristics; throttle handling.	3
8	Manufacturing Processes.		3
9	Test Bed Procedures (OEL Test Bed)		2
10	Fuel System Review and Troubleshooting.		12
11	General Review and Discussion.		6
12	Test and Test Review.		3
		TOTAL HOURS	120

BASIC TURBOJET PRINCIPLES

All aircraft are propelled by jets. The conventional propeller drives a plane through the air by the reaction from the airstream which it accelerates rearward. The turbojet engine produces thrust by accelerating the gases flowing through it. Rockets are propelled by the reaction of the jet of hot gases produced by the combustion of the propellant.

One conception of jet engine thrust is that the jet stream pushes against the surrounding atmosphere and the reaction of the 'push' gives the aircraft its forward thrust. This conception is disproved if we visualize the jet issuing into a vacuum, in which case there is nothing to push against, yet the forward thrust of the engine would, in fact, be greater because of the LACK of atmospheric pressure in the exhaust area.

Large masses of air pass through the jet engine - about nine tons per minute in the case of the Iroquois. Through the processes of compression, combustion, expansion and exhaust, the engine internally accelerates the air mass rearward. To any applied force there is an equal and opposite reaction and the reaction is the forward thrust which is felt throughout the internal structure of the engine.

To assist in this conception it may be useful to review the principles of weight, mass, acceleration and momentum, and so follow through the operating processes of the turbojet engine.

MASS - INERTIA - MOMENTUM

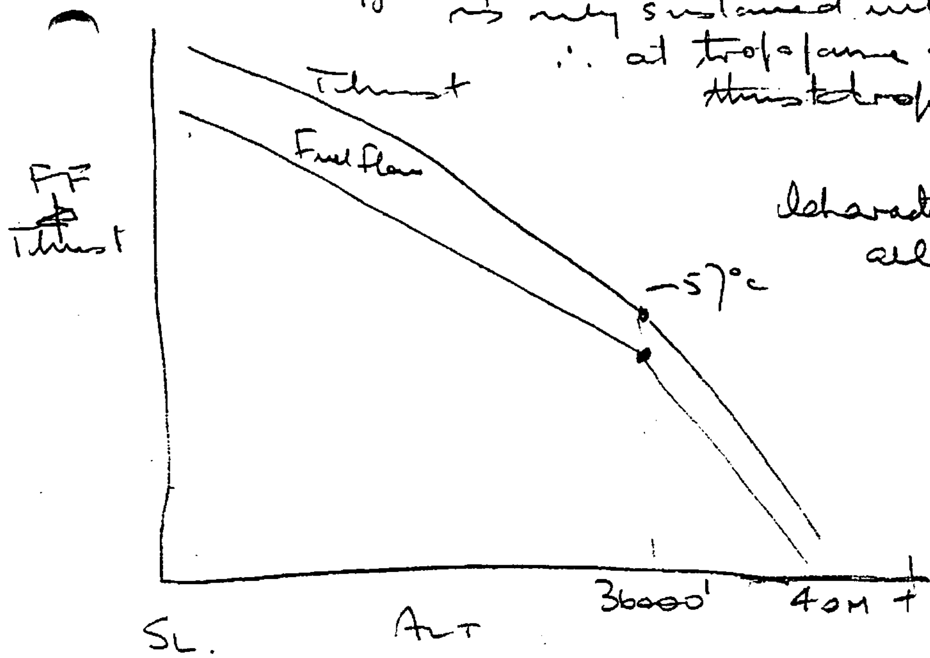
The term MASS refers to that property of matter which in everyday language is described by the word INERTIA. We know that an object at rest will never start to move by itself - a push or pull must be exerted by some other body. Similarly a force is required to speed up or slow down a body that is already in motion and a sidewise force must be exerted on a moving body to deviate it from straight line motion.

A body in motion possesses Momentum and Momentum is defined as the product of Mass (inertia) and Velocity.

$$\text{Momentum} = M \times V$$

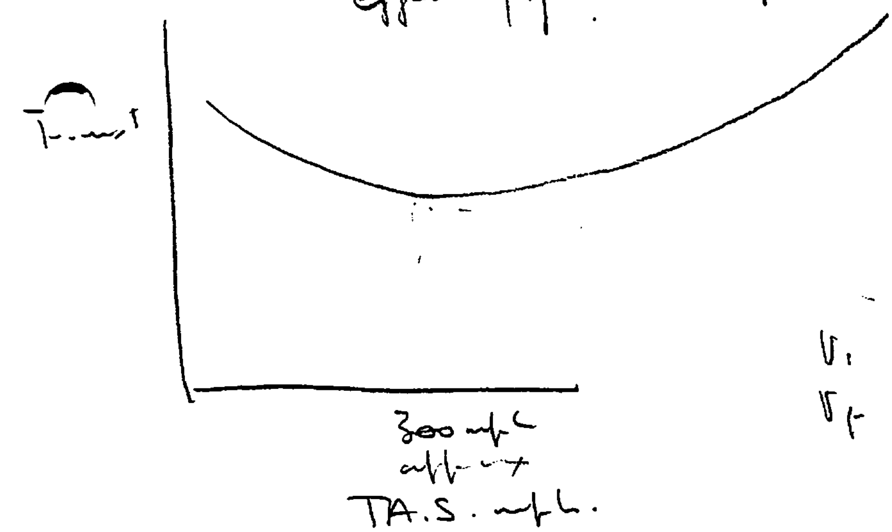
Obviously, speeding up, slowing down, or changing direction, involve a change in velocity or in other words an acceleration or deceleration. For example, if a body with an initial speed, V_1 feet per second, is speeded up to a final velocity, V_2 fps, then the change in velocity would be $(V_2 - V_1)$ fps over a certain period, or its acceleration would be $(V_2 - V_1)/t$ fps each second, where 't' is the time in seconds between the recording of V_1 to V_2.

Effect of reduced temp on thrust is only sustained whilst dropping ∴ at tropopause this ceases & thrust drops sharply.



Characteristic of all Jet engines.

Effect of forward speed



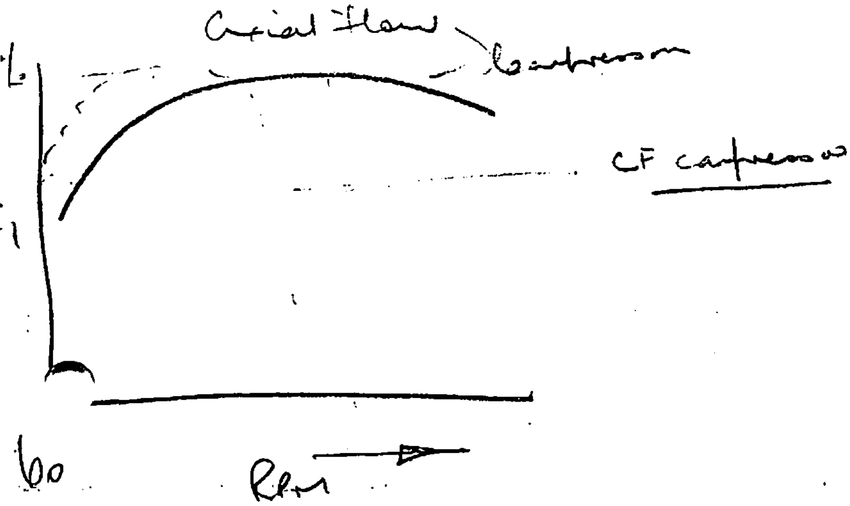
Efficiency }

Prop efficiency =

$$\frac{2V_1}{V_1 + V_f}$$

V_1 = initial velocity

V_f = final velocity



These notes are intended as a supplement to the lecture programme only and possess no authority over existing RCAF Engineering Orders or Bulletins published by Orenda Engines Limited.

The notes are arranged in sequence with the Training Schedule. The TS-1 to 9 series apply to the General Maintenance Course on the Iroquois engine and its accessory systems.

The change in momentum, therefore, would be $M (V_2 - V_1) / t$; and if we give acceleration the symbol 'a' then the change in momentum = Ma.

REACTION

Momentum is the inertial property of a body which resists a change from a state of rest, or of uniform motion in a straight line, and shows up in the form of a reacting force. Any body applying force to another body will be met with a force acting in the opposite direction. If the applied force changes the momentum of the body then the reacting force will be equal to the change in momentum of the body, or the reacting force, $F = Ma = M (V_2 - V_1) / t$.

UNITS OF MEASUREMENT

For simplification the units of measurement of force, mass, and acceleration, should be chosen so that the unit of force, imparts unit acceleration to unit mass.

In the British and American gravitational system the unit of force is the imperial standard pound, which is defined as the weight of a standard unit of force, i.e. the force equal to the force of gravity which the earth exerts on the pound.

The unit of mass in this simplified system must then be of such magnitude that when acted upon by a force of one pound, its acceleration will be one foot per second, per second. That unit of mass is called one SLUG. To clarify, consider a mass of one slug falling freely. Its acceleration will be equal to the acceleration of gravity, g, (32.2 ft per sec, per sec). By definition a unit force of one pound imparts to one slug mass an acceleration of only one foot per second, per second. Since the freely falling slug has an acceleration of g ft per sec, per sec, the force must be g times as great as the unit force, or g pounds. In other words, one slug weighs g pounds, so if we have

W = weight of the body in pounds
M = mass in slugs
g = acceleration of gravity, ft per sec, per sec

Then $M = W/g$ slugs

and $F = Ma = Wa/g = W (V_2 - V_1)/gt$ lb

AIR MASS FLOW

In the considerations so far we have been visualizing a solid body of a certain weight, W lb. A flow of air may be considered as a stream of minute bodies flowing through a duct, say, at W lb per sec, and the same considerations apply. A moving mass of air possesses inertia and momentum and if we apply a force to the air to change its momentum, i.e. speed it up, we will experience a reaction and that reaction will be numerically equal to the change in momentum of the air. We know that for a constant inlet flow in a duct the air mass that leaves the duct will be equal to the mass that enters, regardless of the internal shape, and similarly the mass flow past any station in the duct will equal that past any other station. Thus if by means we can accelerate the flow somewhere within the duct so that its exit velocity V_2, is greater than the inlet velocity V_1, then we can, by measuring the mass flow M, the inlet and exit velocities V_1 and V_2, calculate the change in momentum of the air and therefore, its reaction or thrust on the duct. As an example let us consider the Iroquois engine, without afterburner, as such a duct operating under sea level static conditions, and that we have measured the following:

Air flow, W = 320 lb per sec

Inlet velocity $V_1 = 0$, (engine is stationary)

Jet velocity $V_2 = 2000$ ft per sec

Then the thrust, $F = W (V_2 - V_1)/g = 320 (2000 - 0)/32 = 20,000$ lb

Basically, afterburning increases the final nozzle velocity V_2, by the addition of heat in the tail pipe, and from the above it is obvious that a higher V_2 will give higher thrust.

An investigation of the forces which accelerate the mass flow may be simplified by a familiarization with the common terms and factors which effect changes in the flow throughout the engine.

ENERGY

One of the most common terms used throughout the study of gas flow systems is the term 'energy' and it is worthy of some note. Energy is simply the capacity for doing work and it exists in many different forms, e.g., thermal, mechanical, electrical, and chemical. Heat is energy; compressed air possesses potential (pressure) energy; a fast moving stream or jet of air possesses kinetic (velocity, dynamic) energy due to its motion. All forms of energy are mutually convertible, for instance, charging a storage battery is essentially a conversion of electrical to chemical energy.

The pressure energy of compressed air can be converted to kinetic energy by releasing it through an orifice and then it can be recovered to pressure energy by collecting it through a duct into a receiving container. Internal temperature changes always take place during such airflow processes and these temperature changes can be used as a direct means of measuring the change of internal energy, that is the amount of work done by the air, or the amount of work done on the air. For example when air is compressed, its temperature increases - handling a bicycle pump will confirm this. That increase in temperature is a direct measure of both the increase in potential energy of the air and the work required to perform the compression. Similarly if compressed air is allowed to expand, whether against a piston or through an orifice, it will be doing work in moving the piston or pushing the air through the orifice. In the process, its temperature will drop and this again, is a direct measure of the work the air does - or the energy it expends during the expansion. Also, of course, if heat (energy) is added from an external source to a system which already possesses potential or kinetic energy then the total energy of the system is increased, e.g. combustion in a piston engine at the end of a compression stroke, or combustion in the fast moving stream of the turbojet engine, either in the flame tube or in the afterburner.

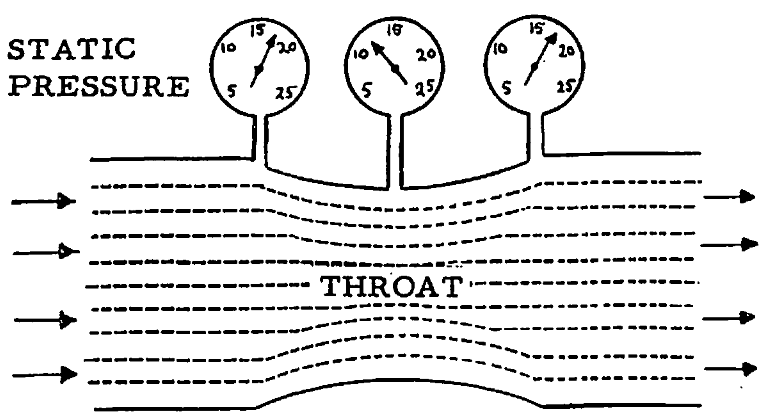
The fact that the various forms of energy are mutually convertible means that a definite relationship exists between them. This relationship is called the mechanical equivalent of heat, i.e.,

778 ft lb (Work) = 1 British Thermal Unit (BThU) (Heat)

It is often convenient to convert various forms of energy to heat units. This in fact is done throughout the design and performance calculations of jet engines.

STATIC PRESSURE

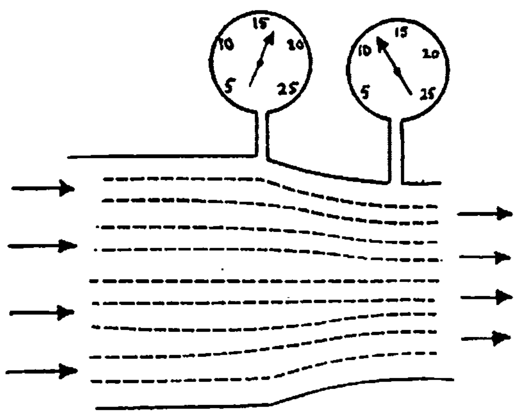
Molecules of any substance are always in a state of motion or vibration among themselves, although such is not readily apparent to us. If we visualize a cube of air for example, the walls of that cube will be under constant bombardment by the vibrating molecules of air. If the cube contains compressed air, more molecules bombard the same surface area of the cube, thus an increased pressure is felt by the walls. If the air in the cube is heated, the molecules are agitated and their increased vibration again results in a higher pressure on the walls. Static pressure is merely a measure of this molecular activity.



VENTURI

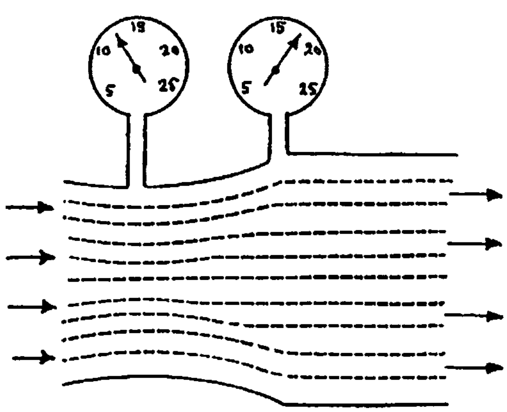
TOTAL PRESSURE

A moving stream of air will possess dynamic pressure due to its velocity as well as static pressure due to the molecular vibration. Dynamic pressure is the force you brace yourself against in the slipstream of an aircraft. For a steady flow through a duct, the sum of the two pressures, i.e., static plus dynamic, will remain unchanged throughout the duct, regardless of its internal shape. A common example of this principle is the venturi tube: while passing through the narrow section (throat) of the tube the static pressure decreases while the dynamic pressure increases, but they will balance so that the sum, static plus dynamic pressure, will remain the same as at the duct inlet. This may be explained by the fact that at the throat the molecules of air become more widely spaced, hence less static pressure, but their velocity increases in order to pass out of the tube the same amount of air that entered - thus the increase in dynamic pressure. This is known as Bernoulli's principle and it plays a major role in the design of the jet engine. The effects can be divided into two important air flow processes that take place throughout the engine, namely, expansion and diffusion.



EXPANSION

When a uniform flow in a duct passes from a larger to a smaller cross sectional area, as in the venturi, it undergoes an expansion process, and the particles become more widely spaced. The static pressure decreases with a corresponding decrease in temperature, and the velocity, hence kinetic energy, increases.



DIFFUSION

When a uniform flow in a duct passes from a smaller to a larger cross section it undergoes a diffusion process. The static pressure increases with a corresponding increase in temperature, and a decrease in velocity.

Diffusion and expansion processes are useful in converting kinetic or velocity energy into pressure energy, as we shall see in the jet engine compressor, and again converting pressure energy to high velocity energy as in the exhaust system of the jet engine.

GAS FLOWS - DESIGN FUNDAMENTALS

GENERAL

Thrust is derived from the reaction of the air mass which the engine internally accelerates rearward. In conjunction with the accompanying gas flow diagram let us consider the processes throughout the engine which provide this acceleration, and some of the design fundamentals involved. The gas flow diagram depicts an Iroquois operating under static, sea level conditions at about maximum output.

The jet engine operates on a constant pressure cycle. This means that if the engine is running, say on a test bed at constant speed, the pressure at any given point in the engine remains constant. Compression, combustion and expansion are continuous processes which take place in three separate sections of the engine instead of in a single section as in the piston engine. Since the flow is continuous, volume changes show up as velocity changes and this is an important fundamental to keep in mind while considering gas flows throughout the engine.

Velocity and pressure changes are governed by diffusion and expansion as mentioned earlier. If these changes take place without loss or gain of heat from external sources, e.g. heat conduction through walls or friction losses, then the flows undergo what are called adiabatic (theoretically ideal) processes. Obviously there will be losses because of heat conduction through duct walls, essential cooling, and friction reducing the velocity energy of the gas flows. Efficiencies of the various components are determined by comparing the acual temperature changes that take place from inlet to outlet of, say, the compressor or combustion chamber, to the temperature changes that would take place if adiabatic conditions prevailed.

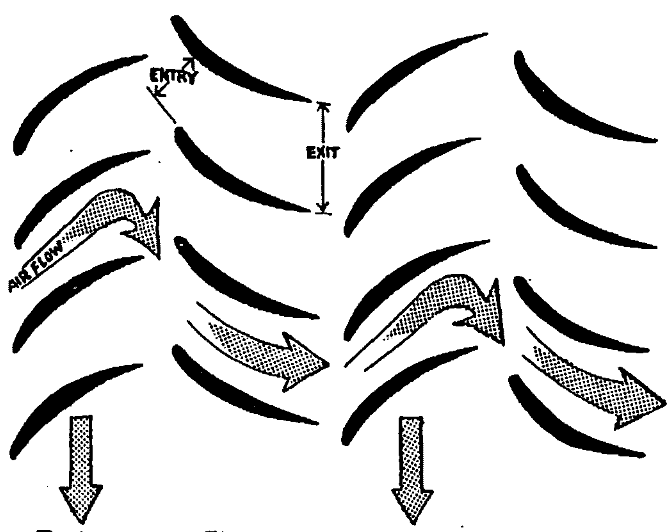


Figure 1 Compressor Blading

THE COMPRESSOR

Ambient air is induced through the intake by the rotating compressor. Each of the ten stages of the compressor comprise a row of stator (stationary) blades and a row of rotating blades. The rotor blades increase the energy of the air mass by imparting a high whirl or tangential velocity to the flow and the stators convert this velocity energy to pressure energy.

Each pair of stator blades is in effect a diffusor, i.e. the entry area is smaller than the exit area (Figure 1); as the flow passes through the stators the velocity energy imparted by the rotating blades is converted to pressure energy by the diffusion action and the flow is redirected to an axial path at a much reduced velocity.

The overall compression ratio of the Iroquois compressor is approximately 8:1 and the temperature rise accompanying each stage of compression is about 21 degrees C and 31 degrees C per stage in the LP and HP sections respectively. Thus, including ambient conditions, the compressor delivery air is at about 105 psi and 300 degrees C.

Because of the relatively slow speed of kerosene or JP4 flame front travel, it is necessary to control the velocity of the air entering the combustion chamber to within a maximum of about 110 fps, otherwise the flame might be blown away before combustion could be completed. This is catered to in the design of the compressor and by use of a diffuser ahead of the combustion inlet. It will be noted from the gas flow diagram that the compressor annulus decreases in area towards the delivery end: since the density of the air increases with compression and since the mass flow is the same at all stations, then the area of the annulus must be reduced in order to maintain compression and at the same time reduce the axial flow velocity. Thus as indicated on the gas flow diagram, the axial flow velocity is reduced from 600 fps at the engine inlet to about 100 fps at the diffuser outlet.

The compressor blades are designed with a twist from root to tip to provide a radial air pressure distribution that will balance the centrifugal effect that the rotor imparts to the air particles.

COMPRESSOR STALL

At this point it might be advantageous to outline the basic mechanics of compressor stall, or surge, as it is sometimes called - a phenomenon that is not confined to any one particular make or type of engine but may be experienced in any turbojet engine if the conditions are right.

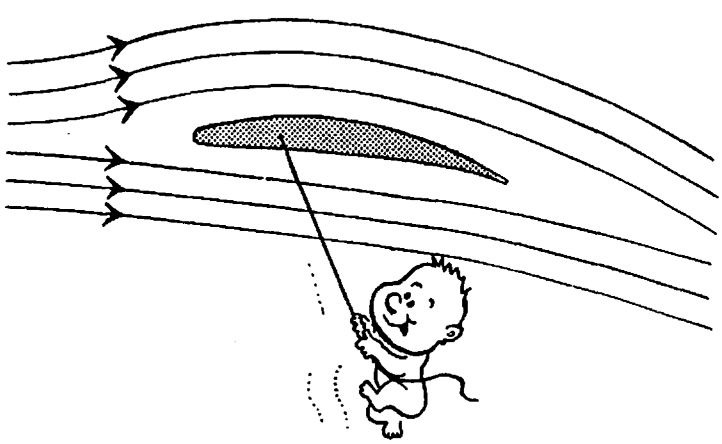


Figure 2

Stalling of the compressor is said to occur when an abrupt decrease or severe fluctuation of the compressor delivery pressure takes place. When this condition arises the compressor blades cease to impel the air uniformly into the combustion system. The severity will depend on whether the stall involves only a portion or the entire compressor.

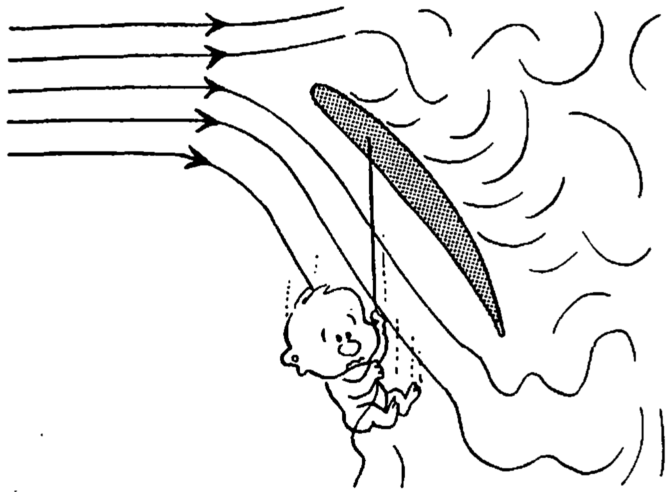


Figure 3

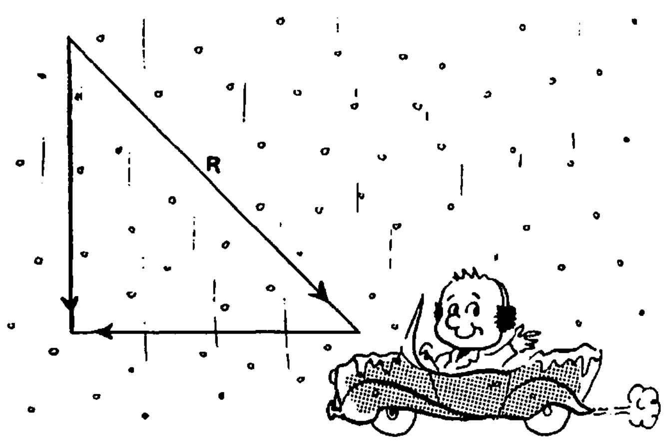


Figure 4

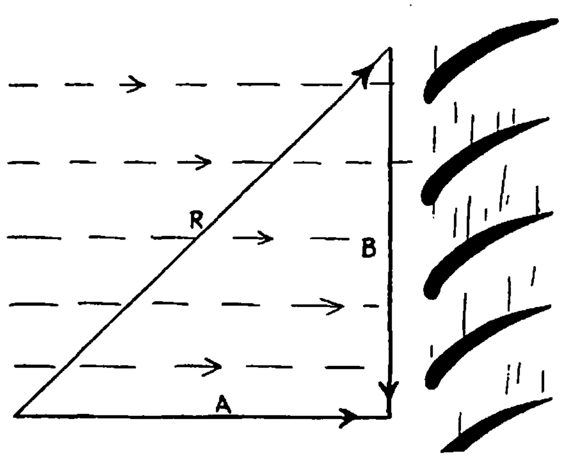


Figure 5

Incipient or partial stall may produce roughness with or without rumble, drone or similar noises. More pronounced stalls may produce explosive noises of varying intensity. Extremely severe stalls may produce violent pulsations accompanied by flame, vapour or smoke at the exhaust or even at the inlet.

Stalls are not normally injurious to the compressor itself. The main source of danger is in the excessive temperature that can be produced in the combustion and to turbine assemblies of the engine during a stall.

The airfoil shaped blades of an axial compressor are subject to the same stall characteristics as an aircraft wing. As long as the angle of attack of the airfoil to the approaching air is within certain limits, the air flow over the wing will be smooth and lift will be produced (Figure 2). However, if the angle of attack exceeds the maximum or critical value, the flow will separate from the upper surface, become turbulent and result in partial or complete loss of lift (Figure 3).

In a compressor the angle of attack is a little more difficult to visualize because it is a combination or resultant of the axial direction of air flow and the direction of travel of the blades. The effect is very similar to watching the snow fall while sitting in a moving car, Figure 4; although the snow is falling vertically, the motion of the car creates the impression that the snow is approaching the car at an angle. The direction of approach, R, is the resultant direction of the snow relative to the car. By rotating Figure 4, 90 degrees anti-clockwise and superimposing the conditions existing in the compressor, we can picture the effective angle of attack of the air onto the blades. In Figure 5, vector A represents the axial air flow through the compressor, its length, to scale, is a measure of the air velocity, similarly vector B represents the direction of rotation and velocity of the compressor blade. The third vector R, then, which completes the triangle, is the resultant direction of the air flow relative to the blade and the angle it makes with the chord of the blade is the resultant or effective angle of attack. As already mentioned, the length of each vector to the chosen scale, represents the velocity of the various components of the triangle. Thus it can be seen that a reduction in the axial air velocity (shorter vector A) while maintaining constant rpm (blade velocity, B) will result in an increased angle of attack.

In other words, at a given rpm, any condition which results in a reduction of velocity of air flow through the compressor can if severe enough lead to compressor stall.

An example of such a condition is the effect of excessive over-fuelling. In this case excessive pressures are created in the combustion chamber, the turbine chokes, and a high back pressure or 'Throttling' effect results in a reduction of the air flow through the compressor. Another example would be blockage of the inlet by flying the aircraft in abnormal attitudes, e.g. severe side slipping.

The various stages of a 'single spool' multi-stage compressor are so matched that each will operate in conjunction with the others at maximum efficiency at a chosen set of rpm and inlet conditions, i.e. the selected design conditions. During 'off-design' operation, individual stages will operate at varying degrees of efficiency and the overall efficiency will drop. For instance, at low rpm the front stages may be unstable or approaching stall while the rear stages may be in a partially choked or 'turbining' condition. Since the design speed is usually in the maximum output range of the engine, then the compressor becomes more susceptible to instability and stall in the lower speed range, particularly during acceleration.

Ever increasing requirements for higher aircraft speeds mean that the operating range of the engine must be greatly increased. Several means are used in modern engines towards this end, among them are variable stators or guide vanes to control the mass flow to suit various inlet-rpm conditions, blow-off valves at the delivery end of the compressor to relieve excessive pressures and prevent choking or turbining conditions, 'split-spool' compressors which reduce the number of stages that must be matched and allow sections of the compressor to operate independently and therefore nearer their peak efficiency over a wider rpm range.

The Iroquois incorporates a two-spool compressor and variable guide vanes at the inlet to the HP compressor. Further engine development may incorporate additional variable inlet vanes in front of the LP compressor.

COMBUSTION

Combustion is a continuous process which, during steady state conditions, takes place at constant pressure. This follows from the fact that the gases are allowed to expand freely rearward, i.e. volume changes show up as velocity changes. Thus all the energy that has been added by compression and combustion will go into increasing the velocity of the air mass and, as we have seen, the greater the velocity increase, the greater will be the change in momentum and therefore thrust.

About 20 per cent of the total air flow delivered by the compressor is metered through the snout of the combustion base plate assembly into the primary zone of the combustion chamber for mixing and burning with the fuel. The remaining 80 per cent cools the combustion liners and outer casings and by means of the dilution holes in the liner, mixes with the hot products of combustion to give a uniform and reduced temperature at the turbine. The flame temperature may be in the vicinity of 2000 degrees C, but with dilution the overall mixture temperature is brought to about 1000 degrees C at the turbine entry.

The combustion base plate acts as a flame holder by creating a wake or recirculation area for the fuel/air mixture. Air passes through the base plate via secondary tubes which are designed to provide the correct air/fuel ratio (about 15:1) and at the same time direct the air so that, in mixing with the fuel vapour issuing upstream from the 'walking sticks', a vortex is created which retains the mixture in the flame area until combustion is completed.

TURBINE

The turbine performs a function which in effect is the reverse of the compressor. Acting in a sense like a windmill it makes use of some of the energy of the gas stream by converting it to mechanical power to drive the compressor. In the case of the Iroquois engine, the turbine provides a total of about 50,000 HP to drive the compressor at maximum rpm, at sea level static conditions.

The nozzle guide vanes (or turbine stators) are inclined to the flow to give a whirl to the gas stream in the direction of turbine rotation. In addition, the passages between the stators converge to form nozzles (Figure 6) which accelerate the flow before impingement on the rotor blades.

The turbine rotor blades take the whirl out of the flow that has been imparted by the stators and deflect it to a nearly axial direction. In so doing, the turbine rotates with a torque that is proportional to the change in whirl velocity (momentum) of the flow as it passes through the rotor blades. The turbine blades, like the stators, also form expansion nozzles to accelerate the flow passing through them and this results in an additional reaction on the rotor which adds to the torque.

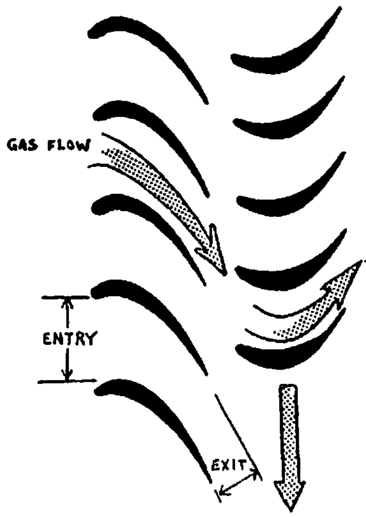


Figure 6 Turbine Blading

Since the gases must have sufficient energy to provide a propulsive jet through the exhaust system, the energy extracted by the turbine is only about one third of the total stream energy - in other words, the expansion ratio across the turbine is only a portion of the compression ratio across the compressor. The power extracted by the turbine is represented by the drop in pressure from approximately 100 to 22 psig and the accompanying temperature drop from 1000 to about 745 degrees C as shown on the gas flow diagram.

EXHAUST PIPE AND FINAL NOZZLE

For simplicity let us first consider the basic exhaust duct and final nozzle, i.e. without afterburner.

Any whirl left in the flow by the turbine is straightened to an axial direction by the aft frame vanes, this stabilizes the flow and improves the net axial thrust. Also, to build up pressure in the pipe to a value which will give maximum final nozzle velocity and at the same time reduce axial velocities and therefore losses due to skin friction within the pipe, the turbine exit area of the duct is designed with a diffusing (divergent) passage.

The final nozzle of the engine is the mechanism for converting pressure energy into velocity or thrust energy - in other words, it is the heart of the problem in so far as accelerating the flow from engine intake to exhaust is concerned. Once the type or contour of the nozzle is fixed, then the area is the determining factor. Too large an area will result in excessive pressure drop across the turbine, low jet velocities and hence low thrust, while too small an area will result in back pressure, excessive temperatures and compressor instability.

For a given set of engine operating conditions the final nozzle velocity and therefore thrust will increase with decreasing final nozzle area until sonic velocity is reached at the nozzle throat. Since sonic velocity cannot be exceeded at the throat, then for the conventional convergent nozzle, this becomes the limiting point, which is known as the 'choked' condition, wherein the mass flow, the final nozzle velocity and therefore the thrust, have reached maximum. The choked condition will be attained when the ratio of the pressure in the pipe to ambient pressure (P_7/Pa), reaches a certain value and this value is called the critical pressure ratio.

The speed of sound in a gas increases with the temperature of the gas, therefore if the exhaust temperature can be increased, then the final nozzle velocity can be increased before choking occurs. In other words, for maximum thrust, the final nozzle area should be adjusted to give the highest exhaust pipe temperature acceptable to safe engine operation.

We have noted the changes in pressures and velocities that take place according to Bernouilli's principle for subsonic flows, through venturis, nozzles and diffusers. However, when the flow becomes supersonic these changes reverse, i.e., a converging section of the duct will decrease the velocity (which invariably becomes Mach 1 at the throat) and a divergent passage will increase the velocity. Thus when the pressure available at the final nozzle is sufficient to produce a supersonic jet, or when P_7/Pa is above critical, then a convergent-divergent nozzle should be used to obtain maximum acceleration of the jet, hence maximum thrust. A fixed nozzle of this type however, would be efficient within only a narrow band of operating conditions and would become a disadvantage outside of this band.

Additionally, with a fixed nozzle configuration the only method by which the pilot can vary the thrust is by varying fuel/air ratio, i.e. engine rpm. This means that off-design speeds will be used at the expense of efficiency.

The Iroquois incorporates a variable area nozzle which allows the engine to operate at best efficiencies over a wide range of engine rpm and inlet conditions as well as providing for cooler starts, faster accelerations and improved throttle response. Also at high flight speeds (with ram pressure) the nozzle pressure is sufficient to produce a supersonic jet, the nacelle cooling air ejector functions as an aerodynamic, convergent-divergent nozzle to improve the acceleration of the exhaust gases.

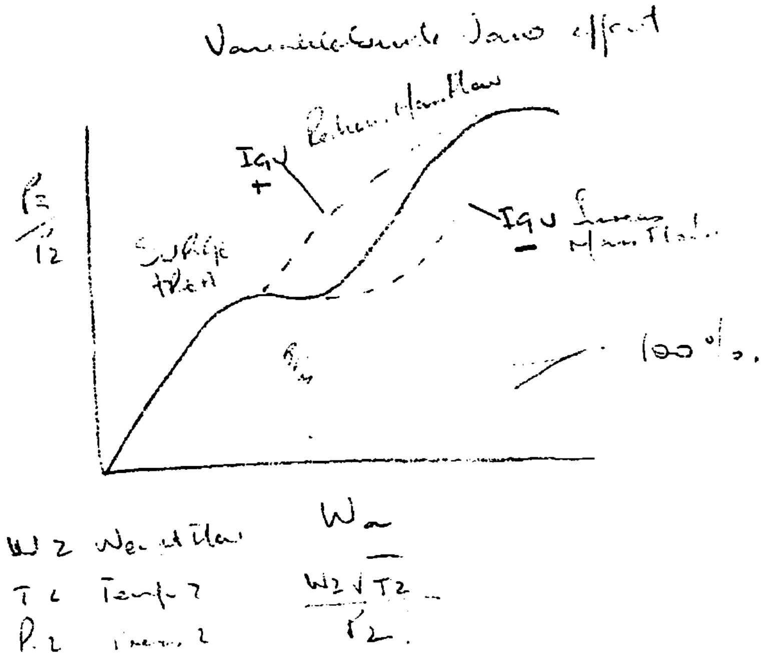
AFTERBURNER

Afterburning is a method of thrust augmentation which consists of introducing and burning fuel in the exhaust gases as they pass from the turbine to the final nozzle of the engine. The increased temperature results in greater expansion of the gases, a higher jet velocity and hence increased thrust.

Since only about 20 per cent of the total air delivered by the compressor is utilized for burning in the combustion chamber, a considerable amount of unburnt oxygen is still available for further combustion in the jet pipe. However, because the temperature of the afterburner flame may be in excess of 2000 degrees C, the combustion process must be concentrated around the axis of the pipe. This results in only a portion of the unburnt air being used in afterburner combustion, the remainder flows along the walls of the pipe to maintain skin temperatures within safe material limits.

Since the flame propagation speed of kerosene, or JP-4, at normal mixture ratios is only a few feet per second, any flame lit in the free air stream of the jet pipe would be blown away. Therefore, downstream of the afterburner fuel injection, a form of flame stabilizer or 'flame holder' is necessary to provide a wake or region of recirculation, whereby hot combustion gases can be fed back to ignite the fresh fuel-gas mixture in a continuous process. In addition, the lower the Mach number and the higher the static pressure of the gas stream in the afterburner region, the higher will be the reheat efficiency, hence the turbine outlet flow must be diffused for afterburning. The amount of thrust boost gained depends largely on the amount of diffusion and this in turn is limited by the maximum specified diameter of the pipe. The Iroquois afterburner provides about 25 per cent thrust increase.

The gas flow diagram indicates the changes in temperatures, pressures, and velocities through the exhaust pipe and final nozzle during both 'dry engine' and afterburner operation. The final nozzle is wide open during engine starts and accelerations. In the normal operating ranges the nozzle area is automatically controlled to maintain a constant pressure drop across the turbine for both afterburner and non-afterburner conditions, the main differences between the two conditions being: the larger final nozzle area during afterburning to accommodate the increased expansion, the higher exhaust temperature because of afterburning, and the higher final nozzle velocity created by afterburning.



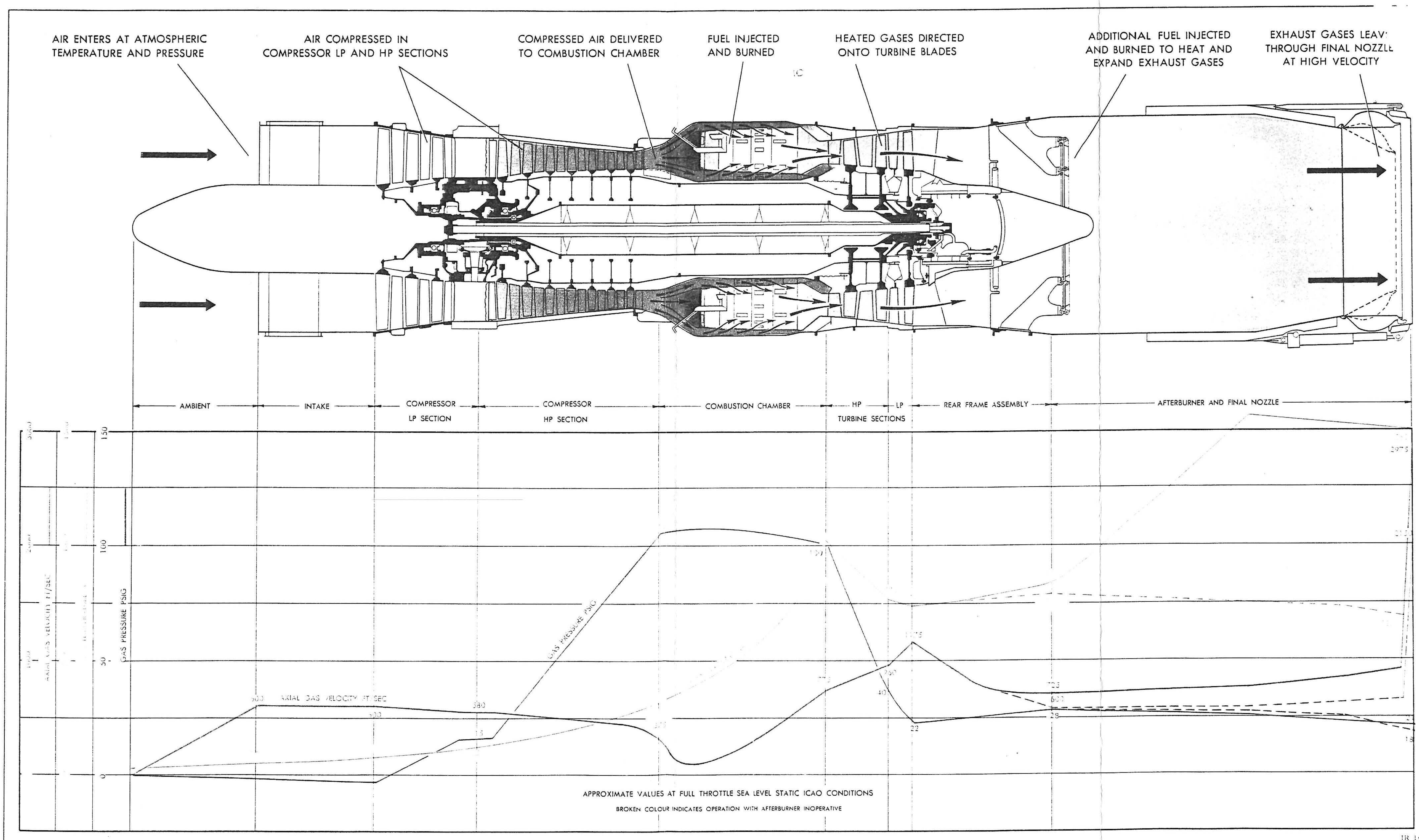
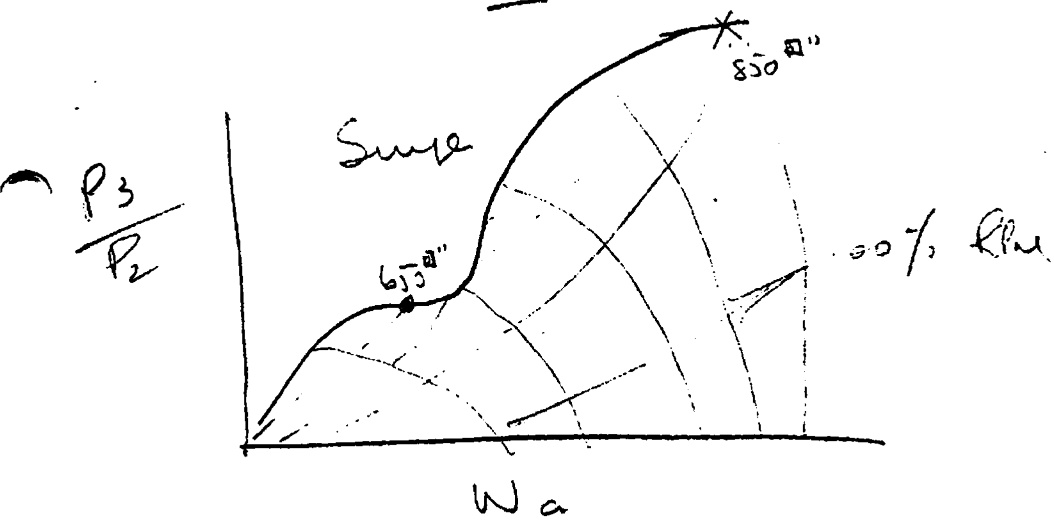


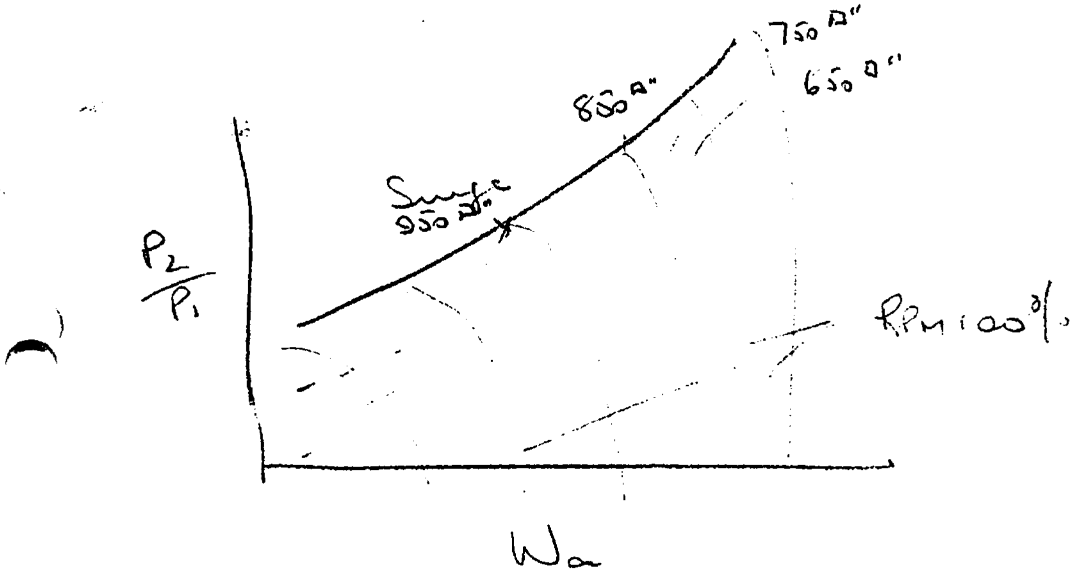
Figure 1-1-7 Gas Flow Diagram

HP [illegible]

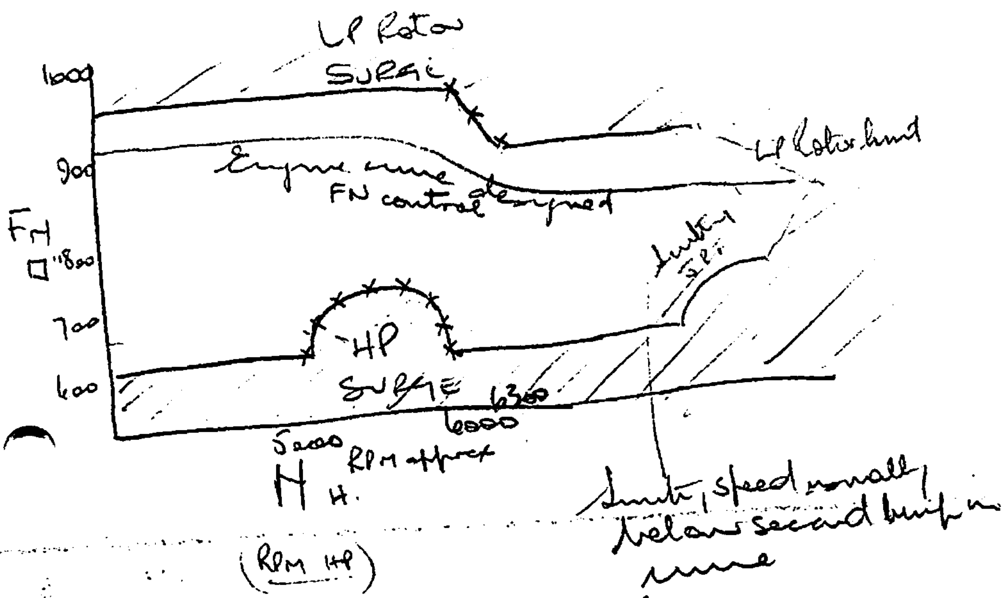


Surge
850
650
100% RPM
P_3/P_2
Wa

LP [illegible]



750
650
850
Surge
950
RPM 100%
P_2/P_1
Wa



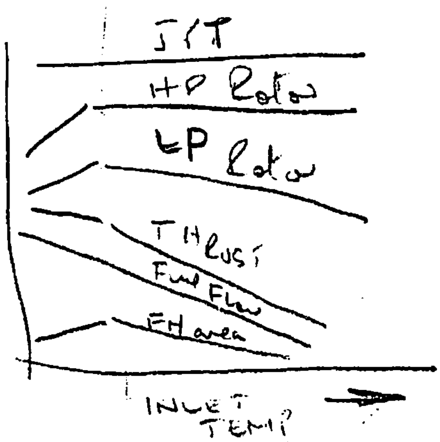
LP Rotor
SURGE
LP Rotor limit
Engine curve
FN control designed
1000
900
800
700
600
FN
HP
SURGE
6300
6000
5000
RPM approx
H
(RPM HP)

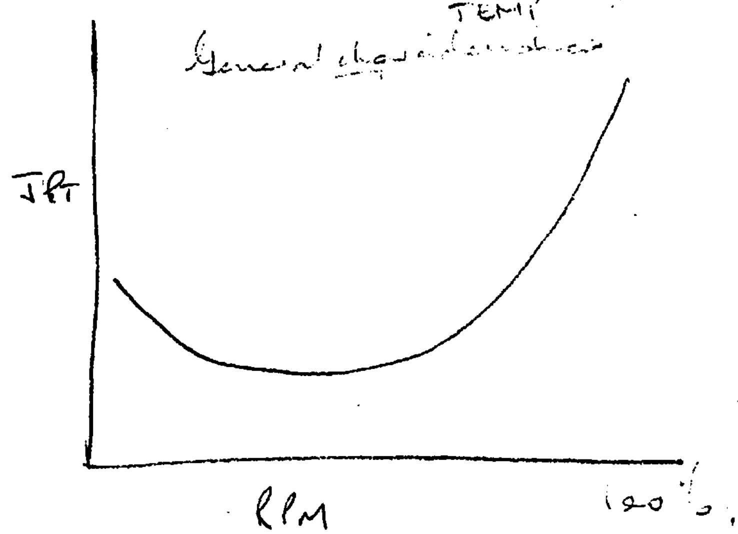
IROQUOIS

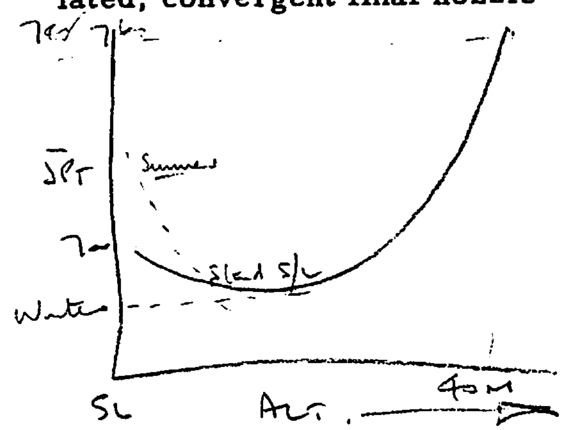
LEADING PARTICULARS

GENERAL	
Name of Engine	Iroquois 2
Type of Engine	Two-spool, axial flow, turbojet with integral afterburner
Dry Weight	4500 lb (maximum)
Direction of Rotation	Anti-clockwise as viewed from the rear of the engine
Compressor	
(LP Section)	3-stage, axial flow
(HP Section)	7-stage, axial flow
Combustion	Annular vaporizing type chamber
Turbine	
(HP Section)	2-stage, axial flow
(LP Section)	Single-stage, axial flow
Afterburner	Integral design with fully modulated, convergent final nozzle

TRIPLE POINT







Reason for rise ... not aerodynamic but effect of constant requirement in HP of accessories must be considered.

Thrust HP THP $= \frac{T \times V}{375}$

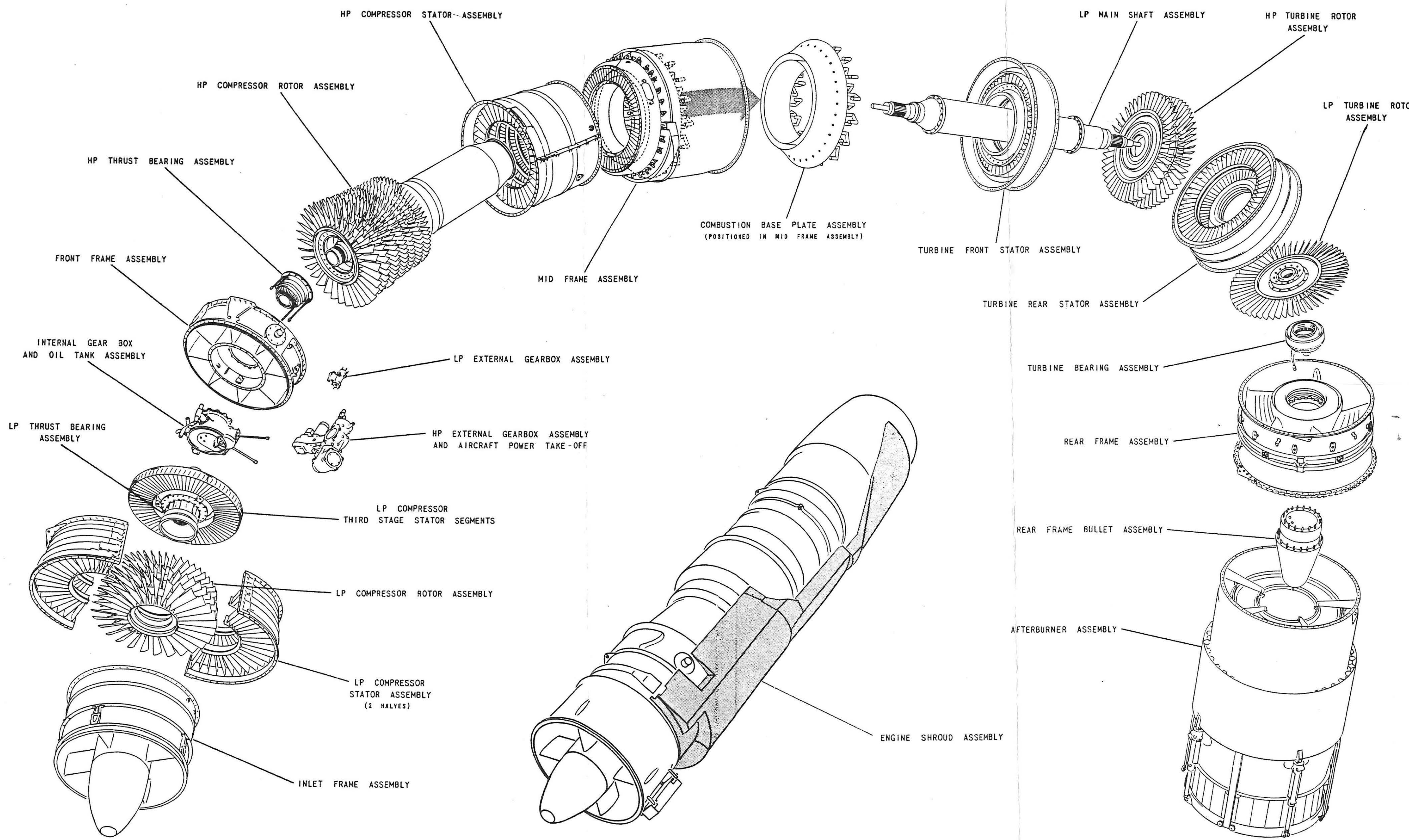


Figure 2-1 Exploded View of Engine

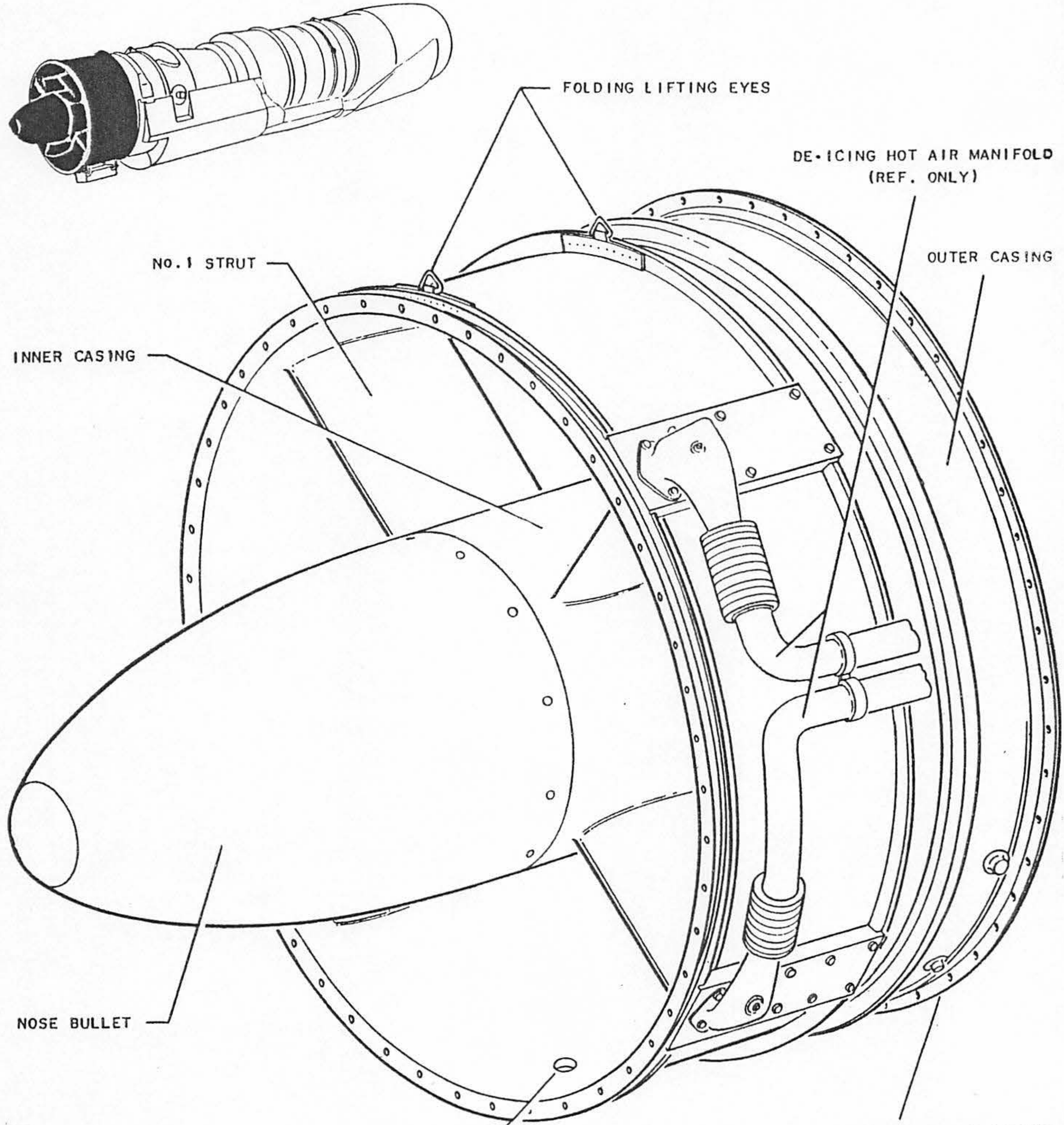


Figure 2-2 Inlet Frame Assembly

IROQUOIS

GENERAL ARRANGEMENT AND CONSTRUCTION

In this section of the notes the basic construction of the Iroquois is described under separate headings which represent its major assembly and sub-assembly sections. Ancillary systems and equipment involved in the control and operation of the engine are described in other individual sections of the notes.

GENERAL

The engine comprises essentially a two-spool compressor, an annular type combustion chamber, an integral afterburner - exhaust system with variable final nozzle and two main support frames. The front frame, interposed between the two compressor sections accommodates the main thrust bearings and front engine mounts. The rear frame, between the turbine and exhaust pipe, houses the turbine bearings and provides the engine rear mounting attachments.

INLET FRAME

The inlet frame assembly is a fabricated structure consisting of a cylindrical outer casing, four aerofoil struts, a cylindrical inner casing, and a nose bullet. The main function of this assembly is to house and support an alternator/constant speed unit combination which is mounted within the inner casing. The drive for these units is provided by suitable shafting extending forward from the LP compressor rotor.

The outer casing is fabricated from aluminum sheet with a machined aluminum flange ring at the rear, which serves as the mounting attachment to the forward flange of the LP compressor stator casing. Two folding lifting eyes are provided at the top of the casing, and at the bottom provision is made for the inclusion of the ram temperature sensing probe. A dual pressure probe is fitted on the outer casing, just below No. 2 strut for obtaining the P_1 total pressure and the P_1 reverse pressure; these probes are connected into the fuel system.

The inner casing consists of a magnesium casting shrouded by aluminum sheet. Four radial, hollow, equi-spaced struts locate the inner casing in the centre of the outer casing. The struts are numbered anti-clockwise, No. 1 strut being at approximately the ten o'clock position when viewed from the front of the engine. A temperature sensitive thermistor is fitted to the outer skin of the inner casing between No. 1 and 4 struts, to sense icing conditions at the engine intake.

The nose bullet, fitted to the front of the inner casing, the four struts, and the

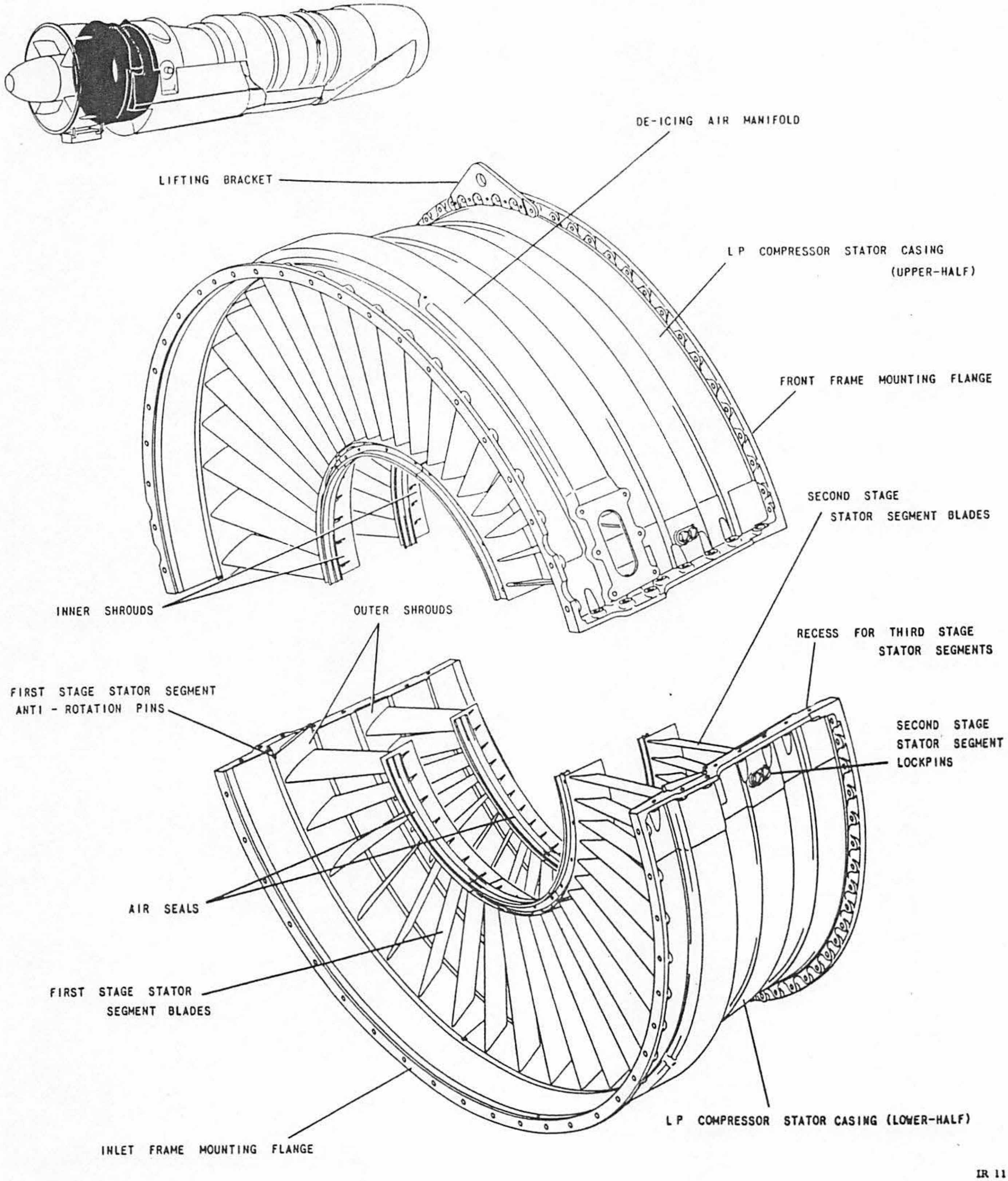


Figure 2-3 Low Pressure Compressor Stator Assembly

inner casing are all double skinned, thus providing passage for a supply of compressor delivery air for anti-icing purposes. The anti-icing system is explained in a separate section of the notes.

The hollow struts also carry electrical and thermocouple leads, the oil supply, return, and vent pipes for the aircraft constant speed unit, together with other aircraft services.

LOW PRESSURE COMPRESSOR STATOR ASSEMBLY

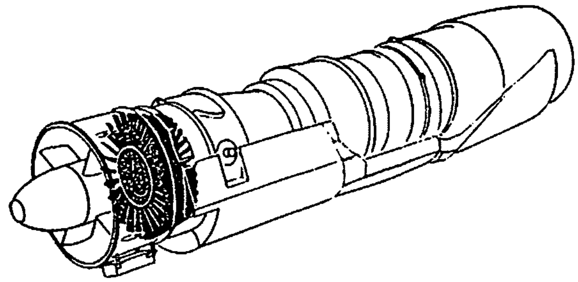
The LP compressor stator assembly is a horizontally split, magnesium casing containing three stages of stator blades. Recesses and 'T' slots are machined round the inside diameter, the front recess being designed to provide a manifold for the distribution of compressor delivery air for blade anti-icing purposes. A coating of friable material is applied to the casing inner surfaces opposite the tips of the compressor rotor blades to minimize damage which may occur should a rotor blade tip rub develop. A machined flange at each end of the casing serves as the mounting attachment to the inlet frame and front frame respectively. Nuts, and bolts with externally splined heads, are used at the split line to secure the two halves of the casing. A lifting bracket is secured with the casing joint bolts and by shear pins interposed between the bolts.

The stator blades are manufactured from stainless steel sheet, and have stainless steel corrugated stiffeners through each blade core. Inner and outer shrouds of stainless steel are brazed to the blades to form stator segments. Each stage has six segments, the first, second and third stage segments having seven, ten and twelve blades respectively.

The first and second stage stator segments are located in the 'T' slots and are prevented from rotating by anti-rotational pins and lockpins respectively. The third stage segments are mounted on the LP thrust bearing housing outer flange, and are retained and locked against the forward outer and inner flange faces of the front frame by pins and bolts when the LP thrust bearing assembly is installed on the front frame. When the front frame is attached to the stator casing the segments fit into the rear recess in the stator casing. Stainless steel air seals are riveted to the inner shrouds of the first and second stage stator segments and mate with seal sleeves on the LP compressor rotor spacers.

LOW PRESSURE COMPRESSOR ROTOR ASSEMBLY

The low pressure compressor rotor is composed of three stages of blades, three discs, two spacers, and two peripheral seals, all of which are machined from titanium. The driving flange of the first stage disc is remotely positioned to the rear of the disc hub. The remaining stages are cantilevered in a rearward direction, with the spacer rings bolted directly to the discs to form a continuous drum.



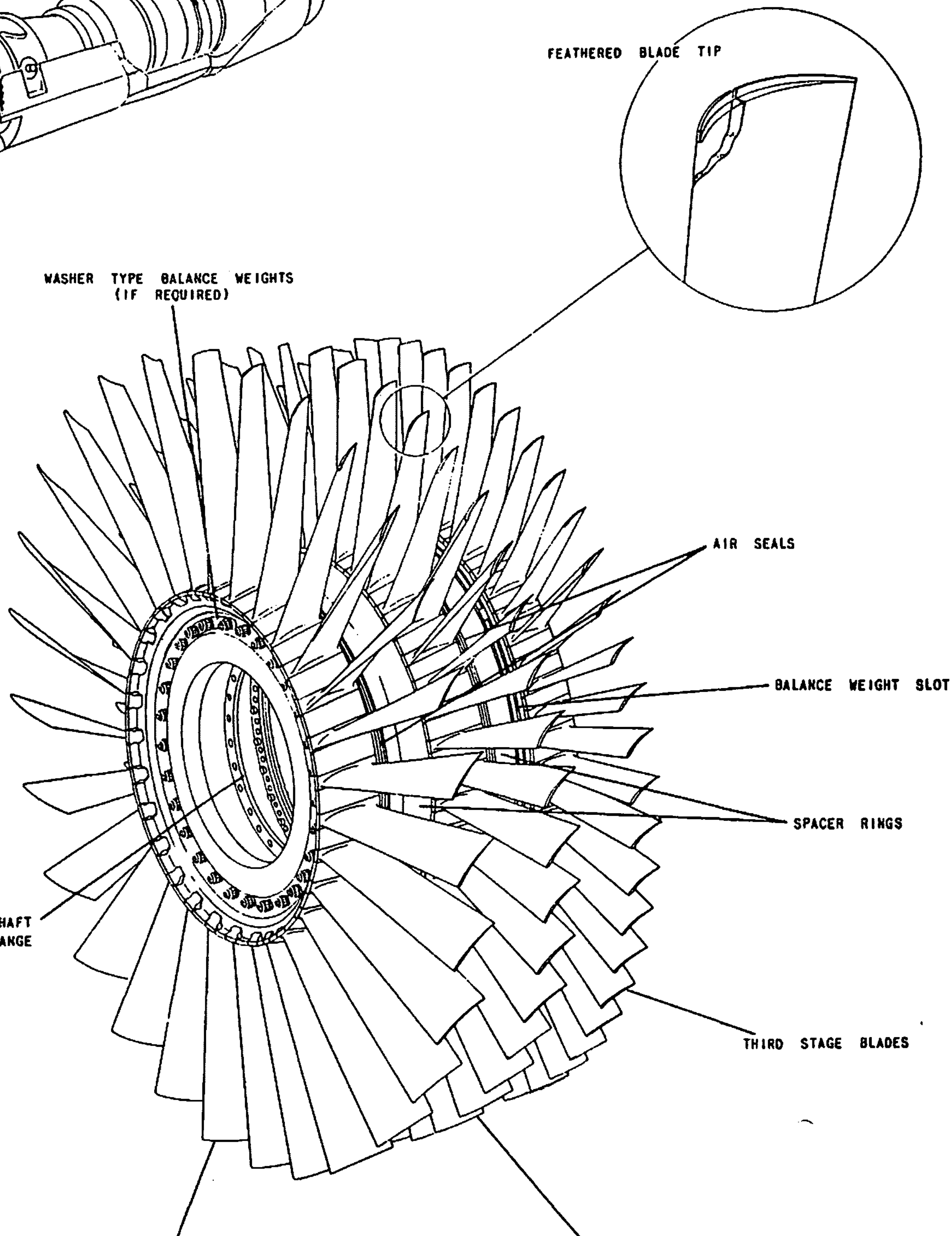


Figure 2-4 Low Pressure Compressor Rotor Assembly

A fourteen groove, labyrinth type, peripheral seal is riveted in position in a machined groove in the spacer rings immediately behind the first and second stage blade roots. The mating parts of the seals are fitted to the inner shrouds of the compressor first and second stage stators respectively. Immediately in front of the third stage rotor blades, a machined slot is provided in the spacer ring for the addition of balance weights which may be required during dynamic balancing. At the forward face of the first stage disc, additional weights in the form of strip washers may be added under the disc securing nuts to attain the correct balance.

The rotor blades are machined from forgings, all three stages having feathered edges on their outer extremities to reduce the possibility of damage to the stator casing if a tip rub should develop. The first and third stage blades are secured to the disc by a two branch fir tree root while the second stage employs a cylindrical seat dovetail fixing. On the rear face of the blades a positive stop is machined into the blade roots to prevent the blades from being knocked loose should a failure occur in a preceding stage.

LOW PRESSURE THRUST BEARING ASSEMBLY

The LP thrust bearing assembly supports the LP compressor shaft and absorbs the net axial thrust of the LP compressor rotor and LP turbine rotor. The assembly consists of the LP compressor shaft, a bearing housing, a ball thrust bearing, and the LP compressor third stage stator segments. The shaft is balanced by the insertion of balance weights in the two machined grooves on the shaft.

The LP compressor shaft is of nickel-plated steel. Internal splines at the front and rear of the shaft provide the drive to an accessories gearbox mounted in the inlet frame, and transmit the torque from the LP mainshaft to the LP compressor rotor respectively. The shaft is bolted to the driving flange of the LP compressor rotor first stage disc at engine assembly. A seat and shoulder on the external diameter of the shaft, and located immediately forward of the shaft mid-point, accommodates the LP ball thrust bearing.

The bearing housing is of magnesium and is mounted on the front inner flange of the front frame. The LP compressor third stage stator segments are mounted on the outer periphery of the housing and are retained by the housing attachment bolts passing through the inner shrouds which are flanged inwards. Pins on the front outer flange of the front frame position the outer shrouds of the segments when the bearing assembly is installed on the front frame. Three hydraulically operated actuators in the bearing housing are mechanically connected to the variable HP compressor inlet guide vanes. A transfer ring on the front inner face of the bearing housing collects and directs hydraulic fluid into internal passages in the bearing housing to supply the actuators.

The bearing is a single row, ball thrust bearing with a split inner track. It is

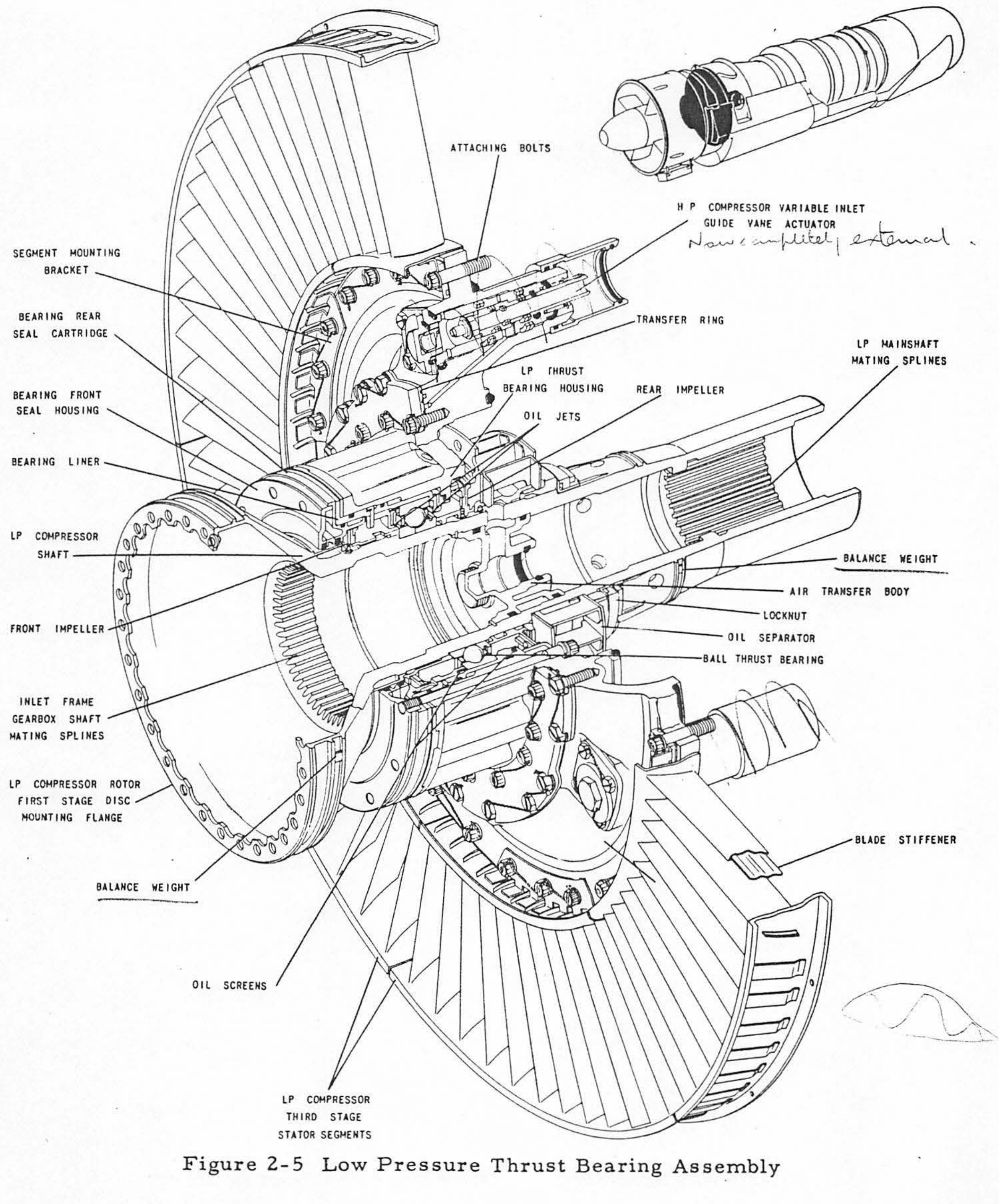


Figure 2-5 Low Pressure Thrust Bearing Assembly

installed in a steel liner in the bearing housing and is held in position by a locknut. An oil screen is located between the bearing locknut and the front face of the bearing outer race. A second oil screen is located at the rear of the bearing, between the steel liner and the housing. The effects of thermal differential expansion between the housing magnesium casting and the steel liner is counteracted by a locating ring with lugs on its rear face, the ring being located in slots on the front inner face of the bearing housing and secured to the steel liner, thus centralizing the liner.

An air transfer body of cast magnesium is fitted inside the LP compressor shaft and shrouds the forward end of the LP mainshaft to permit an exchange of oil and air between the internal gearbox and the rear oil sump. Aligned holes in the air transfer body and the LP compressor shaft provide passage to an oil separator for the oil and air mixture delivered by the oil return tube of the LP mainshaft.

Forward of the bearing a seal sleeve is fitted to the LP compressor shaft, the outer diameter of the sleeve forming a seat for the seal. A double floating ring, pressure balance type seal with a spacer ring between the seals, is mounted between the seal front housing and the seal rear cartridge. Compressor seventh stage air is led through drilled passages to pressurize the cavity between the seals.

Two radially vaned impellers, one on either side of the bearing, return oil to the internal oil tank in the front frame via drillings in the bearing housing. The front impeller is shrouded by a rearward extended flange on the rear seal cartridge, and the rear impeller rotates within an annulus ring fitted to the rear of the bearing housing and in front of the oil separator. A spacer ring and the seal front housing are also fitted forward of the bearing, and these, together with the seal rear cartridge, are located around the circumference of the bearing seal sleeve.

FRONT FRAME ASSEMBLY

The front frame assembly is the main structural number of the engine and consists basically of magnesium casing, a set of variable incidence HP compressor inlet guide vanes, and hollow struts which are used to carry engine services. The assembly forms the structural connection between the LP and HP compressors and supports the internal gearbox and oil tank assembly, and the LP and HP thrust bearings; the loads of these bearings are transmitted to the airframe via trunnion mounts on the front frame casing.

The front frame casing is cast in the form of a cylindrical outer casing joined to an inner casing by eight integrally casting struts. The outer casing is provided with front and rear mounting flanges for attachment of the LP and HP compressor stator casings respectively. A mounting is fitted to a pad machined at the top of the outer casing and transmits the net axial thrust load and tangential side loads from the engine to the airframe. Trunnion mounting pads fitted to the outer ends of the horizontal struts transmit tangential vertical loads only. Dependent upon aircraft

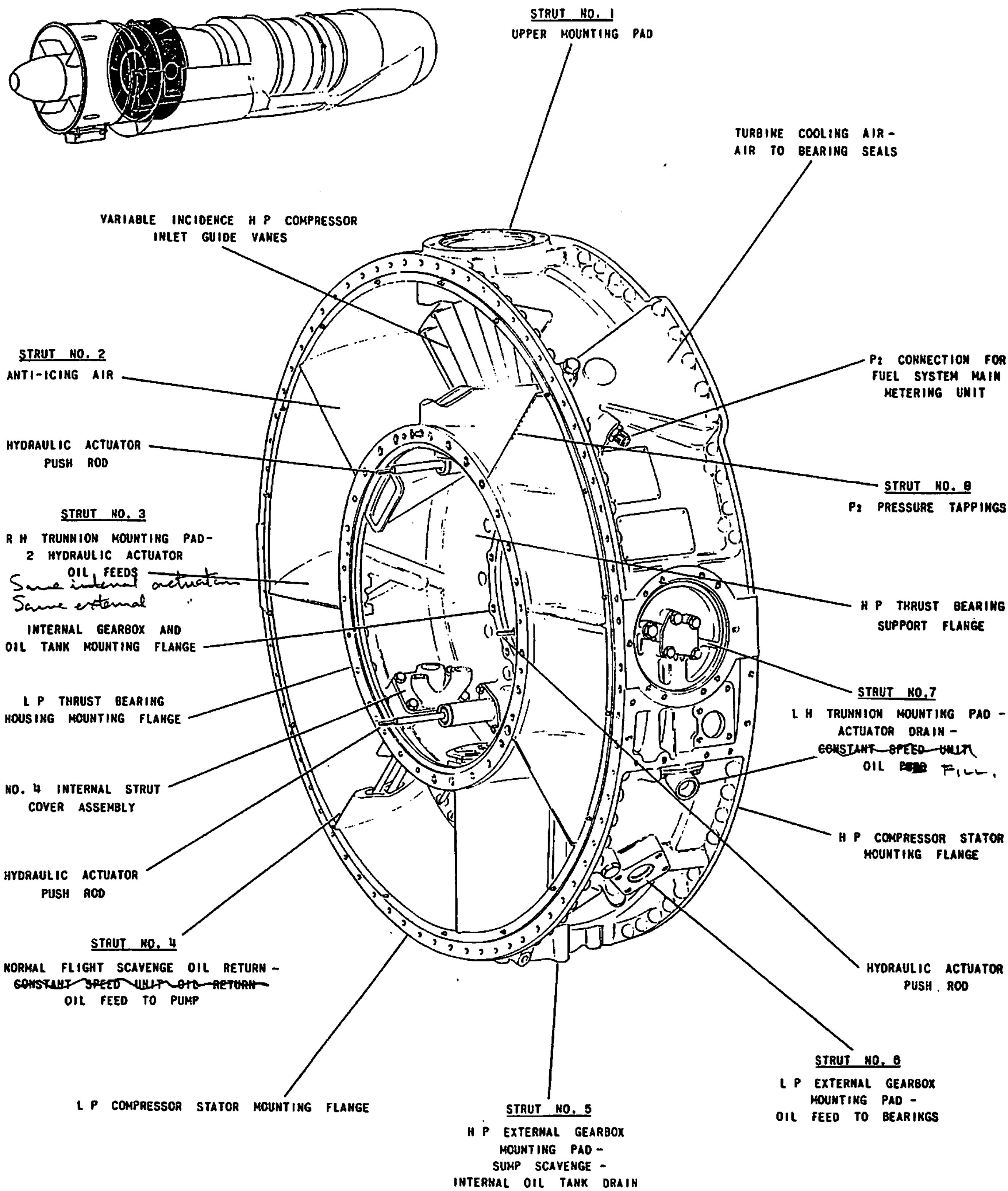


Figure 2-6 Front Frame Assembly

installation, either one or both of the trunnion side mounts may be used in conjunction with the thrust mount without imposing any undue strains on the front frame casing. The LP and HP external gearboxes are mounted on machined bosses in the outer casing at the ends of the struts in the four o'clock and bottom positions respectively.

The eight equi-spaced, hollow struts are aerofoil-shaped and extend radially outwards from the inner casing to the outer casing. Starting with No. 1 strut at the top, the struts are numbered anti-clockwise as viewed from the front of the engine and carry the following services:

No. 2 Strut - Anti-icing air (If required).

No. 3 Strut - Two hydraulic actuator oil feeds.

No. 4 Strut - Normal flight scavenge oil return and oil tank overflow (common line).
- Constant speed unit oil return and oil tank fill pipe (common line).
- Feed from oil tank to oil pump.

No. 5 Strut - Drive for HP external gearbox.
- Front sump scavenge oil.
- Internal oil tank drain.

No. 6 Strut - Oil feed to bearings.
- Drive for LP external gearbox.

No. 7 Strut - Hydraulic actuator oil drain.
- Constant speed unit oil feed.

No. 8 Strut - Seventh stage air for pressurizing oil seals and turbine cooling.
- Inter-compressor pressure (P_2) tapping for fuel system main metering unit.

A machined flange at the front of the cylindrical inner casing provides a mounting for the LP thrust bearing assembly. The HP thrust bearing housing is formed by an inward projecting support flange integrally cast at the rear of the inner casing. A machined flange on the forward face of the bearing housing provides a mounting for the rear of the internal gearbox and oil tank assembly. A cast aluminum cover, embodying three fittings, is attached to the inner end of No. 4 strut. The front fitting is the oil return from the constant speed unit, the large centre fitting is the feed to the oil pump, and the rear fitting is the normal flight scavenge oil return. The three fittings are connected by piping to the internal oil tank. Three machined bosses located round the inside of the inner casing, provide attachment for spring housings and push rods which are directly connected to the hydraulic actuators in the LP thrust bearing assembly.

Air flowing from the outlet of the LP compressor passes through the annular duct formed by the inner and outer casings. The outer periphery of the inner casing is slightly concave between the struts to improve the air flow characteristics through the casing. A ring of variable incidence inlet guide vanes at the rear of the duct direct the air into the HP compressor. The outer ends of the hollow stainless steel guide vanes are mounted in steel support bushings located in an outer ring. This ring is located in a recess in the rear flange of the front frame outer casing and is retained axially when the HP compressor stator casing is assembled to the front frame. Spindles on the inner ends of the vanes locate in an inner ring which is bolted to the rear face of the inner casing. The bolts retaining the inner ring also carry a four groove air seal which mates with the peripheral seal on the front face of the HP compressor rotor. Small levers pinned to the inner ends of the vane spindles engage with slots in a roller-supported unison ring which in turn engages with three bearing-supported bell cranks. Each bell crank is linked to a hydraulic actuator by its respective push rod in a manner which transforms linear movement of the push rod into rotary movement of the unison ring and hence results in a uniform change in the angle of incidence of each inlet guide vane. The total rotational movement of the vanes is governed by internal stops in the hydraulic actuators.

Internal passages in the front frame deliver oil to the jets on the fore and aft faces of the HP thrust bearing. Similar passages carry oil from the HP thrust bearing scavenge impellers to the internal oil tank via a spring-loaded check valve. Seventh stage air is piped into No. 8 strut and is tapped off at the front face of the inner casing to supply air pressure to the LP bearing oil seal. The remainder of the air from No. 8 strut bleeds into the interior of the HP compressor rotor to pressurize the HP thrust bearing oil seal and cool the turbine assemblies.

INTERNAL GEARBOX AND OIL TANK ASSEMBLY

The internal gearbox and oil tank assembly is housed within the front frame inner casing. The assembly consists basically of a gearbox casing, and two gear trains which supply power to the externally mounted LP and HP gearboxes. When installed in the front frame, the gearbox serves as the outer wall of the front sump of the engine lubrication system, with the casing also forming the inner circular wall of the oil tank. The front and rear walls of the oil tank are provided by the LP and HP thrust bearing housings respectively, when the bearing assemblies are installed in the front frame.

The internal gearbox casing is cast from magnesium, with a flange on the front and rear faces. The front flange butts against a flange on the rear face of the LP thrust bearing housing, with an 'O' ring interposed in the joint. The rear flange is bolted to a flange on the forward face of the integrally cast HP thrust bearing housing in the front frame casing. The bottom of the internal gearbox casing seats on the inner end of the front frame No. 5 strut and is bolted to the inner end of the front frame No. 6 strut; an 'O' ring seal is interposed at each of these joints.

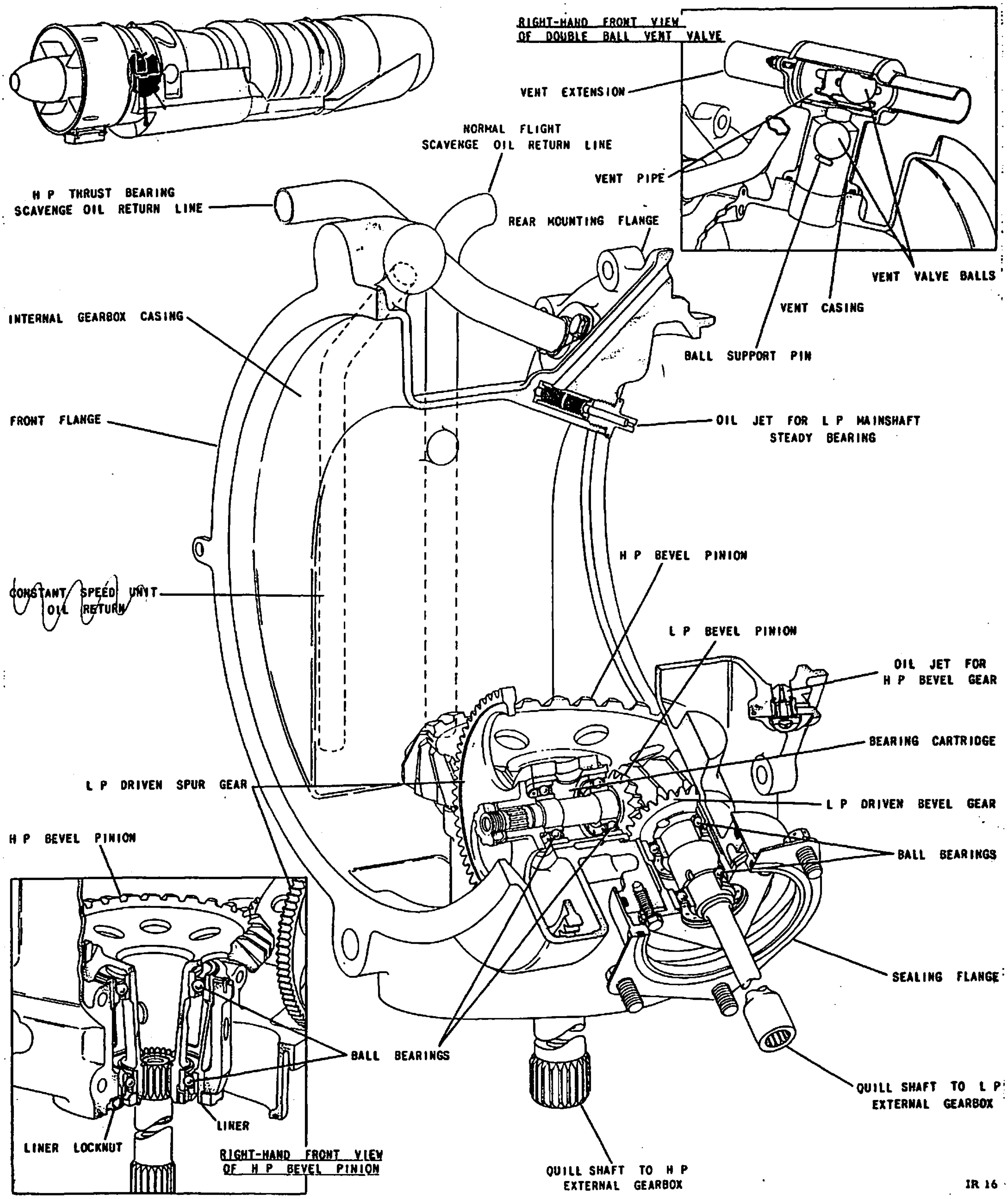


Figure 2-7 Internal Gearbox and Oil Tank Assembly

LP Gear Train

A drive gear mounted on the LP compressor shaft meshes with a spur gear within the gearbox casing. A small bevel pinion gear is splined to the hub of the spur gear and is retained by a locknut and lockwasher. The spur gear and bevel pinion gear combination is supported by two, single row ball bearings, the bearing cartridge being housed within an integrally cast boss in the gearbox casing. The drive is turned through ninety degrees by a second bevel gear which is internally splined at its outer end to receive a quillshaft which extends through the front frame No. 6 strut to drive the LP external gearbox. Two single row ball bearings are used to support the driven bevel gear.

HP Gear Train

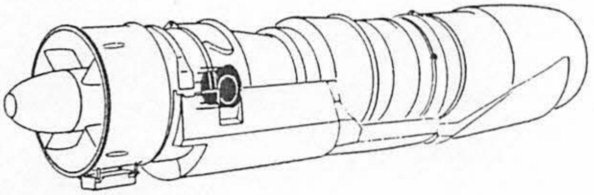
A bevel pinion gear meshes with the bevel gear splined to the HP compressor front stub shaft, and is supported by two, single row ball bearings in an integral housing in the bottom of the casing. The upper bearing is pressed onto the bevel pinion gearshaft and mounted in a steel liner. The lower bearing is pressed into the liner, and is retained on the gearshaft by a washer and locknut. A flange on the upper end of the liner locates the liner against the inner face of the housing and serves as a retainer when the liner locknut is tightened. Rotational movement of the liner is prevented by a dowel pin which is fitted through the face of the flange into the housing. A quillshaft, splined to the bottom of the gearshaft, carries the drive to the HP external gearbox and aircraft power take-off assembly which is mounted on the outer end of the front frame No. 5 strut.

Lubrication

Two oil jets, located in the rear of the casing, provide lubrication for the LP mainshaft steady bearing and the HP bevel pinion gear respectively. On the right-hand side of the casing are two vertical pipes, connected at the lower ends to the No. 4 strut internal cover in the front frame, one carries normal flight scavenge oil to the oil tank and also acts as an overflow line during tank refilling. The other is the oil tank fill line. An oil pipe and fitting, mounted on a boss at the top of the casing returns scavenge oil from the HP thrust bearing to the oil tank. Air transferred to the tank by the scavenge return system is vented to the front sump by a double ball vent valve. The valve also prevents an oil flow from the tank to the front sump in any flight attitude.

HIGH PRESSURE THRUST BEARING ASSEMBLY

The HP thrust bearing assembly supports the forward end of the HP compressor rotor, and absorbs the net axial thrust of the HP compressor rotor and the HP turbine rotor. The assembly consists basically of a single row ball thrust bearing,



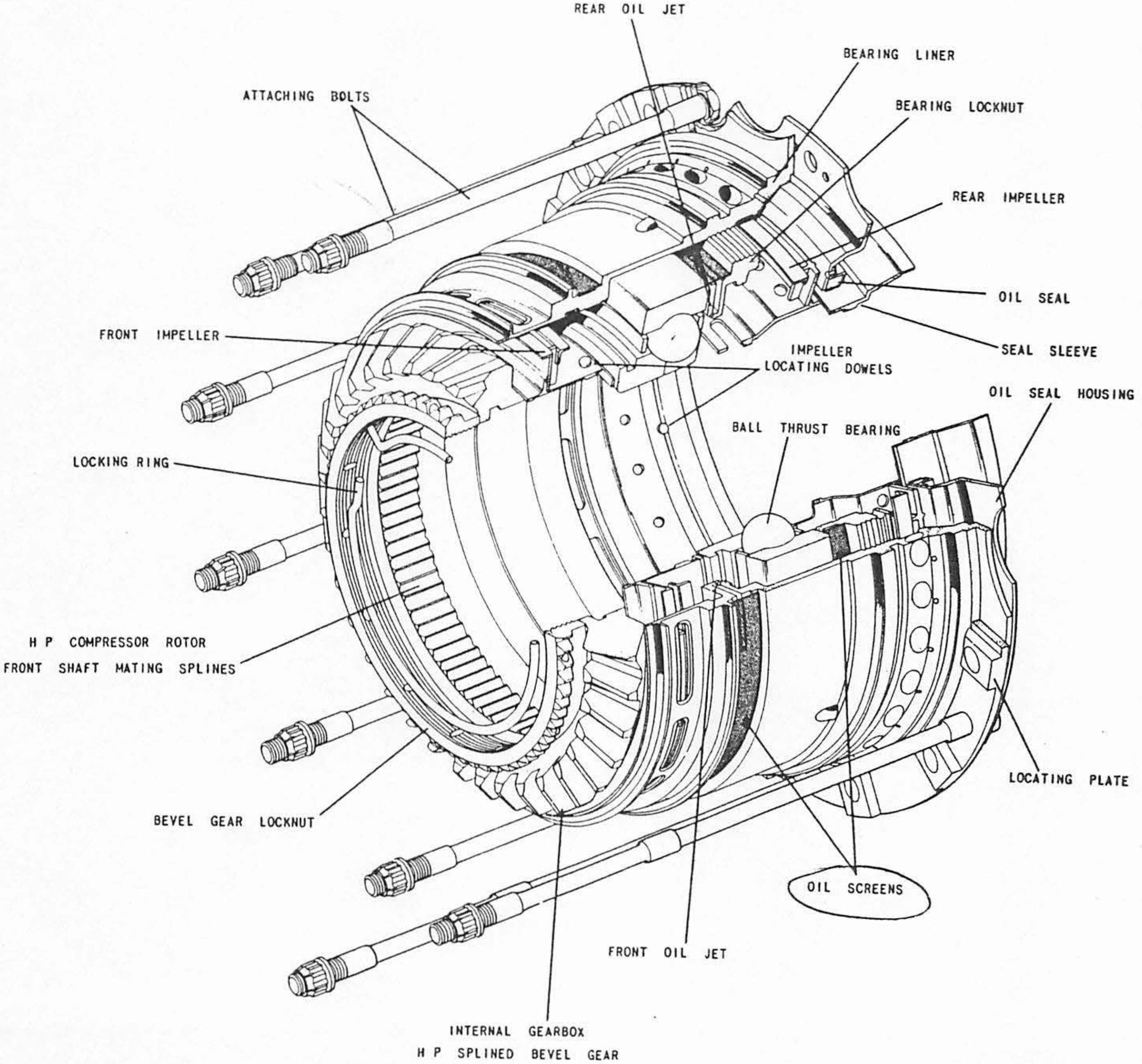


Figure 2-8 High Pressure Thrust Bearing Assembly

IR 14

a bearing liner, the internal gearbox HP bevel gear and locknut, and bearing lubrication system components. When assembled in the front frame it is housed within an inward projecting integrally cast support flange at the rear of the front frame.

The bearing is retained in the steel liner by a pinned locknut which bears against the rear face of the bearing outer race. The liner is in turn retained within the support flange of the front frame by long steel bolts which also serve to retain the internal gearbox casing in position. The split inner race of the bearing butts against a step machined in the HP compressor rotor front stub shaft, and is held in position by the internal gearbox HP bevel gear. The bevel gear is splined to the front stub shaft and is retained axially by a locknut which is in turn positively locked by the tang of a steel locking ring which engages with holes in the front stub shaft and the locknut.

A locating plate, with radially disposed teeth machined on its forward face, is fitted between the bearing liner and the rear face of the front frame. The teeth engage with slots in the front frame, and counteract the effects of thermal differential expansion between the magnesium front frame and the steel liner by keeping the liner centralized.

Lubrication of the bearing is by six oil jets. Three of the jets, located at the front of the bearing, are formed by drillings in the bearing liner while the remaining three, located at the rear of the bearing, are formed by drillings in the bearing locknut. Pressure oil is fed through channelling in the front frame to annular cavities adjacent to each set of oil jets, the front annulus being formed between the front frame and the bearing liner, and the rear annulus between the liner and the bearing locknut. Each annulus is provided with a protective oil screen on the upstream side of the jets.

Two radially vaned impellers, one fitted on each side of the bearing, scavenge the oil from the bearing assembly by centrifugal action. Holes in the bearing liner admit scavenge oil from the impellers to annular cavities which are formed between the liner and the front frame, and which are interconnected to the internal oil tank by channelling in the front frame. The scavenge oil cavities are isolated from the pressure oil cavities by 'O' ring seals.

A single carbon ring oil seal is mounted in an oil seal housing located on the rear face of the bearing liner. A steel seal sleeve which butts against a shoulder on the HP compressor rotor front stub shaft, forms a seat for the oil seal. The rear of the oil seal is pressurized with compressor seventh stage air to prevent oil seepage past the seal in a rearward direction.

Displacement of the seal sleeve and the bearing rear impeller is prevented by a small locating dowel in the front stub shaft. The bearing front impeller is retained by a similar dowel which locates in a slot on the shaft of the internal gearbox HP bevel gear.

HIGH PRESSURE COMPRESSOR ROTOR ASSEMBLY

The HP compressor rotor assembly is the front rotating component of the engine HP spool, and revolves around the LP mainshaft assembly. Together with the stator casing, the HP compressor rotor forms a converging annular passage where further compression of the LP compressor delivery air takes place.

The assembly consists essentially of seven bladed rotor discs and six spacer rings and is supported at the forward end by a stub shaft carried in the HP thrust bearing assembly. The spacer rings and rotor discs are bolted together to form a large diameter drum which transmits axial thrust loadings to the HP thrust bearing. The discs absorb radial loads imposed by the centrifugal blade forces arising from high rotational speeds. The extension shaft and HP mainshaft, extend rearwards from the compressor drum, forming a direct drive from the HP turbine rotor.

The rotor discs and spacer rings are machined from titanium forgings. Development engines have a steel compressor fourth stage disc, but the use of a titanium disc in this stage is proposed for production engines. The fourth stage spacer ring and disc are cantilevered forward from the fifth stage disc. The fifth stage disc is bolted to the fifth stage spacer ring, which has a conical extension bolted to the front stub shaft.

The HP compressor rotor blades are titanium forgings and have an aerofoil section, with the blade tips feathered to aid in heat dissipation and to minimize damage should a tip rub develop. Blade stops are machined in the rear portion of the blade roots of stages four to nine inclusive, to prevent axial movement of the blades in a forward direction. Blades in the tenth stage have a stop on the front portion of the root to prevent rearward movement. The fourth and tenth stage blades are secured to the respective discs by two-branch fir tree roots, while the remaining stages employ a cylindrical seat dovetail fixing.

Steel finally aluminum

Inter-stage air seals of titanium are riveted to flanges on the spacer rings. The forward extending projections on the seal rings butt against the rotor blade roots in the preceding stage and prevent rearward axial movement of the blades. Forward movement of the tenth stage rotor blades is prevented by a similar projection extending from the rear of the ninth stage seal ring. A peripheral air seal is fitted on the front face of the fourth stage disc to prevent excessive air leakage through the clearance between the HP compressor rotor and the front frame.

The bell-shaped front stub shaft is bolted to the conical extension of the fifth stage spacer ring. A seat and shoulder, machined on the outside diameter of the shaft near the mid-point, provides a location for the HP thrust bearing. The outer

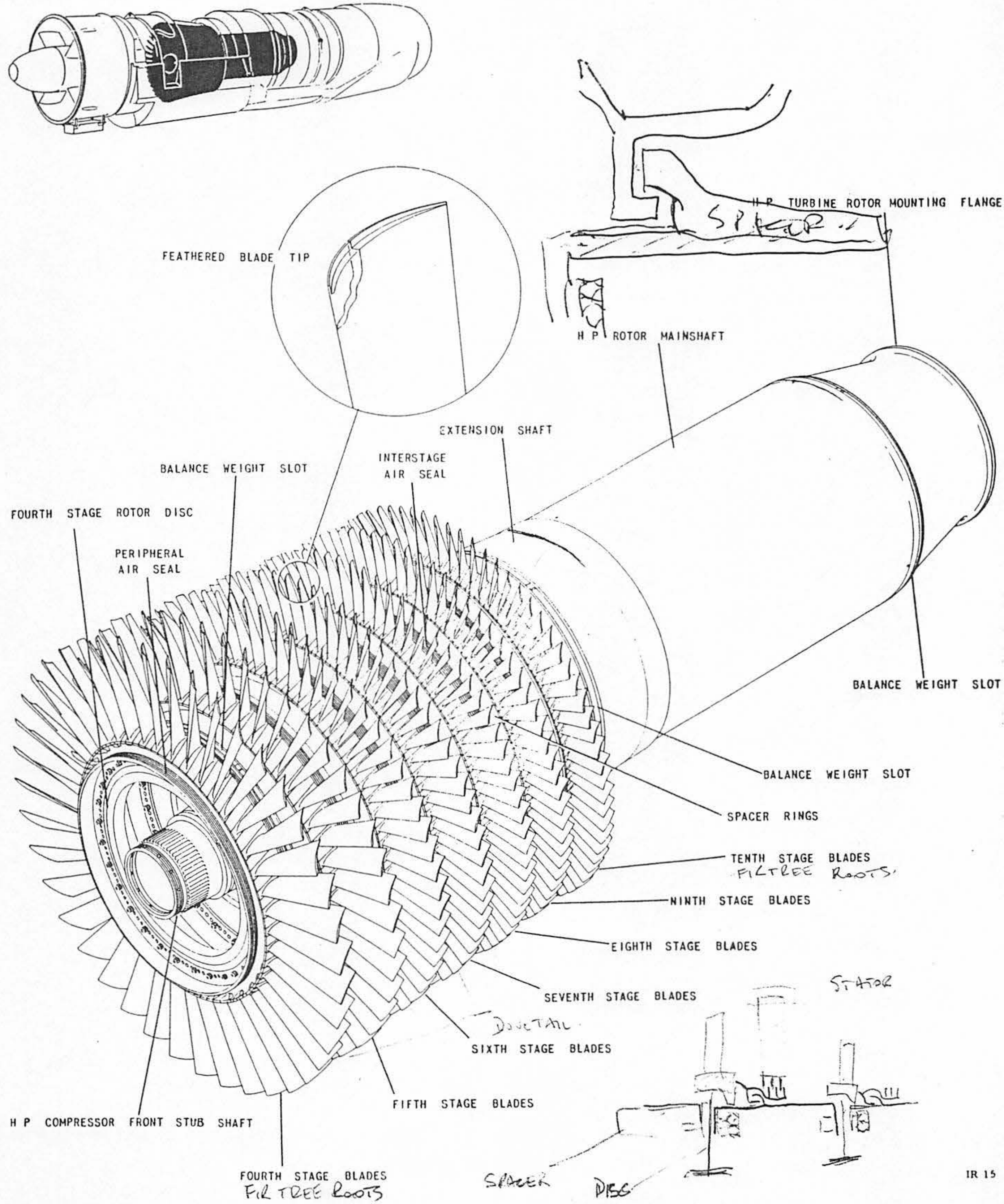


Figure 2-9 High Pressure Compressor Rotor Assembly

diameter of the forward end of the shaft is splined to carry the bevel gear which drives the external gearbox. The bevel gear is retained by a locknut and lockring, the latter being fitted to the inside of the shaft with a tang protruding through the shaft into the nut. A series of large diameter holes in the conical portion of the front stub shaft, permit the entrance of seventh stage compressor air to the interior of the HP compressor rotor.

A steady bearing, consisting of a single roller bearing, is mounted inside the front stub shaft. The inner race of the bearing is located against a shoulder on the front section of the LP mainshaft and is retained by the LP compressor shaft and the LP mainshaft locknut.

The extension shaft links the compressor drum to the HP mainshaft and the HP mainshaft is secured to the HP turbine rotor by stainless steel tension bolts and a clamp ring fitted to the rear flange of the mainshaft.

Balancing of the HP compressor rotor assembly is done by the addition of weights installed in three circular grooves, one located on the rear portion of the HP mainshaft, one at the centre of the extension shaft, and another at the fourth stage spacer ring. The outer balance weights in each group are secured by peening. Later engines will incorporate balancing assemblies on the rotors which will be accessible and adjustable with the engine on a test bed.

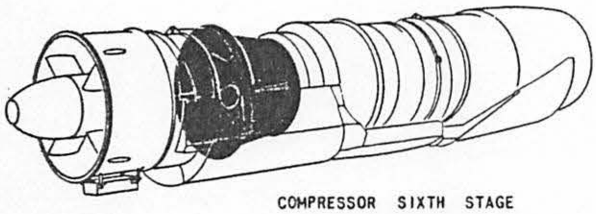
NOTE

Although the LP and HP compressors have been treated as separate units, the numbering of the stages has been consecutive from the front of the engine through the two units. This is to be consistent with the numbering used in the engine functional descriptions.

HP COMPRESSOR STATOR ASSEMBLY

The HP compressor stator assembly forms the structural connection between the front frame and the mid frame, and encases the HP compressor rotor drum. The assembly comprises mainly an outer casing, six stages of stator segments and seven spacer rings.

The outer casing consists of an unsplit, conical, sheet steel shroud with a flange welded to each end for bolting the casing to the front and mid frames respectively. A circular retaining ring, integral with the outer casing, has a machined recess on its inner front face, which mates with a hook-shaped projection on the seventh stage spacer ring. The retaining ring serves as a stiffener for the outer casing to prevent ovality occurring in the stator assembly. Two triangular-shaped instrumentation bosses are located near the bottom of the outer casing immediately aft of the forward mounting flange. A sheet metal flange is welded to each side of



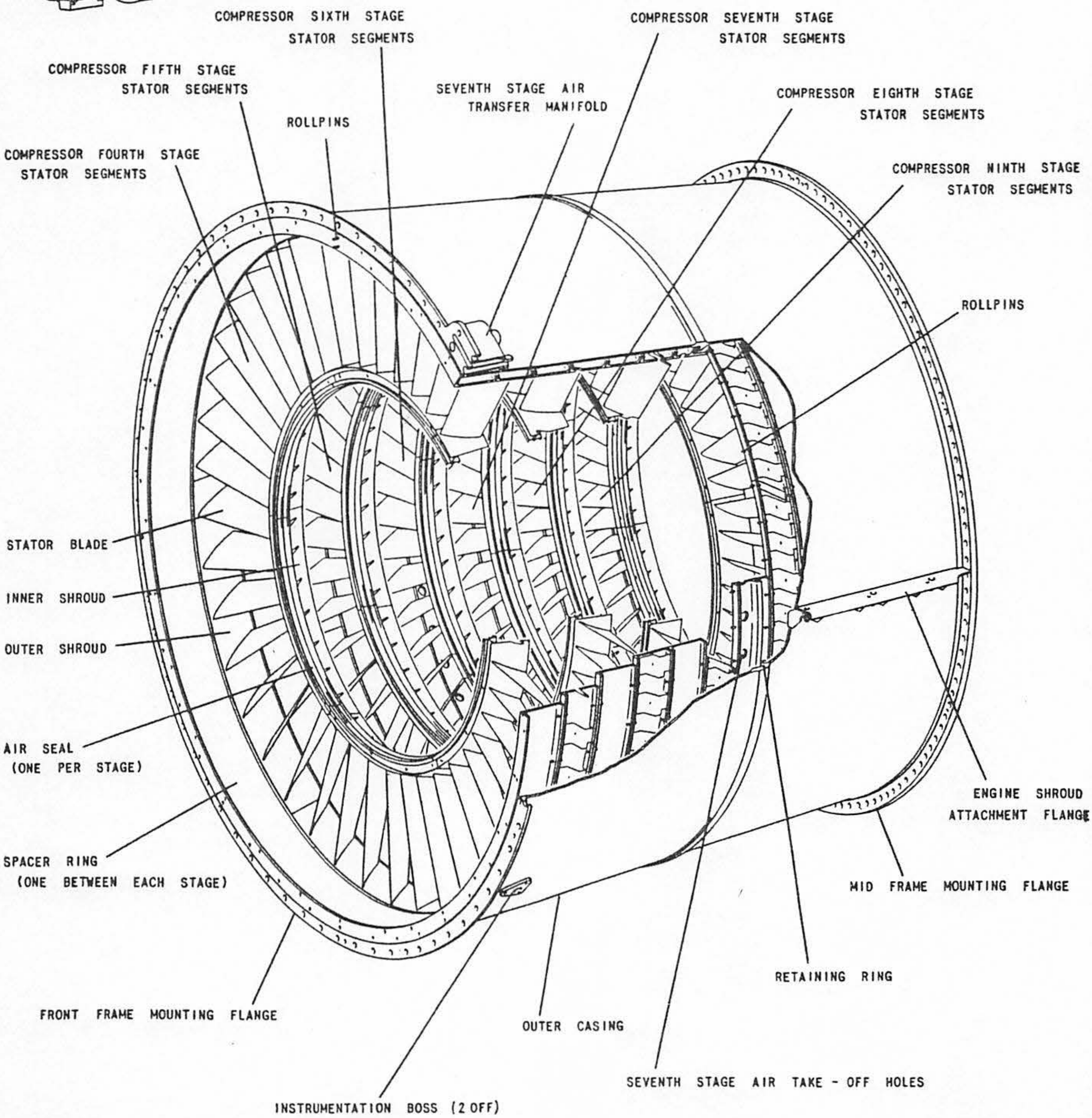


Figure 2-10 High Pressure Compressor Stator Assembly

the casing, along the horizontal centre line, for attachment of the engine shroud.

Each stage of stators is composed of six segments which, when assembled in position, form a continuous ring of stator blades. The number of blades per segment in the fourth to ninth stages inclusive is seven, eight, nine, nine, eleven and eleven respectively. The tenth stage stator segments form part of and are dowelled directly to the mid frame assembly. The stator segments are of stainless steel and consist of inner and outer shrouds brazed to the ends of hollow stator blades. Each aerofoil-shaped stator blade is fabricated from two sheet steel pressings strengthened by a corrugated stiffener inserted in the blade core. The outer end of each blade is sealed with a steel plate brazed to the outer shroud; the inner end is left open to vent the blade core. An air seal of the two-groove type is riveted to a flange on the inner shroud of each segment to mate with the inter-stage air seals fitted to the HP compressor rotor spacer rings.

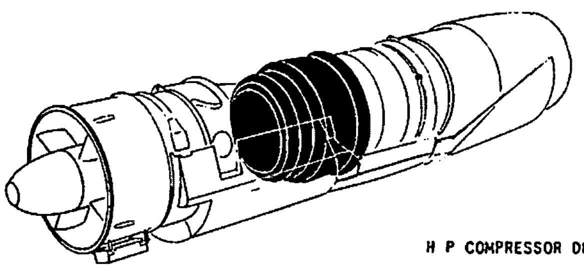
The circular spacer rings are of stainless steel, and are located intermittently between each stage of stator segments. Like the segment outer shrouds, the spacer rings are channel-shaped to provide maximum rigidity of the assembly. The flanges of the segment outer shrouds and the spacer rings butt together and are retained by single rows of equi-spaced rollpins. The foremost spacer ring is pinned to the rear of the front frame by two rows of equi-spaced rollpins. A coating of friable material is applied to the inner diameter of each spacer ring to minimize damage that may be caused by the development of a tip rub.

A series of holes in the seventh stage spacer ring, forward of the hook-shaped projection, permits air to be bled off into the annulus formed by the spacer ring and segment assembly, and the outer casing. An air transfer manifold, located on the outer casing adjacent to the front frame mounting flange, transfers seventh stage bleed air from this annulus to the passage provided through the front frame No. 8 strut. This flow of bleed air is used for pressurizing oil seals in the main bearing assemblies and for turbine cooling.

THE MID FRAME

The mid frame assembly houses the annular combustion chamber and forms the structural connection between the compressor and turbine HP sections. The assembly comprises mainly the inner and outer casings, the annular combustion chamber liners, the compressor tenth stage stator segments and the primary fuel injectors. The combustion base plate assembly is described separately under the next heading. HP compressor delivery air enters the annular opening at the front of the mid frame and is divided into primary and secondary flows; the primary flow passes through the snout of the combustion base plate assembly, and the secondary flow passes into the inner and outer annular spaces formed between the combustion chamber liners and the inner and outer casings.

Annular gives large combustion area and small frontal area.



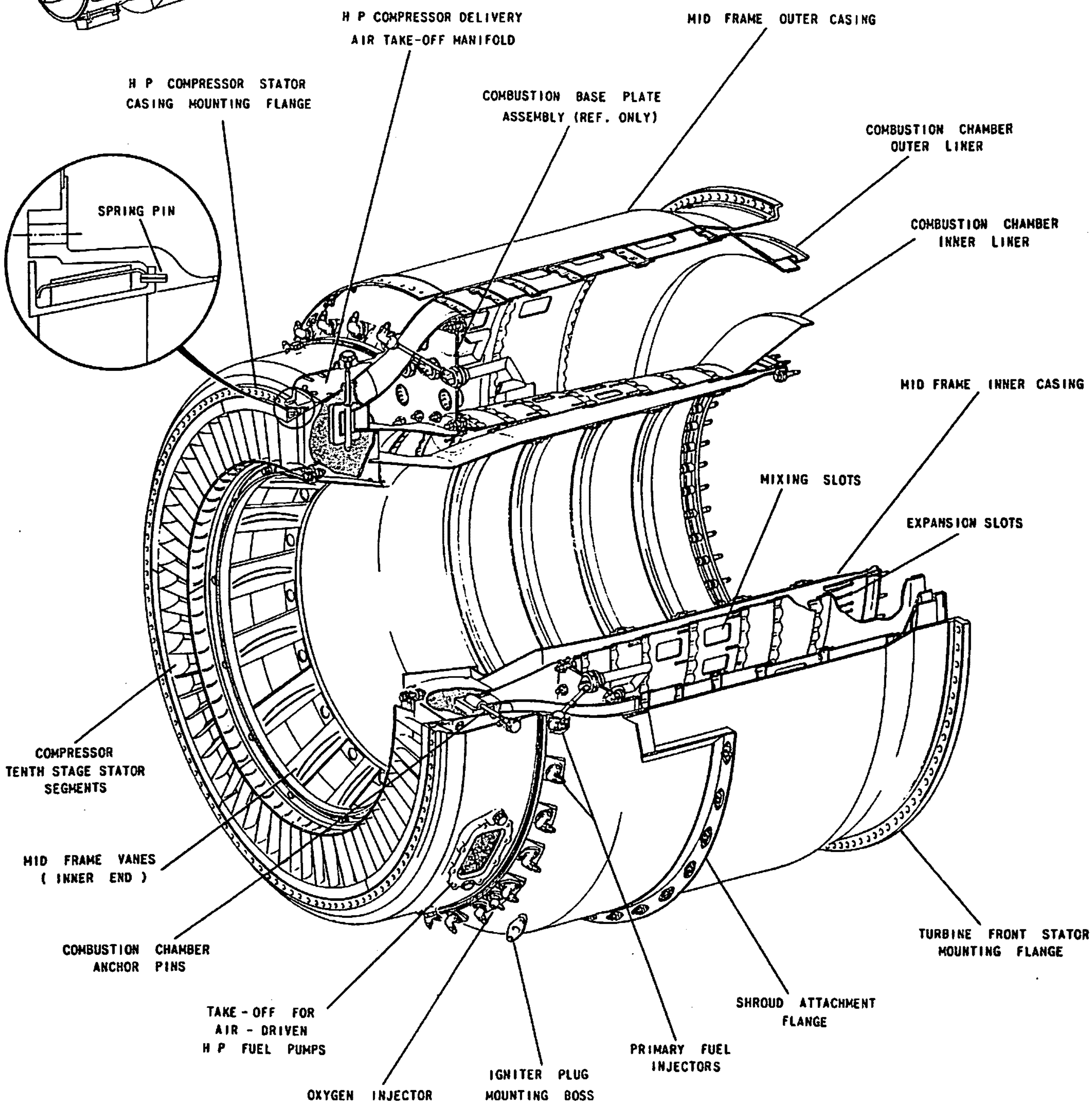


Figure 2-11 Mid-frame Assembly

The concentrically mounted inner and outer casings are fabricated from stainless sheet steel; the outer casing is of heavier gauge material to withstand structural loadings and the high combustion chamber gas pressures to which it is subjected. The inner casing separates the combustion chamber from the HP compressor mainshaft. The inner and outer casings are joined at the front by 16 hollow, aerofoil section vanes, the trailing edges of the vanes is recessed to accommodate the combustion base plate assembly frame. Machined bolting flanges on the outer casing provide for attachment of the mid frame to the HP compressor stator casing at the front and to the turbine front stator casing at the rear. A similar flange at the rear of the inner casing is used for attachment of the turbine first stage stator segment inner support. Two bosses, at the bottom of the outer casing, accommodate combustion drain fittings.

An air take-off manifold is located round the periphery of the outer casing immediately aft of the front mounting flange. Holes in the outer casing, at the rear of the tenth stage compressor stators, permit a bleed of HP compressor delivery (tenth stage) air into the manifold. Two take-off pads, one located on each side of the vertical centre line of the manifold, provide compressor tenth stage air for aircraft services; one pad is used, and the other blanked off, dependant upon engine installation requirements. The air take-off pad for the engine anti-icing system is located on the top RH side of the manifold. On the lower LH side of the manifold a pad is provided for an air take-off to drive the air-driven HP fuel pumps. A small take-off pad on the RH side of the manifold provides a P_3 pressure tapping for use in the engine fuel and final nozzle control systems.

Sixteen radially disposed combustion chamber anchor pins engage with holes in channel-shaped supports which are integral with the mid frame vanes. The anchor pins pass through guides on the forward end of the combustion base plate assembly to support and retain the assembly. Access holes in the air take-off manifold are located in line with the outer ends of the anchor pins and are plugged to prevent radial displacement and hence disengagement of the pins.

The combustion chamber inner and outer liners are fabricated from cylindrical sections of stainless sheet steel which are joined together by spot welded corrugated jointing strips. When assembled in position, the liners form a slightly divergent annulus; a conical section welded to the rear of each liner forms a convergent outlet from the combustion chamber. The openings formed between the corrugations of the jointing strips permit a relatively cool flow of secondary air over the hot surfaces of the liners. Mixing slots in the liners allow a proportion of the secondary air flow to enter the combustion chamber, and mix with and dilute the products of combustion prior to their entry into the turbine. Expansion slots in the inner liner compensate for differential thermal expansion of the cylindrical sections.

The front end of the inner liner is a sliding fit in a support sleeve on the inner rear flange of the combustion base plate assembly; the rear end of the liner is supported by a welded projection on the turbine first stage stator segment inner support. Axial displacement of the inner liner is prevented by a slotted sheet metal projection which is welded round the rear conical section of the liner. A lip on the projection engages with a recess in the rear bolting flange of the mid frame inner casing and is retained by the bolting flange of the turbine first stage stator segment inner support. The outer liner is bolted at the front, to the outer rear flange of the combustion base plate assembly; a support welded to the rear conical section of the outer liner is a sliding fit on the turbine first stage stator segment outer platforms.

The compressor tenth stage stators consist of six segments of 11 stainless steel blades each. The blades have a hollow, aerofoil section, with corrugated stiffeners through the core of each blade. The outer ends of the blades are brazed to stainless steel, outer shrouds. Stainless steel plates are fitted between the flanges of the outer shrouds, over the ends of the blades. Steel inner shrouds are brazed to the inner ends of the blades; the inner ends of the blades are left open to vent the blade cores.

The stator segments are pinned to a recess in the front face of the mid frame outer casing by spring pins and retained in position by the rear bolting flange of the HP compressor stator assembly when the engine is assembled. A segment retaining ring is bolted to the inner front face of the mid frame vanes to retain the inner shrouds of the stator segments.

A primary fuel injector is mounted on each of the 32 bosses located round the mid frame outer casing immediately to the rear of the air take-off manifold. The inner ends of the fuel injectors are centred in flared receptacles at the inlets of the combustion tubes of the combustion base plate assembly. The outer end of each injector is carried in a spherical seating which permits self-alignment of the injectors. A fuel metering orifice in each injector is carefully matched to ensure uniform fuel distribution to the combustion tubes.

Two bosses, near the bottom of the mid frame casing in line with the fuel injectors, provide a mounting for the oxygen injectors. When assembled to the mid frame, the inner ends of the oxygen injectors engage with the two hemispherical igniter chambers in the combustion base plate assembly. The outer ends are supported in spherical seatings similar to those used for the primary fuel injectors. An igniter plug is fitted to an angled boss on each side of the mid frame casing at the bottom so that the plugs line up with the igniter chambers. During normal ground starting, only the igniter plugs are used to ignite the fuel/air mixture round the combustion tubes. Relight at altitude is assisted by injecting oxygen into the igniter chambers through the oxygen injectors.

Sheet metal flanges, welded round the lower half of the mid frame outer casing, are fitted with anchor nuts for attachment of the engine shroud assembly.

COMBUSTION BASE PLATE ASSEMBLY

The combustion base plate assembly, which forms the head section of the annular combustion chamber, is located inside the front portion of the mid frame. The assembly consists primarily of a combustion base plate mounted in a support frame, and serves to meter the correct proportion of HP compressor delivery air into the primary zone of the combustion chamber. Combustion tubes and secondary air tubes, extending rearwards from the combustion base plate, create proper conditions for efficient burning of the air and fuel mixture in the combustion chamber.

The combustion support frame comprises circular inner and outer sheet steel plates, contoured to form a snout-shaped divergent annulus. The support frame divides the incoming HP compressor delivery air into a primary flow and a secondary flow. The primary air flow enters through the snout opening and is diffused in the support frame prior to entering the primary combustion zone. The secondary air flow passes rearwards through inner and outer annular spaces formed between the combustion liners and the mid frame casing, and is admitted into the combustion chamber through openings in the liners to assist combustion in the primary zone and cool the burnt gases prior to their entry into the turbine.

Sixteen streamlined, combustion chamber anchor pin guides are welded equidistantly round the snout opening, between the support frame inner and outer skins. These guides, with the anchor pins, support the forward end of the base plate assembly and keep it concentric with the mid frame casings.

Flanges at the rear of the support frame provide for attachment of the circular combustion base plate to the support frame, and of the complete base plate assembly to the forward ends of the combustion chamber liners. Together with a support sleeve, the base plate is bolted directly to the inner flange at the rear of the support frame, the support sleeve at this location being a sliding fit on the forward end of the combustion chamber inner liner. The base plate is retained to the support frame outer rear flange by the bolts which secure the combustion base plate assembly to the forward end of the combustion chamber outer liner.

An inner and outer circle of stainless steel primary combustion tubes of 16 tubes in each circle, are fitted to the combustion base plate. Each tube is in the form of a 'walking stick' which extends rearwards from the base plate, and is angled through two 90° turns so that the outlet of each tube is facing upstream. This configuration produces a flame which passes rearwards over the outer surface of the tubes, resulting in complete vapourization of the fuel as it passes through the heated tubes. Fuel is introduced into the front end of these tubes by means of 32 injectors which pass through dimpled holes in the support frame outer skin and locate in flared receptacles on the tube inlets. The primary air flow through the tubes mixes with and carries the injected fuel to the tube outlets.

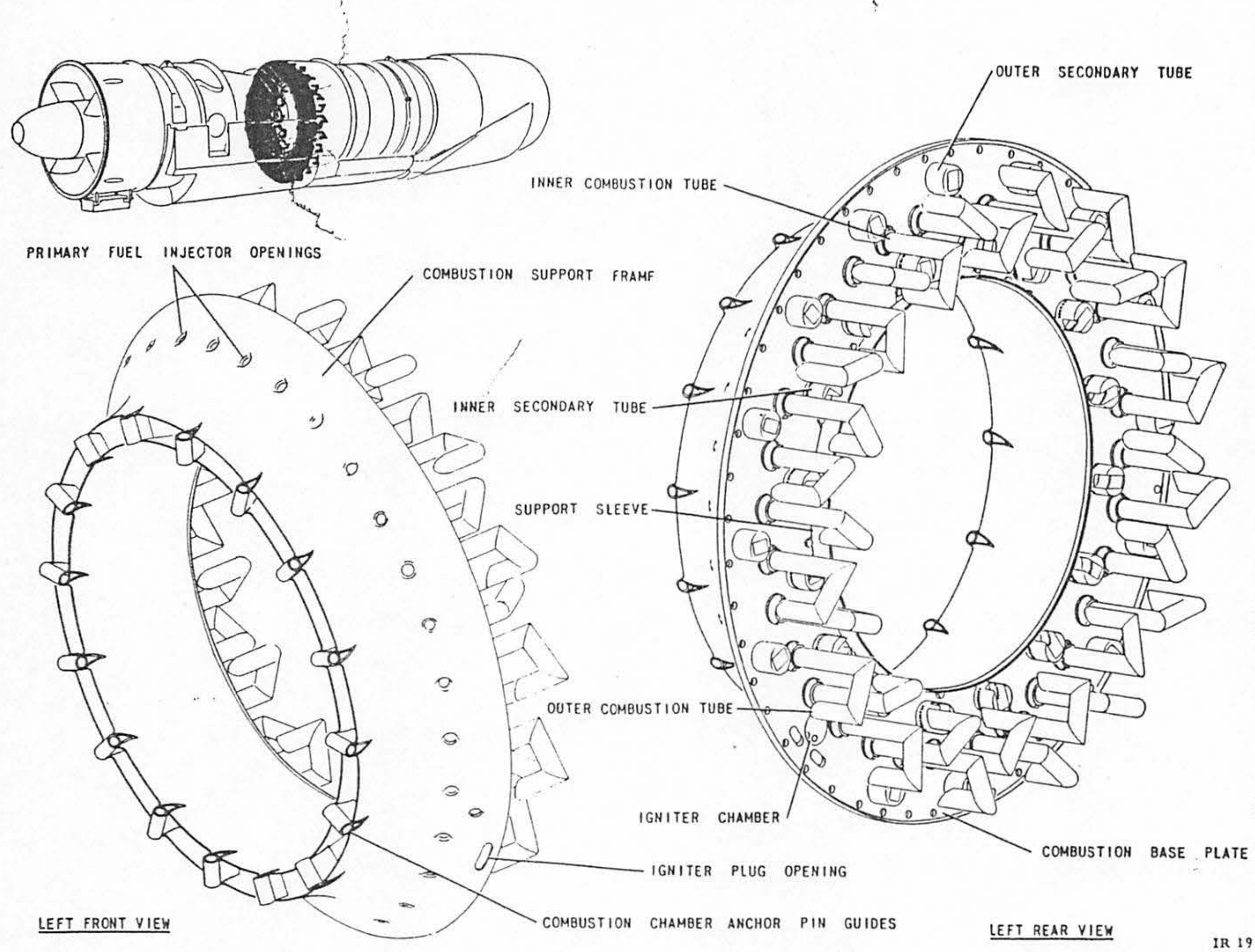
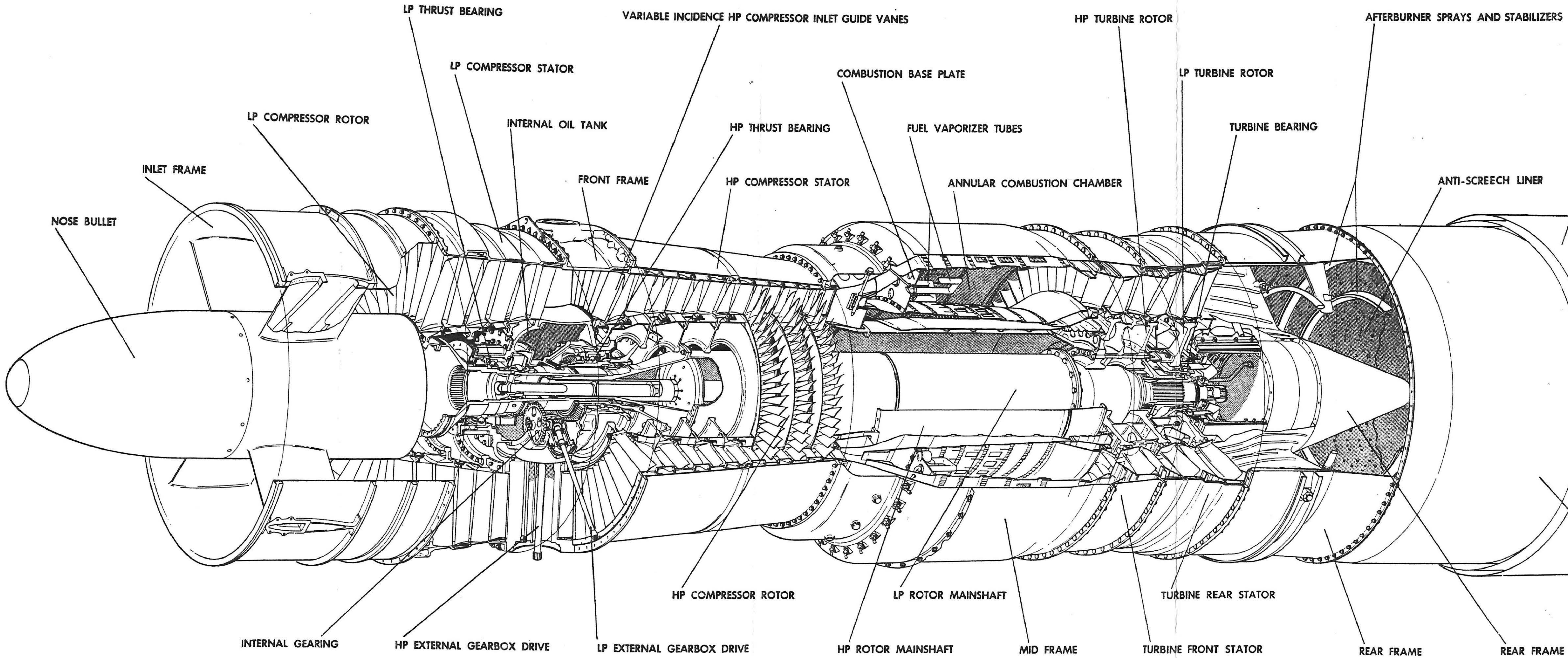


Figure 2-12 Combustion Base Plate Assembly

A double circle of secondary tubes, fitted on the rear face of the base plate, are arranged so that the air passing through these tubes meets with the fuel vapour expelling from the combustion tubes, and form vortexes about which the combustion flame stabilizes. Each of the 14 secondary pipes in the outer circle consists of a cylindrical cup-shaped pressing with a slotted opening in the rear end; the 16 tubes of the inner circle are similar except for the inclusion of a row of small holes round the base of each pipe.

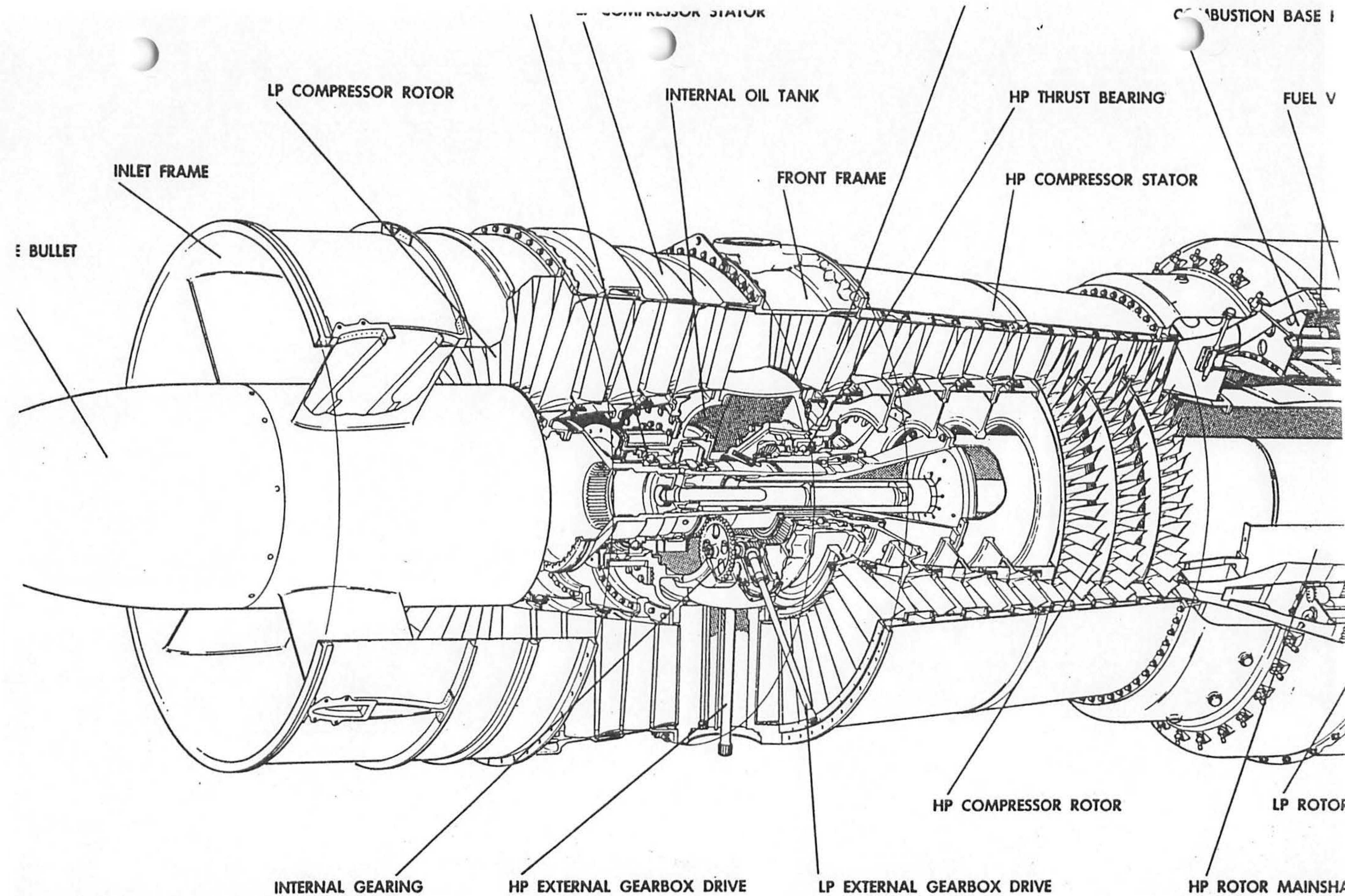
Two hemispherical igniter chambers, each consisting of a slotted dome-shaped pressing, are located on the lower front face of the base plate. Two flared receptacles in each chamber accommodate the operating ends of an igniter plug, and an oxygen injector. The two igniter plugs are of the low voltage surface gap type, and provide the spark which initiates combustion during the engine starting cycle. The oxygen injectors are provided to improve the re-light characteristics of the engine at altitude.



NOTE

THIS ILLUSTRATION IS BASED ON THE IROQUOIS 4th PROGRAMME PRE-PRODUCTION CONFIGURATION AND SHOULD ONLY BE USED TO OBTAIN A GENERAL CONCEPTION OF THE ENGINE. FOR DETAIL REFERENCE USE THE LATEST GENERAL ARRANGEMENT DRAWINGS. THIS ILLUSTRATION WILL BE PROGRESSIVELY REVISED AS LATER INFORMATION BECOMES AVAILABLE.

Figure 2-24 Sectioned View of Engine (Excluding Engine Shroud and External Accessories)



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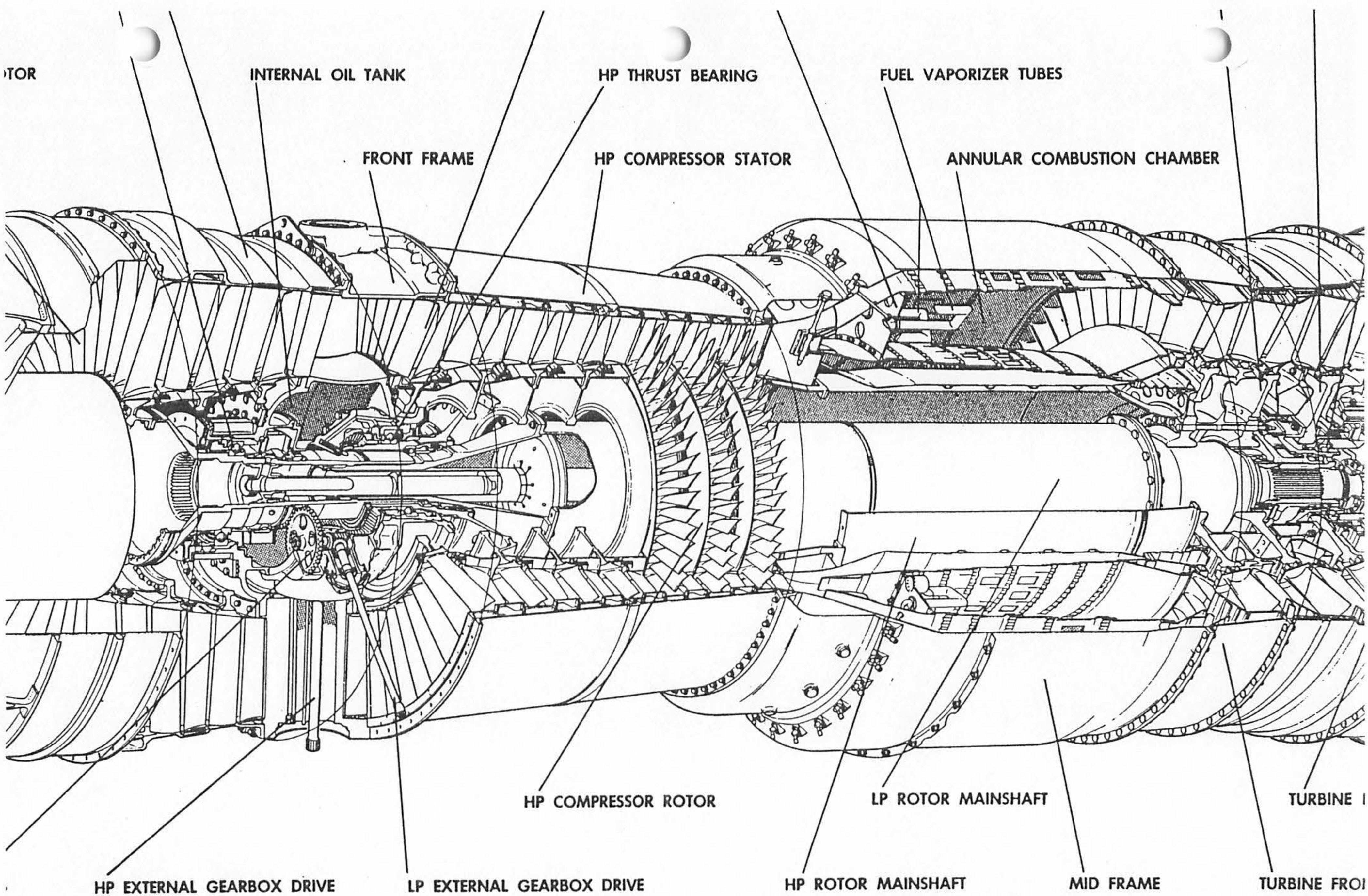
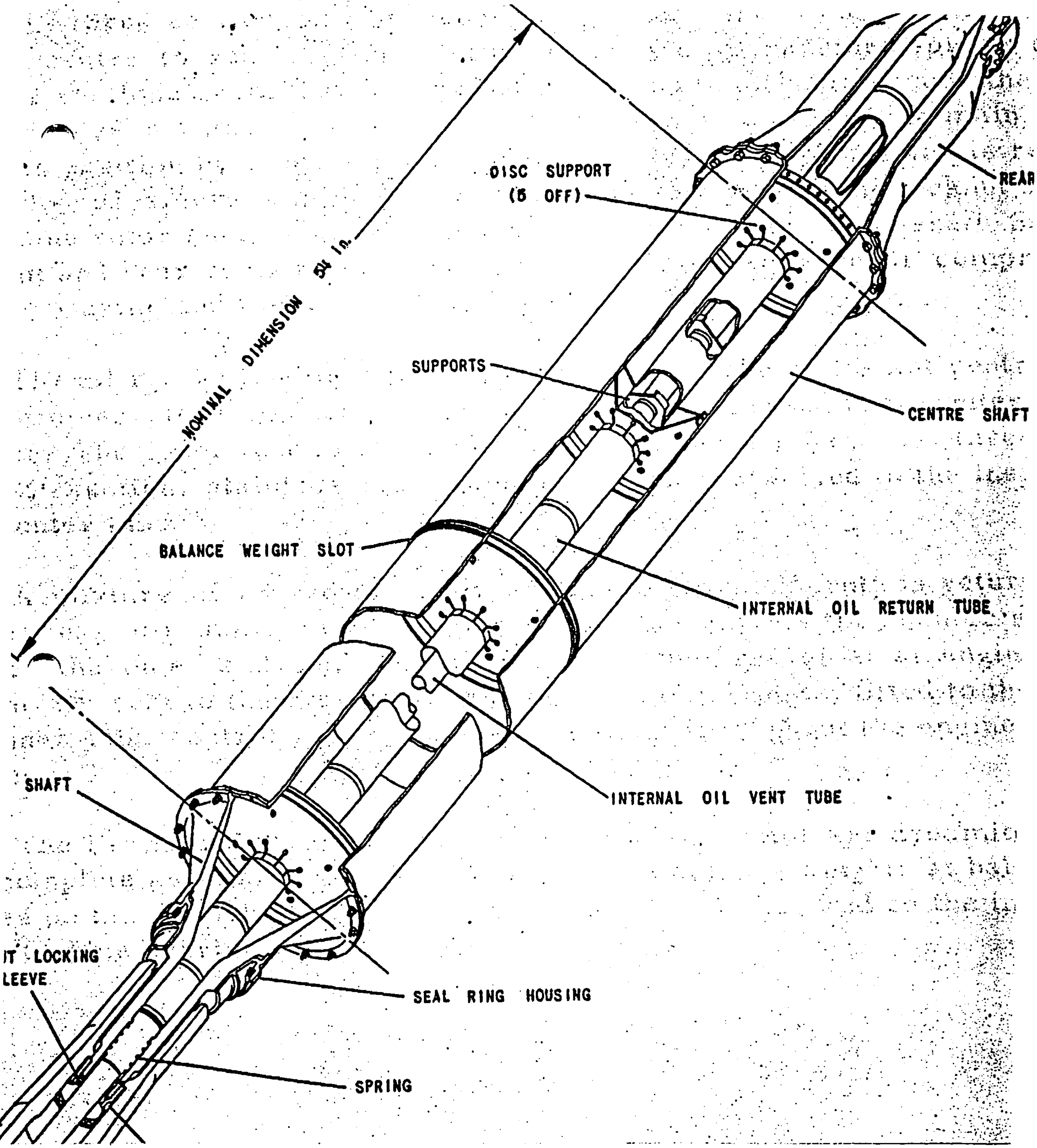


TOR
INTERNAL OIL TANK
HP THRUST BEARING
FUEL VAPORIZER TUBES
FRONT FRAME
HP COMPRESSOR STATOR
ANNULAR COMBUSTION CHAMBER
HP COMPRESSOR ROTOR
LP ROTOR MAINSHAFT
TURBINE
HP EXTERNAL GEARBOX DRIVE
LP EXTERNAL GEARBOX DRIVE
HP ROTOR MAINSHAFT
MID FRAME
TURBINE FRO


NOMINAL DIMENSION 54 in.
DISC SUPPORT
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REAR
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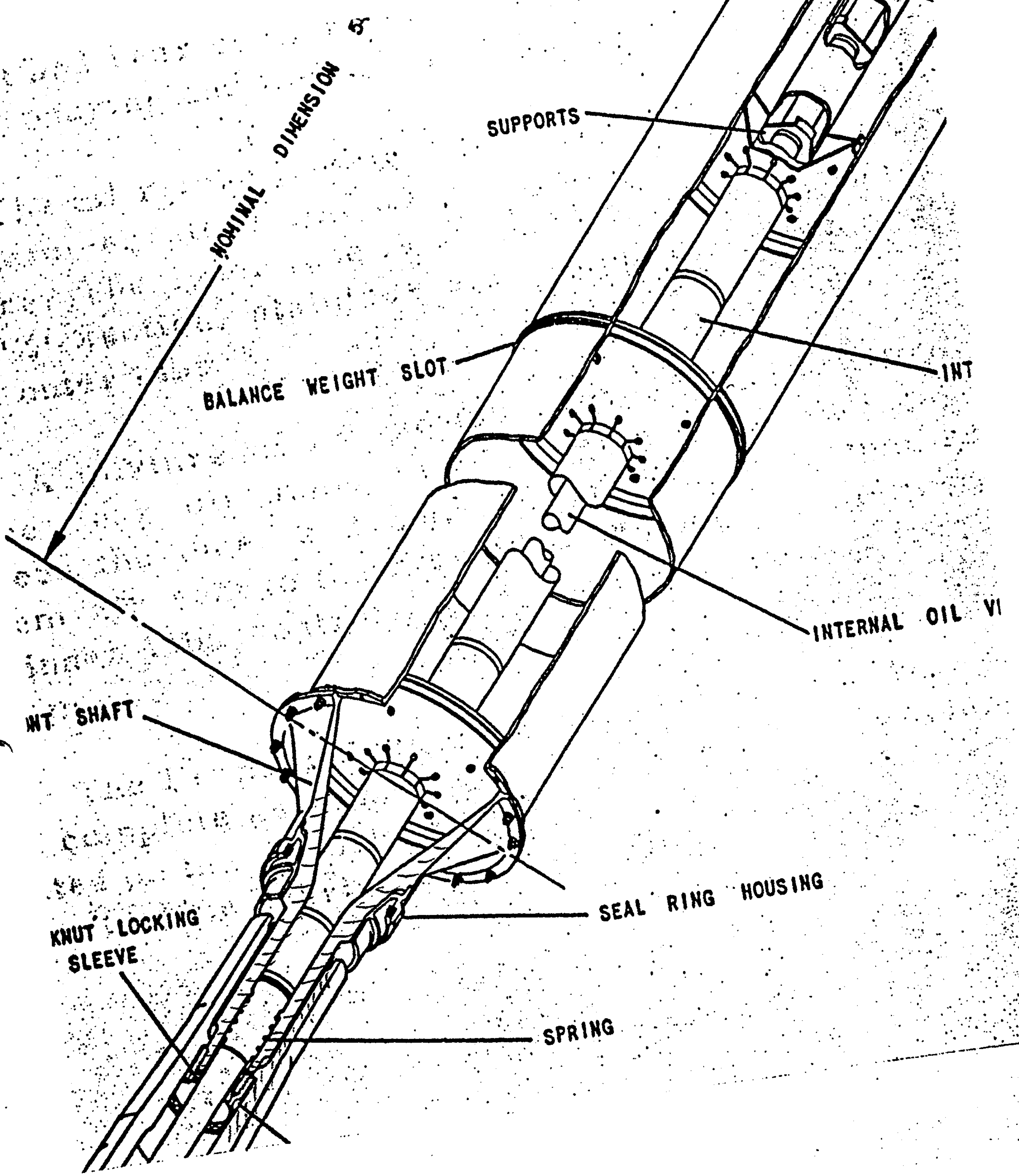
IT LOCKING
LEEVE
SEAL RING HOUSING
SPRING


NOMINAL DIMENSION
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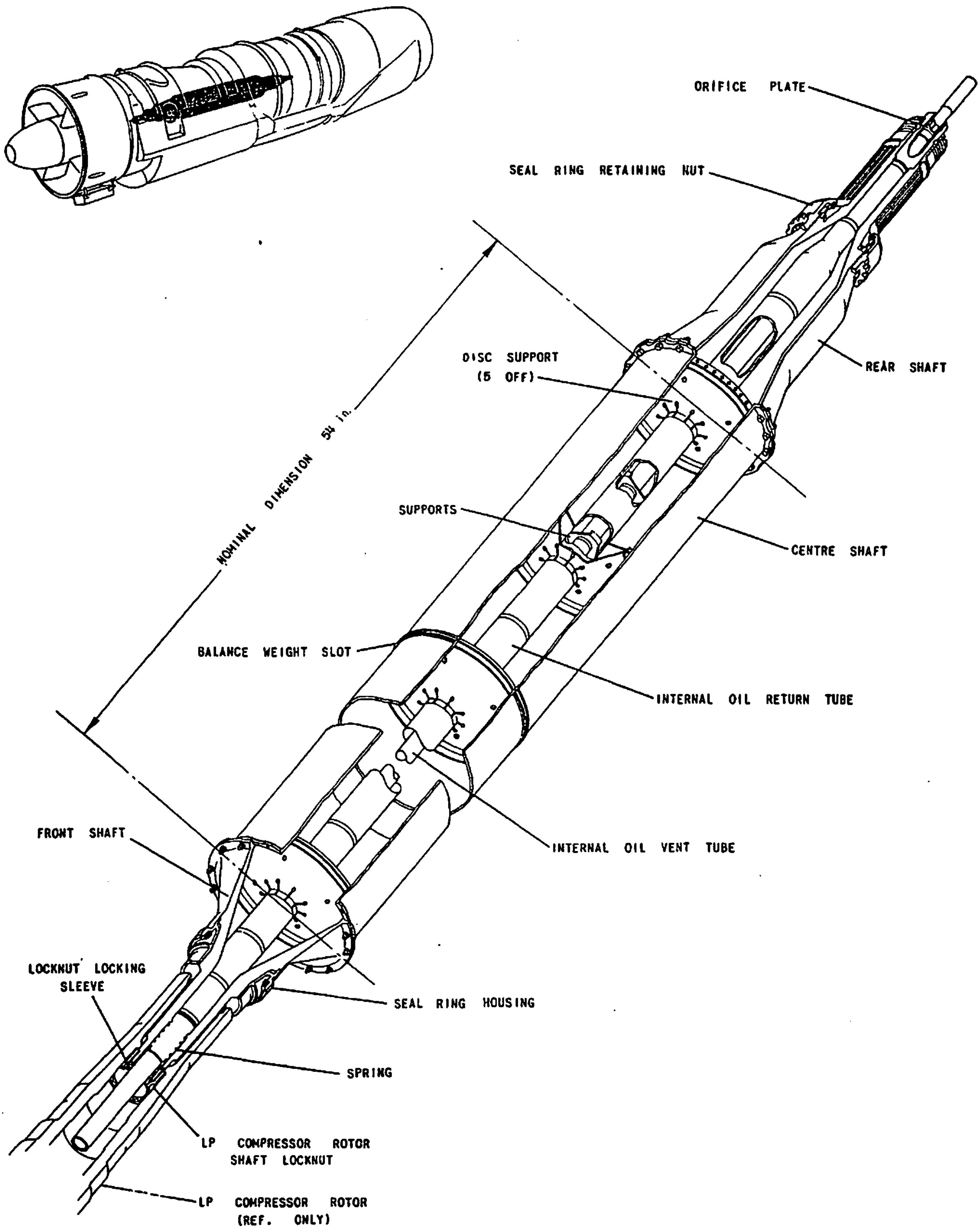


Figure 2-13 Low Pressure Mainshaft Assembly

LP MAINSHAFT

The LP mainshaft is in three sections. The shaft extends forward from the LP turbine rotor to drive the LP compressor rotor. Two concentrically mounted tubes within the shaft, namely the internal oil return tube and the internal oil vent tube, form part of the engine lubrication system.

The three sections are of nickel-plated steel, the front to centre section joint and the centre to rear section joint being located by machined spigots on the front and rear sections respectively; stainless steel bolts and nuts secure the joints. The LP compressor rotor shaft is splined to the front section of the main shaft and is locked in position by a locknut. The locknut is internally splined and is retained by a spring-loaded sleeve which slides forward to engage with the locknut splines. The LP turbine rotor shaft is splined to the rear section of the main shaft. Seal rings on the front and rear sections prevent the leakage of oil from the HP compressor rotor 'steady' bearing and the turbine bearing respectively.

The oil return tube is centrally located inside the mainshaft centre section by five stainless steel disc supports which are silver brazed to the outer diameter of the tube. The inner tube is supported within the outer tube by 12, three point location, 'Z' section, stainless steel supports which are welded to the inside diameter of the outer tube.

A mixture of air and lubricating oil in the form of froth is returned from the engine rear oil sump to the engine front oil sump via the annular space formed between the inner and outer tubes. The inner tube serves as an engine lubrication system air vent to the rear of the engine. An orifice plate, fitted to the rear end of the inner tube controls the air pressure drop throughout the engine oil scavenge system.

The LP mainshaft assembly and the LP turbine rotor are dynamically balanced as a complete assembly. Correction is made by balance weights at balancing planes located on the mainshaft, the rear face of the turbine disc and on the inner diameter of the turbine bearing outer seal ring.

TURBINE FRONT STATOR ASSEMBLY

The turbine front stator assembly is the structural connection between the mid frame and the turbine rear stator. It consists mainly of an outer casing, inner and outer supports for the turbine first stage stator segments, and a segmented shroud ring for the turbine first stage rotor. An air seal on the stator segment inner support permits a controlled flow of tenth stage cooling air (HP compressor delivery bleed air) to pass over the front face of the turbine first stage rotor disc. The turbine first stage stator blades increase the velocity of and direct the gas stream onto the turbine first stage rotor blades.

The outer casing is a stainless sheet steel fabrication with machined bolting flanges welded to the front and rear of the casing. The rear flange is stepped to provide support for the front of the turbine second stage stator segments in the rear stator assembly. Twelve holes in the step are fitted with shear pins which engage with slots in the second stage stator segments. In addition, twelve 'T' headed pins are provided on the flange for location of the second stage stator segments. A small spring pin in the rear bolting flange ensures correct positioning of the turbine rear stator during assembly.

The turbine stator segment outer support is a conical-shaped stainless sheet steel fabrication having a front bolting flange and rear section welded to it, the rear section being channelled to form the outer front support for the stator segments. Five holes are drilled through the channelled section at each segment location and these, in conjunction with five holes drilled at a slightly different spacing pitch in the corresponding segment flange, permit a vernier adjustment to be obtained. A securing pin is inserted through the holes showing the best alignment.

The turbine stator segment inner support is bolted to the rear flange of the mid frame inner casing, and serves as a support for the inner ends of the stator segments. A welded projection on the inner support provides radial support for the rear of the combustion chamber inner liner; holes in the projection allow a flow of cooling air over the segment inner platforms. The three-gland air seal ring which mates with the front air seal of the HP turbine rotor assembly, is riveted to the inner cylindrical portion of the support.

Each of the twelve first stage stator segments consists of four hollow aerofoil section blades nicro-brazed between an inner and outer platform. Circumferential ribs on the segment inner and outer platforms ensure an even distribution of the blade stresses. A lip, machined on the foremost rib of the outer platform, mates with and is secured to the channelled section of the stator segment outer support. The ends of the segment outer platforms overlap to permit expansion and to minimize the escape of hot gases between the joints. A projection on the front face of the segment outer platform provides a support for the rear end of the combustion

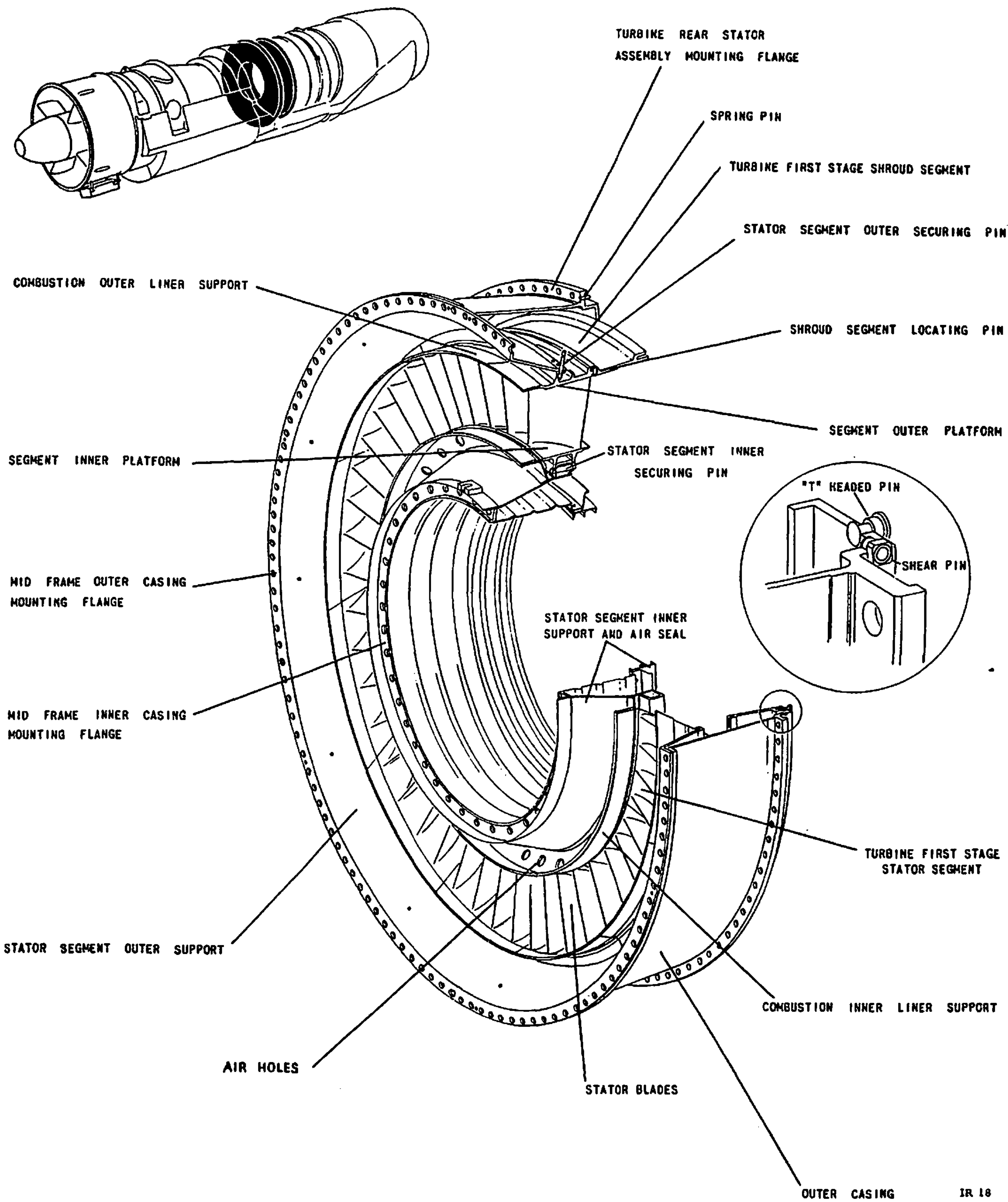


Figure 2-14 Turbine Front Stator Assembly

chamber outer liner. A spigot on the rear face of the outer platform supports the front of the turbine first stage shroud segments.

The ribs on the stator segment inner platform are slotted and fit over an axially drilled top-hat section welded to the stator segment inner support. A securing pin with flats on each end, locates in the top-hat section holes and engages with the slots in the inner platform ribs to retain the inner ends of the stator segments circumferentially while permitting free radial movement to allow for thermal expansion.

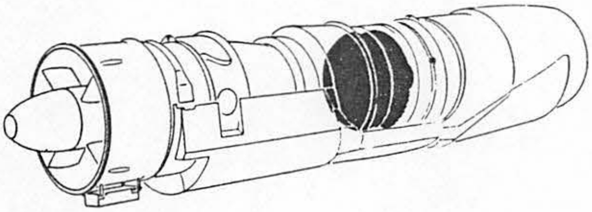
The twelve turbine first stage shroud segments are arranged circumferentially round the inner diameter of the casing immediately aft of the stator segments. As in the case of the stator segments, the shroud segments overlap each other, to allow for thermal expansion and contraction of the segments without serious effects on the turbine blade tip clearances. The front and rear faces of each shroud segment are channelled and mate with lips on the rear face of the stator segments and the front face of the outer casing rear flange. Circumferential displacement of the shroud segments is prevented by a locating pin passing through each stator/shroud segment joint.

HIGH PRESSURE TURBINE ROTOR ASSEMBLY

The HP turbine rotor assembly is of a two-stage design which comprises mainly two bladed rotor discs, separated by a spacer, and mounted on a stub shaft which is supported in the HP turbine bearing. The complete assembly is secured to the rear flange of the HP compressor rotor mainshaft by fifteen stainless steel tension bolts and self-locking nuts. The HP turbine rotor assembly absorbs the required proportion of power from the expanding gases passing from the combustion chamber and transmits it as torque to drive the HP compressor rotor and external accessories.

The turbine first and second stage blades are solid cast from heat resistant Inconel material. There are forty-seven blades in the first stage and fifty-three in the second stage. Each blade has a fir tree root for attachment to its respective disc, and an extended neck between the root and the blade platform. The blade tips of both stages are feathered to minimize damage should a tip rub develop. Conventional blade locking strips are used to prevent forward axial displacement of the first stage blades and rearward axial displacement of the second stage blades.

Nicro-brazed stainless steel air baffles are interposed between the extended necks of the blade roots. The baffles are designed to meter a flow of cooling air between the necks and hence reduce the heat transfer from the blades to the discs. Axial movement of the baffles is prevented by tangs which engage with slots in the rims of the turbine discs. The platform on each blade prevents radial displacement of the baffles under centrifugal loads.



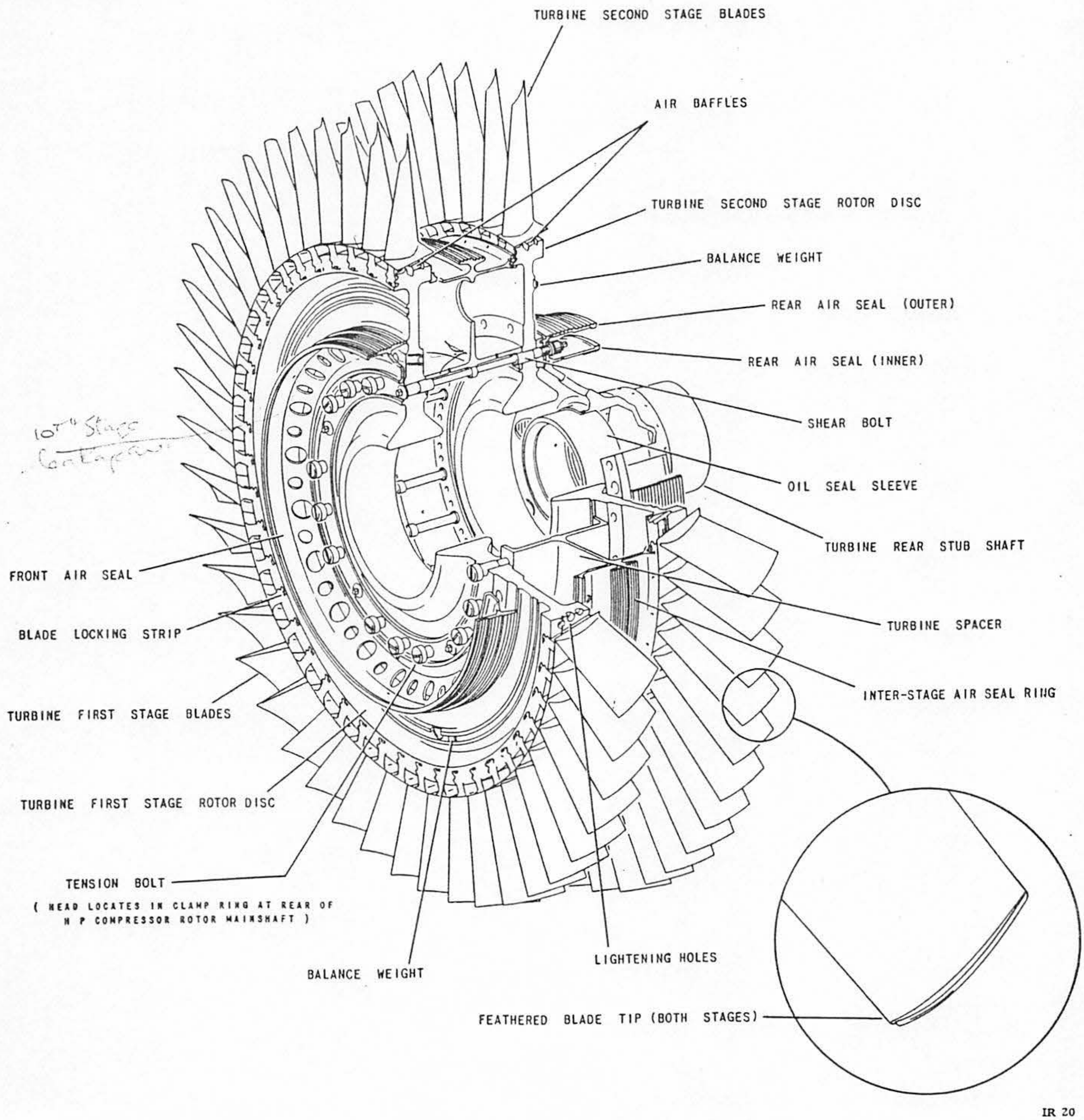


Figure 2-15 High Pressure Turbine Rotor Assembly

Both the first and second stage turbine discs are machined from heat resistant Inconel forgings to withstand the high operating temperature at the disc rims. The discs are of conventional design, and cooling air is passed through holes in each disc web. Lightening holes are drilled in the periphery of the first stage disc between each blade slot.

A spacer which is in the form of cylindrical inner and outer drums interconnected by an integral web is interposed between the first and second stage turbine discs. The outer drum locages between the rims of the discs and embodies an integral inter-stage air seal ring. The inner drum serves as a spacer shaft between the hubs of the discs. Holes in the web and outer drum provide a passage for cooling air through the inter-disc cavity. A cup-shaped flange at the rear of the outer drum butts against the rim of the second stage turbine disc and forms an annulus around the front face of the blade roots. The flange is drilled to allow cooling air to circulate through the annulus and cool the blade roots. In addition the outer drum portion of the spacer prevents rearward and forward axial displacement of the turbine first and second stage blades respectively.

The turbine rear shaft is a robustly designed, nickel plated steel forging. The rear external diameter is nitrided and ground to provide the inner race of the HP turbine bearing. Eight holes drilled in the forward conical section of the shaft provide a passage for cooling air. A finely ground oil seal sleeve is riveted to a spigot on the inner front section of the shaft and mates with the carbon seal ring on the rear section of the LP mainshaft assembly.

In addition to the inter-stage air seal on the turbine spacer, labyrinth-type seals are used at the front and rear of the rotor assembly to separate and apportion the cooling air flows through the turbine assemblies. The front air seal, which butts against the front face of the turbine first stage rotor disc, permits a controlled flow of tenth stage cooling air (HP compressor delivery bleed air) to pass radially outwards over the front face of the disc. Relatively large holes in the web of the front air seal allow the main flow of tenth stage cooling air to pass into the inter-disc cavity. The front air seal is retained in a channel on the rear flange of the HP compressor rotor mainshaft, rotational displacement of the seal being prevented by two lockpins which engage with slots in the mainshaft flange.

The inner and outer air seals, fitted at the rear of the turbine second stage rotor disc, are retained by the HP turbine rotor shear bolts. The inner rear seal separates the seventh and tenth stage cooling air flows; the outer seal provides a controlled flow of tenth stage air to the front face of the turbine third stage rotor disc.

With the exception of the front air seal, the components of the HP turbine rotor assembly are bolted together with five stainless steel shear bolts and high temperature self-locking nuts. The heads of the shear bolts locate in mating holes in the rear flange of the HP compressor rotor mainshaft. One bolt head, being of a larger diameter, ensures correct location of the complete turbine assembly on the mainshaft.

Balancing of the HP turbine rotor assembly is effected by the addition of weights installed in two circular grooves, one on the front face of the turbine first stage disc and the other on the rear face of the turbine second stage disc. The outer balance weights in each group are secured by peening.

LOW PRESSURE TURBINE ROTOR ASSEMBLY

The LP turbine rotor assembly is a single stage design comprising a bladed disc bolted to the LP turbine shaft which is supported by the turbine bearing assembly. The complete assembly, together with the HP turbine bearing, is splined to the LP mainshaft and is centralized at the front and rear by conical seatings. The assembly is retained by a locknut which seats against the rear face of the turbine shaft cone and locked by the tang of a steel lockring which engages with a hole in the LP mainshaft and a slot in the locknut.

The LP turbine rotor assembly absorbs power from the gases passing from the turbine second stage, and transmits it as torque, via the LP mainshaft, to drive the LP compressor rotor, the LP external gearbox, and the constant speed unit in the inlet frame assembly.

The turbine third stage rotor disc is machined from a stainless steel forging and has fifty-nine solid cast Inconel blades arranged around its outer periphery. The blades are retained in the disc rim by fir tree roots and are retained axially by a tanged locking strip, the tangs being bent outwards against the front and rear face of the disc. The blades have feathered tips to reduce the incidence of damage in the event of blade tip rub. Nicro-brazed stainless steel air baffles are fitted between the extended necks of the blade roots, the baffles being retained axially by tangs which engage in slots in the rim of the disc, and radially by the blade platforms. The function of these baffles is to meter cooling air through the extended necks of the blade roots to reduce the heat transfer from the blades to the disc.

A dovetail slot machined on the rear face of the rotor disc provides for the addition of balance weights during balancing operations. Using the same point of attachment to the LP turbine shaft as the disc, a labyrinth-type rear air seal extends to the rear and mates with a seal extending forward from the turbine bearing assembly. This seal provides a controlled flow of seventh stage cooling air over the rear face of the disc.

The LP turbine shaft is a nickel plated steel forging with the rear external diameter nitrided to form an integral inner race for the LP turbine bearing. On the

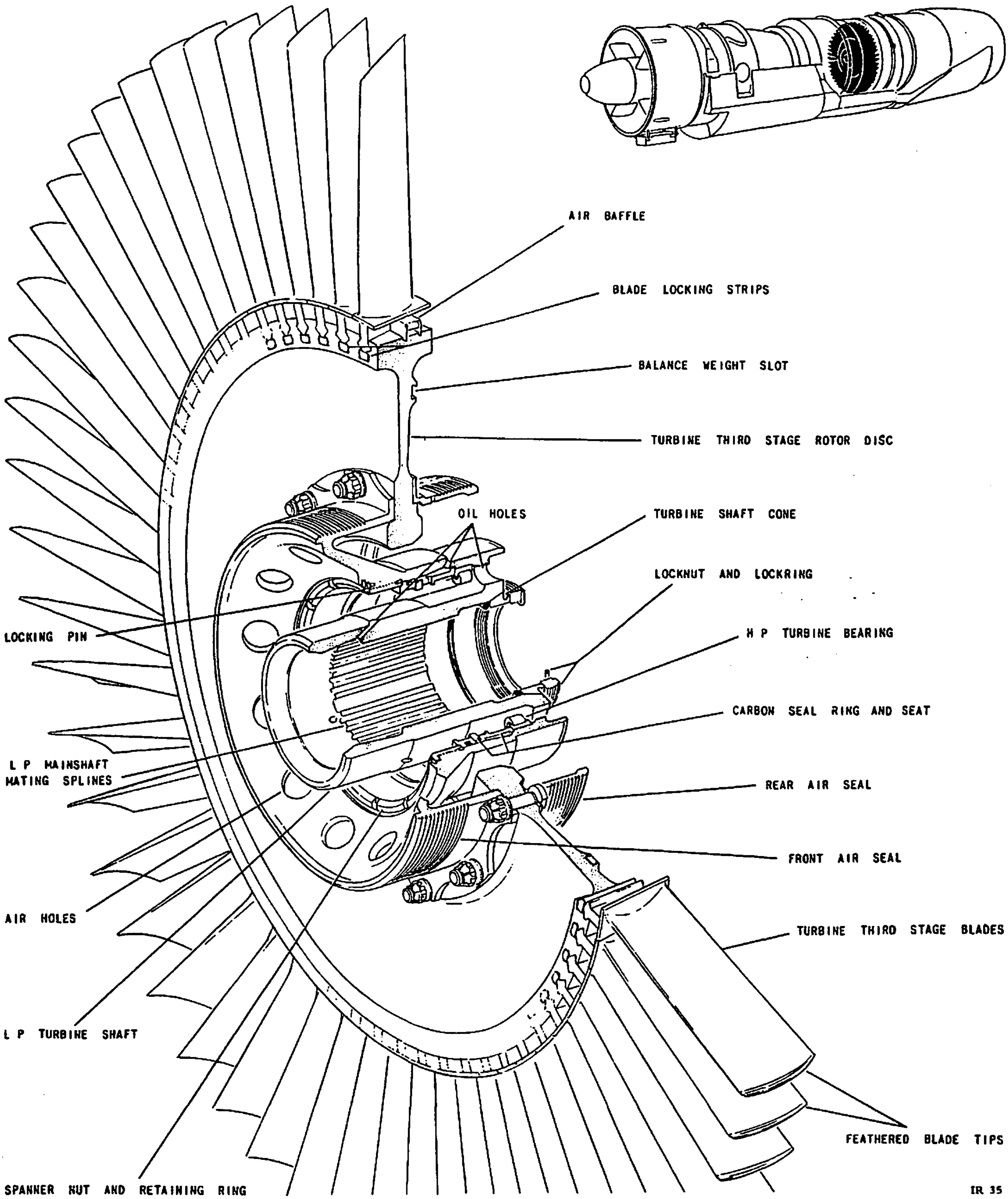


Figure 2-16 Low Pressure Turbine Rotor Assembly

outer periphery of the shaft, an integral labyrinth-type front air seal mates with the inner rear air seal of the HP turbine rotor assembly.

Housed within the reversed portion of the LP turbine shaft are the outer race and rollers of the HP turbine bearing. This bearing utilizes, when the engine is assembled, the hardened surface of the HP turbine shaft as an inner race. Also housed in the reversed portion of the LP turbine shaft is a carbon seal ring which, together with the HP turbine bearing, and carbon seal ring seat, is held in place by a spanner nut. A locking pin, which is inserted in one of the nut slots, locates in a drilling in the shaft, and is held by a retaining ring housed in a groove machined in the shaft.

TURBINE REAR STATOR ASSEMBLY

The turbine rear stator assembly is the structural connection between the turbine front stator assembly and the rear frame assembly. It consists mainly of a machined stainless steel outer casing, the turbine second and third stage stator segments, a turbine stator disc and seal assembly, and segmented shroud rings for the turbine second and third stage rotor blades.

Each of the twelve turbine second stage stator segments and sixteen turbine third stage stator segments consist of four hollow aerofoil section blades, nicro-brazed to inner and outer platforms. Each stage of stator segments is positioned circumferentially around the inner diameter of the outer casing. The ends of the segment outer platforms overlap to permit expansion and to minimize the escape of hot gases between the joints. Circumferential ribs on the segment inner and outer platforms ensure an even distribution of the blade stresses. The outer platforms of the turbine second stage stator segments are localed at the front by locating pins positioned around a step machined on the rear face of the turbine front stator casing; shear pins which engage with the slots in the front lip of the segment outer platforms and holes in the rear face of the turbine front stator casing, prevent circumferential displacement of the segments. A machined groove in the outer casing mates with and retains the rear lip of the segment outer platforms. Two sheet metal glands are riveted to the ribs on the stator segment inner platform and mate with the inter-stage air seal ring which is integral with the HP turbine rotor spacer disc.

The outer platforms of the turbine third stage segments have forward projecting lips machined on the front and rear faces; these lips engage with retaining grooves machined on the inner surface of the outer casing. Circumferential displacement of the turbine third stage stator segments is prevented by shear pins which pass through holes in the outer casing and engage with slots in the segment outer platform front lips.

The turbine second stage shroud segments and third stage shroud segments

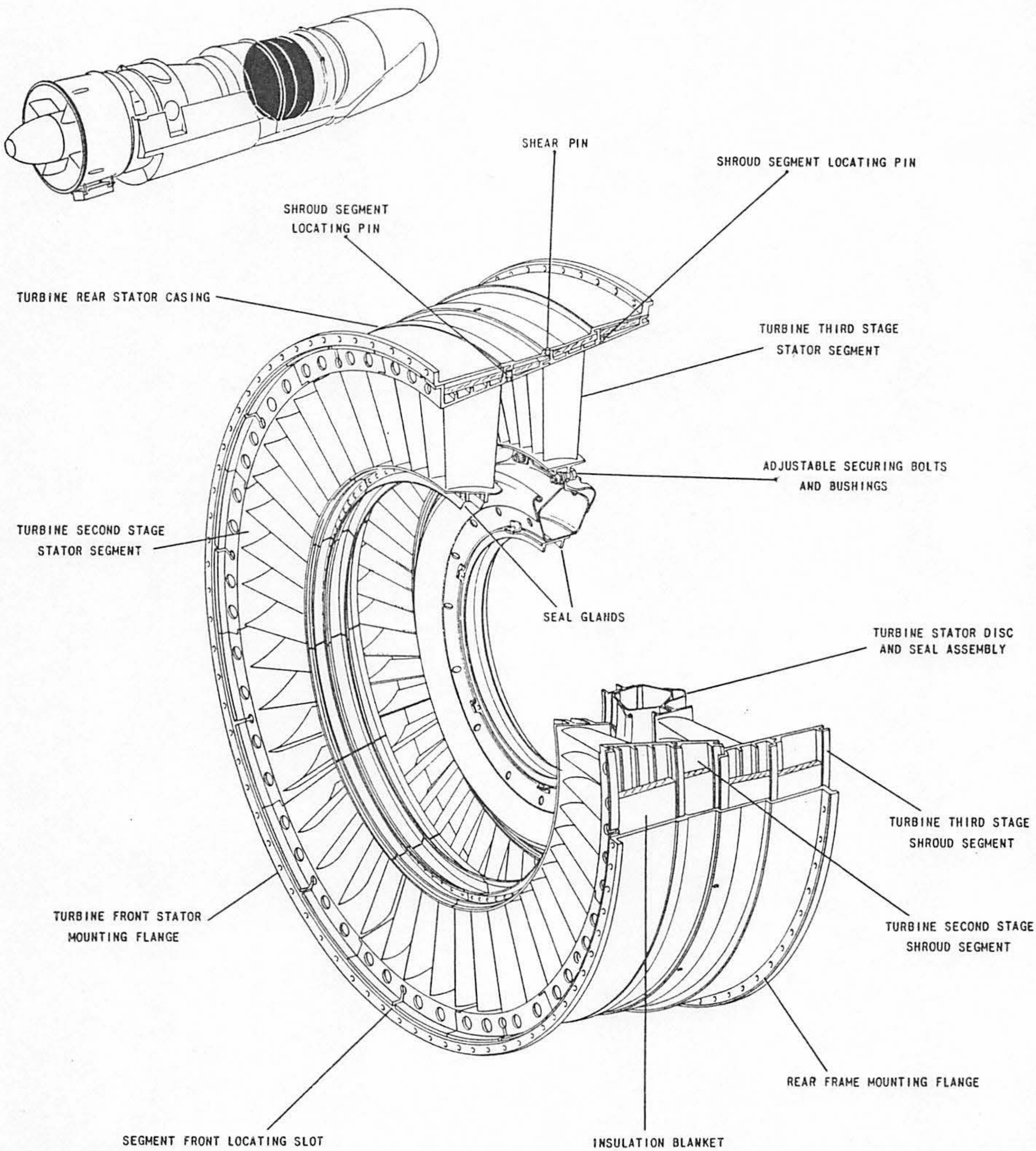


Figure 2-17 Turbine Rear Stator Assembly

are arranged circumferentially around the inner diameter of the casing immediately aft of their respective stator segments. The shroud segments overlap each other to allow for thermal expansion and contraction of the segments without serious effects on the turbine rotor blade tip clearances. The front and rear faces of each shroud segment are channelled; the second stage segment channels mate with lips on the front and rear faces of the turbine third and second stage stator segment outer platforms respectively; the third stage shroud segments locate on lips machined on the rear face of the turbine third stage stator segment outer platforms and on the front bolting flange of the rear frame assembly. Circumferential displacement of the shroud segments is prevented by a locating pin passing through each stator/shroud segment joint.

The turbine stator disc and seal assembly is bolted to the rearmost rib of each turbine third stage stator segment inner platform. True concentricity is obtained at the seal glands by means of adjustable eccentrically drilled hexagonal bushings. The seal glands are riveted to the stator disc inner diameter and mate with the outer rear air seal of the HP turbine rotor assembly. The stator disc is suitably contoured and ported, and in conjunction with the air seal, apportions the flow of tenth stage cooling air (HP compressor delivery bleed air) over the faces of the turbine second and third stage rotor discs and the third stage stator segment inner platforms.

Insulation blankets are fitted between the segments and the stator casing, to reduce thermal expansion of the casing and to minimize heat transfer to the aircraft structure.

L.P. TURBINE BEARING & OIL SUMP ASSEMBLY

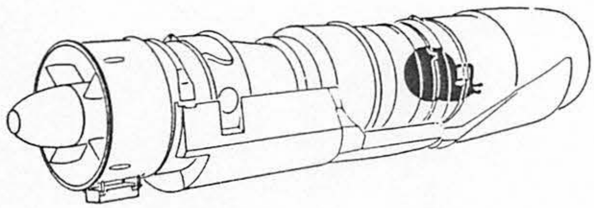
The LP turbine bearing assembly is mounted within the rear frame housing and comprises mainly the housing for the LP turbine bearing, and the engine rear sump. The assembly also embodies features for the distribution and scavenging of oil to and from the LP and HP turbine bearings.

Turbine Bearing Housing

The turbine bearing housing is fabricated from three steel parts welded together to form a mounting for the LP turbine bearing and the location for a large diameter carbon ring type oil seal. The part which houses the bearing is drilled and channelled at three equidistant points to accommodate the oil tubes which carry lubricating oil to the front face of the bearing. A machined flange on the oil seal housing, provides for attachment of the housing to the forward bolting flange on the inner diameter of the rear frame housing.

LP Turbine Bearing

The LP turbine bearing comprises a single row of crowned rollers spaced



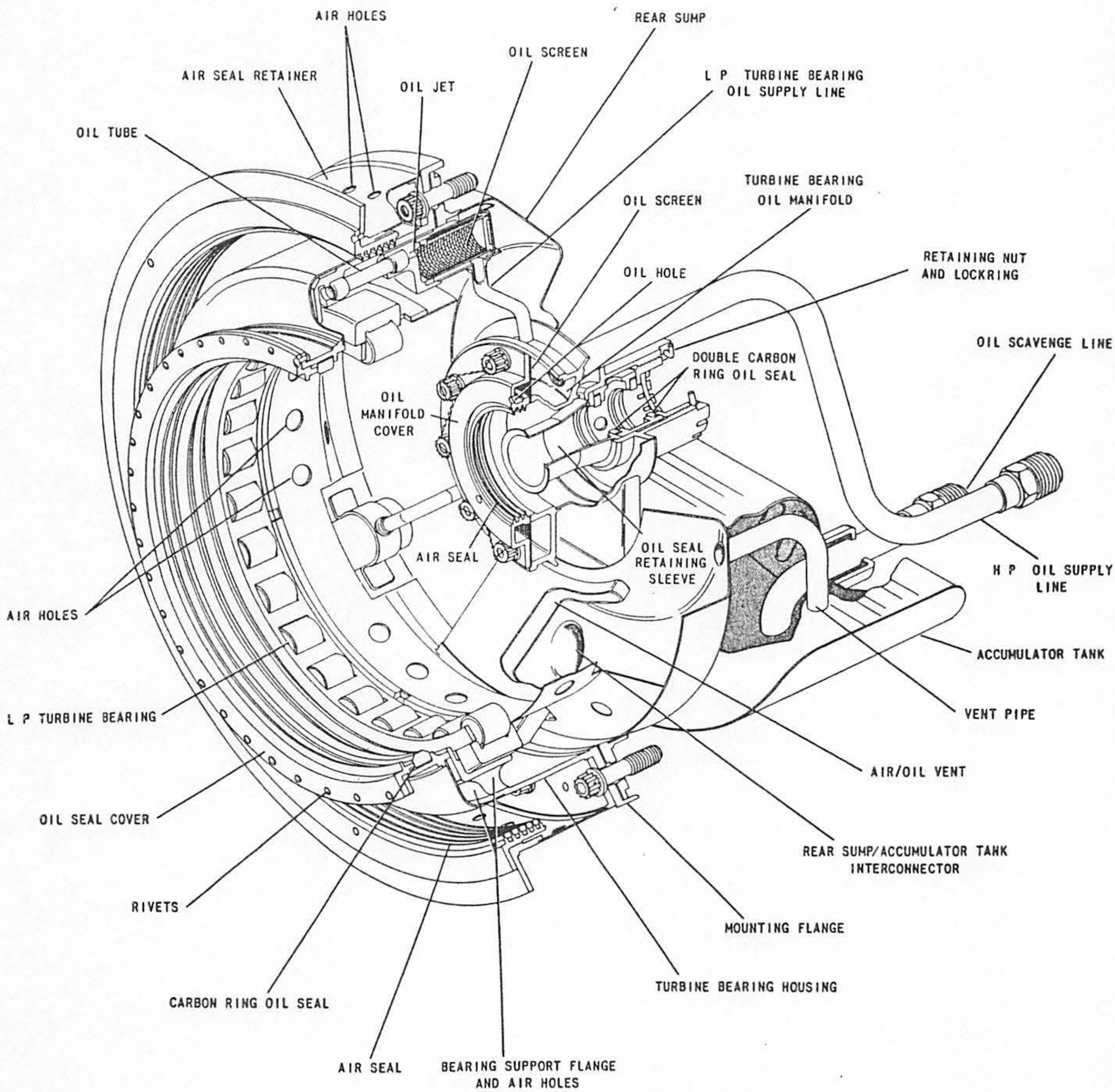


Figure 2-18 Low Pressure Turbine Bearing Assembly

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evenly in the bearing outer race by a silver-plated cage. The bearing outer race locates against a flange on the bearing housing and is retained in position by a flanged extension on the front of the rear sump. Air holes in the extension and the bearing outer race support flange permit a free circulation of cooling air around the bearing.

The LP turbine bearing supports the LP turbine shaft which houses the HP turbine bearing. The HP bearing supports the HP turbine rear stub shaft. The arrangement of mounting the bearings one inside the other in the same transverse plane, reduces the interaction of shaft vibrations.

Rear Sump and Oil Manifold

The rear sump is a welded stainless steel assembly which forms the rear cover of the turbine bearing chamber and serves as a collector for the scavenge oil from the bearings and the air/oil mist created by the rotation of the bearing rollers. The hub of the rear sump houses the rear end of the LP mainshaft internal vent tube and accommodates an annular turbine bearing oil manifold.

The turbine bearing oil manifold is supplied through a HP oil line which is connected to the inner section oil tubes in the rear frame bullet assembly. From the manifold, oil is supplied to the LP and HP turbine bearings in two separate flows. The oil flow to the LP bearing passes radially outward from the manifold through supply lines to three oil screen housings spaced evenly on the inside of the rear sump outer cylindrical portion. After passing through the screens, the oil flows through oil jets into the oil tubes which lead to the front face of the LP turbine bearing. The flow to the front face of the HP turbine bearing is delivered from the oil manifold through three screen-protected holes in the manifold cover and thence by centrifugal action through drillings and passages in the HP turbine shafting.

An interconnector at the bottom of the rear sump allows scavenge oil from the turbine bearings to drain into an accumulator tank which is welded to the lower rear face of the sump. The oil is scavenged from the accumulator tank through a scavenge line which is connected to the inner section oil tubes in the rear frame bullet assembly. A vent pipe between the top of the accumulator tank and the rear sump ensures proper scavenging from the tank in the event that the oil level in the sump rises above the rear sump/accumulator tank interconnector opening.

Air/oil mist in the bearing cavity is ducted from the bottom of the rear sump, through an air/oil vent, to the annulus formed between the rear sump hub and the internal vent tube. From ths annulus the air/oil mist is transferred to the internal oil return tube and thence to the front sump.

A machined flange welded to the outer cylindrical section of the rear sump is provided to secure the rear sump and turbine bearing housing to the forward flange of the rear frame housing.

Oil Seals

The large diameter carbon ring oil seal at the front of the turbine bearing housing seats on the reversed portion of the LP turbine shaft and, together with the oil seal rings on the LP turbine shaft and the rear shaft of the LP mainshaft assembly, seals the front of the rear sump. The seal ring is retained by an oil seal cover which is riveted to the front part of the bearing housing. A double carbon ring oil seal in the hub of the rear sump seats on the internal vent tube and is pressurized with compressor seventh stage air to complete the sealing of the rear sump. The double seal is retained in the hub by a retaining nut and lockring.

Air Seals

Two labyrinth type air seals are provided, one being a small diameter seal integral with the turbine bearing oil manifold cover, the other a large diameter seal which mates with the seal ring bolted on the rear face of the turbine third stage rotor disc. The small seal on the manifold cover seats against the internal oil return tube. The large seal is riveted to the forward end of a cylindrical sheet metal support which uses the same point of attachment as the bearing housing and rear sump. Holes in the seal support apportion the cooling flow of seventh stage air between the rear frame baffle and cover. The seal itself permits a controlled flow of seventh stage air to pass over and cool the rear face of the turbine third stage rotor disc.

REAR FRAME ASSEMBLY

The rear frame assembly is the main rear structural unit of the engine. It provides a suitable structure for attachment of the engine rear mounts and supports the bearings of both turbine rotors. The assembly consists mainly of an outer casing, a rear frame housing which forms the hub of the assembly, five struts, and the rear frame fairing.

Outer Casing

The outer casing comprises a cylindrical front section and a conical rear section which are welded to a channel section main support ring. The support ring provides three points of suspension, one at the top and one on each side, for mounting the rear of the engine. The top mounting is an aircraft fitting which is bolted through holes in the flanges of the support ring, while the side mounts, located above the horizontal centre line of the engine, are used for the attachment of the rear mounting struts. Each side mount comprises a rear mounting pin which is supported in lined drillings in the support ring flanges. Displacement of the mounting pin is prevented by a cotter pin which passes through holes in the head of the mounting pin and a retaining clip. The support ring also carries the outer ends of the rear frame housing support struts.

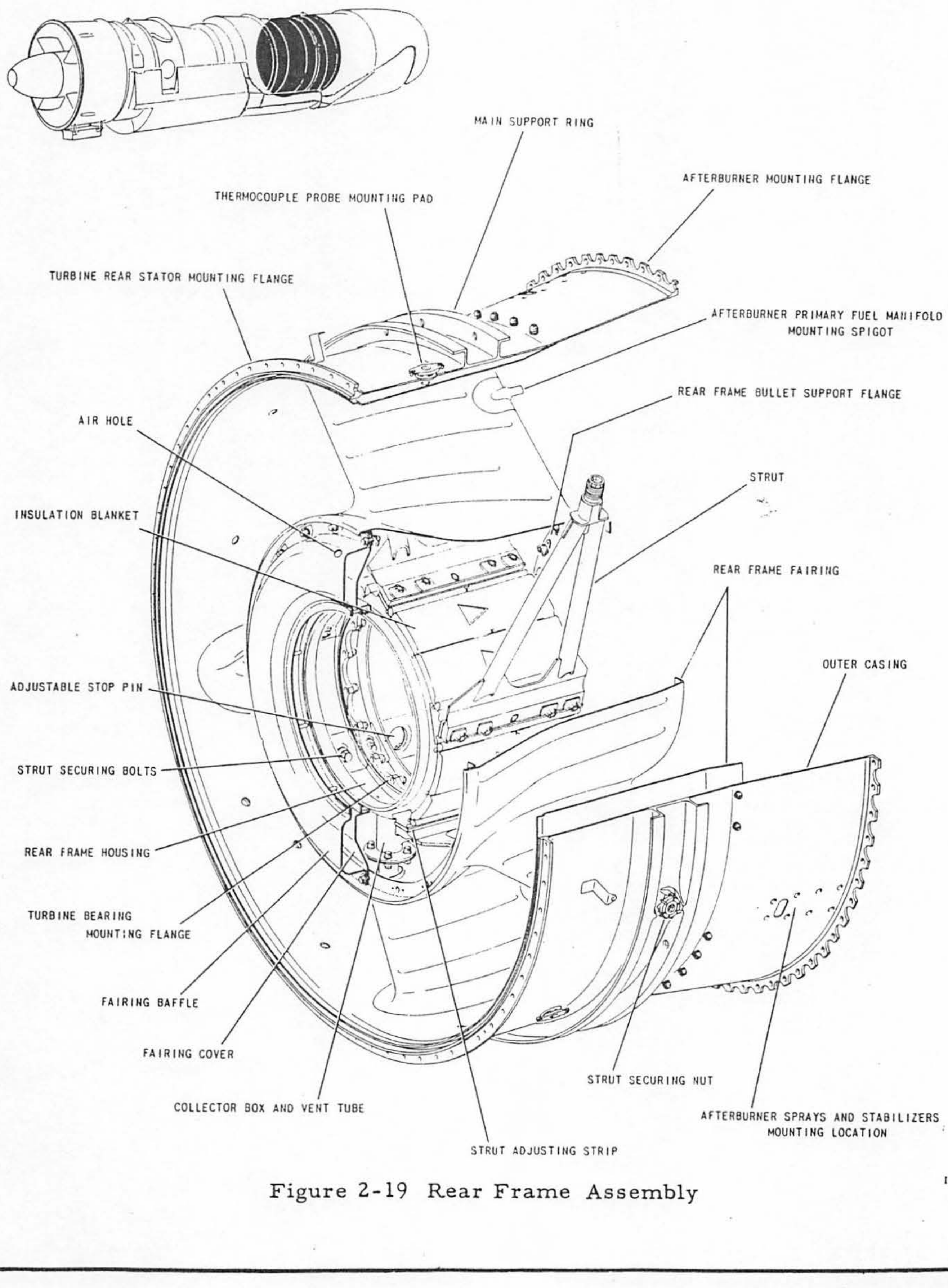

Figure 2-19 Rear Frame Assembly

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Machined bolting flanges welded to the front and rear of the outer casing, provide a mounting for the turbine rear stator assembly at the front and the afterburner assembly at the rear. A spigot on the front mounting flange supports the rear of the turbine third stage shroud segments when the turbine rear stator and rear frame assemblies are bolted together. Flanged pads spaced around the periphery of the casing front section provide a mounting for the eight probe, dual thermocouple system used for exhaust gas temperature measurement. Two additional pads on the lower right-hand side of the casing provide alternative mounting positions for the afterburner hot streak relay jet; a third pad at this location accommodates the P_7 pressure probe. At the bottom of the casing immediately aft of the main support ring are two bosses, the foremost of which accommodates the oil supply and scavenge lines for the turbine bearing assembly, the rearmost being a mounting for the fuel supply line to the afterburner primary fuel manifold. Provision is made at ten points on the rear section of the casing for attachment of the internally mounted afterburner spray and stabilizer group.

Rear Frame Housing

The rear frame housing encases and supports the turbine bearing assembly and associated sub-assemblies. The housing is a stainless steel machined forging, secured centrally and axially by the struts. The turbine bearing assembly is bolted to a flange machined on the inner diameter of the housing. A collector box for seepage oil and waste air is riveted to the lower outside surface of the housing. The collector box is vented to the gas stream through a vent pipe which is attached to the front of the box and passes through an opening in the fairing hub. Four insulation blankets, which serve as a heat shield for the turbine bearing assembly, are positioned and wirelocked around the periphery of the housing between the struts; the fifth space is occupied by the collector box.

Struts

The five struts are designed to transmit bearing loads to the main support ring, and are arranged to compensate for differential thermal expansion of the casting and rear frame housing. Each strut comprises a solid front leg and a tubular rear leg welded at their outer ends to a shouldered and threaded stub which forms the outer support for the strut, and is secured to the rear frame casing by a castellated nut locked by a cotter pin. The inner ends of the legs are welded to plates which are mounted tangentially on the rear frame housing. Serrations machined on the mating faces of the strut plate, an adjusting strip, and the rear frame housing, permit location of the housing concentrically within the rear frame casing. Elongated holes in the adjusting strip permit it to be moved axially along the bolts to obtain correct bedding of the mating serrations. The adjusting strips compensate for variations between the strut and housing due to tolerance stack-up or service distortion of the parts. An eccentric hexagonal-headed stop pin which is positioned through the housing, adjusting strip and strut plate, is held in position by a cover plate and a retaining ring. The pin absorbs axial loads between each strut and the housing.

Rear Frame Fairing

The rear frame fairing provides a smooth passage for the exhaust gases passing through the rear frame by streamlining the struts and smoothly contouring the inner surface of the casing and the outer surface of the rear frame housing. The five aerofoil-shaped vanes of the fairing form streamlined housings for the struts and are integrally formed with the hub of the fairing. The outer ends of the vanes are welded to the cylindrical outer portion of the fairing which is secured to the casing rear section by screws, locking nuts, and washers. The hub of the fairing is supported at the front by the fairing cover and the fairing baffle, which in turn are supported at their inner diameters by a spigot machined on the front face of the rear frame housing.

An opening at the bottom of the fairing hub accommodates the vent pipe from the rear frame collector box. Holes in the outer cylindrical portion of the fairing provide openings through which pass the thermocouple probes, the outer ends of the struts, the turbine bearing oil supply and scavenge lines, the afterburner hot streak relay jet, and the P probe. A boss is welded on the trailing edge of each vane to support the afterburner primary fuel manifold. Welded around the front flange and riveted to the rear lip of the fairing hub is a series of self-locking anchor nuts. The nuts are used for attachment of the front cover and baffle, and the rear frame bullet assembly respectively. The fairing is strengthened by embossed fluting on the surfaces of the vanes and hub.

REAR FRAME BULLET ASSEMBLY

The rear frame bullet assembly is located at the rear of the inner housing of the rear frame. The main function of the assembly is to provide a smoothly transitioned passage for the engine exhaust gases leaving the rear frame, and to seal the rear end of the rear frame hub cavity into which the internal air vent tube exhausts. In addition, the bullet assembly forms the rear supporting member of the rear frame fairing and houses the inner section of the rear bearing oil tubes. The assembly comprises mainly a fairing support cylinder and ring, a rear frame housing cover, a bullet, and the inner and outer sections of the rear bearing oil tubes.

Fairing Support Cylinder and Ring

The fairing support cylinder is a stainless sheet steel fabrication with machined flanges welded to the front and rear faces, the front flange being used to attach the fairing support cylinder to the rear bolting flange of the rear frame housing. The fairing support ring consists of an angled circular collar, which locates in the extended portion of the fairing support cylinder rear flange. Together with the rear

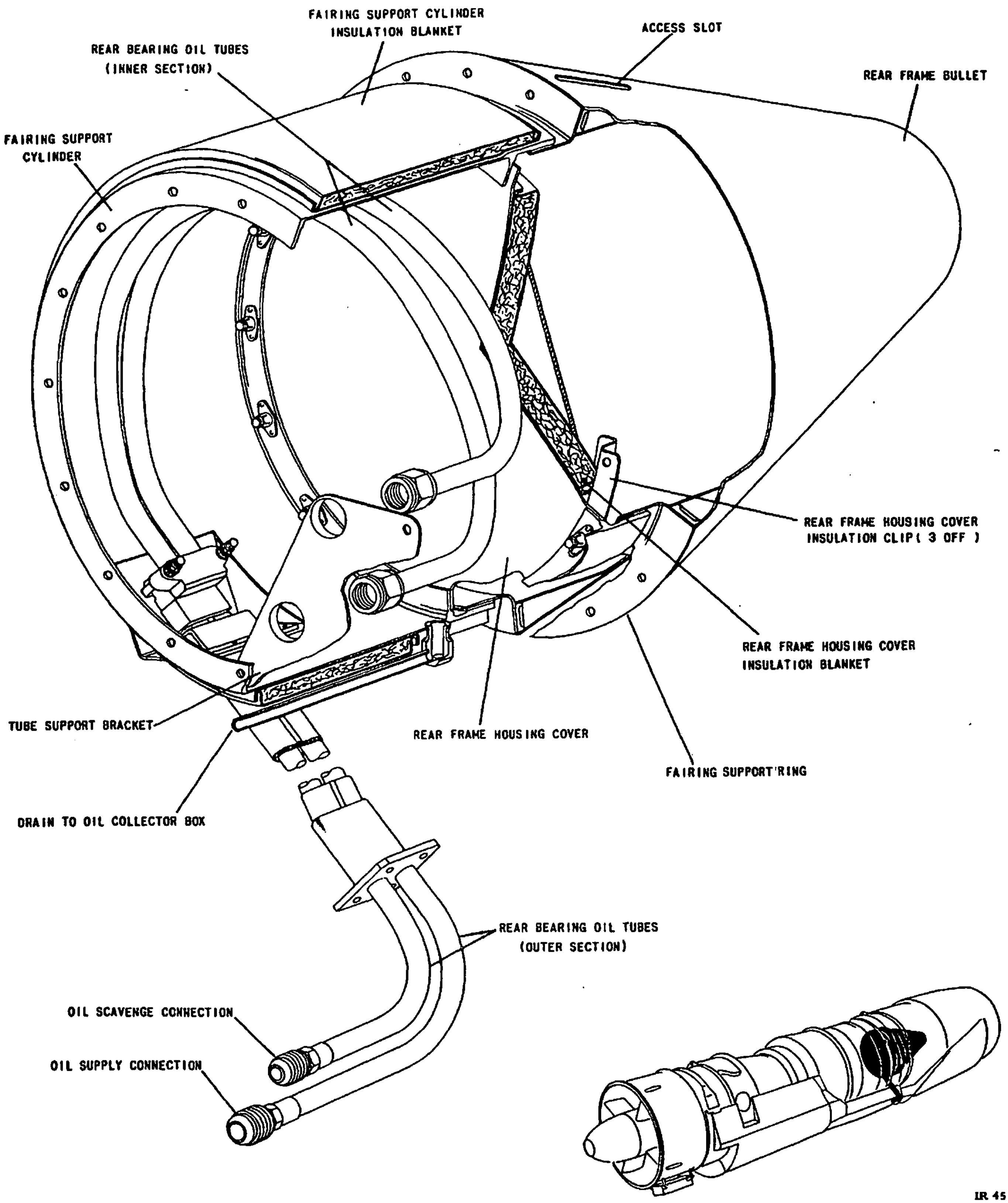

Figure 2-20 Rear Frame Bullet Assembly

frame bullet, the support ring is bolted to the inward projecting rear flange of the rear frame fairing. An insulation blanket is wrapped around the periphery of the fairing support cylinder to minimize the transfer of heat to the rear frame hub cavity. A drain, located at the bottom of the fairing support cylinder and immediately forward of the rear flange, removes any residual oil which tends to collect in the rear frame hub cavity. The oil is drained forward into a collector box in the rear frame, from which it is vented into the engine exhaust stream.

Rear Frame Housing Cover

The rear frame housing cover, comprising a concave stainless steel plate, is bolted to the inward projecting rear flange of the fairing support cylinder and is covered on the rear face by an insulation blanket. The rear frame cover and insulation blanket separate the rear frame hub cavity from the relatively high temperatures inside the rear frame bullet. The insulation blanket consists of a pad of insulation sandwiched between two layers of Inconel foil and is positioned by three equi-spaced insulation clips which are bolted to the rear face of the cover. The blanket is firmly retained against the rear frame housing cover by locking wire which passes between the three insulation clips.

Rear Frame Bullet

The rear frame bullet is a conical steel pressing with a mounting flange welded to the front face for bolting the bullet to the rear flange of the rear frame fairing. Access to the mounting bolt heads is gained through slots in the bullet skin immediately behind the front face. An inner skin adjacent to the slots minimizes the entry of hot exhaust gases into the bullet cavity. A small vent hole in the inner skin maintains equal gas pressure loads on each side of the bullet skin.

Rear Bearing Oil Tubes

The rear bearing oil supply and scavenge tubes both consist of an outer section which passes inward from two external connections, through the bottom vane of the rear frame, and an inner section which is housed within the fairing support cylinder. The outer section is secured to a mounting boss at the bottom of the rear frame outer casing and terminates at a junction on the bottom of the support cylinder. The inner section passes from the junction and terminates immediately behind a bracket which supports the oil tube connections of the turbine bearing assembly.

AFTERBURNER ASSEMBLY

General

Afterburning is a method of thrust augmentation used primarily to improve the performance of an aircraft under take-off, climb or combat conditions. Since only a proportion of the total air entering the combustion chamber is used in the engine combustion cycle, a considerable amount of axygen is available in the exhaust gas stream to assist further combustion. Fuel is injected and burned in the exhaust gas stream at a point between the turbine and the final nozzle thus increasing the temperature and exit velocity of the exhaust gases which results in an increase in thrust. The mixture must be easy to ignite under all flight conditions and altitudes, and a stable flame which will burn steadily over a wide range of mixture strengths and gas flows is necessary to ensure smooth and efficient operation of the afterburner.

The Iroquois afterburner assembly is integral with the engine and is designed to fulfil the general requirements detailed in the preceding paragraph. In addition to the final nozzle, spray, and stabilizer group, the afterburner consists of a casing, an anti-screech liner, and a shroud. A hot streak ignition system is used to start combustion of the afterburner fuel flow.

Due to the increased volume of exhaust gases when the afterburner is in operation, an increase in final nozzle area is required to avoid excessive exhaust gas temperature and high turbine back pressure. To meet this and other requirements, the engine is equipped with a variable area final nozzle, operated by four automatically controlled hydraulic actuators. The fuel supply to the afterburner is also automatically controlled to provide full thrust modulation over the entire range of afterburner operation. Details of the final nozzle control system and afterburner fuel control system will be issued at a later date.

Afterburner Casing

The afterburner casing is a hollow cylinder of stainless steel sheet which extends rearward from the engine rear frame assembly. The casing is fabricated from five circular sections which are overlapped and welded together. The divergent conical front section of the casing has a machined bolting flange welded to the forward edge for attachment of the complete afterburner assembly to the engine rear frame. The convergent conical rear section of the casing has a circular hinge ring welded to its rear edge. A contoured groove machined on the rear face of the hinge ring accommodates the forward ends of the variable nozzle segments. At approximately the centre of the afterburner casing, a circle of shroud spacers is welded to the outside of the casing for attaching the forward end of an afterburner shroud. The stainless steel spacers have a hexagonal section, and are designed to accommodate the differential thermal expansion between the afterburner casing and the shroud, and to transmit the axial loads of the final nozzle hydraulic actuators attached to the rear of the shroud. The front portion of the afterburner casing is covered by insulation blankets which minimize heat radiation to the adjacent airframe structure.

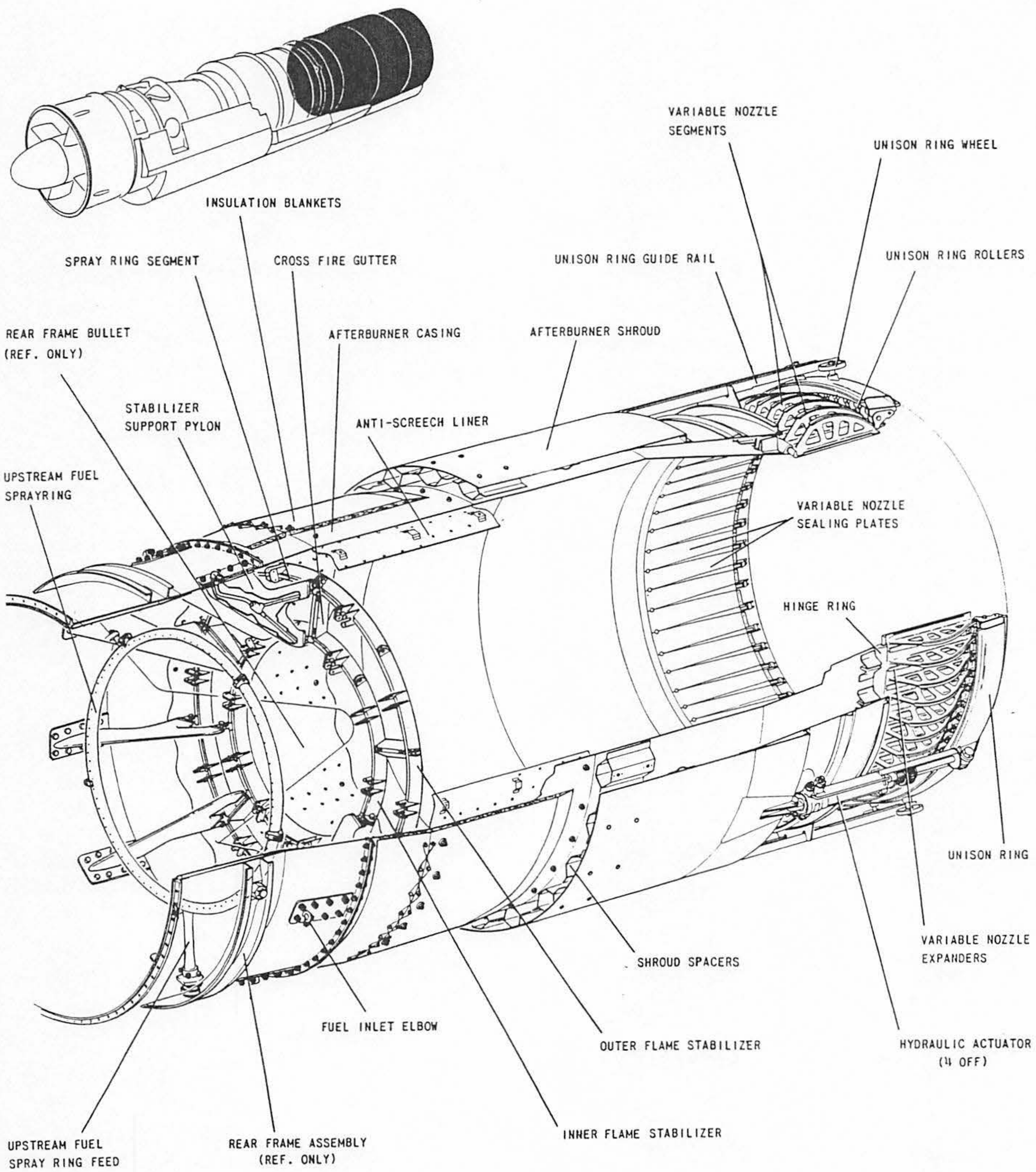

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Figure 2-21 Afterburner Assembly

Anti-screech Liner

Forward of the mounting spacers, two circles of holes around the casing accommodate the attaching bolts of an anti-screech liner. The anti-screech liner comprises a perforated stainless steel sheet with axial corrugations, and is located round the inner wall of the afterburner casing. The liner prevents high amplitude resonant vibrations in the afterburner casing by absorbing and dissipating the high frequency gas pressure fluctuations which originate in the region of the flame stabilizer during afterburner operation.

Afterburner Shroud

The afterburner shroud encases the rear half of the afterburner casing, and is fabricated from circular spotwelded sections of stainless steel sheet. The front of the shroud is bolted to hexagonal shroud spacers fitted to the afterburner casing, whilst at the rear, the shroud is supported by a circle of guides which engage with 'M' section spacers welded on the rear of the afterburner casing forward of the final nozzle. The guides permit free axial displacement of the rear section of the shroud resulting from the differential thermal expansion of the shroud and casing.

A flow of relatively cool air is directed over the exterior of the afterburner casing by the shroud. The cooling air flow dissipates and carries away heat from the afterburner casing and thus maintains the temperature of the casing at a safe value during afterburner operation. From the shroud, the air is then directed over the variable nozzle segments to cool the final nozzle outer surfaces and reduce heat radiation to the final nozzle actuators.

Final Nozzle

The final nozzle, fitted to the rear of the afterburner casing, is of the segmented, variable area type. The final nozzle area is varied by four hydraulic actuators, the forward ends of which are retained by fork-ends at the rear of the afterburner shroud. The actuators are equi-spaced round the final nozzle at 45 degrees from the vertical, and are designed for operation in the relatively high ambient temperatures at the final nozzle. The rear ends of the actuators are eye-bolted to a unison ring. Four double-sided roller guide rails, welded to the rear of the afterburner shroud and equally spaced between the actuators, accommodate four pairs of rollers that are mounted on and carry the unison ring. The rollers ensure that the unison ring is centrally located during fore and aft movements.

The unison ring is a rectangular sectioned ring, fabricated from stainless steel sheet. Four brackets, spaced at 45 degrees from the vertical on the outer diameter of the unison ring, mate with eyebolts on the rear of the actuators. Inward projecting brackets, welded to the inner face of the unison ring, carry sixty stainless steel unison ring rollers which bear against hardened cam faces on the outer edges of the final nozzle segments.

The forward ends of the sixty cast stainless steel final nozzle segments are located round the rear of the afterburner casing in the contoured groove of the hinge ring. A braided Inconel tape is spotwelded in the hinge ring groove and locates the segment ends securely while cushioning the movement of the segments and forming a gas tight seal. Sealing plates and flexible metal expanders are interposed between the nozzle segments to form a closely knit, gas tight, fully expandable nozzle. Insulation is packed in each expander and is retained in position by covers which are pinned to the rear ends of the sealing plates.

The final nozzle area is varied by the axial movement of the unison ring which causes the unison ring rollers to move along the segment cam faces and pivot the segments about the hinge ring. During actuation, the unison ring is held normal to the final nozzle axis by four flexible cables which interconnect the hydraulic actuators. A worm gear arrangement in each actuator drives the flexible cables and ensures synchronous movement of the actuators and hence correct axial positioning of the unison ring.

Spray and Stabilizer Group

The spray and stabilizer group consists of ten stabilizer support pylons, an inner and an outer stabilizer and segmented spray ring, and an upstream fuel spray ring. The upstream fuel spray ring is an elliptical sectioned circular stainless steel tube designed to deliver a large proportion of the fuel supply to the stabilizers. Drilled orifices in the leading edge of the spray ring are angled alternately inward and outward so that the fuel spray is broken into a wide band of small droplets which avoids overloading of the stabilizers with raw fuel and hence improves the rich limit characteristics of the afterburner and reduces the possibility of flame out. Brackets on the forward face of the spray ring are bolted to a spigot on the trailing edge of each rear frame vane. The spray ring has a single feed which is slightly offset from the vertical at the bottom of the rear frame outer casing.

The stabilizer support pylons, are equally spaced round the inner diameter of the engine rear frame. The outer ends of the pylons are bolted to the engine rear frame outer casing immediately forward of the rear flange. The 'Y' shaped pylons are of stainless steel sheet with an aerofoil section and extend inward and rearward into the inlet of the afterburner assembly. A fuel inlet elbow on the outer end of each pylon transfers fuel to a 'Y' shaped pipe located inside the pylon. The aft ends of the pipes are welded to inner and outer spray ring segments. The segments are made from short lengths of stainless steel tubing which, when assembled in position, form two segmented fuel spray rings. Equi-spaced fuel orifices are drilled in the downstream face of the spray ring segments.

Two flame stabilizers are fitted immediately downstream of the segmented spray rings, the outer stabilizer being supported by all ten pylons, the inner stabilizer by five of the pylons. Both stabilizers are of stainless steel to withstand oxidation and distortion, and have a 'V' cross-section, the apex of which points upstream. A circular disced groove in the apex of each stabilizer collects the fuel from the spray ring segments and evenly distributes it over the stabilizer regardless of any slight distortion or displacements of the spray rings relative to the stabilizer. Five radially disposed, stainless steel cross fire gutters with a cross section similar to that of the flame stabilizers, are equally spaced between the inner and outer stabilizers to assist in an even distribution of flame during afterburner operation.

Afterburner Ignition System

The hot streak method is used to light up the afterburner. A hot streak igniter valve operates automatically on commencement of the light up sequence, and introduces intermittent pulses of metered fuel into the primary combustion zone of the engine combustion chamber. The injected fuel ignites and produces a core of extremely hot gases which extends through the HP and LP turbine sections and ignites the afterburner fuel at the flame stabilizers. The core of hot gases is augmented by additional fuel injected through a relay jet located on the engine rear frame outer casing. Full details of the operation of the hot streak igniter valve will be issued later in Section 2 of this part.

LOW PRESSURE EXTERNAL GEARBOX ASSEMBLY

The LP external gearbox assembly is located on a mounting pad at the outer end of the front frame No.6 strut, the drive to the gearbox being supplied by shafting from the internal gearbox. The cast magnesium gearbox casing is T shaped in section with a flange at each end of the T head for mounting the LP tachometer generator and the LP speed governor. The tachometer generator and the speed governor measure and limit respectively the rpm of the LP compressor rotor.

The drive is transmitted from a radially positioned shaft to one parallel to the engine axis by means of a pair of case hardened, phosphated, steel, bevel gears. The bevel drive gear on the radial axis is centrally located by two ball bearings in a flanged, steel cartridge, which in turn is retained in the gearbox casing by two machine screws. The screws also locate and retain the steel shim between the flanges of the gearbox casing and the bearing cartridge.

The driven gear is dowelled to a hollow gearshaft and meshes with the drive gear on the radial axis. The shaft and gear combination is mounted in two ball bearings located in bearing cartridges in the gearbox casing. The front and rear ends of the driven gearshaft are machined to accommodate the square-ended driveshafts of the LP tachometer generator and the LP speed governor respectively.

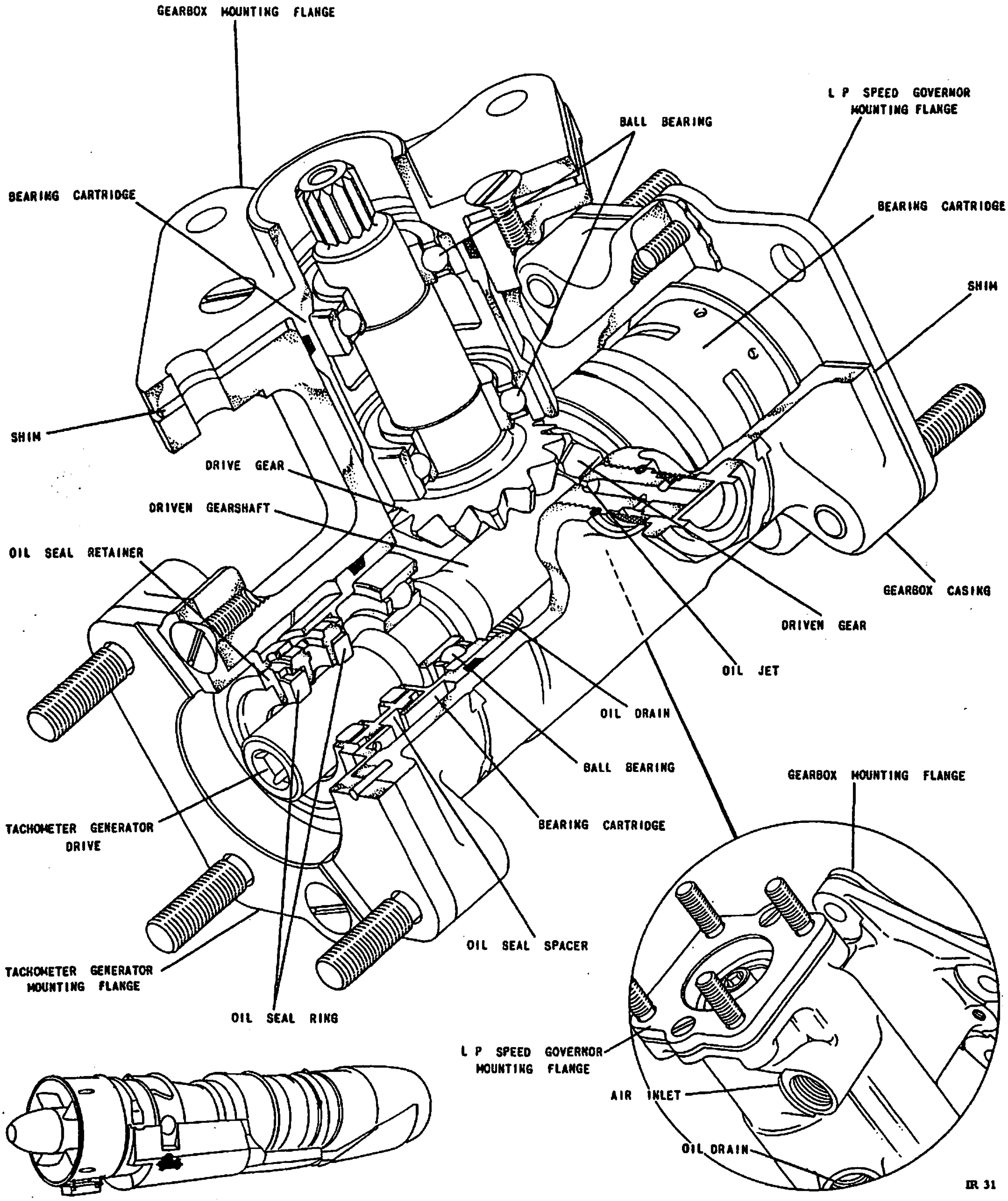


Figure 2-22 Low Pressure External Gearbox Assembly

Lubrication of the bevel gears is by a screened oil jet positioned in the outer casing, which supplies a spray of oil to the contact point of the gears. The ball bearings in the gearbox are splash lubricated. External piping delivers pressure oil to the gearbox and carries scavenge oil to the HP external gearbox sump. A pair of oil seal rings, separated by a steel spacer, are fitted at each end of the driven gearshaft to prevent oil seepage from the gearbox. The cavity between the oil seal rings is pressurized with HP compressor air supplied through an external fitting and channelling in the gearbox casing.

HIGH PRESSURE EXTERNAL GEARBOX

General

The HP external gearbox assembly is located on the mounting pad at the outer end of No. 5 strut in the front frame casing. When viewed from the rear, the gearbox follows the contour of the front frame, through approximately 40 degrees, up the right-hand side of the engine. The drive for the gearbox is transmitted from the HP compressor front stub shaft to the HP gear train in the internal gearbox assembly and then by a quillshaft to the external gearbox. The assembly consists of a gearbox casing and front cover, together with the gear train required to drive an aircraft power take-off assembly and engine auxiliary components mounted on the gearbox.

The front face of the gearbox has three pads which accommodate the aircraft power take-off assembly, the engine oil scavenge pump, and the HP speed governor. The rear face has four mounting pads to which are mounted the engine starter, the hydraulic pump, the oil lubrication pump, and the tachometer generator and HP speed senser.

Gearbox Casing and Front Cover

The gearbox is a crescent shaped magnesium casting with drillings to supply lubricating oil to the gears and oil jets. Forward of the rear face, an integral inner wall houses the bearing cartridges supporting the rear ends of the gearshafts. The cavity formed between the rear face and the inner wall is sealed and serves as an air gallery for compressor seventh stage air which is used to pressurize the oil seals and bearings. The front cover, which is bolted to the front face of the gearbox with an 'O' ring interposed in the joint, is also of cast magnesium. The cover supports the forward end bearings of the hydraulic pump gear, the scavenge pump gear and the lubrication pump gear, and provides a mounting flange for the oil scavenge pump. When the gearbox is installed on the engine it is supported by a bracket fitted on a mounting boss at the upper right-hand end of the casing; the bracket in turn is secured to the front frame front mounting flange. Four bosses located on the left-hand side of the casing below the gearbox mounting face accommodate the engine oil temperature regulator mounting brackets. The mounting pads for the engine auxiliaries are machined so that their axes are parallel to the engine axis.

Input Bevel Gear Assembly

The input bevel gear assembly consists of a forged steel bevel gear supported by two single row angular contact bearings installed in bearing mountings. The upper mounting is screwed into the lower mounting, and retained in position by a tanged, steel locking ring. The lower mounting is retained in position on the gearbox mounting face by two fillister head screws.
Po

Power Take-off Input Shaft and Bearing Cartridge

The power take-off input shaft assembly comprises three concentrically mounted shafts with integral gears supported by two ball bearings housed in a flanged steel cartridge. The bearing cartridge mounting flange is fitted between the gearbox casing and the mounting flange of the aircraft power take-off casing with the forward end of the cartridge extending into the aircraft power take-off casing. The three gears comprise an auxiliary drive pinion, an input bevel pinion, and a power take-off bevel pinion. The input bevel gear meshes with the power take-off bevel pinion, the shaft of which extends forward and is supported by a ball bearing at each end. The power take-off bevel pinion shaft is hollow with rows of lightening holes drilled round the circumference. Internal splines at the rear of the shaft, inside the pinion, mate with splines on the input bevel pinion shaft and the auxiliary drive pinion shaft.

The input bevel pinion shaft is hollow and drilled with lightening holes. A close tolerance diameter at the rear of the input bevel pinion teeth locates the shaft centrally inside the forward end of the power take-off bevel pinion shaft. The input bevel pinion delivers the drive to the aircraft power take-off output bevel gear.

The auxiliary drive pinion shaft is the inner member of the three concentrically mounted shafts. It has a spur gear at the rear; threads on the forward end accommodate a locknut and washer to secure the three shafts together. Internal splines at the rear of the shaft mate with the splines on the forward end of the starter shaft.

Aircraft Power Take-off Assembly

The aircraft power take-off assembly is bolted to the front face of the mounting pad on the front of the HP external gearbox, see Figure 2-23. The assembly consists mainly of a casing, an output shaft assembly, an output safety shaft, an oil seal housing, a power take-off drive oil seal and an output shaft access plug.

The power take-off casing is of cast magnesium, is roughly 'V' shaped in section and is installed on the HP external gearbox in either one of two positions,

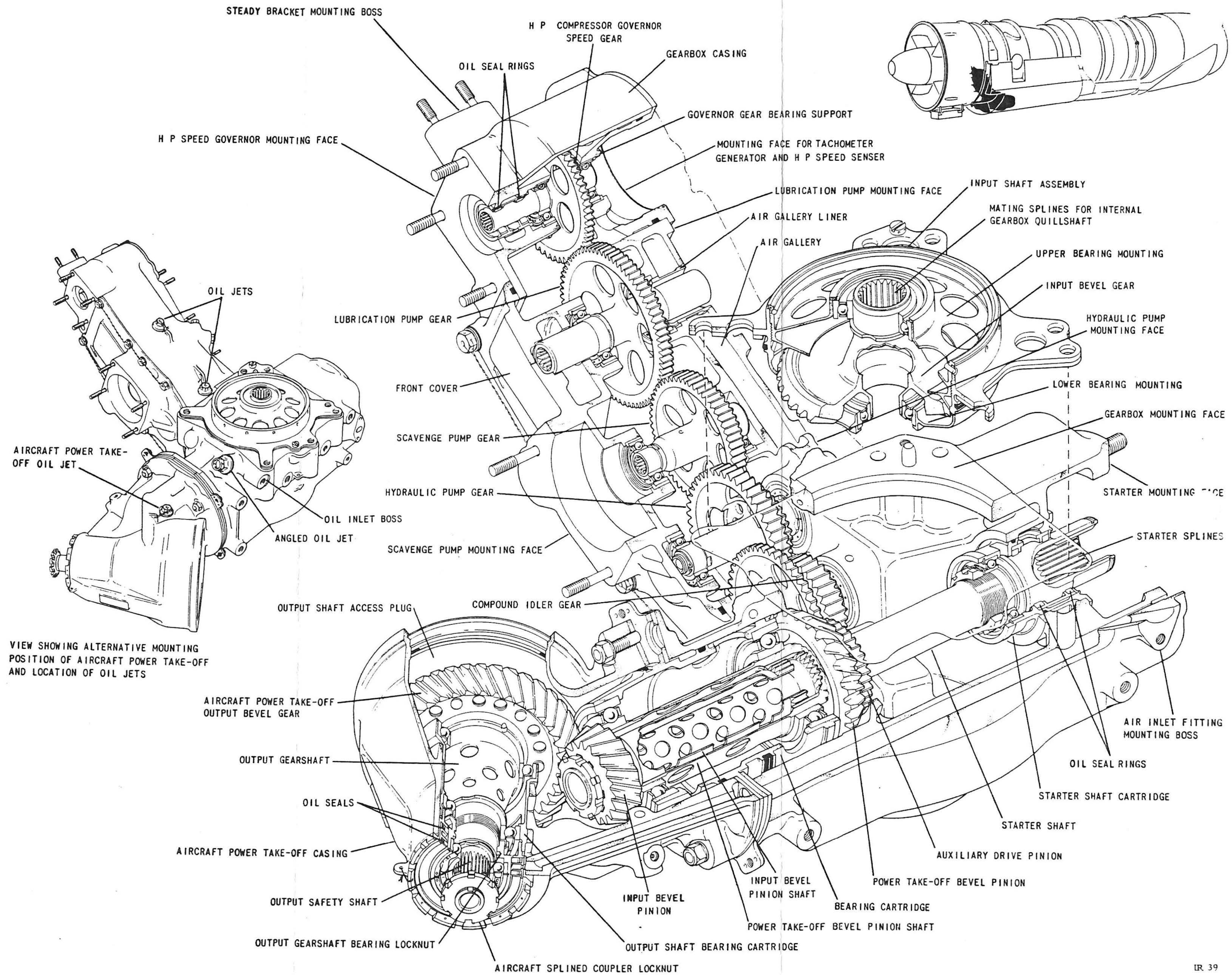


Figure 2-23 High Pressure External Gearbox

approximately 180 degrees apart, dependent upon the engine installation. One leg of the 'V' has a flange for mounting the power take-off to the HP external gearbox, whilst the other leg houses the output sub-assembly to which is fitted an aircraft splined coupler.

The output shaft sub-assembly consists of an output bevel gear bolted to the flanged end of the output gearshaft, the gear meshes with the input bevel pinion. The output gearshaft and bevel gear are supported by two single row ball bearings mounted in a bearing cartridge. The output gearshaft has splines machined on the inner diameter of the forward end which mate with the splines on the rear of the output safety shaft. Three rows of lightening holes are drilled round the body of the shaft. A blanking plug is installed at the rear of the internal splines and is retained by an internal circlip. Threads on the shaft accommodate a special type of sleeved locknut which has a ball bearing fitted at the forward end of its inner diameter to support the output safety shaft; a circlip at the front of the bearing retains the bearing and safety shaft in position. The forward outer diameter of the locknut is smooth and provides the seat for a double carbon ring oil seal.

The output safety shaft, which is hollow except for the centre waisted section, fits into the internal splines in the forward end of the output gearshaft. The shaft is located axially by its shoulder, which butts against the front bearing, and the plug in the output gearshaft. The waisted section is to provide a shear spot which, in the event of over-load conditions, will shear and thus prevent damage to the gearbox and/or engine. A locknut and washer are fitted to the forward end of the safety shaft to retain the aircraft splined coupler.

An oil seal housing is screwed into the forward end of the output gearshaft bearing cartridge. Slots in the forward face of the housing accommodate a wrench for tightening, and a row of holes drilled in the large diameter of the housing provide passage for compressor seventh stage air for pressurizing the oil seals.

Access to the output gearshaft is provided by a plug installed in the casing at the rear of the bevel gear. The plug is of cast magnesium and is a sliding fit in the casing. It is retained by a retaining ring, and an 'O' ring forms an oil seal between the plug and the casing. Three equi-spaced integrally cast lugs are provided for withdrawing the plug.

Starter Shaft

The starter shaft is a steel forging which is bored out from the front and rear with a small section left solid near the forward end. External splines on the forward end of the shaft mate with the internal splines at the rear of the auxiliary drive pinion. The shaft is supported by a ball bearing mounted at the front end of a bearing cartridge at which point the shaft is retained by a washer and locknut. The bearing is retained in the cartridge by a retaining ring. Slots in the cartridge permit oil to be sprayed from a projection on the shaft, the oil being returned to scavenge. A double, carbon ring oil seal is mounted in the rear of the cartridge, the shaft outer diameter providing the seat for the seal rings. Compressor seventh stage air passes through slots in the cartridge to pressurize the cavity between the seal rings. Internal splines at the rear of the shaft are for mating with the engine starter driveshaft.

Compound Idler Gears

The compound idler gears consist of two spur gears mounted on a hollow shaft, the larger gear meshing with the auxiliary drive pinion and the smaller one with the hydraulic pump gear. Each end of the idler shaft is supported by a ball bearing mounted in a cartridge which is an interference fit in the external gearbox casing. At the front of the shaft, a plug and an 'O' ring are fitted in the gearbox casing and retained by an internal circlip. At the rear, the bearing and cartridge are located in position by an air gallery plug, the flange of which butts against a shoulder in the gearbox casing. The air gallery plug is of magnesium and has a top hat section with four cross grooves to permit air circulation. Compressor seventh stage air is fed to the plug, via internal drillings in the casing, to pressurize the outer ends of both bearings; air passes from the air gallery plug and flows through the hollow shaft to the outer end of the front bearing.

Hydraulic Pump Gear

The hydraulic pump gear consists of two spur gears integral with a shaft. The larger gear meshes with and is driven by the smaller of the compound idler gears whilst the smaller gear meshes with and drives the scavenge pump gear. The gearshaft, which is hollow, is mounted on two ball bearings, the inner races of which butt against shoulders on the gearshaft. Internal splines towards the rear end of a gearshaft, mate with those on the hydraulic pump driveshaft, which fits inside the gearshaft.

The bearing at the rear of the hydraulic pump gearshaft is carried in a steel cartridge which is an interference fit in the gearbox casing. The front bearing is mounted in a steel cartridge fitted to the gearbox front cover. Immediately behind the rear bearing cartridge is an oil seal cartridge and oil seal. The oil seal is of the double carbon ring type and the gearshaft serves as the seat for the seal rings. Compressor seventh stage air from the air gallery is fed through slots in the cartridge to pressurize the cavity between the seals.

Scavenge Pump Gear

The scavenge pump gear is a single spur gear with an integral hollow shaft. It is supported at the front by a ball bearing mounted in a cartridge housed in the gearbox front cover, and at the rear by a bearing mounted in a cartridge fitted into the inner wall of the gearbox. Internal splines at the front of the shaft provide the drive for the oil scavenge pump, and the spur gear meshes with and drives the lubrication pump gear.

Lubrication Pump Gear

The lubrication pump gear is a single spur gear with an integral hollow shaft which is supported by a ball bearing at each end. At the rear, the shaft passes through a steel air gallery liner which is installed between the internal inner wall and the rear face of the gearbox to prevent leakage of compressor seventh stage air from the air gallery. Internal splines at the rear end of the shaft provide the drive for the oil lubrication pump, and the spur gear meshes with and drives the HP compressor speed governor gear.

HP Compressor Speed Governor Gear

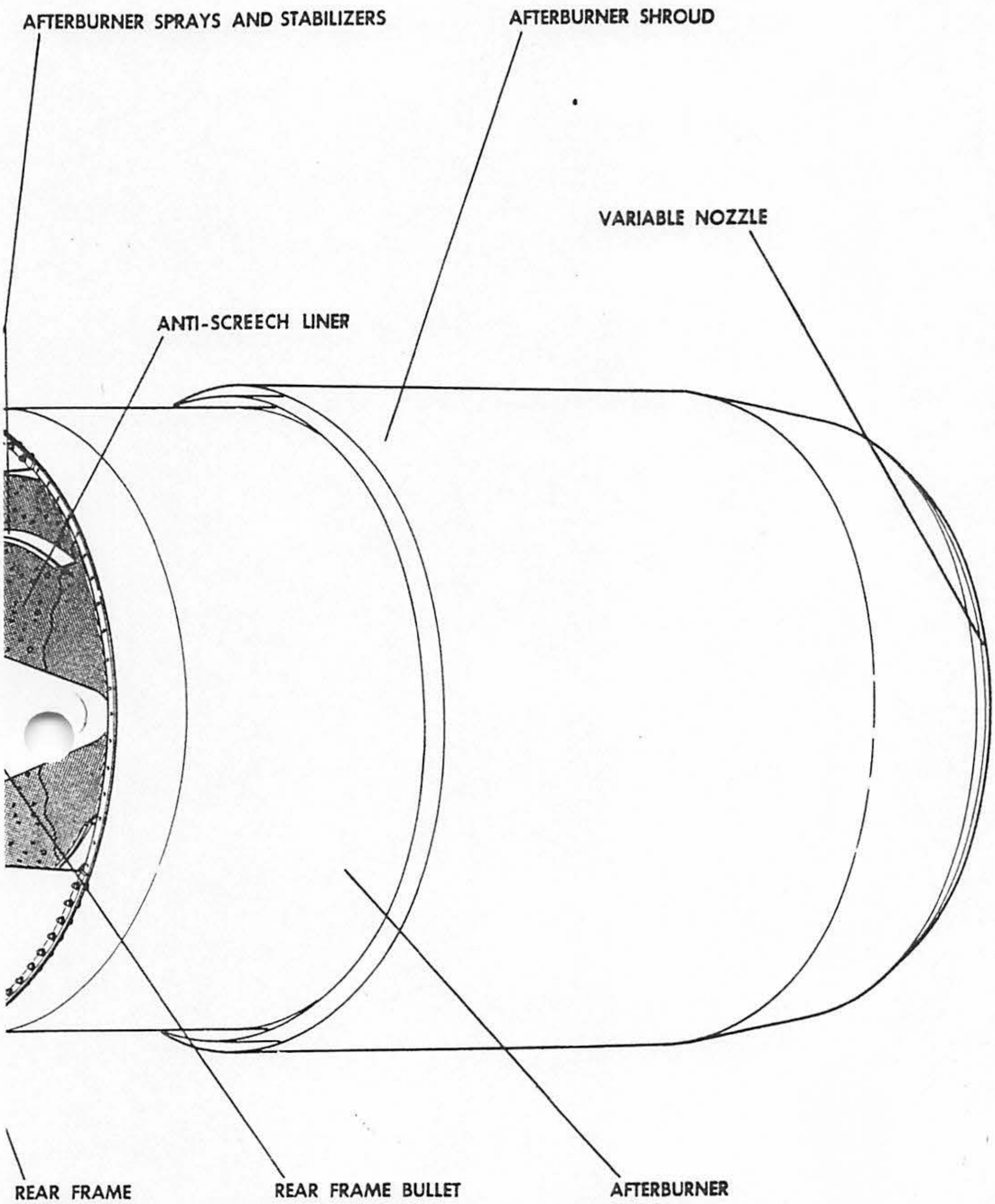
The governor spur gear has an integral hollow shaft and is supported by two ball bearings. The rear bearing is mounted in a steel flanged cartridge which is housed in a bearing support installed in the gearbox rear face. The forward section of the shaft is necked and a double carbon ring type oil seal, mounted in a cartridge, seats on this portion of the shaft. Compressor seventh stage air is fed through holes in the cartridge to pressurize the cavity between the seal rings.

Internal splines at the front and rear of the shaft transmit the drive to a HP compressor speed governor, a tachometer generator and a HP speed senser respectively.

Lubrication

Lubrication of the gearbox is effected by a combination of oil jets supplied from the engine oil system, supplemented by oil mist. Two oil jets, positioned at the top of the casing, approximately six inches apart, spray oil onto the main gear train. A third oil jet of the angled type is located near the top of the gearbox just forward of the gearbox mounting pad, and sprays oil at the meshing point of the input bevel gear and the power take-off bevel pinion. A fourth jet is mounted in the aircraft power take-off for lubrication of the bevel gears. Each oil jet is screened to protect the jet orifices from blockage by foreign matter.

Two oil bobbins are positioned in the oil feed and return passages at the joint between the HP external gearbox and the aircraft power take-off. The bobbin at the top of the joint is in the oil feed passage to the aircraft power take-off, and the one at the bottom is in the oil return passage. Adequate lubrication of the gearbox bearings is provided by a considerable oil mist circulation within the gearbox.


AFTERBURNER SPRAYS AND STABILIZERS
AFTERBURNER SHROUD
VARIABLE NOZZLE
ANTI-SCREECH LINER
REAR FRAME
REAR FRAME BULLET
AFTERBURNER

AIR SYSTEM

General

The Iroquois air system is divided primarily into two distinct internal flows, the air supply being bled from the compressor seventh and tenth stages. Seventh stage air is used to pressurize the carbon ring oil seals in the front and rear oil sumps, and in the LP and HP external gearboxes; this air also scavenges oil mist from the rear sump to the front sump through an internal oil return tube. The tenth stage air is used to cool the turbine rotor discs and blades, and tappings on the HP compressor delivery air take-off manifold supply tenth stage air for the engine anti-icing system, the air driven fuel pumps, and aircraft services.

Seventh Stage Air

Seventh stage air is bled through holes in the compressor seventh stage spacer ring immediately upstream of the seventh stage rotor blade tips, and passes forward through an annulus formed in the HP compressor stator casing, to a manifold cast on the front frame outer casing at the No. 8 strut location. The air then flows radially inwards through the hollow core of the strut and divides into two flows. A drilled passage in the LP compressor bearing housing permits one flow to pass to the annulus formed between the double carbon ring oil seal at the front of the LP thrust bearing. Some of this air leaks through the rearmost seal into the front sump, thus preventing oil seepage past the seal ring; the remainder leaks forward through the front seal into the cavity formed by the LP bearing housing and LP compressor rotor.

The second flow, which is the main stream of seventh stage air, is ducted from the base of No. 8 strut, through cored passages in the HP thrust bearing support flange on the rear of the front frame inner casing, into the cavity ahead of the HP compressor rotor stub shaft. This air pressurizes the carbon ring oil seal at the rear of the HP thrust bearing assembly; a small portion of the air leaks through the seal to the front sump. A further small quantity of air leaks out past the peripheral air seal at the rear of the variable incidence HP compressor inlet guide vanes. The main seventh stage air flow continues rearward through holes in the HP compressor stub shaft and completes the sealing of the front sump by pressurizing the LP mainshaft steady bearing seal.

The remainder of the air flows to the rear of the engine through the annulus formed between the HP and LP mainshafts to the front inner oil seal of the HP turbine rotor assembly, where it divides, some to pressurize and leak past this seal to the rear sump, the remainder passing through holes in the HP turbine shaft to the space between the HP and LP turbine shafts. At this point a proportion of the air pressurizes and leaks past the front intermediate oil seal to the rear sump, while a second flow passes radially outwards through an air seal to combine with tenth stage air and cool the front face of the turbine third stage rotor disc; this air is exhausted to the main gas stream at the rim of the disc. The remainder flows through holes in the LP turbine shaft where some of the air pressurizes and leaks through the front outer oil seal to the rear sump. The flow continues past the hub of the LP turbine disc to the LP turbine rear air seal where some air flows through a tube at the rear of the bearing housing to pressurize the double carbon ring oil seal at the rear of the rear sump, the remaining air passing through the LP turbine rear air seal to cool the rear face of the turbine third stage rotor disc. This flow joins that which passes through the space formed between the rear frame baffle and cover and is exhausted to the main gas stream immediately aft of the turbine third stage rotor disc rim. The air which pressurizes the rear sump seal rings passes forward into the sump and rearward into the rear frame inner housing.

Seventh Stage Air/Oil Mixture

The engine lubricating oil system uses seventh stage air to return scavenge oil mist from the rear sump to the front sump where the oil is separated from the air by a centrifugal air/oil separator. Seventh stage air leaking past the oil seal rings to the rear sump carries oil mist, through a sump vent, to the annulus formed between the internal air vent tube and the internal oil return tube. The air/oil mixture is carried forward to the front sump, where it joins the front sump air/oil mixture. The mixture is vented to the centrifugal air/oil separator and the purified air flows through holes at the base of the vanes in the separator to the internal air vent tube, thence rearwards to the space formed by the rear frame inner housing and the rear frame housing cover. The air cools the housing inner surfaces and, together with any oil vapour that may be present in this region, passes through a series of holes at the bottom of the housing into a collector box from which it is vented to the main gas stream.

External Gearboxes Air Supply

Air is tapped from the seventh stage air take-off annulus and is externally piped to the LP and HP external gearboxes. In the LP external gearbox, the double carbon ring oil seals at each end of the LP external gearbox driven gearshaft are pressurized. In the HP external gearbox, the double ring oil seals on the aft end of the starter shaft, on the aircraft splined coupler end of the power take-off, and on the forward end of the governor gear are pressurized. Seventh stage air is also bled into the hollow shaft of the hydraulic gear to pressurize the outer faces of the hydraulic gear bearings. Some of this air leaks outward past the seals into the engine nacelle, the remainder passing inward to the gearboxes to mix with the oil and be scavenged by the main scavenge pump to the oil tank where the air/oil mixture is vented to the front sump.

Tenth Stage Air

Tenth stage air is bled immediately aft of the roots of the tenth stage rotor

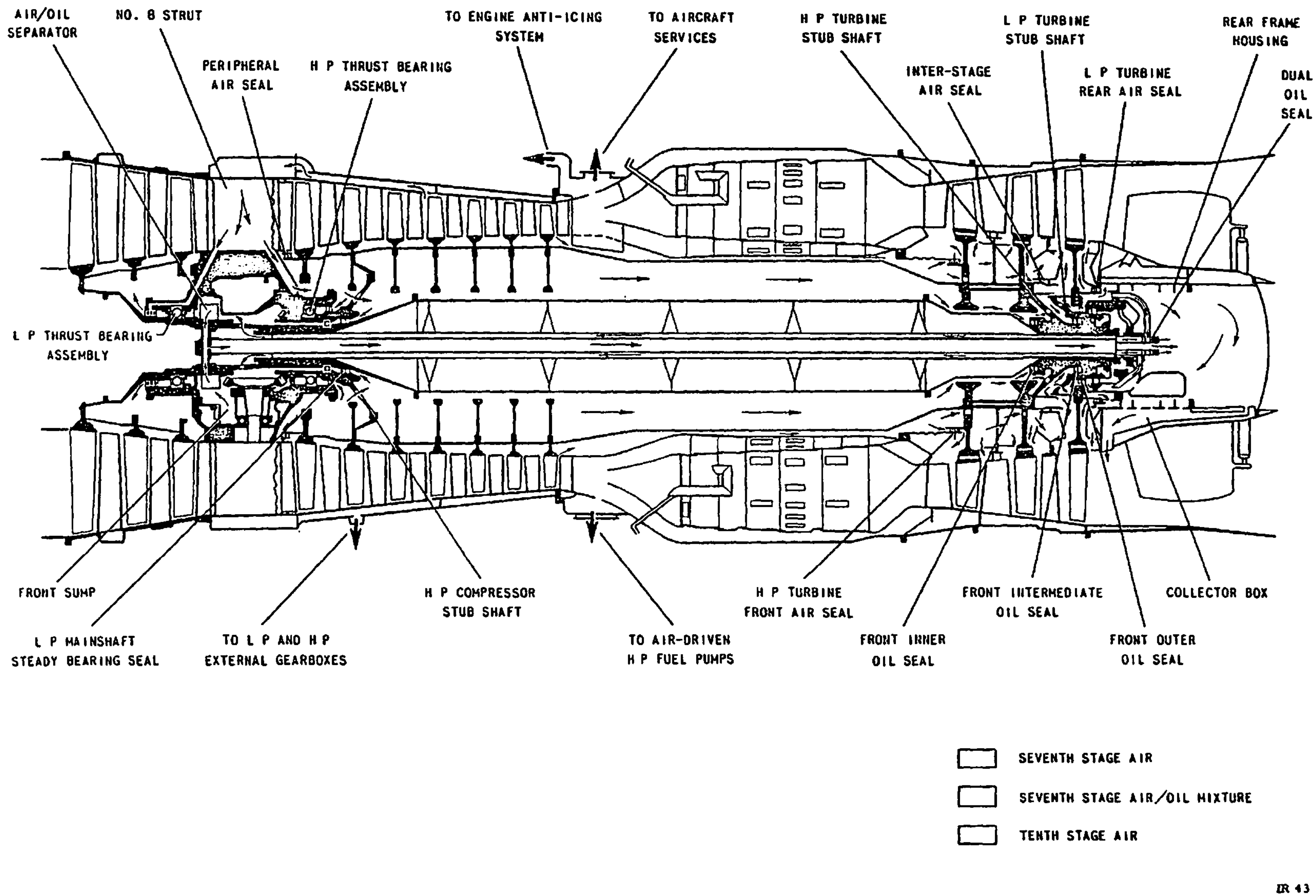


Figure 3-1 Schematic Diagram of Air System

compressor blades and flows rearward through the annulus formed by the HP rotor mainshaft and the mid frame inner casing. When the flow reaches the HP turbine front air seal, some air passes through the seal and cools the front face and rim of the first stage HP turbine disc and the roots of the turbine rotor blades before exhausting to the main gas stream. The remainder flows through axial holes in the turbine first stage rotor disc into the space formed by the turbine spacer disc and the first and second stage turbine discs. From the first chamber some air flows radially outward over, and cools, the rear face of the first stage disc, and is exhausted to the main gas stream through holes in the forward rim of the spacer. The remainder passes through axial holes in the spacer disc web to a second chamber where it again divides into two streams; one flows radially outward through the annulus formed around the rear rim of the spacer to the main gas stream. This annulus is adjacent to the front face of the turbine second stage rotor disc rim and air is permitted to flow through clearances between the blade roots and the disc slots to carry heat away from the area.

The second stream continues to flow rearwards through axial holes in the turbine second stage rotor disc where the flow divides three ways; one stream flows radially outward across the rear face of the second stage turbine disc, the second through holes in the turbine stator disc assembly, and the third through the inter-stage air seal where it joins a portion of seventh stage air, flows radially outward over the front face of the third stage turbine disc, and exhausts into the main gas stream.

Engine Services

Tenth stage air supplied to the engine anti-icing system, and the air driven fuel pumps is drawn from the HP compressor delivery air take-off manifold which forms an integral part of the mid frame assembly. This manifold is supplied with tenth stage air bled into it through holes in the mid frame outer casing. The take-off point for the anti-icing system is at the upper right-hand side of the manifold; for the fuel pumps it is at the lower left-hand side of the manifold.

Aircraft Services

Aircraft services, including air conditioning of the equipment bays, cockpit heating and pressurizing, windshield de-frosting, pressurizing of rocket pods, fuel tanks, and anti-g suits, are supplied with tenth stage air from one of two take-off pads situated at either side of the top manifold centre line, depending on the installation.

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 psi.

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. Since the temperature and pressure of the tapped air increase with engine speed, only a small proportion of the available supply of air is required during high speed operation. Provision is made to compensate for these variations in temperatures and pressures.

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 round the LP compressor stator casing, thence to air ducts at 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 inwards 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 strut. Holes in the strut inner skin direct the air flow against the leading edge inner surface and into the annular space formed between the inner and outer skins. The air flows rearward through the annular space and is then led through a cavity at the rear of each strut into 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 annular space formed by the double skin of the bullet. The air flows rearward through the annular space to a corresponding annular space 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

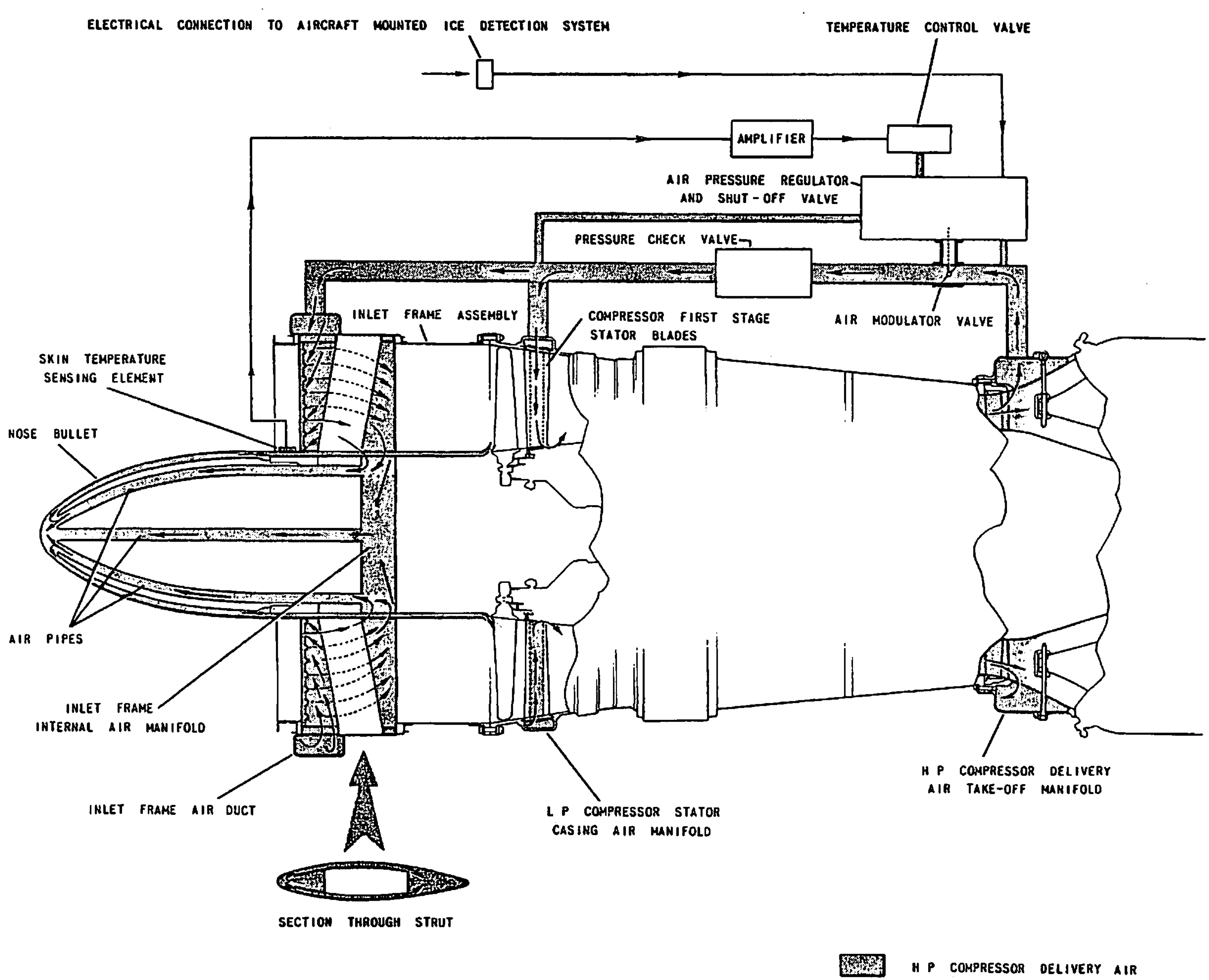


Figure 3-2 Schematic Diagram of Anti-icing System

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 underside 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 cycle is repeated.

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 towards 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 to the set value, the temperature 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.

LUBRICATING OIL SYSTEM

LEADING PARTICULARS

Type of System	Self-contained, dry sump.
Type of Oil	MIL-L-7808C
Oil Tank Capacity	4 Imp. Gal. plus 1 Imp. Gal. air space
Minimum Oil Pressure (Idling)	2.5 ± 2.0 psi.
Nominal Oil Pressure Across Oil Jets at Normal (Cruise) rpm	30 psi with oil at 71°C.
Maximum Oil Pressure (During cold weather engine starting)	150 psi at the oil temperature regulator.
Main Pressure Pump	Single element, Gerotor type (Nichols) cap. 10 gpm.
Scavenge Pump	Two element, Gerotor type (Nichols) capacities 12 and 8 gpm.
Oil Temperature Regulator	Combined lubricating and hydraulic oil sections using fuel as coolant (United Aircraft Products)
Maximum Scavenge Temperature	150°C (302°F).
Maximum Oil Delivery Temperature	15°C (59°F) at oil temperature regulator by-pass valve.
Maximum Oil Consumption (Normal Rated)	2.5 Imp. pt per hr.
(Military Rated)	3.0 Imp. pt per hr.

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 oil pressure supply pumps and a scavenge pump, an oil temperature regulator, and the necessary components for oil distribution.

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 is not a normal military requirement and no special provision is made for anti-g conditions. However, 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 and HP bearing housings 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 near the oil filter 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 the positive displacement type employing a special form of internal-external gear system known as Gerotor mechanism. The inner gerotor is keyed to the pump shaft and drives the outer gerotor through specially shaped teeth. The outer gerotor revolves in a ring which is eccentric to the pump shaft so that when one tooth of the inner gerotor is meshing fully with the outer gerotor the opposite tooth is disengaged. Circular ported plates are fitted to each side of the gerotor mechanism and as each tooth of the inner gerotor is always in contact with

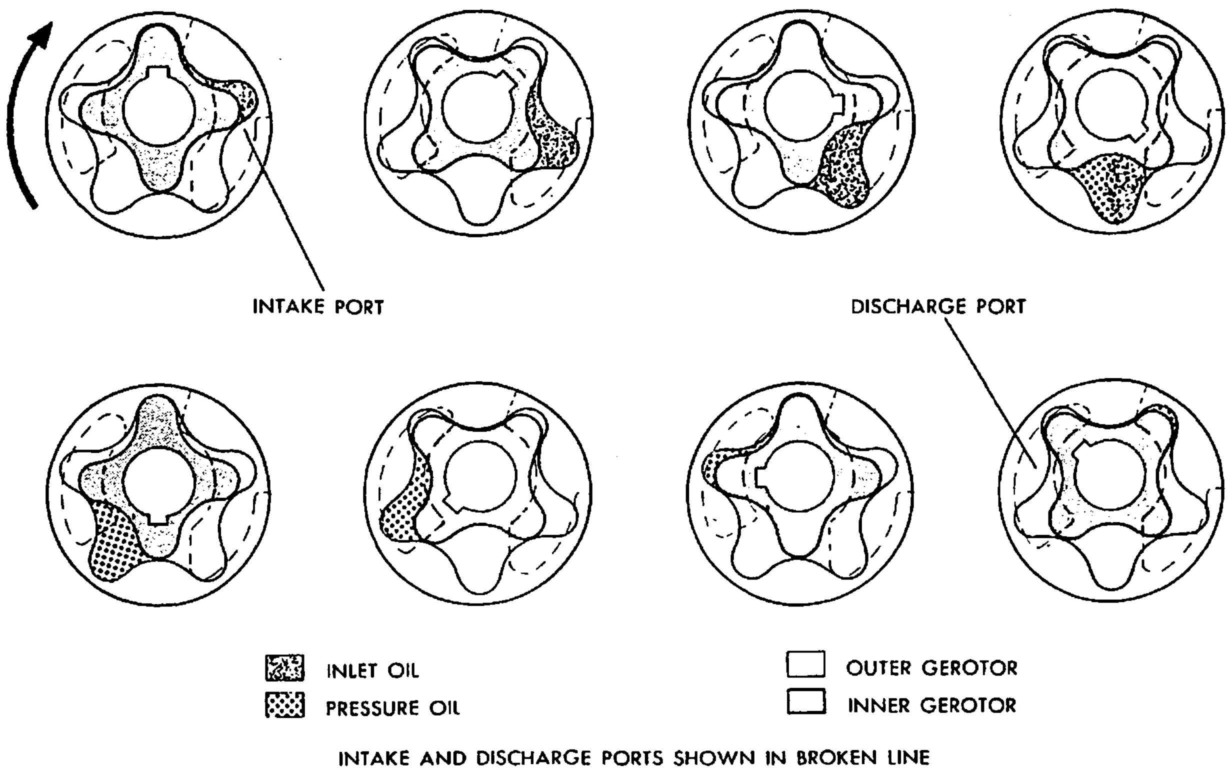


Figure 4-1 Pump Gerotor Mechanism

the outer gerotor, fluid tight pockets are formed, the opening and closing of these pockets providing the pumping action, see Figure 4-1.

The two pumps employed in the engine lubricating system are mounted on the HP external gearbox. A single element pressure pump of about 10 gph capacity supplies the main engine requirements. The scavenge pump unit consists of two elements in a single casting; one of about 12 gph capacity returns oil to the tank from the HP external gearbox, while the other of 8 gph capacity acts as an auxiliary scavenge pump for the rear sump during engine rundown. An oil flow 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 retained 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

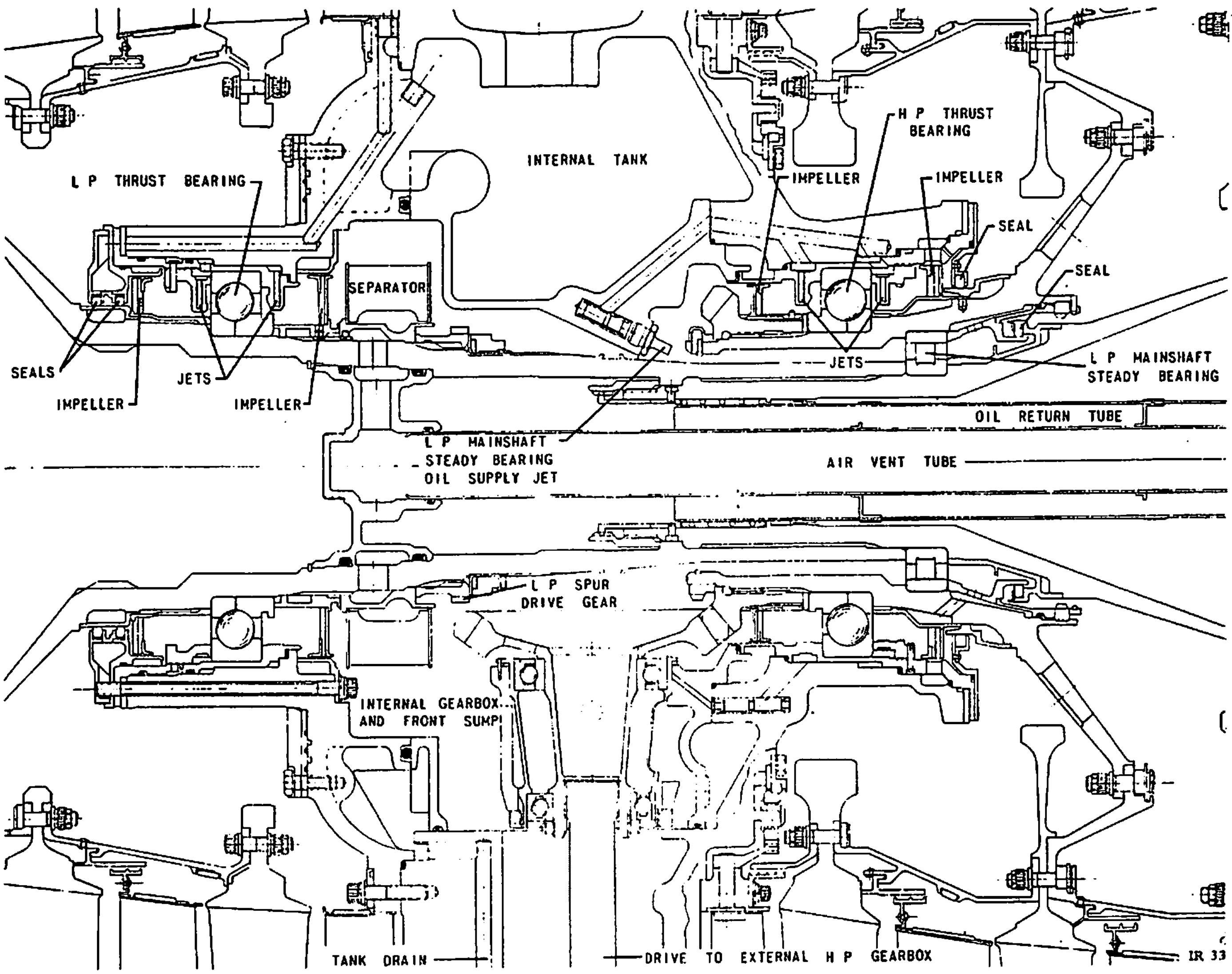


Figure 4-2 View Showing Front Frame Internal Details

adaptor prevents oil drainage 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°C (59°F) or pressure differences above 40 psi, allowing oil to by-pass the cooling element to achieve minimum warm up time.

A check valve in the outlet fitting of the temperature regulator prevents seepage of oil into the sumps after engine shutdown.

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 to both 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 bearings is supplied with oil through the main pressure line. In the case 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 either 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.

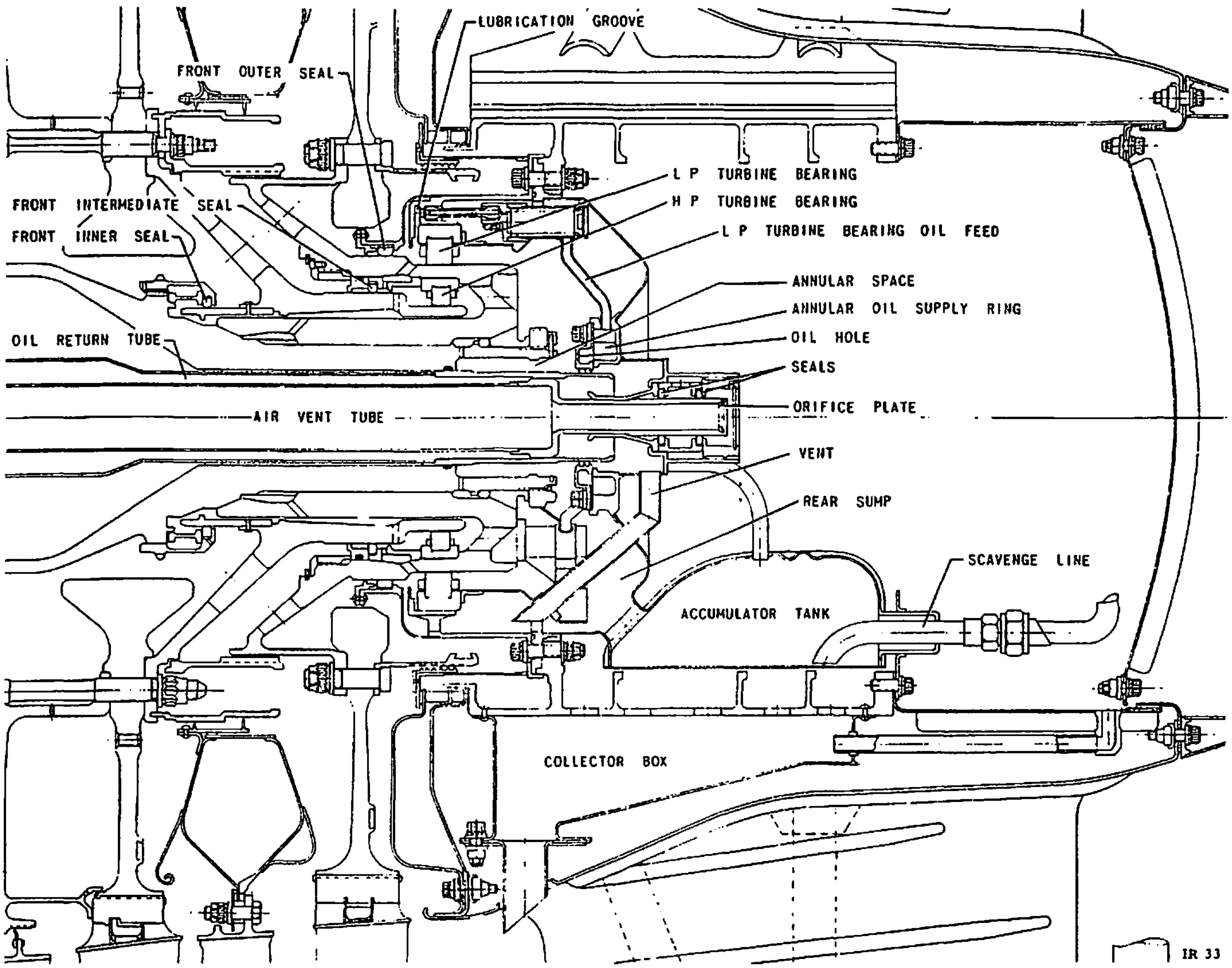


Figure 4-3 View Showing Rear Frame Internal Details

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, see Figure 3. 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, by centrifugal force, makes its way through drillings and spline clearances to the forward face 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 Figure 4. 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 mixture 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 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 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 greater into the front sump. The air flows inward through the separator vanes into the air vent tube, then rearwards 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 into the main exhaust gas stream. Any oil accumulating in the exhaust bullet 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, when the air pressure may be inadequate for efficient scavenging of the sump through the oil return tube.

OEL-1140

ENGINE HYDRAULIC 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 schematic arrangement of the system is shown in Figure 4/1-1.

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°C (185°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.

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 by-pass valve which operates at 50 psi pressure difference, to prevent excessive flow restriction in the event of the filter element becoming clogged with foreign matter.

Accumulator

A 25 cub. 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 a result of a control valve opening or closing. The accumulator is pre-charged with nitrogen gas to a pressure within 1500 to 2000 psi. A connection is provided for pressure checks, and for recharging the accumulator should this be found necessary during ground servicing.

Reservoir

A 90 cub. 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 3,000 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 value.

A LP relief valve is incorporated in the reservoir to prevent excessive pressure build-up of the LP oil due to thermal expansion effects and overfilling. 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 adjacent engine panel is removed. The pointer is connected to the reservoir piston by a flexible cable.

HP Relief Valve

A 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 3,300 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

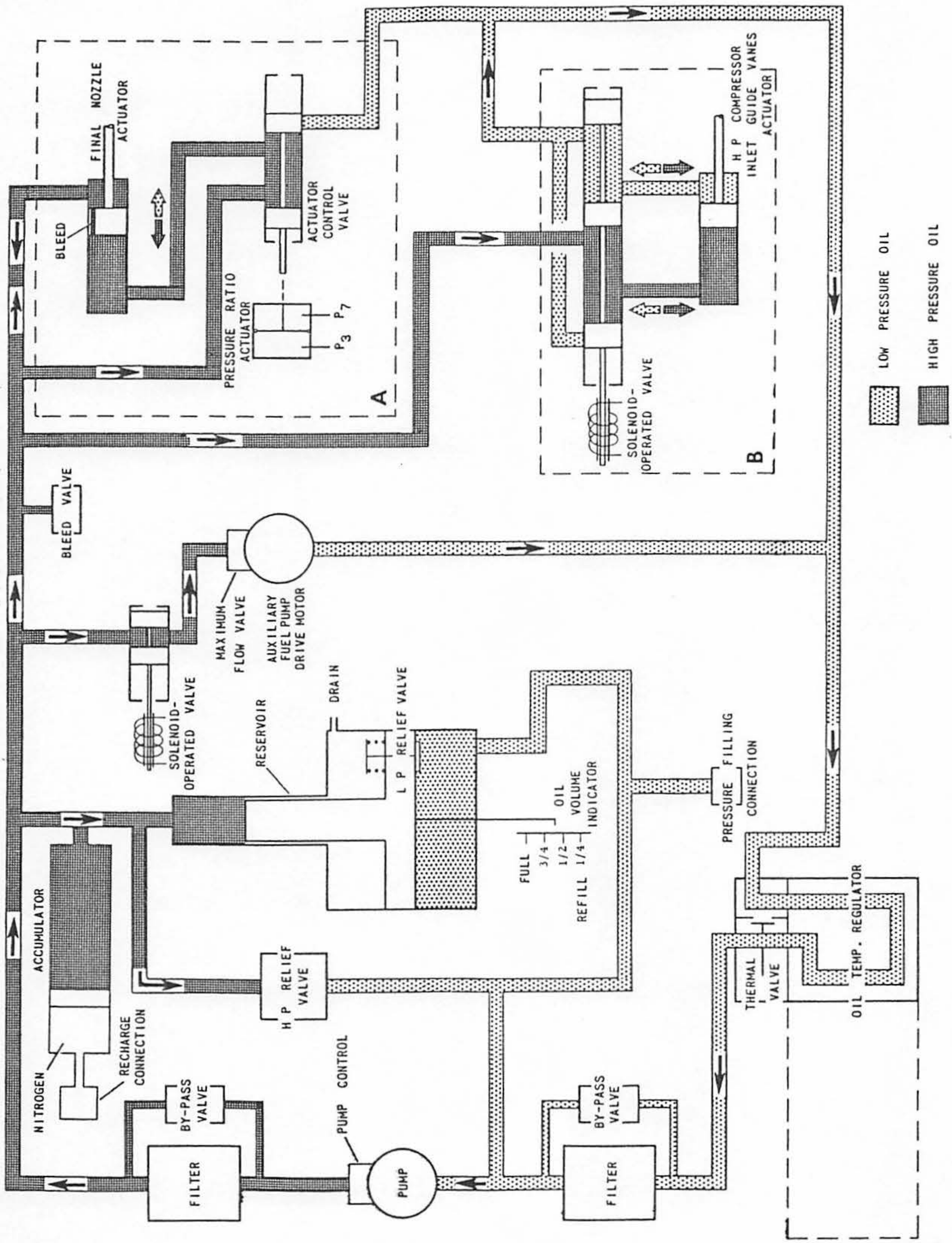


Figure 4/1-1 Schematic Diagram of Hydraulic System the entry of air during the filling operation.

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_3) to afterburner 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 varying engine requirements. Referring to 'A' of Figure 4/1-1, it will be seen that variations in the P_3 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 the left of the final nozzle actuator piston. HP oil is fed to the space on the right of the final nozzle actuator piston at all times. When the actuator control valve is positioned as shown, HP oil is fed to both sides of the final nozzle actuator piston. 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_3 and P_7 pressure ratio occurs the actuator control valve moves to the right, and reduces the HP oil supply to the control. Simultaneously the LP port is uncovered and the pressure acting on the left side of the final nozzle actuator piston is relieved, causing the actuator piston to move to the left and reduce the engine final nozzle area.

A bleed through each of the actuator pistons maintains a continuous circulation of oil through the actuators for cooling purposes. A hydraulic clutch 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 details shown in 'B', Figure 4/1-1, 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 rotor speed.

On engine start, the speed switch in the amplifier unit energizes the solenoid to move the valve to the right, as shown. HP oil is fed to the space on the left of the actuator piston, the space on the right of the piston being opened to LP oil The actuator piston therefore moves to the right and rotates the guide vanes to the plus 25 degree position. The guide vanes remain at this setting until the HP compressor exceeds the speed at which the switch in the amplifier is set. The switch then opens to break the supply of current to the solenoid, and the valve moves to the left. HP oil is now fed to the space on the right of the actuator piston, the space on the left of the piston being open to LP oil. The actuator piston therefore moves to the left and rotates the guide vanes to the minus 15 degrees position. The guide vanes remain at the minus 15 degrees position until the HP compressor speed falls below the speed switch setting, when the switch closes to energize the solenoid, and return the guide vanes to the plus 25 degrees position.

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.

FUEL SYSTEM

Introduction

The following pages give but an introduction to the Iroquois fuel system and a brief description of the basic principles contained therein. Revisions to this section will be issued later, and will treat the system in greater detail and explain how the units operate to obtain optimum performance from the engine over its full range of operating conditions.

General Principles of Engine Control

The thrust of a turbojet engine is related to the increase in momentum of the gases passing through it, this increase being brought about by burning fuel in the air flow passing through the engine. Thus, air and fuel are the two essential substances consumed by the engine in its operation. Therefore, basic control of the engine power output can be effected by regulating either the mass air flow through the engine or the fuel supply to the engine combustion chamber.

The complex mechanical arrangement and resultant performance losses and weight penalties necessary to effect control of the large quantities of air consumed by a turbojet engine, renders this method of engine control impracticable in aircraft applications, thus, regulation of the engine fuel flow is selected as the most suitable method of control.

In addition to regulating the engine fuel flow, it is desirable to control the area of the final (propelling) nozzle of the turbojet to obtain maximum performance from the engine. A detailed description of the variable area final nozzle used on the Iroquois, and the control principles involved, will be issued later.

Basic Control Variables

Certain engine conditions vary as a result of changing the quantity of fuel burned in the engine. For example, when the pilot's power control lever is advanced to obtain a higher thrust output, the fuel flow to the engine is increased. This in turn results in higher gas temperatures in the combustion chamber and an increased acceleration of the gases through the turbine causing an increase in engine rpm and mass airflow. In addition, corresponding increases in gas pressure occur throughout the engine. Thus, any one of these variables, - engine thrust, gas pressures and temperatures, or engine rpm, could be used as a basis on which to establish fuel schedules.

Engine rpm is ideally suited as the primary controlled operating variable in that it bears a definite relationship to thrust output and the method of cockpit indication of engine rpm is relatively simple. The rpm indicator in the cockpit readily provides the pilot with an indication that the engine is satisfactorily responding to his power setting.

The graph in Figure 5-1 shows the linear relationship between the position of the power control lever (throttle angle) and thrust. The graph also shows a typical engine rpm curve required to produce the desired thrust/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, 75% rpm produces 50% of the available thrust output at sea level conditions, the same engine rpm will produce approximately 50% of the available thrust at altitude. Thus, the setting of the power control lever establishes a demand for a certain percentage of the 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, schedule the exact fuel flow required to produce this rpm.

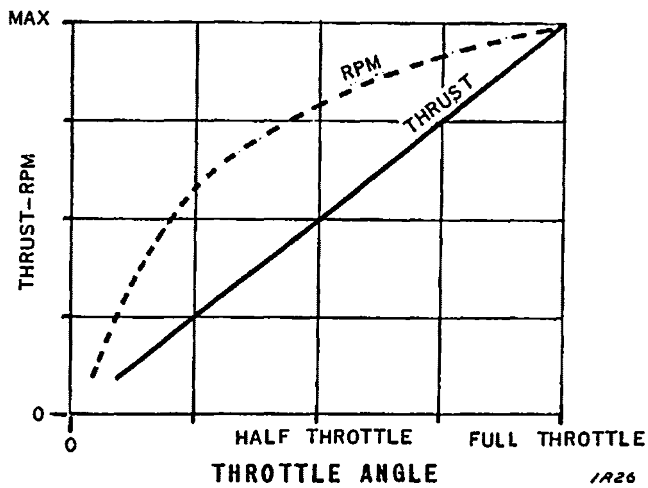


Figure 5-1 Typical Thrust - RPM/Throttle Angle Relationship

Temperature and pressure sensing devices are used extensively in the Iroquois fuel system to apply trims on the basic fuel schedule. These 'override' controls modify the fuel flow to keep the engine within its operating requirements and limitations. Figure 5-2 shows the source and terminology of the control variables used on the Iroquois engine.

Fuel Control Requirements

It is essential that the engines in modern high performance aircraft be automatically controlled to provide the thrust output determined by the setting of the power control lever. Furthermore, it is desirous to have a single control lever per engine which enables the pilot to request thrust modulation over the entire engine and afterburner operating range, and which can be used to effect engine shutdown.

As the turbojet engine is frequently operated at or near its maximum operating conditions, a system of controls is necessary which will override and modulate the basic fuel schedule when maximum rotor speed or structural and temperature limitations are reached or exceeded. On the Iroquois engine these trimming controls

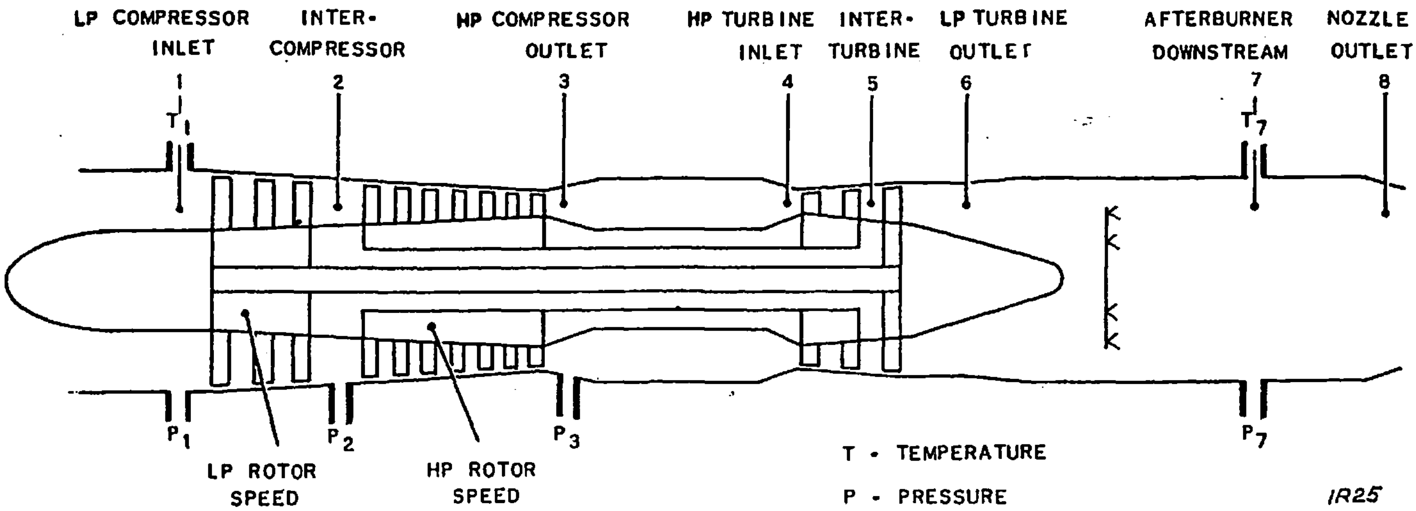


Figure 5-2 Engine Control Variables and Pick-up Points

are fully automatic and protect the engine from damage under normal 'steady state' operating conditions, during accelerations and decelerations, or as a result of inept handling of the power control lever. In addition, the control units are capable of maintaining the air/fuel ratio at the correct values to sustain safe and efficient combustion of the fuel under all operating conditions. Compensation is also made for changes in altitude and aircraft forward speed to maintain a constant engine speed for a given power control lever setting over the complete range of aircraft operation. The minimum idling fuel flow to the engine is modulated to produce a rising idling speed schedule as altitude increases, and hence maintain sufficient compressor pressure for adequate cabin pressurization and other aircraft services at altitude.

As an added safety feature, provision is made for an emergency system which provides the pilot with full manual control of the fuel flow to the engine should malfunctioning of the normal control units develop.

OUTLINE OF IROQUOIS FUEL CONTROL SYSTEM

General

The Iroquois fuel control system is specifically designed to schedule the engine and afterburner fuel flows according to the varying fuel requirements encountered under all operating conditions. The system is essentially hydromechanical in nature and includes adequate provision to fulfil the general control requirements discussed in the preceding paragraphs.

For the purpose of description the system can be sub-divided into four major groups, namely, an HP fuel supply system, main and emergency control systems for the engine fuel supply, and a control system for the afterburner fuel flow. A brief description of each group is contained in the following paragraphs; more detailed accounts of the components and their functions will be issued later.

HP Fuel Supply

The supply of pressurized fuel required for engine and afterburner operation is provided by two air-driven centrifugal type pumps. Referring to Figure 5-3, the fuel at boost pump pressure is supplied from the aircraft tanks to the HP pumps where it is pressurized to the required operating pressure. From the pumps, the fuel is delivered at high pressure to the appropriate control system units.

The pump pressure control unit depicted in the diagram, controls the pressure output of the pumps by regulating a bleed flow of HP compressor delivery air (P_3) used to drive the air turbines which in turn drive the centrifugal impellers in the pumps. Regulation of the air flow is effected by a hydraulic actuator which receives signals from the pump pressure control unit and which is mechanically linked to a barrel valve in the air inlet duct of the pumps. The pump pressure control unit primarily senses the pressure rise across the engine compressor, this being a measure of the mass air flow through the engine, and hence regulates the pump delivery pressure 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 principle 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.

The fuel discharge pressure required for vaporizer type burners is relatively low in comparison to other types of fuel burners and the maximum operating pressure required in the Iroquois fuel system is about 1150 psi. Fuel pump delivery pressure at ground idling is approximately 200 psi.

As described later in this Section, the actual flow rate of the fuel delivered by the pumps is determined by the metering orifice sizes in the various control units and the pressure drop across these orifices.

As seen in Figure 5-3, 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 HP pumps. The starting pump is hydraulically driven 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 comprises the units which under normal circumstances provide automatic regulation of the fuel flow to the burners in the engine combustion chamber. Referring to Figure 5-3, the main flow of pressurized fuel delivered by the HP pumps passes through the main metering unit and a servo throttle valve. The servo throttle valve is positioned by a servo control pressure originating in the pressure drop unit and trim controls depicted in the diagram.

The servo control pressure acting on the servo throttle valve is modulated by the pressure drop unit which in turn operates primarily according to the setting of the pilot's power control lever and signals received from the HP and LP rotor speed governors and a minimum idling flow valve. A proportional fuel valve included in the pressure drop circuit controls the fuel flow to avoid excessive exhaust temperatures and to provide safe engine accelerations. The servo throttle valve establishes the pressure drop across the metering valve in the main metering unit. The opening of the metering valve is dependent on inter-compressor pressure (P_2), (see Figure 5-2) this being a measure of the mass air flow through the engine. Thus the system is altitude compensated.

From the servo throttle valve the fuel flows unrestricted to the flow distributor, and thence to the burners, via a selector valve which is fully open when the engine is operating on NORMAL. A shut-off valve located upstream of the flow distributor closes when the cockpit power control lever is retarded to the CLOSED position and cuts off the fuel flow to the burners during engine shut-down. The shut-off valve is fully closed between 0° and 3° throttle angle. Above 3° throttle angle, the valve opens sufficiently to allow fuel flow with negligible pressure drop. An igniter valve, located in parallel to the flow distributor, supplies fuel to the torch igniters for starting purposes. The igniter valve will be deleted from later engines.

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 one 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.

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. 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. 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 System

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_3), which, like P_2, is a measure of the mass airflow through the engine. Thus, the afterburner fuel flow is also altitude compensated.

The pressure drop unit establishes the servo control pressure, and hence the fuel flow to the afterburner, primarily, according to the setting of a miniature throttle valve located in the proportional flow by-pass circuit. This throttle valve, being mechanically linked to the power control lever, is opened and closed according to movement of the power control lever in the afterburning range, thus providing full thrust modulation of the afterburner. In addition, a small pressurizing valve in the proportional by-pass circuit establishes the minimum fuel flow required to obtain afterburner light-up and sustain combustion. HP compressor delivery pressure (P_3) is used also in the pressurizing valve as the controlling variable.

Thus it is seen that the afterburner fuel flow is regulated according to power control lever setting in the afterburning range, and as a function of P_3 and hence altitude.

A normally closed, solenoid-operated shut-off valve located downstream of the servo throttle valve, opens to permit the passage of fuel to the afterburner manifolds when the power control lever is advanced into the afterburning range (93° to 110° throttle angle). Afterburner ignition is accomplished by a hot streak igniter system. Premature opening of the shut-off valve and operation of the hot streak system is prevented by a speed lock-out switch which breaks the electrical circuits involved until the minimum safe HP rotor speed is reached. Closing of the shut-off valve, and hence shutdown of the afterburner, is accomplished by retarding the power control lever to a setting below 78° throttle angle.

OPERATING PRINCIPLES OF HYDROMECHANICAL FUEL CONTROL UNITS

Principles of Flow Control

The quantity of fluid flowing through any passage is dependent primarily on the value to which the fluid is pressurized and secondly on the cross sectional area of the passage. Other factors such as fluid viscosity, have a marked effect on flow values, however, these are not considered in the following simplified examples. Thus, there are two primary means of controlling the flow through a passage; that of varying the pressure of the fluid, or, varying the cross sectional area of the passage.

By placing a restriction in a passage, see Figure 5-4, the effective cross sectional area (A_1) upstream of the restriction is reduced. Thus, with applied pressure constant, the flow through the passage is reduced and since the flow rate on both sides of the restriction is obviously equal, and the cross sectional area (A_2) downstream of the restriction is larger than A_1, then the downstream pressure is at a reduced value. This process is termed 'throttling' whereby the pressure and hence flow of a fluid is reduced by varying the area of the passage by the use of a valve or other form of restriction. It will now be seen that one possible means of controlling the flow through a passage is to vary the area of a restriction in the passage.

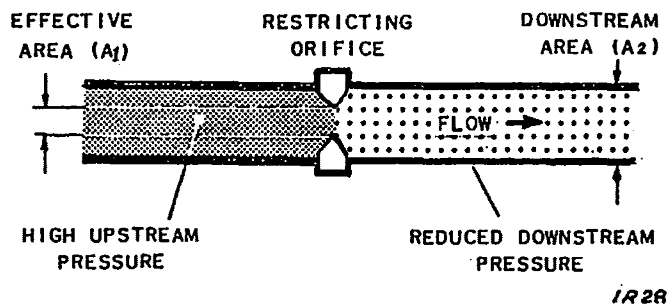


Figure 5-4 Throttling Effect of a Restriction

On the other hand, with a constant upstream pressure and constant area restriction, control of the flow can be effected by varying the pressure downstream of the restriction and hence the pressure drop across the restriction. If by some means the downstream pressure is reduced, this will create a larger pressure drop across the restriction, and result in increased flow. From the relationship between pressure and area it will be seen that an increase or decrease in the downstream pressure can be brought about by decreasing or increasing respectively the effective

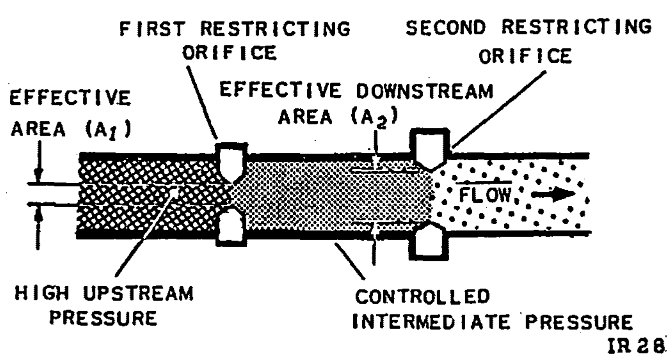


Figure 5-5 Flow Control Using Two Restrictions

cross sectional area of the downstream passage. This is effected by introducing a second restriction downstream of the first restriction, see Figure 5-5. With the effective area of the passage downstream of the first restriction now dependent on the size of the second variable area restriction, the pressure drop across the first fixed area restriction, and hence the flow through the entire passage, can be regulated by varying the area of the second restriction.

Both the main control system and afterburner control system in the Iroquois fuel control adopt this principle of controlling the fuel flow, the metering units and servo throttle valves being the equivalent of the first and second restrictions respectively in the example previously described. The valve openings in these units are determined by the position of servo actuated plungers operating in orifice

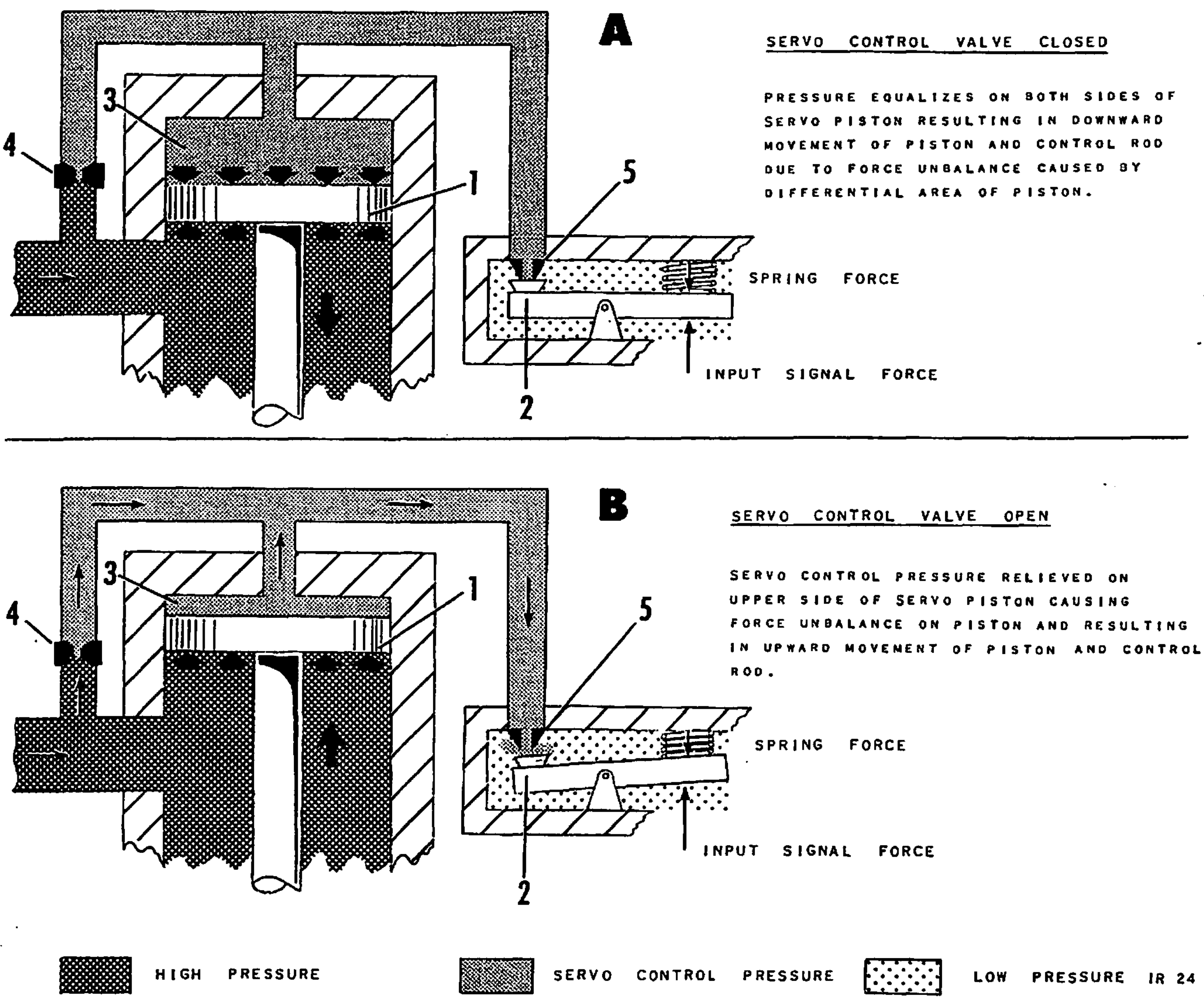


Figure 5-6 Operating Principles of a Servo Motor

sleeves. This brings about another operating principle used in the Iroquois type of control units, the servo motor, which is described in the following paragraphs.

Servo Motors

Servo motors are used extensively in the Iroquois fuel system as a means of actuating metering valves in the control units. A typical servo motor (see Figure 5-6) comprises essentially a control rod and piston (1), and a servo control valve (2), linked by suitable channelling to the chamber (3) above the piston. One side of the servo piston is subjected to high pressure fuel while a servo pressure derived from a small fixed area restriction (4) in a high pressure tapping upstream of one motor is applied to the opposite side of the piston.

A second restricting orifice (5) is placed in the servo circuit downstream of the fixed orifice and is made variable by the action of the servo control valve (2). Thus, the arrangement of these orifices is similar to that previously described, where the second orifice is made variable to regulate the pressure downstream of the first orifice. This regulated or intermediate pressure is termed 'servo control pressure' and being applied to the upper face of the servo piston can be used to create a pressure differential across the piston.

When the servo control valve closes, the pressure in the servo circuit and hence on the upper face of the piston equalizes with the upstream pressure applied on the underside of the piston. The differential area of the piston (caused by the control rod reducing the effective area on the underside of the piston) results in an unbalance of the pressure forces applied to the piston and a downward movement of the piston and hence the control rod results. In certain applications this movement is assisted or opposed, depending on the configuration, by applying appropriate spring forces to the piston. Movement of the piston ceases when balance is restored between the forces applied to the piston.

When the servo control valve opens, a flow occurs in the servo circuit and as previously described, the pressure downstream of the fixed orifice is relieved. This upsets the pressure balance of the servo piston and results in an upward movement of the piston and control rod in the direction of lower pressure. Thus, the pressure in the servo circuit is established by the effective area of the servo control valve.

The degree of opening of the control valve generally varies in proportion to an input signal. The spring-loaded rocker lever type of servo control valve is commonly used in the system, whereby one end of the lever carries the sealing member of the control orifice while a signal force is applied to the other end of the lever. Other types of servo control valves used in the Iroquois fuel system include manually selected valves, slide type valves, diaphragm-operated plate valves, solenoid-operated valves, and valves operated by an arrangement of pressure sensitive bellows.

The piston control rod in a servo motor can be used as an actuator in many different applications. In the main metering unit for example, the end of the control rod is equipped with a profiled plunger which moves up and down in an orifice plate to form a variable area metering valve. A second application of the servo motor in the Iroquois fuel system is the hydraulic actuator used to position the barrel valve located in the inlet of the air duct to the HP fuel pump turbines.

Proportional Flow Control

As previously mentioned, the flow through the fuel system is established by the orifice size of the servo throttle valve which controls the pressure drop across, and hence the fuel flow, through the metering unit. The orifice in the servo throttle valve is made variable by a servo motor which moves according to servo pressure signals originating in a 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.

Referring to Figure 5-7, the proportional by-pass flow in the Iroquois fuel system originates from the HP side of the metering unit and passes through two restrictions in series and located in parallel to the main flow. This arrangement of two restricting orifices is again similar to the general configuration previously described whereby the effective areas of the restrictions establish the flow rate of fuel passing through the circuit and regulate the pressure between the orifices.

In the proportional flow circuit, the intermediate reference pressure between the restrictions is applied on the pressure drop unit which converts the pressure signal into a servo control pressure. The servo control pressure varies inversely to the reference signal to suit the configuration and operates the servo-operated plunger in the servo throttle valve, which in turn regulates the main flow as previously described.

Thus, the main flow is established in proportion to the small by-pass flow by regulating the areas of the restricting orifices (A_1 and A_2) in the proportional flow circuit. It will be seen that a demand for an increase in the main fuel flow can be fulfilled by increasing the effective area of the first restricting orifice (A_1). Enlarging the area of the first restriction results in an increased reference pressure which via the pressure drop unit increases the opening of the servo throttle valve. This in turn increases the pressure drop across and hence the fuel flow through the metering unit. Conversely, increasing the area of the second restriction (A_2) reduces the reference pressure and hence reduces the main flow.

In the main control system, for example, the area of the first restriction corresponds in effect to the combined areas of a miniature throttle valve which is

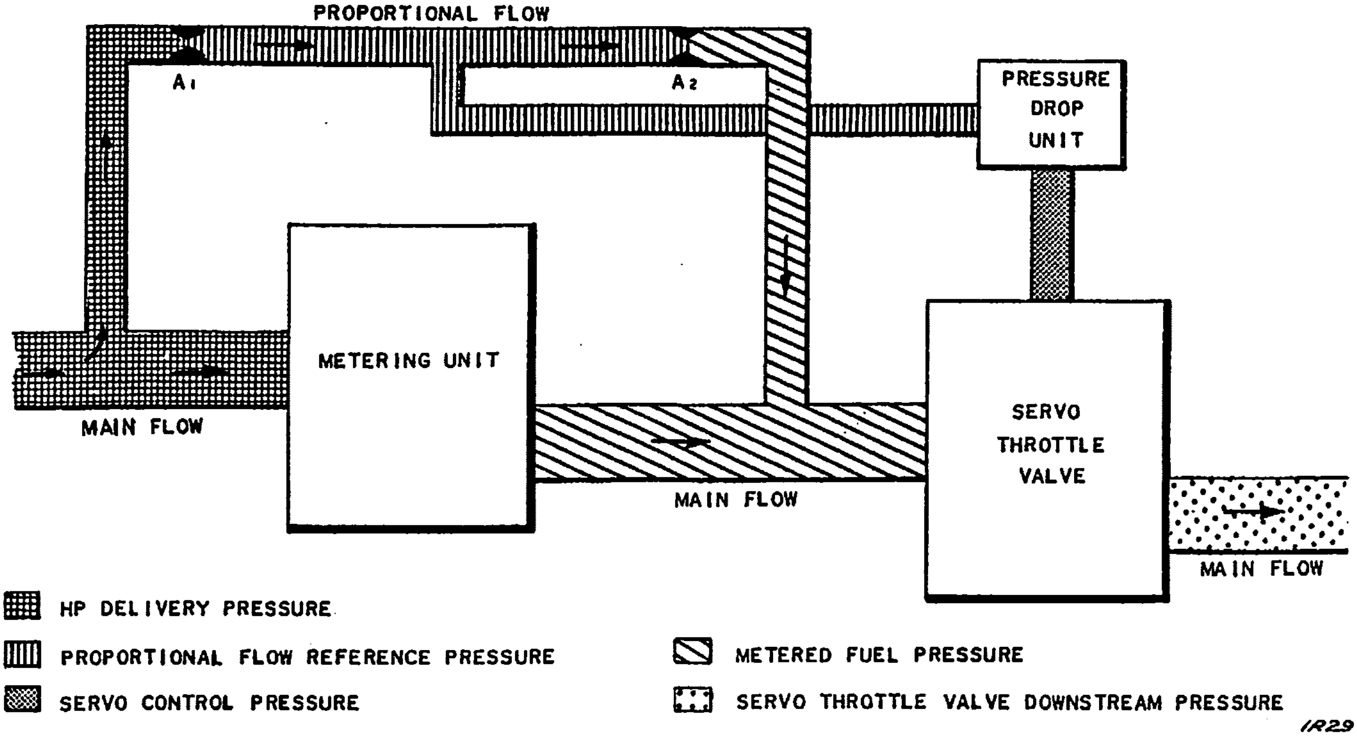
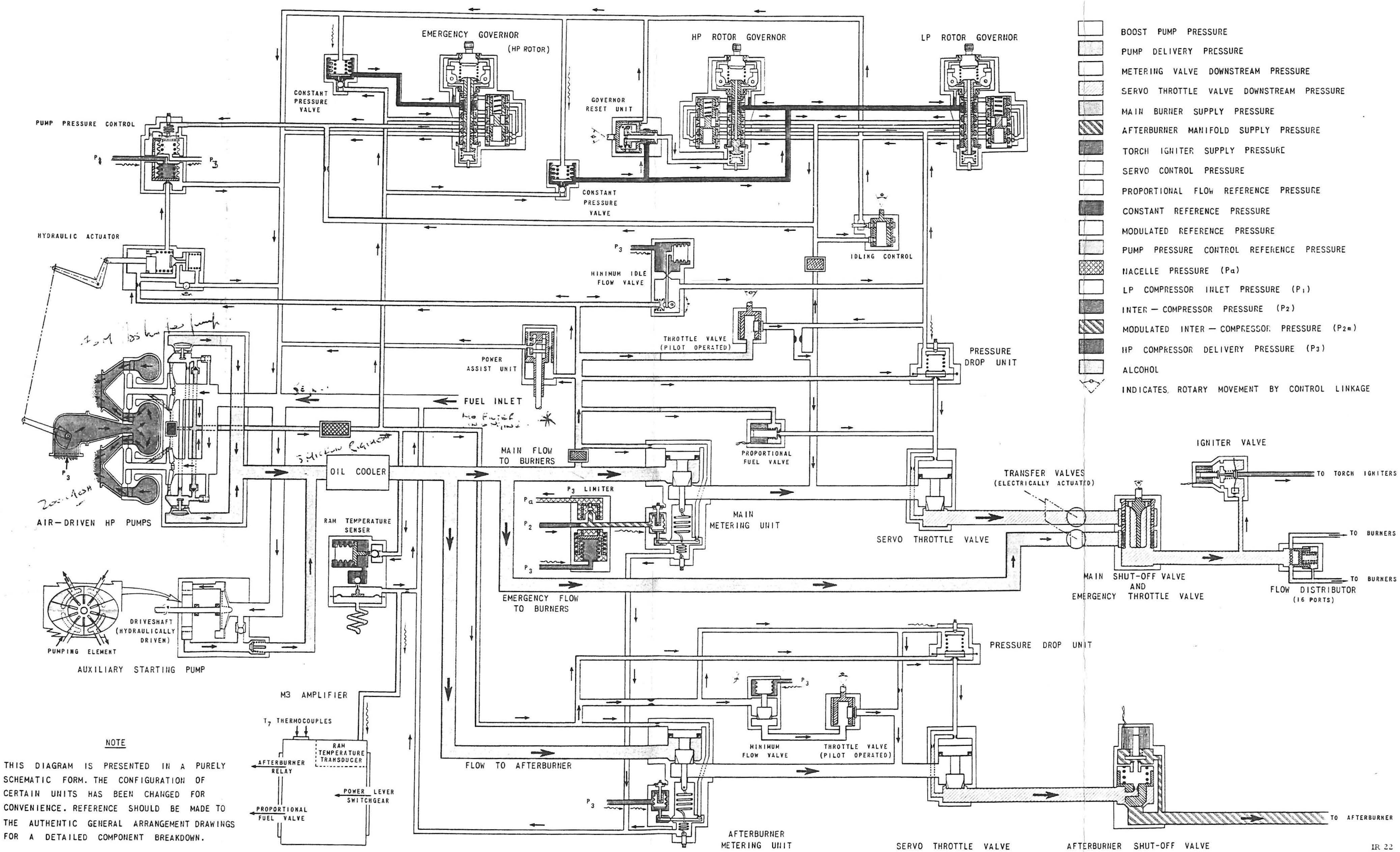


Figure 5-7 Proportional Flow Control

opened or closed by direct mechanical linkage as a function of power control lever movement, and a control orifice in the minimum idling flow valve which opens when an increase in fuel flow is required with an increase in altitude. The effective area of the second orifice corresponds to the size of a fixed area restriction together with the combined areas of the control orifice in the LP and HP rotor speed governors. The fixed area restriction is used in calibrating the proportional system to establish the desired main fuel flows. The governor control valves act essentially as override controls which increase the effective area of the second restriction only when necessary to stabilize or reduce the main flow when the required rotor speeds are reached or exceeded.

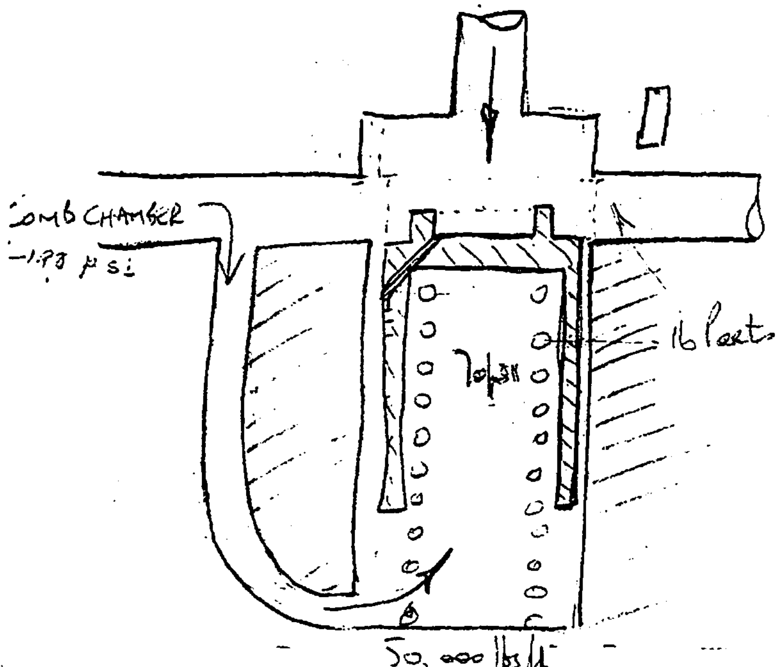


NOTE

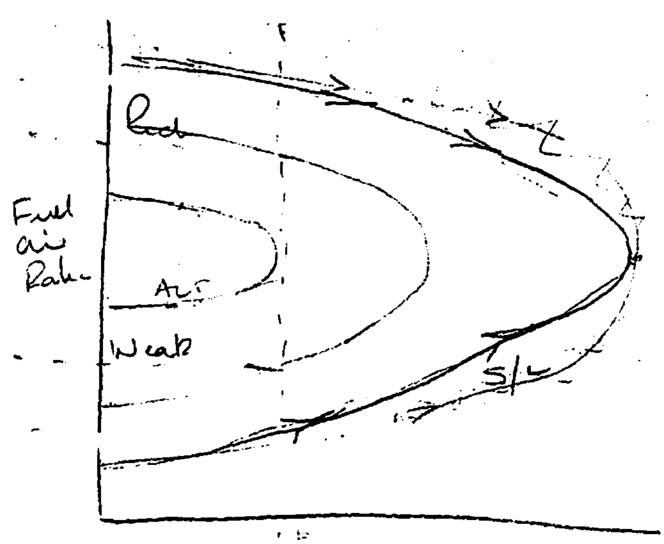
THIS DIAGRAM IS PRESENTED IN A PURELY SCHEMATIC FORM. THE CONFIGURATION OF CERTAIN UNITS HAS BEEN CHANGED FOR CONVENIENCE. REFERENCE SHOULD BE MADE TO THE AUTHENTIC GENERAL ARRANGEMENT DRAWINGS FOR A DETAILED COMPONENT BREAKDOWN.

Figure 5-8 Schematic Diagram of the Fuel System

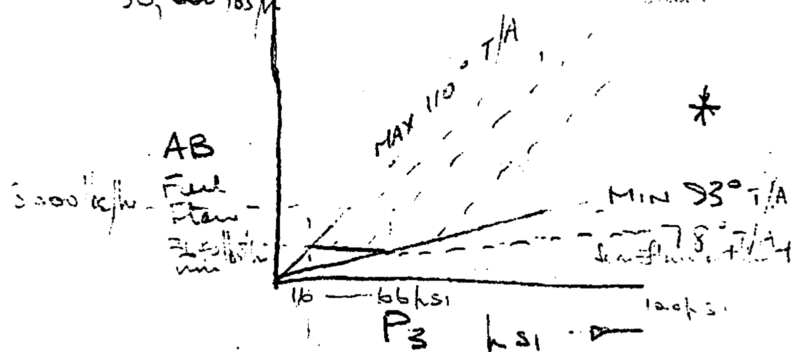
Distributor



Afterburner



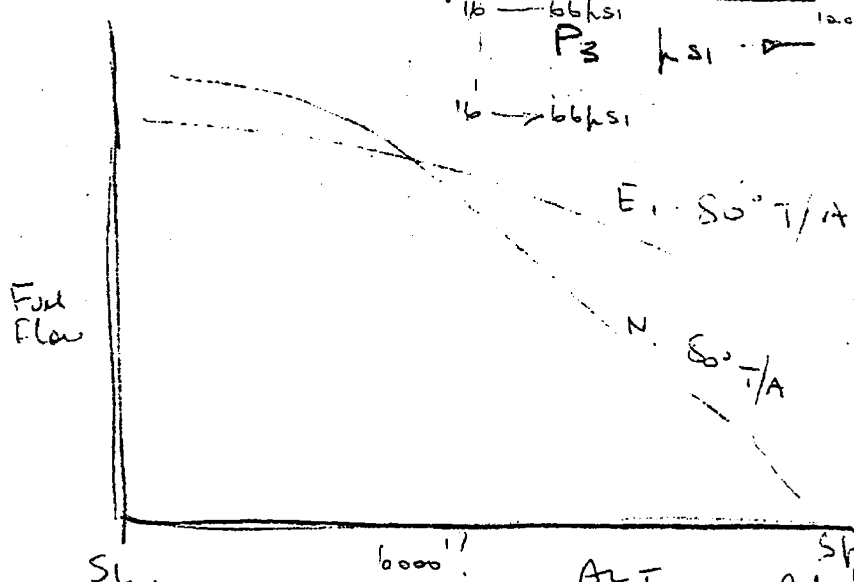
Rich blow-out limit JP Mach #



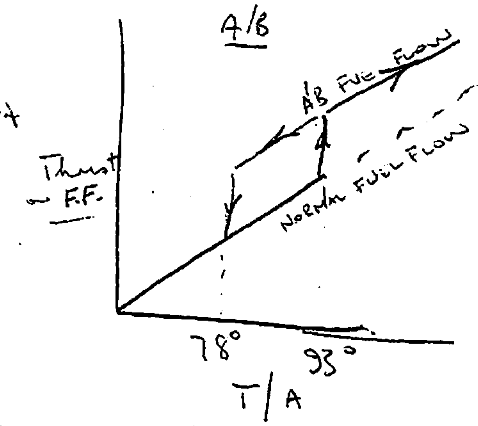
*

When P_3 values are between 16 & 66 psi this valve has control overriding throttle control.

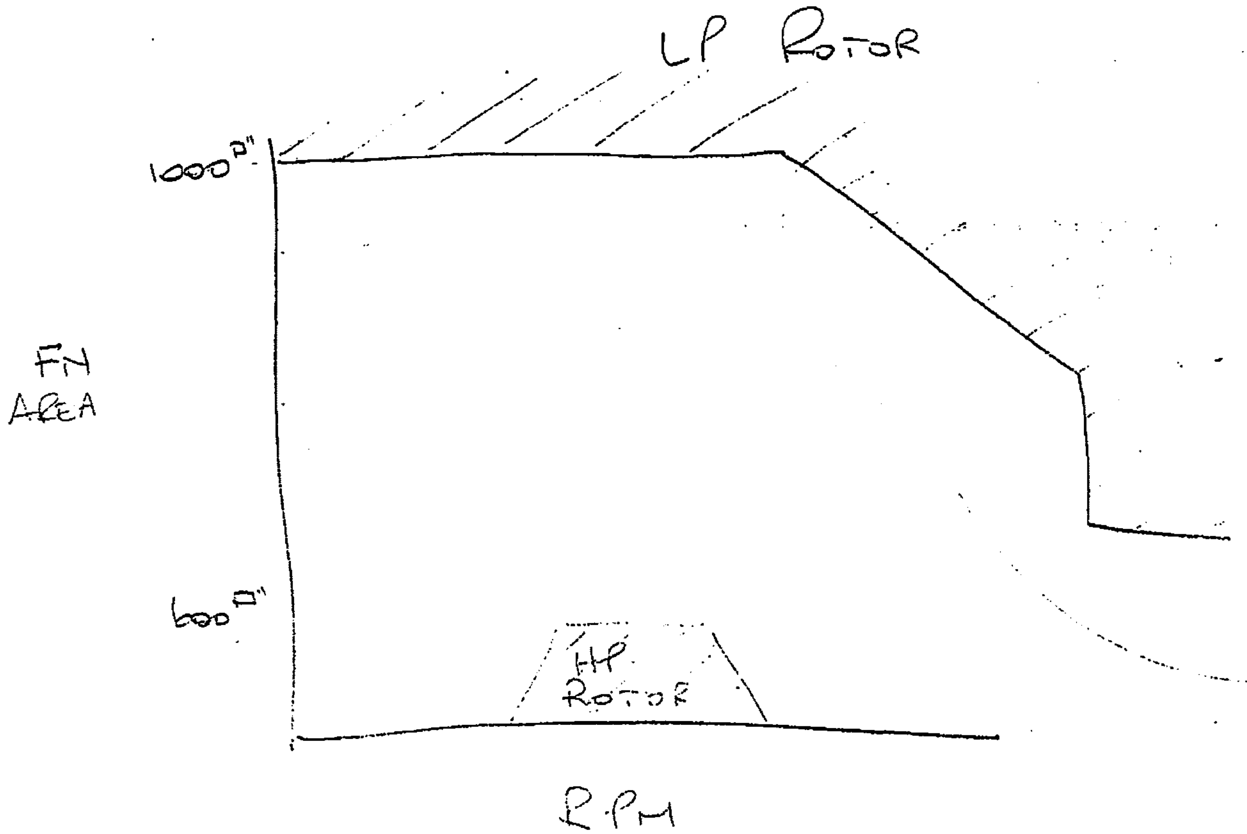
16 → 66 psi



Ideal change over N to E.



Spec. requires ability to maintain 93°/a at S/L (approach conditions)


LP ROTOR
1000 D"
FN
AREA
600 D"
HP
ROTOR
RPM

ELECTRICAL SYSTEM

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 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 and is 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, and indirectly, switches in the power lever switchbox which 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 6-3.

STARTING SYSTEM

Power Supply

The electrical power required to operate the engine starting circuit is drawn from the aircraft emergency DC supply. For ground starting, a starting trolley supplies 28 volts DC and 110 volts AC to the aircraft electrical system, and provides the compressed air required to drive the engine air turbine type starter. The compressed air supply is controlled by a solenoid valve which is connected to the aircraft electrical system, but which is controlled by the engine starting circuit.

Ignition System

A 12 joule high energy ignition system, consisting of a dual input ignition exciter box, two lead assemblies, and two surface discharge type igniter plugs, is used. The plugs are located one on either side of the combustion chamber. The voltage supplied to the exciter box is stepped up within the box by a contact breaker mechanism and transformer, and stored in a capacitor. The stored energy is discharged to each igniter plug through a sealed spark gap in the exciter box at the rate of approximately one discharge in one second. A separate circuit is provided from the DC supply source to each plug, thus ensuring that the failure of one circuit will not impair the functioning of the other.

Ground Starting Circuit

The ground starting circuit is arranged so that an engine start is obtained only if the prescribed operating sequence is followed. It is impossible to energize the circuit by accidental selection of the START-MOTOR switch only, as the starting system is open circuited until the power control lever is advanced beyond the four degrees position.

WARNING

Although the engine cannot be started on the ground by selection of the RELIGHT switch, this action does energize the ignition system. Serious injury or death might result if personnel are handling the HT section of the ignition system at the time the RELIGHT switch is selected.

An oxygen system control valve solenoid is connected to the engine starting circuit; as the oxygen system is not required during ground starting, the solenoid is held in the CLOSED position by the contacts of a scissors switch relay on the aircraft undercarriage when the aircraft is on the ground.

Ground Starting Sequence

NOTE

The starting sequence given here is only for descriptive purposes. For operational use, the Aircraft Operating Instructions must be followed.

The power control lever is advanced from the SHUT-OFF position to the IDLE position, closing a micro-switch in the power lever switchbox as it passes the four degrees position. The START-MOTOR switch is then held in the START position; this completes a circuit from the emergency DC supply through the K1 relay solenoid, the HP compressor 2500 rpm switch on the starter motor, and the four degree switch in the power lever switchbox to ground. The K1 relay closes and the START-MOTOR switch can be released as the relay is held closed by its own hold-in circuit. When the K1 relay closes, current flows from the emergency DC supply to energize the ground starting equipment air valve solenoid, which opens to supply compressed air to the engine air turbine starter. The starter in turn rotates the HP compressor shaft through the HP external gearbox.

A second switch on the starter closes when the HP compressor speed is about 200 rpm, thus causing relay K2 to close. Closure of the K2 relay energizes both the ignition exciter box and the solenoid which controls the starting fuel pump drive motor; the oxygen supply valve solenoid is not energized. The starting fuel pump must remain in operation until the main HP fuel pumps are supplying enough fuel to sustain the engine during the starting cycle, therefore a hold-in circuit, independent of the K2 relay, is provided for the starting fuel pump drive motor solenoid. This hold-in circuit is energized when the flow from the starting fuel pump is sufficient to operate a starting pump check valve switch.

When the HP compressor reaches a speed of approximately 2500 rpm, the associated starter speed switch opens, thus de-energizing relay K1 which de-energizes the ground starting equipment solenoid valve, and relay K2 which de-energizes the ignition exciter box. The check valve hold-in circuit retains the starting fuel pump in operation however, until the output from the main HP fuel pumps reduces the flow through the check valve switch to a predetermined value; the check valve switch then opens to de-energize the starting pump drive motor solenoid. This completes the starting cycle, and the engine will accelerate to idling speed.

Failure to Start

Should light-up not occur during the starting cycle, the starting circuit is reset by returning the power control lever to the OFF position. The micro-switch in the power lever switchbox opens at the four degrees position thus de-energizing relays K1 and K2, and the starting fuel pump drive motor control solenoid.

Failure to reset the circuit can cause overheating and damage to this solenoid and an unnecessary drain of electrical power. Furthermore the open circuit safety feature of the starting system is no longer effective.

Engine Motoring Circuit

In order to permit the engine to be rotated with the remainder of the starter circuit inoperative for inspection and servicing purposes, the START-MOTOR switch is selected to MOTOR. This action energizes the ground equipment air valve solenoid only. The switch is held in this position throughout the motoring operation.

Engine Relighting in Flight

The engine is relit in flight by advancing the power control lever past the four degrees position, whilst the RELIGHT switch is held in the closed position and the engine windmilling speed is controlled within the appropriate rpm range. The RELIGHT switch directly energizes and closes the K2 relay which in turn energizes the ignition exciter box, the starting fuel pump drive motor solenoid, and the oxygen supply valve solenoid (the contacts on the undercarriage scissors switch relay close when the aircraft is airborne). The RELIGHT switch is held closed until a rise in engine exhaust temperature is observed, indicating that combustion is taking place.

The use of oxygen helps to promote rapid engine relights in flight and extends the range of altitudes at which relights can be successfully accomplished.

Failure to Relight

If an engine fails to relight, the relight switch is released and the power control lever returned to the OFF position. When further attempts to relight are made, they should be as specified in the Aircraft Operating Instructions.

CONTROL SYSTEM

General

The engine control system is basically hydromechanical in operation, but some of the engine control variables are more easily sensed and applied electrically. These are used to provide trims and interlocks on the operation of the basic system.

The functions which are controlled electrically are as follows:

(a) Maximum exhaust temperature during start.

(b) Engine acceleration.

(c) HP compressor variable incidence inlet guide vane actuation.

(d) Exhaust temperature control during steady state operation in the upper power range.

(e) Prevention of afterburner light-up below a minimum engine speed.

Operating Principles

The components which provide the trims and interlocks are controlled by the electronic control amplifier. Control variables sensed at various points on the engine are transmitted to the control amplifier as input signals. These signals are inlet temperature (T_1), exhaust temperature (T_7), HP compressor rotor speed (N_H), and the power lever angle through the four degree switch and the afterburner switch. Within the amplifier this data is assimilated into output signals for system operation. The underlying principle behind the operation of the control amplifier is the use of circuits to compare voltages proportional to actual and desired values of various control variables. Where the actual value exceeds the desired value, the difference between the two is used to actuate a control device. Since the power level of input and error signals is generally small, the signal used to actuate the control device is amplified. In the case of modulated control signals amplification is achieved electronically; in the case of interlocks, relays are used.

Power Supply

The control amplifier is powered by 400 cycle AC and 28 volt DC from the aircraft primary power supply. To minimize the number of switches and to maintain the amplifier in a state of readiness, the amplifier is energized when the aircraft master switch is ON. When the aircraft is on the ground with the engine stopped, and it is necessary to energize the amplifier, the AC and DC power is supplied from an external source. A warm-up period of approximately 30 seconds is required for the amplifier to become operational.

Power Lever Switchbox

The engine mounted power lever switchbox contains three micro-switches and a potentiometer; the switches are actuated by a rotary shaft, in the switchbox, which is interconnected with the power control lever. Only two of the three micro-switches are utilized in the electrical system at present; the third switch is provided to accommodate possible future requirements. The central controlling member of the potentiometer is coupled to the end of the rotary shaft, thus the two micro-switches and the potentiometer operate as a function of power control lever travel.

One of the micro-switches is connected into the engine starting circuit and the other into the afterburner circuit; the potentiometer forms part of the exhaust temperature control circuit. The functions and settings of these three controls are detailed in the paragraphs describing the operation of the engine starting and control system.

Exhaust Temperature Control During Engine Start

During engine start the exhaust temperature must not be allowed to rise above a predetermined maximum value, in order to prevent damage to engine components. However, to ensure that the starting time is as brief as possible, it is desirable to provide the richest fuel/air ratio consistent with safe temperature limits, so that maximum torque is available for acceleration to idle.

When engine light-up takes place, the exhaust temperature, as sensed by the thermocouples, is transmitted to the control amplifier as a voltage signal. This signal is compared to a reference voltage, within the amplifier, which is at a level corresponding to the maximum permissible exhaust temperature during starting; the starting reference temperature is represented by Ta in Figure 6-1. When the sensed temperature exceeds the reference temperature, the voltage error, amplified to the required power level, actuates a solenoid controlled proportional fuel valve in the fuel system, to reduce the rate of increase of the engine fuel flow. The solenoid opens the valve to an extent proportional to the degree of excess temperature.

Reference to Figure 5-8 in Section 5, shows that when the proportional fuel valve opens, the servo control pressure acting above the piston in the servo throttle valve tends to increase. The servo throttle valve reduces the fuel flow rate to the engine, with resultant reduction in the exhaust temperature.

As the engine approaches idling speed, a signal from the HP speed sensing generator switches the reference voltage (temperature) in the amplifier to a lower value Tb (see Figure 6-1). It should be noted that the exhaust temperature at idle does not normally approach the value represented by Tb. The prime purpose of the Tb reference setting is to control acceleration from idle to prevent low speed surge.

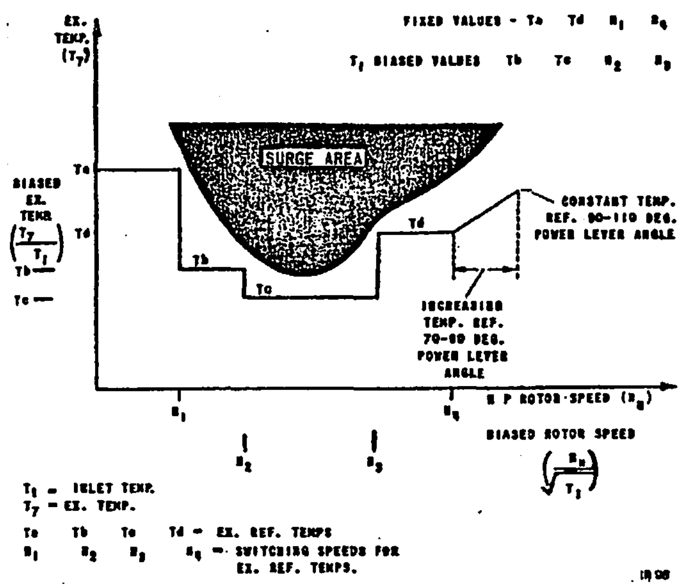


Figure 6-1 Exhaust Temperature Reference Diagram

Factors Determining Maximum Permissible Rate of Acceleration

Acceleration is controlled by scheduling engine fuel flow as a function of exhaust temperature (T_7) and HP compressor rotor speed (N_H). If the exhaust temperatures obtained at different engine speeds during steady state operation of the engine are measured and plotted, a curve known as the engine operating line is produced. It can be shown that as compressor inlet temperature (T_1) varies the curve is displaced in both the vertical and horizontal planes while still maintaining the same general shape. Similarly if the exhaust temperatures which just produce compressor surge at various engine speeds are plotted, a curve defining the surge area is obtained. This too is displaced by variation in inlet temperature, in the same manner as the engine operating line.

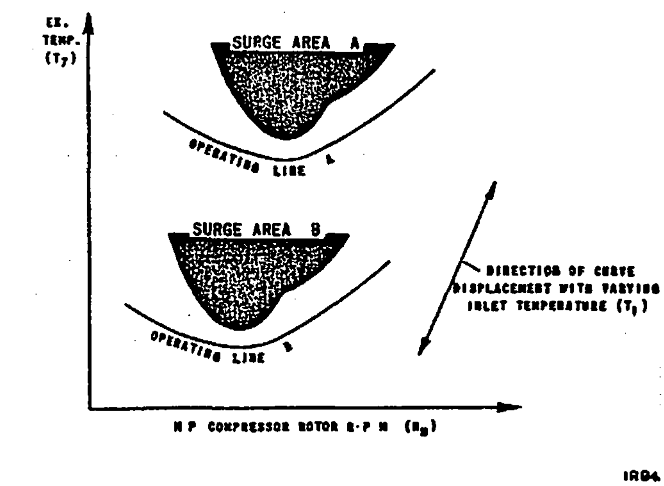


Figure 6-2 Surge Area and Operating Line Relationships

An indication of the general shape of

the surge area and associated operating line are given in Figure 6-2 for two differing compressor inlet temperature conditions. Surge area A and operating line A provide an indication of the curve obtained at the same relatively high inlet temperature, whereas surge area B and operating line B represent the curves obtained at the same relatively low inlet temperature. The general direction of curve displacement with varying inlet temperature is indicated by the arrows to the right. It is possible however to produce a single basic curve defining surge area and a similar basic curve defining the engine operating line, which cover all varying inlet temperature conditions, if T_7/T_1 versus $N_H/\sqrt{T_1}$ are used as the co-ordinates instead of simply T_7 and N_H. The manner in which this T_1 bias is sensed and applied is described later.

The margin between the engine operating line and the surge area is a measure of the excess torque, in terms of exhaust temperature, available to accelerate the engine. Maximum rate of acceleration is achieved when the exhaust temperature approaches, but does not enter the surge area.

An acceleration control which generates a reference temperature curve conforming precisely to the shape of the compressor surge characteristic is difficult to provide. A stepped approximation can however be readily produced by providing a series of fixed reference temperatures to which the engine can be controlled during acceleration, while maintaining a safe margin from the compressor surge area. This is illustrated in Figure 6-1; the four reference temperatures, Ta, Tb, Tc, and Td are selected automatically in the control amplifier.

Engine Control During Acceleration

When the power control lever is advanced from the IDLE position, an increase in fuel flow to the engine takes place, and a corresponding increase in exhaust temperature is sensed by the exhaust thermocouples. As previously described, if the sensed exhaust temperature exceeds the reference temperature (in this case Tb) the proportional fuel valve reduces the rate of increase of the fuel flow to maintain the required temperature limit. When N_2 switching speed is reached, a signal from the HP speed sensing generator switches the reference temperature to Tc. This reduction in reference temperature results in a reduced rate of engine fuelling while skirting the critical surge area. When N_3 switching speed is reached, the reference temperature is switched to a higher value Td, and the rate of fuelling is accordingly increased. At the same time the solenoid valve controlling the HP compressor variable incidence inlet guide vanes is de-energized causing the guide vanes to move to their high speed position, as described in Section 4/1. The reference temperature remains at the Td value until N_4 speed is reached at which point the margin between the engine operating line and surge area increases and acceleration fuel flow is therefore no longer restricted.

The Tb, Tc, N_2 and N_3 reference values are based on thermodynamic considerations and are biased by T_1 in order to compensate for shift in the surge area with inlet temperature; Tb and Tc are biased by $1/T_1$, and N_2 and N_3 are biased by $1/\sqrt{T_1}$. The T_1 bias is provided by the ram temperature senser which transmits a pressure signal proportional to T_1 to the control amplifier. Within the amplifier this pressure input is converted to electrical signals proportional to $1/T_1$ and $1/\sqrt{T_1}$ and applied to these temperature and speed input signals respectively.

Temperature references Ta and Td are established by temperature limits of the engine structure. Since these limits are absolute values, they are not biased by T_1. Switching speeds N_1 and N_4 are not critical, and are therefore not T_1 biased.

Exhaust Temperature Control in the Upper Power Range

When N_4 speed is reached, a speed switch in the control amplifier transfers the amplifier output from the proportional fuel valve to the proportional air valve; the exhaust temperature trim is then applied to the final nozzle area control instead of to the fuel flow.

As described in Section 4/1, the engine final nozzle area is controlled by the pressure ratio actuator to maintain a constant turbine pressure ratio (P_3/P_7). When the exhaust temperature sensed by the thermocouples exceeds the temperature reference, an output signal (voltage) from the control amplifier proportional to the degree of overtemperature, is applied to the proportional air valve, causing it to open. The extent of the valve opening is proportional to the applied voltage. When the valve opens, air is bled from the P_3 chamber of the pressure ratio actuator, thus the sensed P_3 pressure at the pressure ratio actuator will be less than the true P_3 pressure. This artificial reduction of P_3 pressure causes the pressure ratio actuator to reposition the actuator control valve in order to restore the original P_3/P_7 pressure ratio. The final nozzle actuators therefore move to increase the final nozzle area in order to reduce P_7. This also causes a proportional reduction in exhaust temperature (T_7).

When the exhaust temperature trim is transferred to the proportional air valve, the reference temperature (voltage) with which the sensed exhaust temperature is compared, varies with power control lever position. As the power control lever is advanced, the reference temperature is raised progressively through the 70 degrees to 90 degrees power lever range, and then remains constant to 110 degrees power lever angle. This provides automatic exhaust temperature control during both cruise and maximum speed operation in order to maintain high engine efficiency and ensure long engine life. The variable temperature reference is provided by the potentiometer in the power lever switchbox, which produces a variable voltage in the error detecting network of the control amplifier.

Afterburner Operation

Afterburning is initiated by advancing the power control lever beyond the 93 degrees position which closes a micro-switch in the power lever switchbox. A speed switch incorporated in the circuit to the afterburner electrical control delays light-up while the engine is still accelerating until a predetermined minimum speed is reached, in order to achieve minimum acceleration times and reduce the risk of surge. This speed switch closes when N_4 speed is attained. The afterburner control circuit is therefore energized causing a relay in the afterburner electrical control to close. This relay in turn energizes the solenoid-operated afterburner shut-off valve which opens and permits fuel to flow to the afterburner spray rings.

A second circuit in the afterburner control is energized simultaneously with the shut-off valve, and, after a two second delay, to allow for priming of the afterburner fuel manifold, energizes the hot streak igniter valve continuously for fifteen seconds. This time circuit is voltage compensated to ensure that variation in supply voltage will not affect the time cycle.

The hot streak igniter valve has a pulsating action, and injects a metered quantity of fuel at intervals of about three to four seconds into the combustion chamber, with a second injection point downstream of the turbines to serve as a relay. Fuel from the first jet ignites in the combustion chamber and passes through the turbines as a burning stream. This flame, augmented by the relay jet downstream of the turbine, is used to ignite the fuel being supplied to the afterburner spray rings. At the end of the fifteen second period, the hot streak circuit is de-energized; the shut-off valve solenoid, however, remains energized continuously while the afterburner is in operation.

The afterburner fuel flow is varied with power lever position to modulate afterburner thrust. To provide continuity to the thrust versus power lever position curve for certain operating conditions, such as formation flight, a hysteresis loop is provided which causes the shut-off valve to remain energized down to the 78 degree power control lever position. Afterburner fuel flow remains constant between 93 and 78 degrees power lever position and thrust is varied by varying engine basic fuel flow only. When the power lever is retarded below 78 degrees the micro-switch in the power lever switchbox opens. The shut-off valve solenoid is then de-energized and the valve closes to cut off the fuel supply to the afterburner manifold.

Emergency Change-over Control

As described in Section 5, fuel is supplied to the engine through either the NORMAL or EMERGENCY fuel system. Change-over from one system to the other is by means of motorized valves which are controlled by a two position selector switch in the pilot's cockpit. The switch is connected to the aircraft emergency DC supply so that fuel system transfer can be achieved in the event of failure of the main DC supply.

When the selector switch is moved from NORMAL to EMERGENCY or viceversa, it energizes a rotary actuator in the engine control system. The rotary actuator embodies an electric motor and reduction gear which drive two rotary flow transfer valves in the emergency control. These rotary valves are geared so that as one valve closes the other opens. The position of these valves determines whether the fuel flows through the NORMAL or EMERGENCY fuel system. The actuator incorporates cam-operated limit switches to de-energize the motor at the limits of travel and to reverse the direction of motor rotation. Accurate positioning of the transfer valves is ensured by an electromagnetic brake on the electric motor shaft. Current supply to the brake is controlled by the limit switches.

Engine Operation in EMERGENCY

CAUTION

When EMERGENCY is selected the acceleration control is rendered inoperative as well as a number of hydromechanical components in the fuel control system. Automatic control of engine speed and accelerations, and possibly exhaust temperature trim, are no longer in effect; therefore great care must be exercised in handling the power control lever and rapid changes of power lever setting must not be attempted.

Provided the amplifier control is serviceable, operation of the HP compressor variable incidence inlet guide vanes and the final nozzle area control and its associated temperature trim, will continue normally. Afterburning could also be carried out subject to the observance of exhaust temperature limitations.

If the amplifier is unserviceable, the inlet guide vanes move to the high speed position irrespective of engine speed. The engine is therefore more susceptible to surge on accelerating from low speed, and demands particular care in power lever handling. The final nozzle area control will function normally except that exhaust temperature trim is not applied.

NOTE

If the amplifier is unserviceable, afterburning when EMERGENCY is selected is not recommended.

ANTI-ICING SYSTEM

Anti-icing System Operation

The engine anti-icing system uses hot air piped from the HP compressor outlet to prevent ice accumulation at the engine air intake under adverse weather conditions. When icing conditions are encountered, a detector circuit in the aircraft operates a relay which supplies DC current to a solenoid-operated shut-off valve integral with the air pressure regulator, and allows hot air circulation to commence. The magnetic amplifier is simultaneously energized through the relay by the 400 cycle AC supply.

A temperature sensing element senses the inlet frame skin temperature and this temperature signal is amplified into an electrical output by the magnetic amplifier. The temperature control valve converts this output into a pneumatic control signal which modulates the hot air flow through the air pressure regulator as a function of inlet frame skin temperature.

The aircraft ice detector system operates on a time cycle; when icing conditions no longer exist, the relay opens to de-energize the engine anti-icing circuit, causing the flow of hot air to cease.

A detailed description of the engine anti-icing system is provided in Section 3, of these notes.

Serviceability Test of the Anti-icing System

The serviceability of the engine anti-icing system can be checked on the ground by supplying a false ice detection signal to the anti-icing magnetic amplifier. Provision is made in the engine anti-icing circuit for this check, which entails the removal of the flight connector from the engine firewall electrical connector panel and the plugging in of a specially instrumented test connector.

INDICATORS AND INSTRUMENTATION

General

The engine and airframe instrumentation requirements are not yet finalized. However, a projected arrangement, which may not be fully representative of the final layout, is shown in Figure 6-3. The following paragraphs describe the proposed instrumentation details.

Exhaust Temperature Indication

The engine exhaust temperature is indicated continuously in the cockpit, and, as already described, is also one of the signals used to vary the area of the final nozzle. Two separate thermocouple circuits are therefore provided for these two functions.

An eight probe dual thermocouple system senses the exhaust temperature; the probes project radially through the rear frame casing into the exhaust gas stream, downstream of the turbine. Each probe contains two chromel/alumel thermocouples, one of which connects to the cockpit engine performance indicator circuit, and the other to the control amplifier circuit. Each circuit transmits the average temperature sensed by its set of eight thermocouples. The thermocouple leads are shielded and incorporated in a separate harness to prevent induced interference from external electrical sources. The harness is routed forward to the airframe connector and control amplifier.

Nozzle Area Transducer

The incorporation of a nozzle area transducer is proposed as part of the engine performance instrumentation. Detailed information on this equipment is not yet available, but will be supplied at a later date.

RPM Indicator and Overspeed Detector

It is anticipated that an rpm indicator will be installed in the cockpit for the HP compressor, together with an overspeed warning for the LP compressor. To meet these requirements, and to ensure that a ready means of determining HP and LP compressor rotor speed is available when the engine is test run, provision is made for the mounting of tachometer generators on both the HP and LP external gearboxes.

Minimum Hydraulic Oil Reserve Warning

A magnetic switch mounted on the hydraulic oil reservoir operates a warning device when the minimum safe hydraulic oil reserve is reached. This warning enables the pilot to take precautionary measures before serious damage to the engine occurs, caused by loss of final nozzle area control.

Minimum Oil Pressure Detector

A pressure switch senses the pressure differential across the lubricating system supply pump. The switch operates a warning device in the cockpit when the pressure difference across the pump falls below a minimum value.

400 Cycle AC Power Failure Warning

The power failure warning equipment and circuit is part of the aircraft AC electrical system. For details refer to the applicable Arrow 2 Engineering Order. Failure of the 400 cycle AC supply results in the loss of acceleration control and exhaust temperature trim of the final nozzle area. Engine control can still be maintained satisfactorily in either NORMAL or EMERGENCY provided the pilot appreciates the effect of the AC supply failure on engine control, and acts accordingly.

Fuel Low Pressure Warning

If the fuel supply pressure to the fuel pumps falls below a predetermined value, a pressure switch, located on the inlet side of the engine fuel pumps, transmits a signal to an indicator in the pilot's cockpit. This pressure switch is incorporated in the aircraft electrical system.

Emergency System Indicator

When the NORMAL-EMERGENCY selector switch is in EMERGENCY a circuit is energized through one of the limit switches in the rotary actuator to operate an indicator in the pilot's cockpit.

FIRE DETECTION AND AIR BLEED VALVE CIRCUITS

Fire Detection Circuit

Two connectors on the engine firewall electrical connector panel are fitted to the end of the fire detector loop and connect into the aircraft fire detector system. A detailed description of the engine fire detector system is contained in Section 6/1 of these notes.

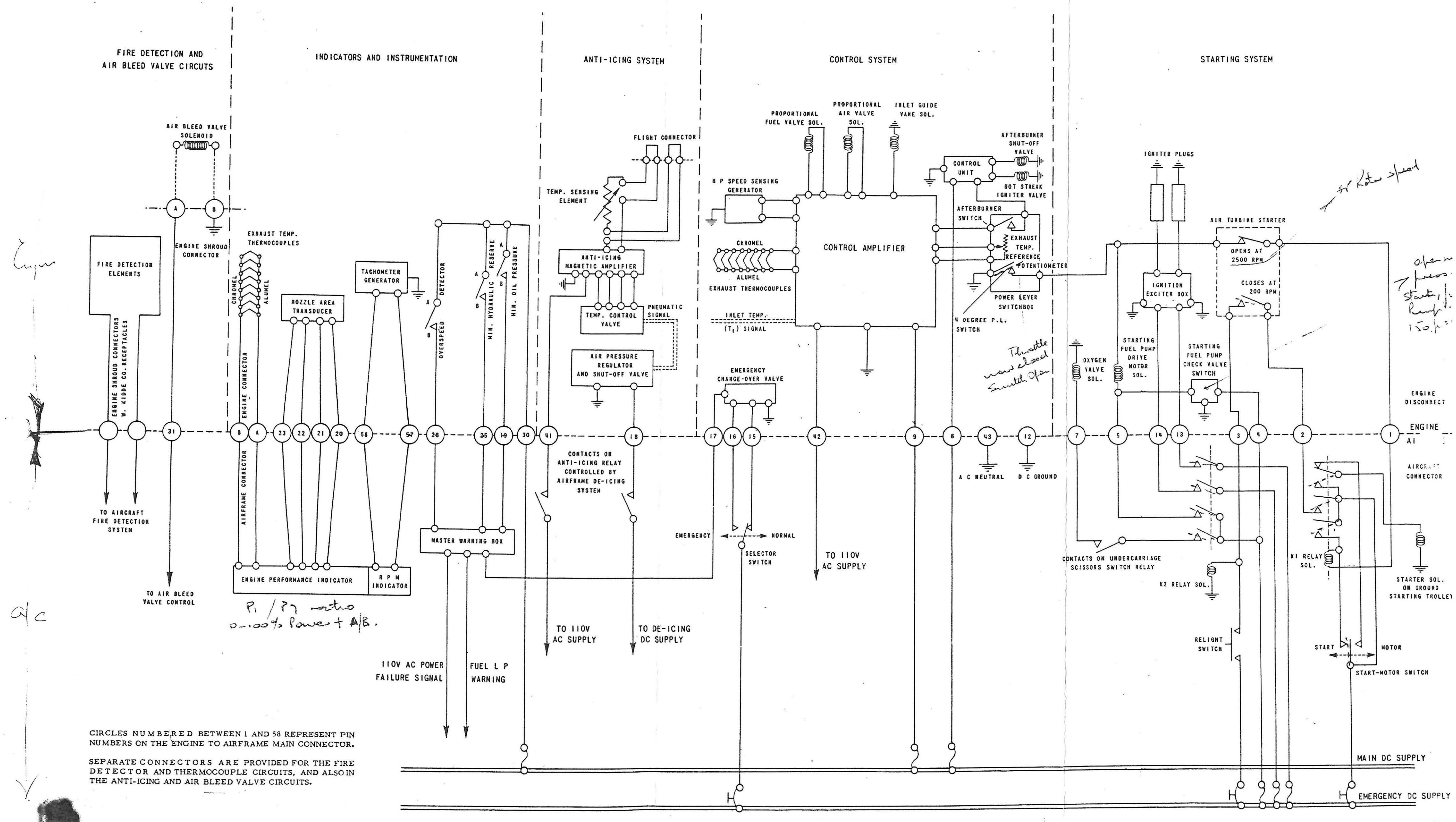
Air Bleed Valve

Prior to installation of the engine in the airframe, a solenoid-operated air bleed valve is mounted on one of the upper two air take-off pads located on the mid frame assembly air take-off manifold. The particular pad to which the valve is fitted is dependent on whether the engine is to be a port or starboard installation. The purpose of the valve is to control the supply of compressor tenth-stage air used for aircraft services such as cabin pressurization, fuel tank pressurization and air conditioning.

The valve is actuated by a control in the aircraft electrical system. This control circuit is routed to the main engine to aircraft connector on the engine firewall electrical connector panel. From this point a lead in the engine harness extends to a connector mounted on the firewall horizontal support member adjacent to the left-hand side of the compressor casing.

A separate harness containing the current supply lead and ground return, extends from the firewall support to the solenoid valve. The length of this harness varies with the disposition of the valve on either the LH or RH air take-off pad.

The only engine supply item in this circuit is the lead from the engine and airframe main electrical connector to the firewall horizontal support. The solenoid valve and harness are supplied by the aircraft contractor.



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.

Figure 6-3 Schematic Diagram of Electrical System

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.

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 towards the bottom centre at the mid-frame, and then rearwards 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; refer to the applicable Arrow 2 Engineering Order for details.

The three heat sensing elements are identical and consist of a semi-flexible Inconel tube enclosing a ceramic thermistor core and one internal wire. 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 wire 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°C (400°F) and a red light fire signal at 288°C (550°F). No signal is given for normal temperature changes up to the average maximum ambient temperature of 177°C (350°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

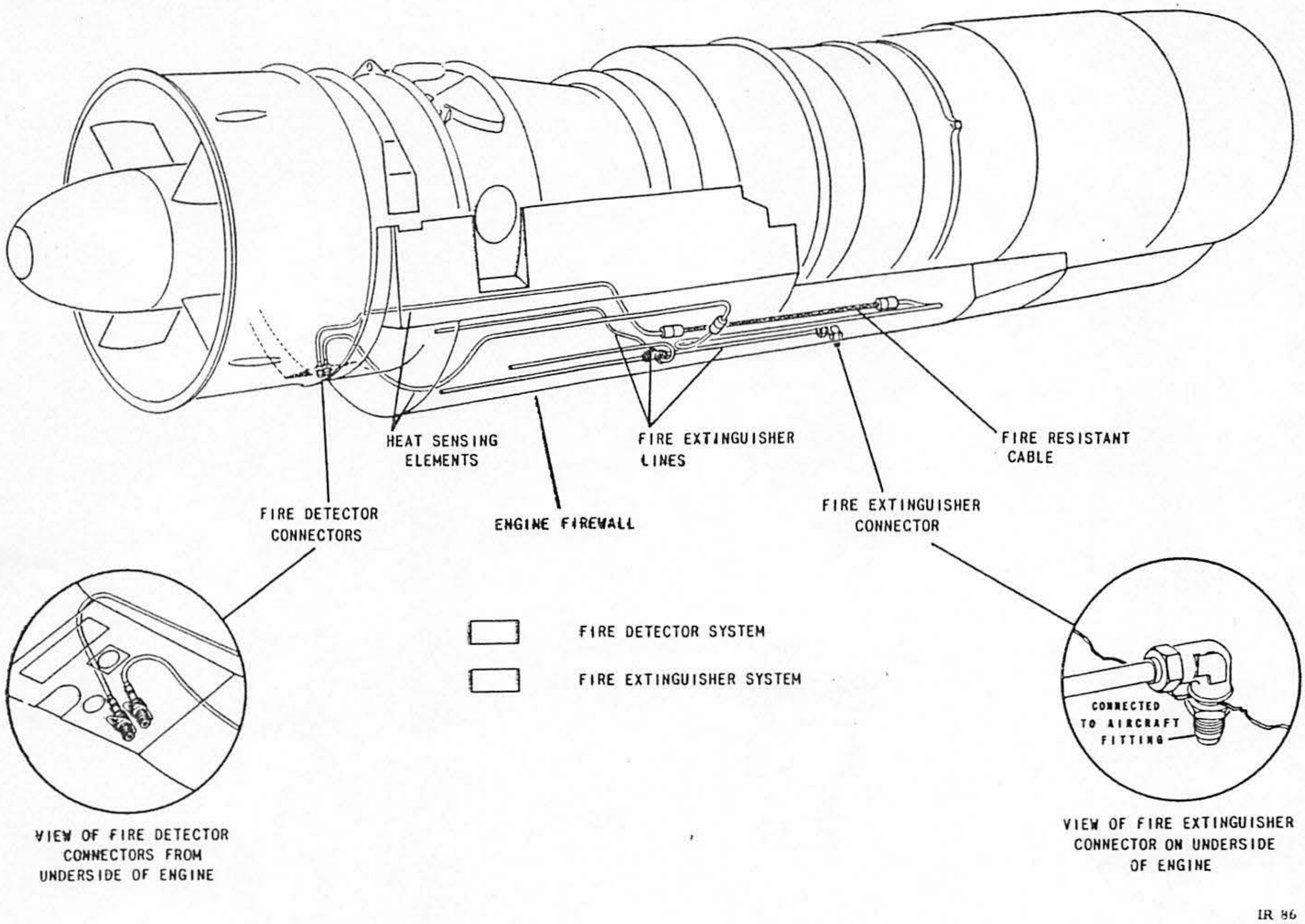


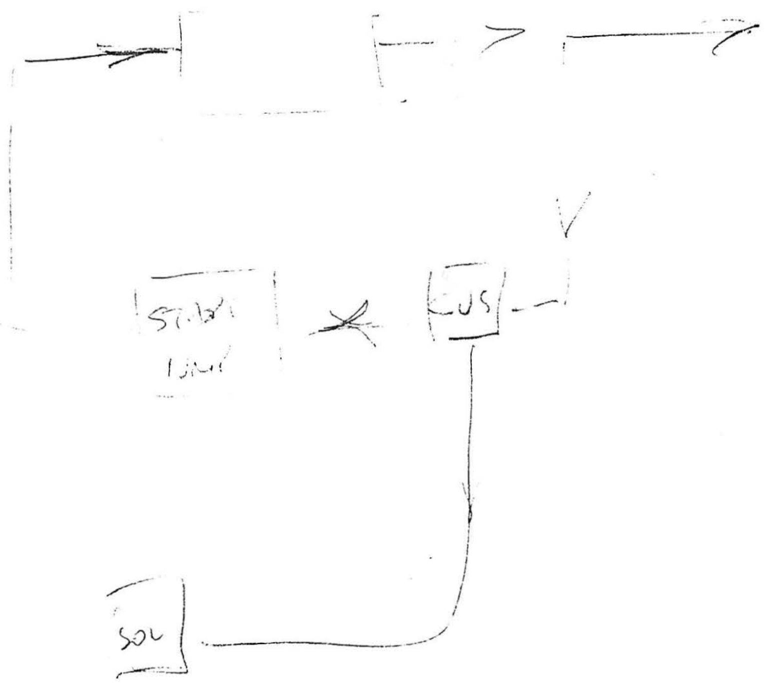
Figure 6/1-1 Fire Detector and Extinguisher Systems

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.

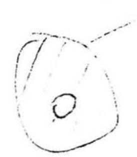
Fire Extinguisher System

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 regions fore and aft of the HP external gearbox. For details of the fire extinguisher bottles and controls, refer to the applicable Arrow 2 Engineering Order.

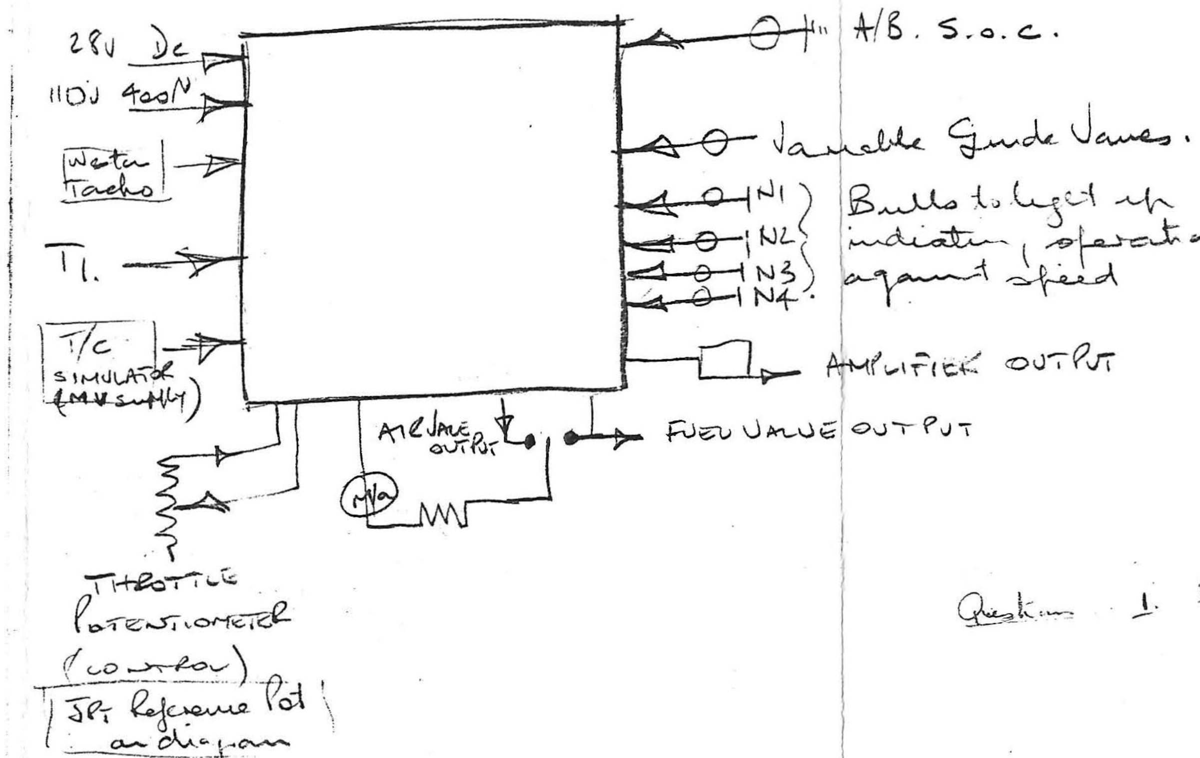
In the event of fire during an engine ground run, provision is made for a manual fire extinguishing procedure. Two fire traps, each consisting of a screened opening, are located one on either side of the engine firewall centre section, near the bottom of the engine. The screened openings line up with openings in the aircraft nacelle structure which permit the insertion of hand or mobile fire extinguisher nozzles.



Thermistor - Property of decreasing elect resistance with heat rise.



Bench Test Requirements



Questions 1. If T/C breaks how does ...