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PROPULSION

COMMERCIAL SUPERSONIC TRANSPORT PROPOSAL JANUARY 15, 1964

THE BOEING COMPANY

D6-2400-12

FAA SECURITY CONTROL
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ERRATA January 29, 1964

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	Page No.	Now Reads	Should Read
1	4/2 Para. 4.2.3	The gross thrust coefficient should read:	

$$C_{Fg} = \frac{\text{Gross thrust - nozzle drag (including ram drag of secondary air)}}{\text{Ideal gross thrust of primary air}}$$

	Page No.	Now Reads	Should Read
2	11/8 Para. 4.2.3	Gross thrust minus drag (C_{Fg}) is defined ...	Gross thrust minus drag (C_{Fg}) is defined as nozzle thrust minus nozzle drag (including ram drag of secondary air and nozzle boattail drag) divided by the ideal thrust of the nozzle primary airflow.

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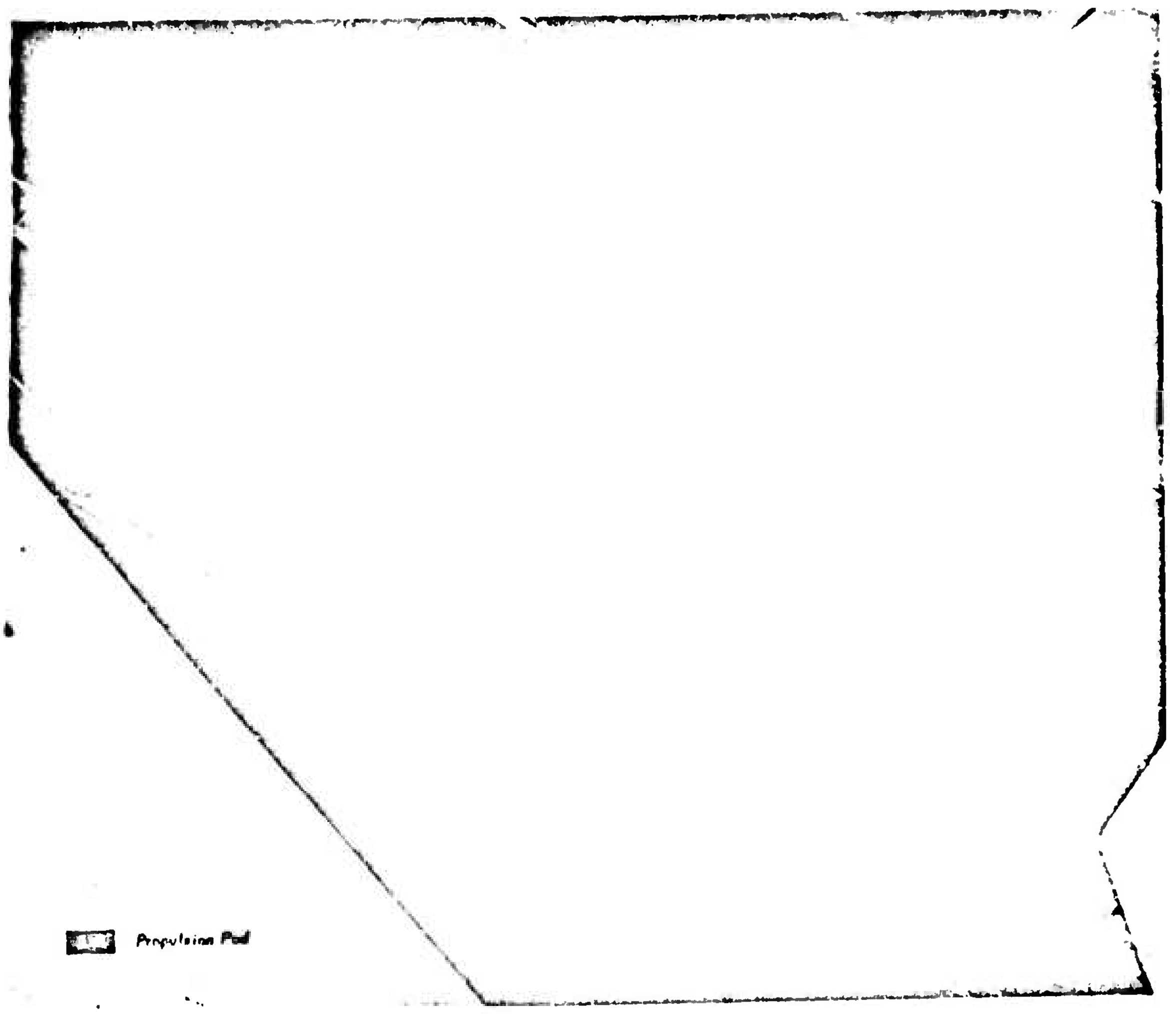

Propulsion Pod

1.0 SUMMARY (RFP 3.2.9)

The Boeing propulsion pod concept selected for the proposal airplane is shown in Fig. 1-1. Significant features incorporated in the pod for maximum performance, safety, reliability, maintainability, and serviceability include:

- A separable, self-contained, self-controlling, high performance, supersonic inlet with a differential, pressure-actuated, secondary air door system to maintain a safe and stable operating condition in the event of an inlet unstart.
- A separable exhaust system assembly which supplies its own ventilating and cooling air and which requires no inputs other than thrust lever angle to control its variable area functions for maximum performance.
- A thrust reverser integrated into the exhaust system which includes a partial reverse position for increased flexibility in airplane speed control during descent, landing, and taxi.
- A flight idle provision to allow the rotor RPM to be reduced as a function of airplane speed during normal descent to provide drag for deceleration and to save airplane fuel.
- A windmill brake to reduce engine rotor RPM in the event of in-flight engine shutdown.
- An accessory compartment containing all engine and airplane accessories in a low temperature environment.
- Non-pressurized conventional cowling which opens as two halves, exposing the complete engine case and accessories for ease of service and maintenance.
- Inherent fire safety due to the non-ventilated, non-pressurized burn-through cowling and the remote engine location provided by strut mounting.
- Freedom and flexibility in adapting the airplane configuration to any selected engine.

1.1 Engine Selection

At this time, a strong argument is not being made for any one of the specific engine offerings submitted in preliminary form on November 15, 1963. The propulsion pod for the proposed airplane is designed around the General Electric GE4/J4C engine. Although this engine appears to be the correct choice, based on the current RFP mission and available engine data, the Boeing configuration, using the propulsion pod concept, lends itself to use of any of the offered engines.

Boeing experience with the Model 707 has had a useful influence on engine selection. That program has shown that the aircraft manufacturer must consider the long-term utilization and growth of the aircraft and not make a point-design evaluation based solely on conditions existing at the outset of any program.

Early efforts to combine the lessons learned in the Model 707 program with detailed SST trade studies in support of airline forecasts led Boeing to the augmented fan as the desired cycle. This choice came about due to the desire for reasonable subsonic specific fuel consumption and low airport noise.

These early judgments have been altered recently, in part by unexpected improvements in turbine technology and by an increased understanding of sonic boom. The requirement for a limiting overpressure of 2.0 psf during transonic acceleration strongly influences engine cycle choice.

Improvements in turbine technology, both in materials and cooling techniques, have led to a reliable forecast that turbine flame temperatures 200° to 300° F. higher than was believed practical three years ago could be used when the SST becomes operational. This consideration raises the flight speed at which the turbojet is still superior to the fan. At a fixed, supersonic speed the fuel advantage of the turbojet is increased.

The importance of simplicity for maintenance, reliability and early availability make the turbojet cycle more attractive.

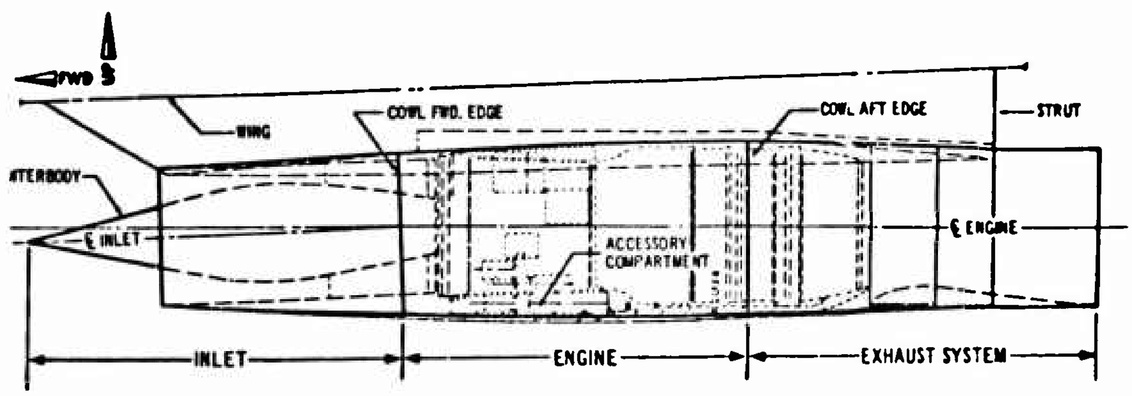
The Boeing Company, in selecting the engine to use in the proposal phase, worked with the engine manufacturers to ensure that the turbojets being offered at this time could be converted either to zero-staged turbojets or to turbofans at some later date. The change would occur if subsequent FAA-sponsored studies or airline needs should necessitate program redirection.

A complete discussion of the proposed engines and the selection of the basic engine for the proposal is contained in Section 11.

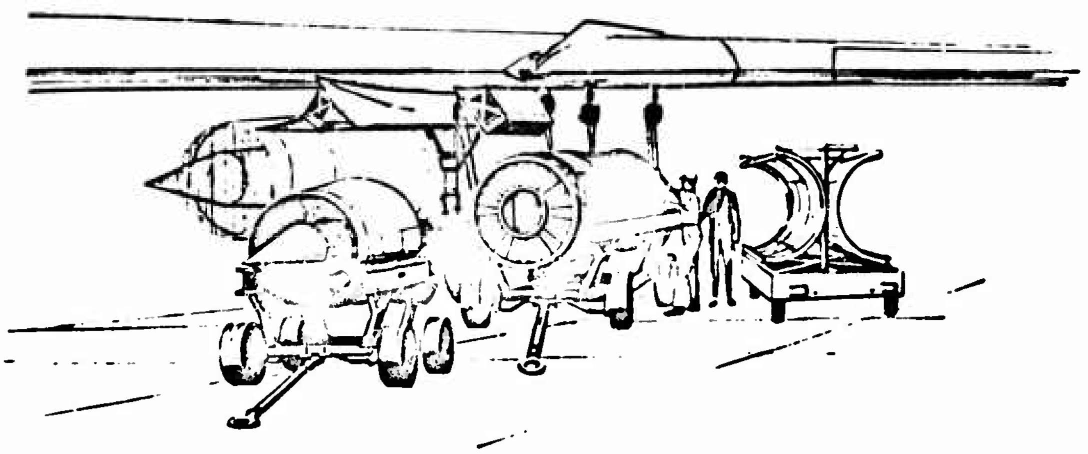
1.2 Propulsion Pods

There are four independent propulsion pods (Fig. 1-2). Each pod is hung by a strut to the underside of the inboard wing torque box. The engine is attached to the strut at three points with cone-type fittings. These fittings are self-aligning to simplify installation. An inlet assembly is bolted to the engine compressor case and an exhaust and reverser assembly is bolted to the engine turbine frame. The inlet and the exhaust sections may be readily removed from the propulsion pod for separate maintenance (Fig. 1-3). Cowling over the engine and strut fairings complete the propulsion pod. The inlet, the engine, and the exhaust section comprise a unit that can be assembled in its entirety for installation on the strut.

Because the unitized exhaust section provides its own cooling, the engine installation is not compromised with large ducts. Conventional two-piece cowling can be used. Airplane and engine accessories are arranged on the



1-2 *Pod General Arrangement*



1-3 *Propulsion Pod Ground Handling*

engine in the usual manner. This permits the same ease of servicing as is typical of existing airplanes. Opening or removal of the cowl panels exposes the entire engine build-up, including the portion under the strut.

1.3 Engine Inlet

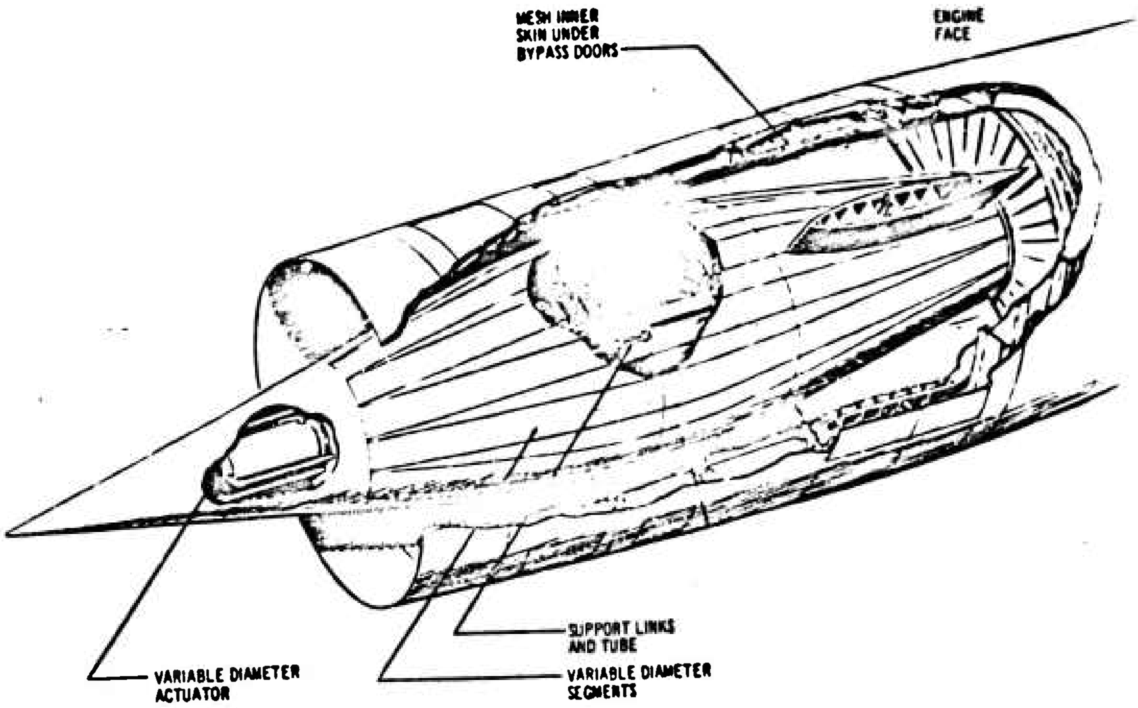
A new concept of a variable [illegible] non-translating centerbody inlet is submitted in this proposal (Fig. 1-4). Noteworthy advances in safety and efficiency are realized by this new design.

Control of the inlet is accomplished by an automatic control system which governs the position of the variable diameter centerbody and the controlled bypass doors. Natural forces act on secondary air inlet doors for takeoff and on secondary bypass doors to arrest shock expulsion. The system, except for the fuel supply pump, is self-contained within the inlet and requires no signal from the flight deck.

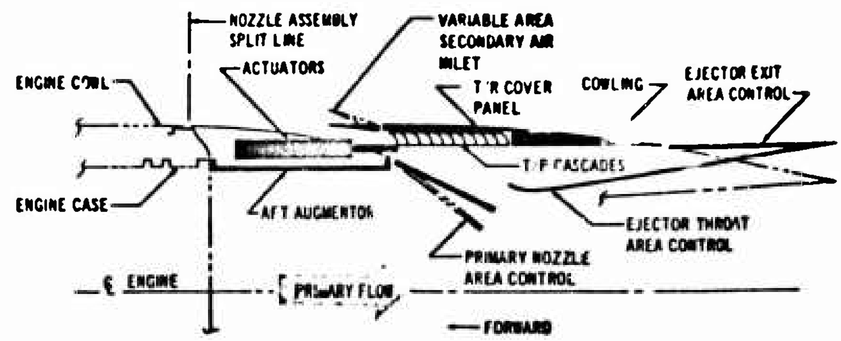
The inlet provides a cruise pressure recovery of 90 percent at a bleed penalty of only 5 percent. Performance is also high during off-design conditions.

1.4 Exhaust System

The complete exhaust nozzle and reverser, including integral cooling provisions, actuation, and cascade covers, is supplied by the engine manufacturer as a unit (Fig. 1-5).



1-4 *Engine Inlet*



1-5 *Left Hand Side View Exhaust System Forward Thrust*

Clear definition of responsibility for development and operation is thus ensured. Development of this hot section as an independent item, completely free of any airframe considerations such as cooling air, is assigned to the engine manufacturer.

For maintenance and serviceability the exhaust nozzle-reverser section is readily detachable from the engine. The engine exhaust system consists of:

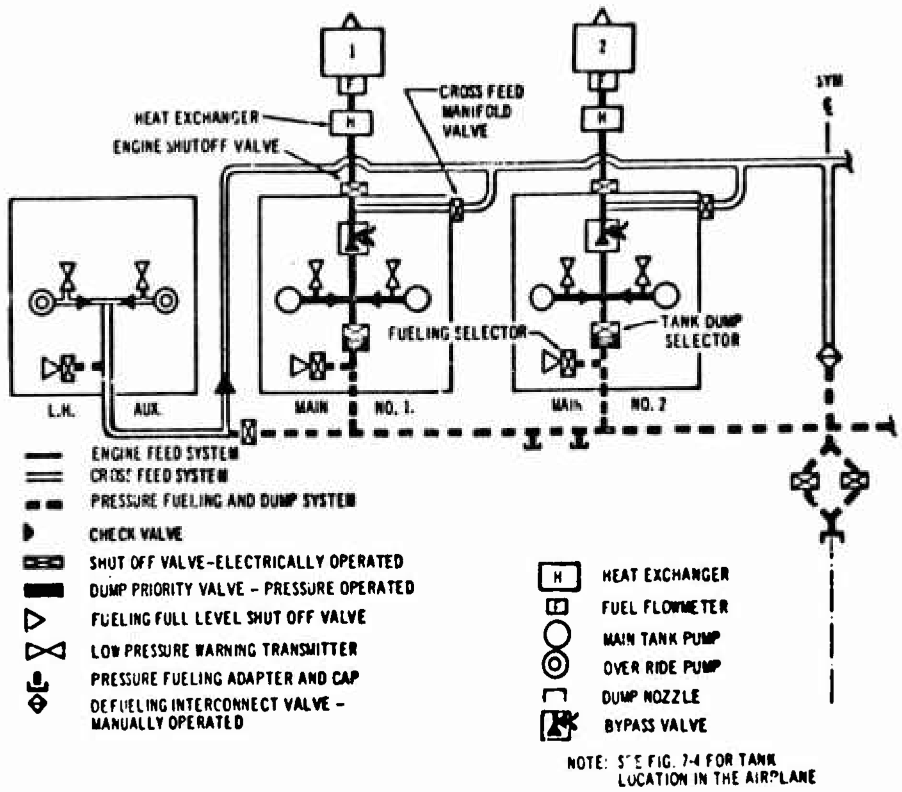
- The aft section of the engine augmentor case
- The variable area convergent-divergent nozzle
- The integrated thrust reverser
- The variable area secondary inlets for nozzle ventilation and cooling air
- Actuators, controls, and associated plumbing
- The exterior cowling from the aft end of the engine cowl panels to the nozzle exit.

The design of the system will be closely coordinated by Boeing and the engine manufacturer. Boeing will control the external lines of the exhaust system and integrate the exhaust and reverser systems into the propulsion pod.

Noise suppressors are not required. The takeoff, landing, and ground noise requirements are satisfied without special hardware. Nevertheless, The Boeing Company is applying its experience in testing, analyzing, and reducing noise throughout the design.

1.5 Fuel System

Simplicity is a feature of the SST fuel system, giving it the desired maintainability and reliability. Only four main and two auxiliary tanks are used (Fig. 1-C). Center of gravity control is maintained by a balanced arrangement of tanks and by feeding fuel directly from the tanks to the engines without monitoring or switching by the flight engineer or by the use of computing devices. The auxiliary tanks use an override pumping system to deliver fuel to the crossfeed manifold and selected engines. Re-



1-6 *Fuel System Schematic*

serves are equally distributed in the main tanks. System design precludes tank-to-tank transfer. Cross feeding from main tanks to an engine is used only to compensate for unusual conditions, such as an engine out or sustained differential fuel consumption.

A combination of thermal insulation, scheduled fuel usage, and natural oxygen depletion through an open-vent system eliminates coking and the formation of tank deposits. The need for inerting and purging is avoided by locating tanks in cooler portions of the airplane and by placing vent exits to avoid full stagnation temperatures. Transient overshoots to Mach 2.8 will not be hazardous.

Pressure fueling and defueling is done from two stations containing nozzle adapters, tank quantity gages, controls, and illumination.

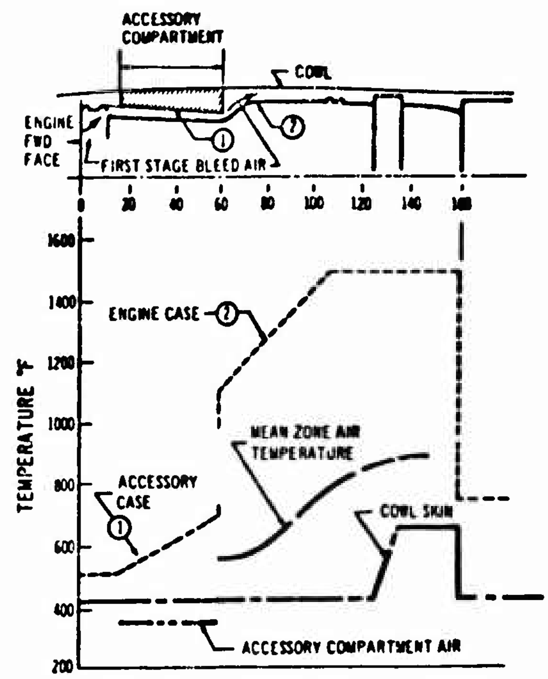
The dump system uses portions of the pressure fueling plumbing and the additional capacity in the engine feed system boost pumps to jettison fuel out of a fixed tube in the body tail cone.

1.6 Other Design Considerations

A pneumatic starter, mounted on the gear box of each engine, is used for engine starting. Air from either a ground source or an operating engine is used to drive the starter. The time required to start is approximately 37 seconds.

The engine oil system is an integral part of the engine and is furnished by the engine manufacturer. The system capacity is sufficient for all flight requirements.

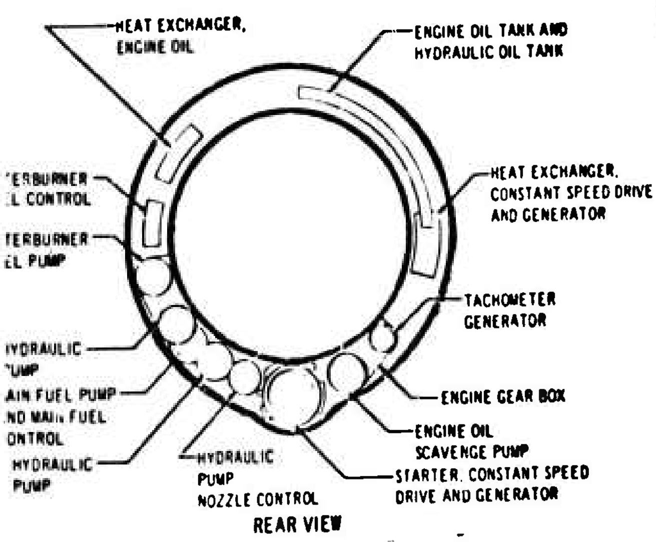
Engine fuel is used for cooling the accessories. The compartment housing the accessories, plumbing, electrical systems, and controls is an annular chamber insulated from the engine case. The outer wall of this compartment is formed by the insulated cowl panels. The aft end of the compartment is a conventional firewall barrier to the aft portion of the engine. The forward wall is a

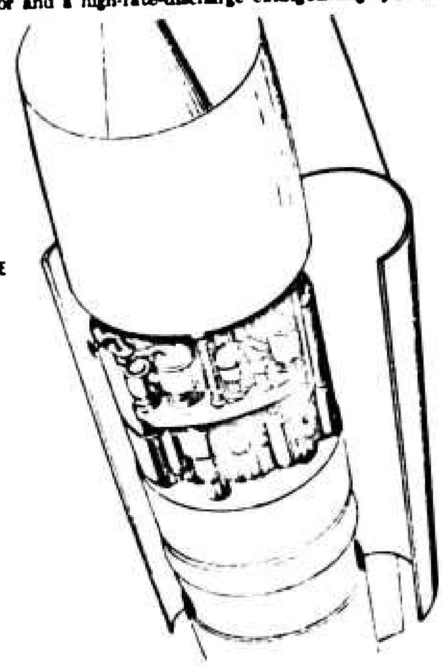


ENGINE COMPARTMENT TEMPERATURES CRUISE

1-7 *Engine Compartment Temperatures Cruise* portion of the inlet. Convection cooling to the many fuel-cooled items and lines establishes the temperature environment of the compartment. The environment will be less severe than that in the usual unshielded engine case accessory sections typical of today's jet transports (Figs. 1-7 and 1-8).

The engine compartment design minimizes the probability of fire or of serious damage if one does occur: the engine cowling consists of two hinged titanium alloy assemblies with aluminum burn-out panels; combustibles are separated from ignition sources; fluid drains are provided; air flow through the compartment is minimized. Fire protection is provided by a continuous-element fire detector and a high-rate-discharge extinguishing system.





1-8 Major Accessories Location

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PROPULSION

2.0 ENGINE INSTALLATION (RFP 3.2.9.5)

2.1 General Description

The major components of the propulsion pod are shown in Fig. 2-1. The pod consists of: (1) the supersonic inlet, (2) the engine section, (3) the exhaust section, and (4) the strut.

Several changes from the General Electric GE4/J4C November 15, 1963, proposal engine were necessary to make the engine compatible with the propulsion system installation. The deviations listed below have been agreed upon by General Electric.

- The engine support system designed to permit a three-point-attachment method.
- The forward flange of the engine designed to support the supersonic inlet through use of a bolted flange arrangement.
- The engine accessory gear box enlarged to include provisions for direct mounting of a starter, two hydraulic pumps, and a generator with its constant speed drive.
- The compressor outlet guide vanes designed to be rotated to an overlapping position by moving the engine-start lever to the cut-off position. This provides a windmilling brake.
- The engine accessory compartment insulated from the engine-case temperatures by an engine-mounted, Boeing supplied annular shell. Boundary layer bleed air from the first stage of the compressor flows at a very low rate between the shell and the engine case. The shell is fitted with thermal insulating blankets.
- The engine fuel control designed to include: (a) an unlocked rotor regime for flight-idle as a standard operating procedure; (b) a partial-reverse-thrust operation band; and (c) a special bias to open the stator angles during reverse-thrust operation.
- The engine exhaust system designed to include variable-area boundary layer air scoops to provide ventilating and cooling air to the exhaust nozzle.
- The exhaust gas exit path for reverse-thrust operation tailored to match the propulsion pod positions on the airplane.

The supersonic inlet is bolted to the forward face of the engine. The aft, or exhaust, section is furnished by the engine manufacturer.

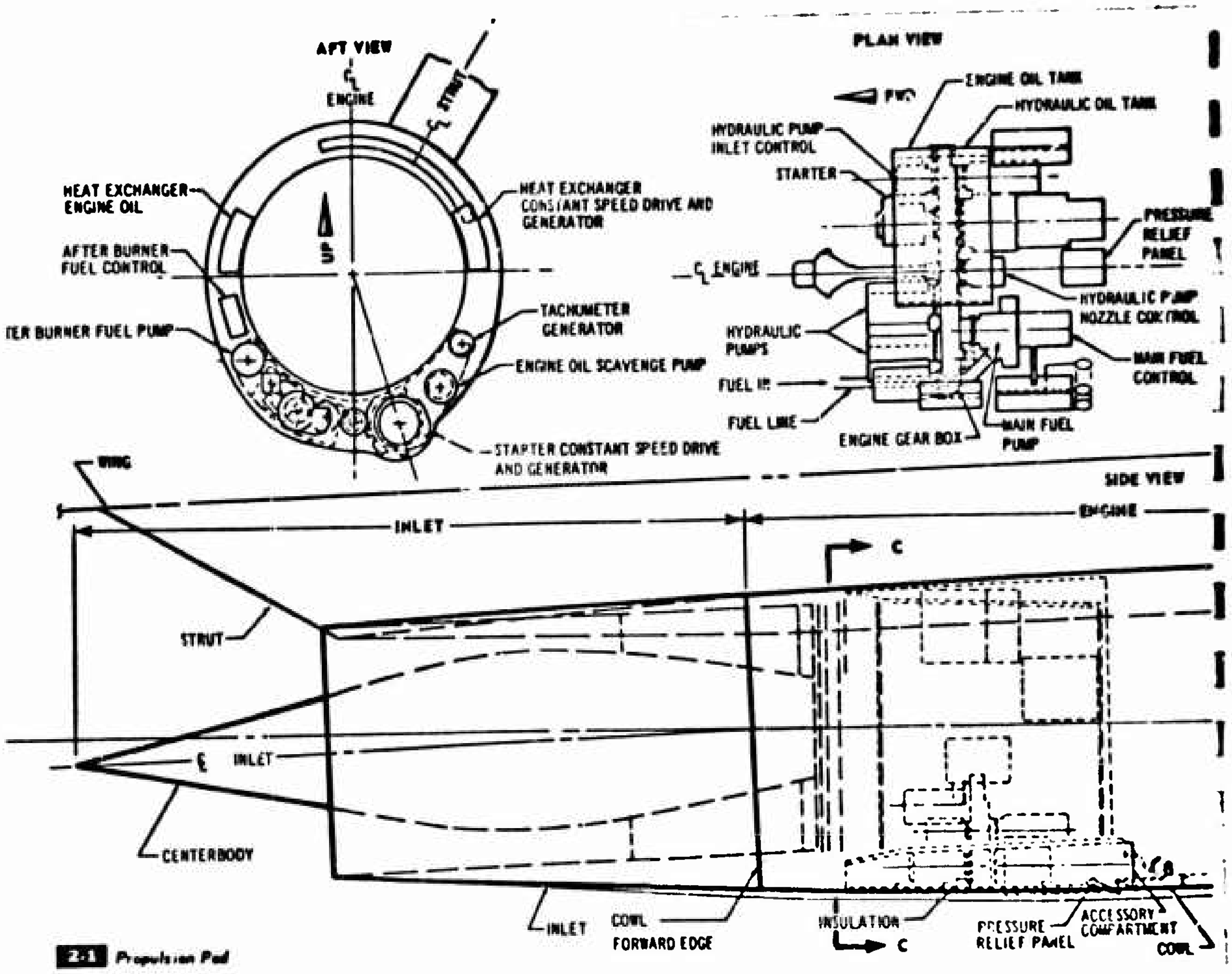
The engine section is the center portion of the propulsion pod. Contained within the engine section are the engine and its mounts, the engine accessories, the engine-driven airframe accessories, and the engine-instrumentation transmitters, together with associated plumbing, wiring and controls. To provide rapid and unhampered access to all areas requiring frequent servicing and maintenance, hinged cowling with quick-release latches enclose the engine section.

This propulsion pod provides the same easy and simple acess to engine components characteristic of subsonic jets. Complete propulsion pods can be built-up in their entirety before installation. By this method, complete pods can be placed at strategic locations throughout the world for use as pool stock.

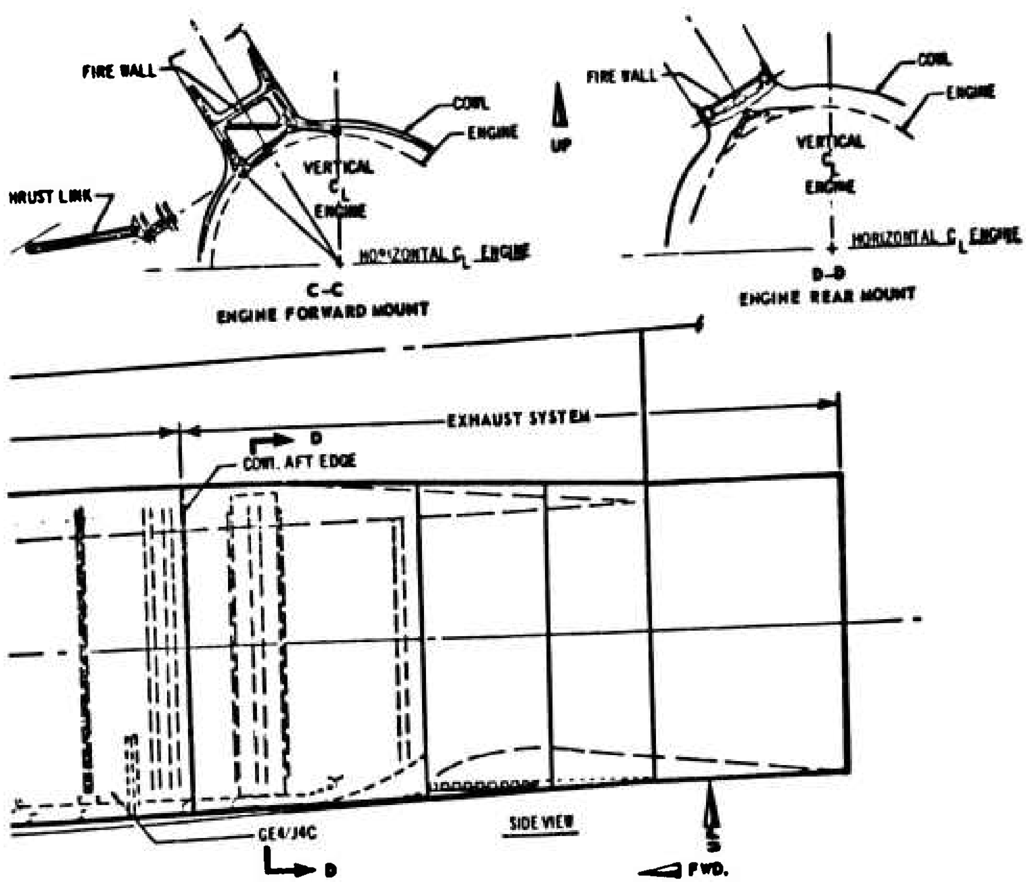
2.2 Mounting

Each propulsion pod is attached to the underside of the inboard wing torque box. The center section or structural portion of the strut contains two structural bulkheads made of heat-treated AISI 17-4PH corrosion-resistant steel. These bulkheads accept the engine-mount loads and transmit them to the wing front and rear spars. Shear fittings are used at the wing-to-strut attach points to provide rapid removal and installation of the strut. Forward and aft of the structural portion of the strut are nonstructural fairings. These are attached to the lower surface of the wing with quick-release fasteners.

The engine is attached to the strut by a conventional three point attach system similar to that used on the Model 707 commercial jet. Details of the system are shown



2-1 *Propulsion Pod*

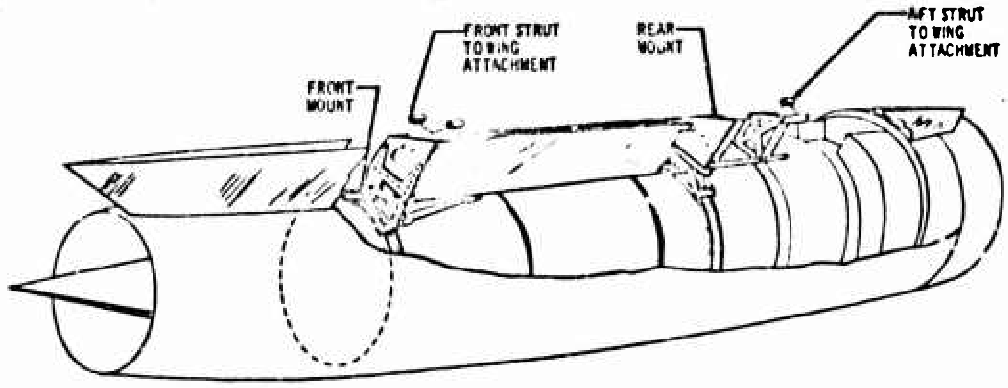

FIRE WALL
COWL
ENGINE
UP
HRUST LINK
VERTICAL
ENGINE
HORIZONTAL C_L ENGINE
C–C
ENGINE FORWARD MOUNT
FIRE WALL
COWL
ENGINE
VERTICAL
ENGINE
HORIZONTAL C_L ENGINE
D–D
ENGINE REAR MOUNT
EXHAUST SYSTEM
D
COWL AFT EDGE
GE4/J4C
D
SIDE VIEW
FWD.

in Fig. 2-2. Two attach points are on the engine forward mount ring and one attach point is on the engine rear mount ring. The engine is installed by seating a cone bolt at each of the three attach points and torquing a nut on each bolt. Within the cone bolt, engine vertical loads are taken in bolt tension; thrust and side loads are taken in bearing on the cone socket. The cone bolts are self-aligning and simplify engine installation by eliminating the need for precise alignment of matching holes before a bolt can be inserted. Two cone bolts are attached to the engine by links which transmit the engine loads tangentially from the engine case. Two forgings attach the cone bolts to the strut structure. The forging for the forward attach points is fixed and is part of the strut. The forging for the aft attach point, also part of the strut, is hinged to allow for engine expansion. The cone bolts and the links are made of heat-treated AISI 17-4PH corrosion resistant steel as are the strut forgings.

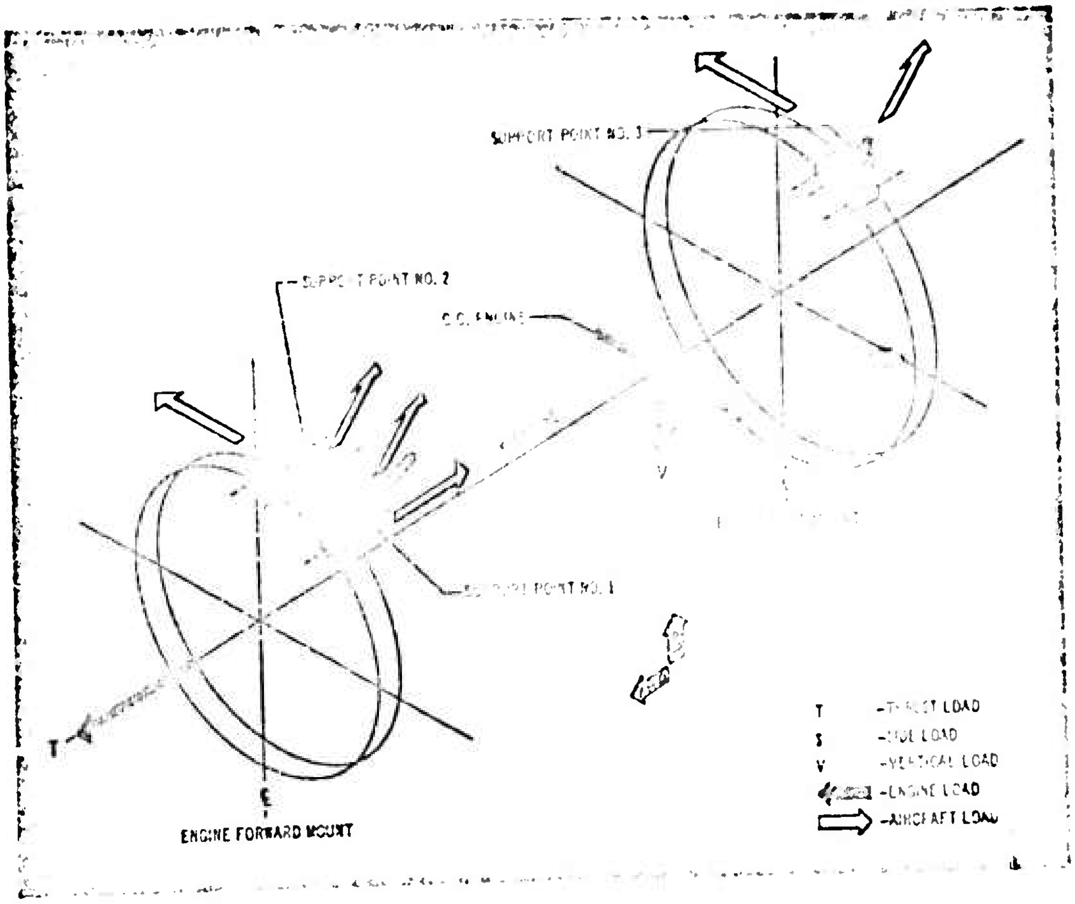
The load diagram for the three point mounting system is shown in Fig. 2-3. Engine thrust is taken totally at Point 1. Engine side and vertical loads are taken at Points 1, 2 and 3. Engine seizure loads are taken at Points 1 and 2.

2.3 Engine Oil System (RFP 2.10; 2.25.2)

The engine oil system (Fig. 2-4) provides for engine lubrication and is an integral part of the engine. It is furnished



2-2 Engine Mounting



2-3 ***Engine Load Diagram***

by the engine manufacturer. The system is comprised of the following items: an oil tank of nine gallons total oil capacity, which also contains a deaerator; a fuel oil heat exchanger, which cools the oil; pumps which provide for scavenging and pressurizing the lubrication system; and an oil filter, which is bypassed should the filter become clogged.

The instrumentation required to monitor the oil system (Fig. 2-7) is furnished and installed on the engine by Boeing. Components of this instrumentation are:

- Oil quantity probe
- Oil pressure transmitter
- Oil low-pressure warning switch
- Oil temperature probe
- Oil filter pressure transmitter
- Oil breather pressure transmitter.

The oil flow circuit for engine lubrication begins with the oil being scavenged from the engine bearings and pumped into the fuel oil heat exchanger where it is cooled. From the heat exchanger, the oil flows through a filter and check valve to a tank where it is deaerated. From the oil tank, the oil passes in series through a pressure pump, a filter, a check valve, and on to the various bearings requiring lubrication. The system operates at a normal pressure of 40 to 70 psig with pressure relief occurring at 100 psig. The oil-in temperature limits are −40° F. to 425° F. For engine starts at lower temperatures, some external heating will be required.

The type of oil used is per General Electric specification, GEA 60T 20A. An experimental oil, Esso WSX-5435, is being qualified to the General Electric specifications. The nine gallon tank is more than enough for the maximum endurance of the airplane at the guaranteed maximum oil consumption of 0.50 gallons per hour.

2.4 Accessories

To provide maximum reliability and the best possible location for maintenance and serviceability, the main hydraulic pumps and the constant speed drive electrical generator are mounted directly on the engine. These airframe accessories are adjacent to the engine starter and also to the engine accessories. Fig. 2-5 shows the location of the major accessories mounted directly on the engine and readily accessible through the open cowl panels.

The hydraulic pumps are cooled by the fluid that is passing through them. The constant speed drive and the generator are both cooled by the constant speed drive oil. The oil is in turn cooled by a fuel-to-oil heat exchanger. A schematic diagram of this cooling system is shown in Fig. 2-6.

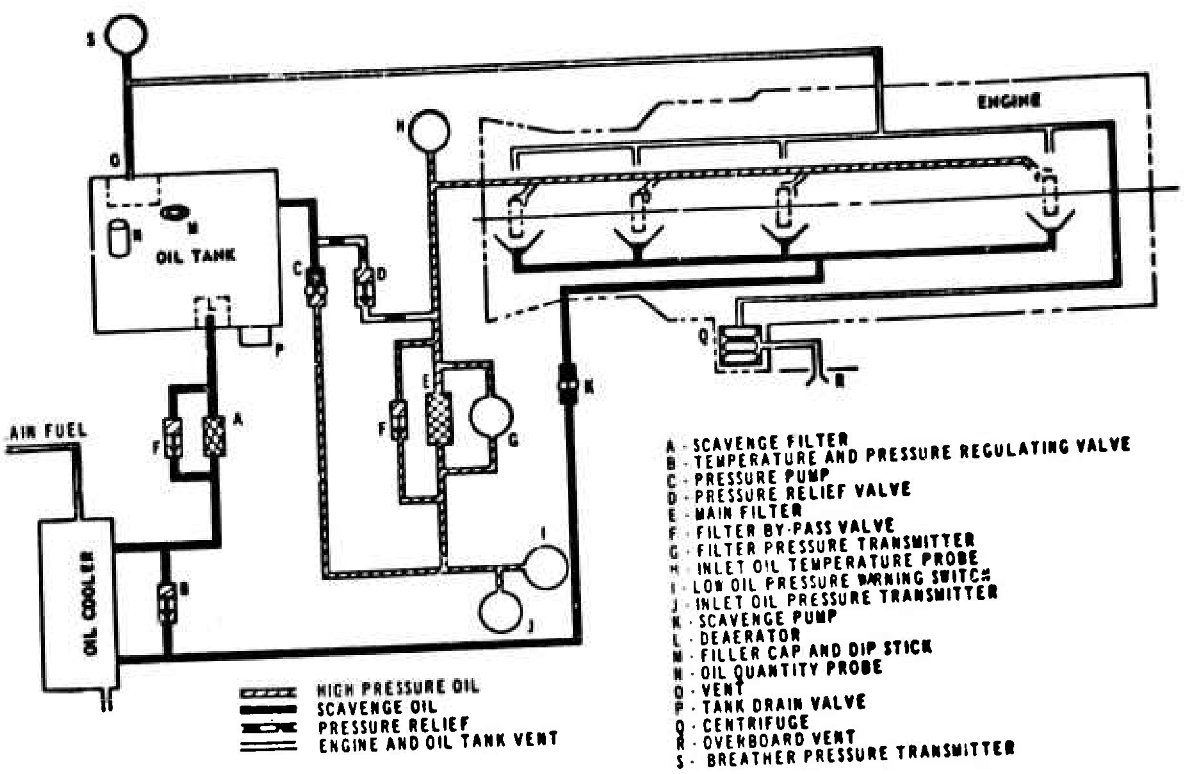
2.5 Instrumentation and Engine Analyzer

2.5.1 INSTRUMENTATION (RFP 3.2.11.5)

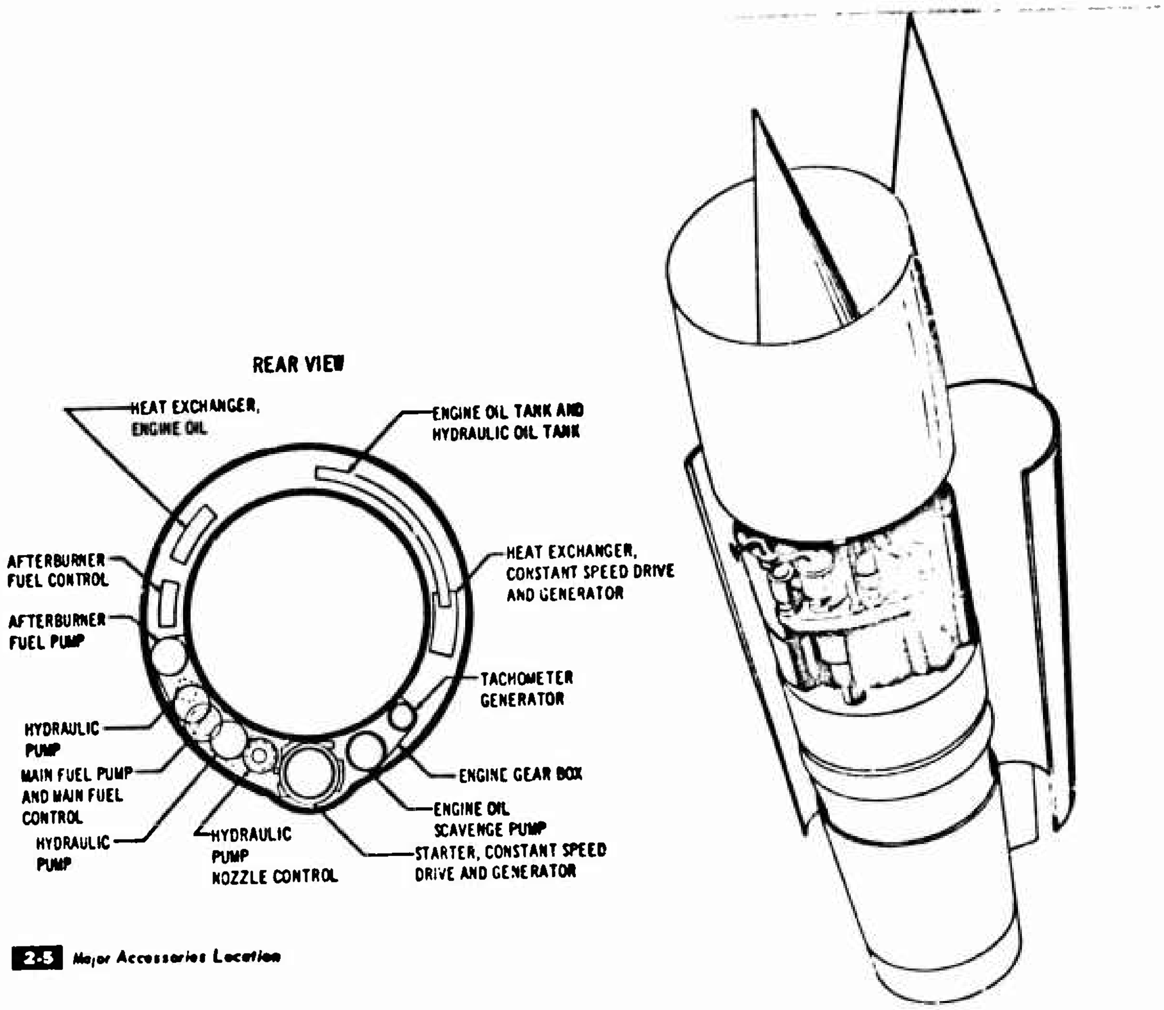
Complete and accurate indications of all important engine functions are provided at the flight deck by signals from electrical transmitters mounted on the engine.(The instrument panel arrangement is shown in Volume A-VII.) Fig. 2-7 lists for each function the type of indication, monitor location, and the type of pickup used.

With the exception of thrust indication, all of the indicators and transmitters are production items used on present day aircraft. A schematic diagram of the thrust indication system is shown in Fig. 2-8. The system proposed for the SST will react electrically to changes in the nozzle area as well as changes in the ratio of the total exhaust gas pressure to the inlet total pressure. This system is very similar to present engine pressure ratio systems. The only difference is that the nozzle area variable does not exist on present fixed area exhaust systems for subsonic jets.

Another method of thrust indication, which has been used for flight test purposes, is under consideration for the SST. This method will give a direct reading of pod net thrust by electrically reading the force exerted by the pod on the thrust link. The Boeing SST engine mount-



2-4 *Engine Oil System*



2-5 *Major Accessories Location*

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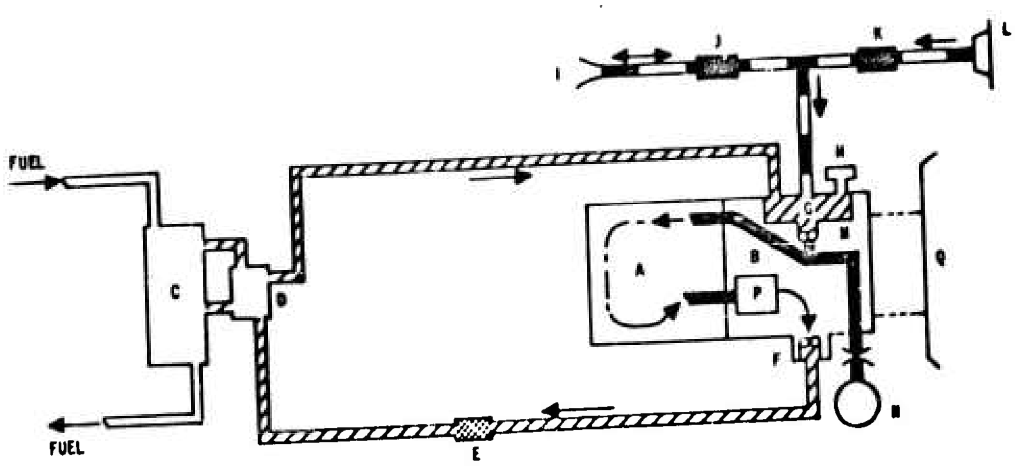


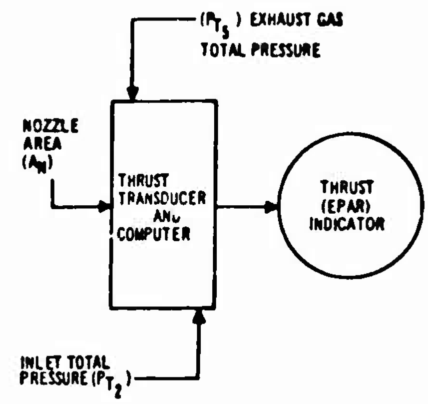
A – GENERATOR
B – CONSTANT SPEED DRIVE
C – FUEL - OIL HEAT EXCHANGER
D – TEMPERATURE AND REGULATING VALVE
E – OIL FILTER
F – OIL SUMP PUMP
G – INTEGRAL OIL TANK
H – FILL CAP
I – OVERBOARD VENT
J – PRESSURE RELIEF VALVE
K – AIR FILTER AND RESTRICTOR
L – PRESSURIZING AIR SOURCE
M – PRESSURE PUMP
N – OIL PRESSURE WARNING TRANSMITTER
P – C.S.D. BLOCK
Q – ENGINE GEAR BOX

PRESSURE OIL
SCAVENGE OIL
ENGINE FUEL
VENT AND PRESSURIZING

2-6 *Generator and C.S.D. Cooling System*

	FUNCTION	LOCATION		TYPE OF INDICATOR	TYPE OF PICKUP
		PILOT	FLT ENG.		
1	RPM	X		DIAL	TACHOMETER GENERATOR
2	EXHAUST GAS TEMPERATURE (EGT)	X		DIAL	THERMOCOUPLE
3	THRUST (EPAR)	X		DIAL	TRANSMITTER
4	TANK FUEL TEMPERATURE - MAINS.		X	DIAL	THERMOCOUPLE
5	TANK FUEL TEMPERATURE - AUXILIARY		X	DIAL	THERMOCOUPLE
6	FUEL FLOW	X		DIAL	TRANSMITTER
7	INLET FUEL TEMPERATURE		X	DIAL	RESISTANCE BULB
8	OIL PRESSURE		X	DIAL	TRANSMITTER
9	OIL BREATHER PRESSURE		X	DIAL	TRANSMITTER
10	OIL TEMPERATURE		X	DIAL	THERMOCOUPLE
11	OIL QUANTITY		X	DIAL	CAPACITANCE TYPE PROBE
12	LOW OIL PRESSURE		X	LIGHT	PRESSURE SWITCH
13	OIL FILTER CONDITION		X	LIGHT	PRESSURE SWITCH
14	VIBRATION		X	DIAL	MASS ACCELERATION
15	ANTI-ICING		X	LIGHT	VALVE POSITION SWITCH
16	INLET TOTAL PRESSURE		X	DIAL	TRANSMITTER
17	INLET POSITION		X	DIAL	TRANSMITTER
18	AUTO. INLET CONTROL		X	LIGHT	POSITION SWITCH
19	NOZZLE POSITION		X	DIAL	TRANSMITTER
20	FIRE DETECTOR	X		BELL AND LIGHT	CONTINUOUS ELEMENT
21	THRUST REVERSER POSITION	X		LIGHT	POSITION SWITCH
22	CSD OIL INLET TEMPERATURE		X	DIAL	THERMOCOUPLE
23	FUEL QUANTITY EACH TANK		X	DIAL	CAPACITANCE TYPE PROBE
24	TOTAL FUEL REMAINING		X	DIAL	TRANSMITTER
25	FUEL PUMP LOW PRESSURE		X	LIGHT	PRESSURE SWITCH
26	FUEL VALVE IN-TRANSIT		X	LIGHT	POSITION SWITCH
27	STARTER MANIFOLD PRESSURE		X	DIAL	TRANSMITTER
28	FUEL CONSUMED FLOWMETER		X	DIAL	TRANSMITTER

2-7 *Engine Instrumentation*



2-8 *Diagram Engine Thrust Indication*

ing system is easily adaptable to this method of thrust measurement.

Careful detail design will eliminate the possibility of any thrust being transmitted by other links, fluid lines, ducts, or engine cowling. Elongation of the thrust link will be measured electrically to give the thrust load. Compensation for airplane attitude and "g" loading will be accomplished within the system. This approach to thrust measurement can be developed to give a true reading of actual engine thrust with an accuracy equivalent to existing engine pressure ratio systems.

2.5.2 ENGINE ANALYZER-MAINTENANCE ANALYSIS AND RECORDING

As a means of reducing maintenance costs and improving schedule adherence, a flight maintenance analysis and recording system is under consideration for the engines and engine accessories used on the SST. By monitoring and analyzing engine performance, this system assists in pin-pointing probable failures for preventive action. The potential benefits may be considerable but further evaluation of effectiveness and reliability is required.

The system provides information in two forms. First, the on-board display affords a quick look at the data being accumulated and indicates any significant out-of-tolerance conditions to the flight crew. Second, the data are recorded on magnetic tape for later detailed analysis at ground facilities by general purpose computers, such as those generally available at airline installations.

2.6 Build-Up

Engine build-up is the installation on the engine of plumbing, wiring, and components by the airplane manufacturer. Only plumbing and wiring will be discussed here; components are discussed in other parts of Section 2.

The plumbing is conventional, uncomplicated, and readily installed or removed by standard wrenches. Wherever possible, tubing bends are used to control thermal expansion, eliminating the need for hoses.

Fuel and oil lines on Boeing subsonic jet engines are in areas where engine case temperatures reach 750° F., well above the autoignition temperature of aircraft fluids. Yet, with more than 16 million hours of engine time, autoignition resulting from fluid leakage is not a problem. The SST environment forward of the firewall is not different from that of the subsonic jet. Since the leakage potential is no greater, the use of end fittings for tubing on the SST does not increase fire hazard. The gains associated with welded fittings in engine build-up do not appear to warrant the added cost of stocking spare welded tube assemblies where the use of raw stock tubing will suffice. Standard end fittings are therefore used in the SST engine build-up.

To facilitate engine changes, hydraulic tubing runs are terminated in self-sealing couplings (quick-disconnect fittings) at a common disconnect bracket. All tubing and end fittings are made of corrosion resistant steel. Tubing runs are supported by clamps made of corrosion resistant steel containing a cushion of asbestos-impregnated teflon reinforced with wire.

Tubing less than 1.0 inch in diameter uses flareless-end fittings with conventional "B" nuts. Tubing assemblies from 1.0 to 2.0 inches in diameter have flared-end fittings with "B" nuts.

Tubing greater than 2.0 inches in diameter, such as the main fuel line and the engine bleed ducts, terminates at disconnect points which are flexible to allow for misalignment and thermal growth. These tube sizes use bolted-flange end fittings.

High temperature wire and connectors suitable for the environment are used in the engine electrical installation. The wiring is routed in bundles from equipment on the engine to flame-resistant connectors at the strut-firewall disconnect points. Large bundles of wires which would otherwise be exposed to chafing or damage by maintenance personnel are protected by channel raceways. Conventional high temperature loop clamps are used to attach the wiring to the engine. In systems such as the oil system, where more than one instrument reading is taken, the wiring from each transmitter is run in different bundles to prevent complete loss of instrumentation of that system should a wire bundle be damaged. Ground buses are installed from the basic engine structure to the aircraft to maintain electrical continuity without depending on the engine support fittings.

2.7 Accessory Compartment Environment

The engine compartment, which contains both airframe accessories and engine accessories, is maintained at a relatively cool temperature and a minimum rate of ventilation. This provides an environment that ensures reliability and long life of the accessories. It also gives the maximum protection from fire and permits a simple, lightweight, and effective extinguishing system.

A suitable environment, with acceptable temperature limits for accessories, is provided by use of fuel cooling of components within the compartment and by insulating the compartment from the engine case.

The compartment is insulated by two features. First, the engine case proper is enclosed within an engine-mounted annular shell. Boundary layer bleed air from the first stage of the compressor slowly flows between the engine case and the annular shell. This air maintains the temperature inside the shell below 550° F. Second, the outer surface of this shell is insulated by a thermal blanket which maintains the compartment temperature at an acceptable level (Fig. 2-9).

Airflow in the accessory zone is held to the minimum amount required for venting to compensate for altitude changes. This is done for three reasons: there is no appreciable gain in zone cooling with airflow; fire extinguishing system effectiveness deteriorates as airflow increases; and, fire temperatures are limited when fires are oxygen-starved.

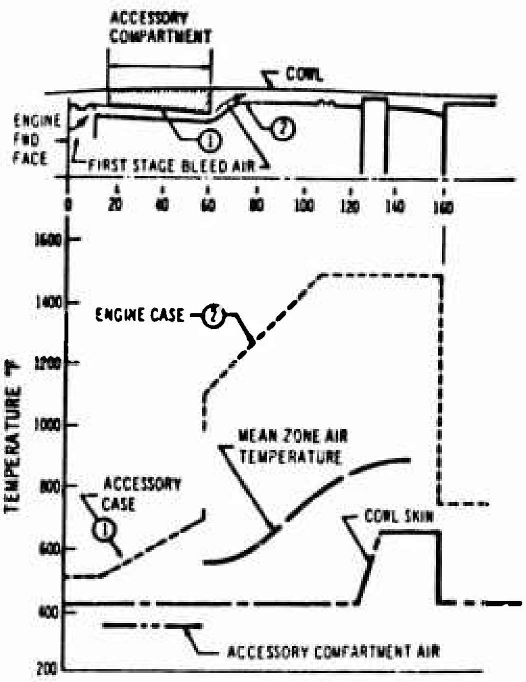
The essential features of the compartment are shown in Fig. 2-10.

2.8 Drain System

Drainage in the pod falls into three categories: cowl drainage, engine pad and equipment drainage, and large-volume fuel drainage. Danger from flammable fluids and contamination of the engine compartment is minimized by these drains. A list of the items drained is shown in Fig. 2-11. A diagram of the drain system is shown in Fig. 2-12.

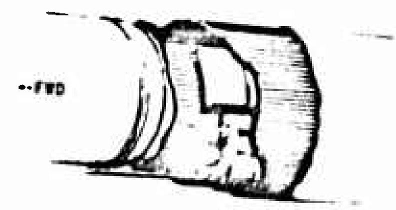
2.8.1 COWL DRAINAGE

Leakage of rain, fuel, oil, and hydraulic fluid within the pod cowling is drained overboard at the cowl low point. To provide a good flow path for fluid, each circumferential frame of the cowling has a large offset. The fluid flows be-



ENGINE COMPARTMENT TEMPERATURES CRUISE

2-9 *Engine Compartment Temperatures Cruise*



2-10 *Engine Accessory Compartment*

tween the cowl skin and the frame offset to the low point. The fluid drains overboard through a 0.5-inch diameter hole in the cowl at that point.

2.8.2 ENGINE PAD AND EQUIPMENT DRAINAGE

The engine drain system removes from the engine compartment the fluids resulting from leakage or overflow of the engine components. Individual drain tubes are run from each engine component and accessory requiring drainage. The discharge points of the drains are clustered, providing a convenient point to check accessories for excessive leakage. The drain tube outlets are located at the aft end of the cowling pressure-relief panel. The cowl exit is shaped to direct drainage overboard and prevent its re-entry. This same method of drainage is used on the Model 707 airplanes.

2.8.3 FUEL DRAINAGE

Fuel drainage from the main fuel manifold, the augmentor manifold, and combustors is collected in a drain can. This drainage would be objectionable if discharged directly on the ground. Pressurization of the can by ram air forces the fuel overboard through the drain exit in the engine cowling.

ITEM NO.	ITEM TO BE DRAINED	TYPE OF FLUID DISCHARGED	DRAIN TUBE SIZE AND MATERIAL	FREQUENCY OF DISCHARGE
1.	Fuel Pump Pad	Fuel	Size - 0.25 O.D. Mat'l - 321 CRES	Wear or Failure
2.	Fuel Control Pad	Oil	Size - 0.25 O.D. Mat'l - 321 CRES	Wear or Failure
3.	Hydraulic Pump Drive Pad	Oil-Fuel	Size - 0.25 O.D. Mat'l - 321 CRES	Wear or Failure
4.	Engine Oil Breather	Oil	Size - 1.25 O.D. Mat'l - 321 CRES	Each Engine Run-Up
5.	Fuel Pressurizing and Dump Valve	Fuel	Size - 0.75 O.D. Mat'l - 321 CRES	Each Engine Run-Up
6.	Engine Combustion Chamber Drain	Fuel	Size - 0.375 O.D. Mat'l - 321 CRES	Each Engine Run-Up
7.	Augmented Thrust Combustion Chamber Drain	Fuel	Size - 0.375 O.D. Mat'l - 321 CRES	Each Engine Run-Up
8.	Engine Oil Tank Scupper Drain	Oil	Size - 0.50 O.D. Mat'l - 321 CRES	Spillage From Oil Tank Filling
9.	Strut Drain	Oil-Fuel-Water	Size - 0.50 O.D. Mat'l - 321 CRES	Each Flight

2-11 *Engine Drain Chart*

2.9 Engine Cowling

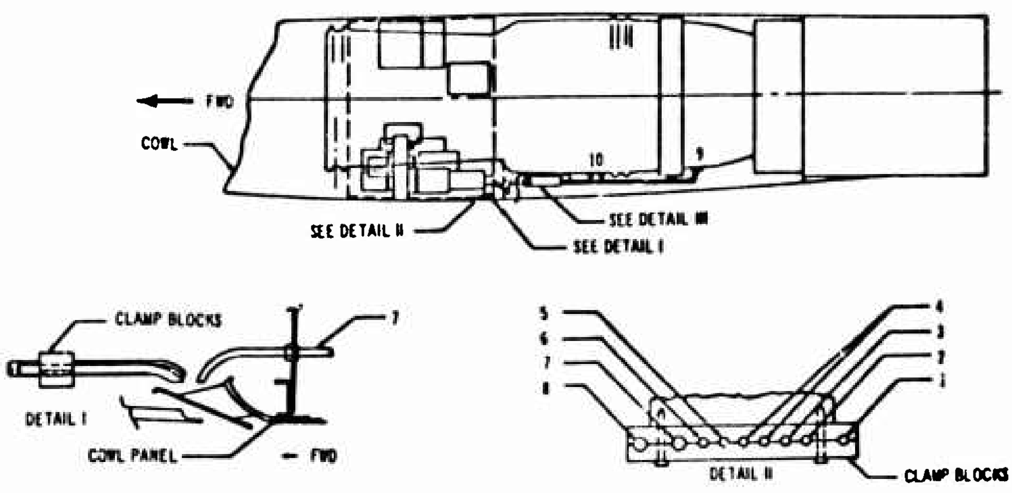
The engine cowling (Fig. 2-13) extends from the aft end of the outer surface of the engine inlet to the forward end of the engine exhaust nozzle. Opening the cowling provides access to the engine mounting, plumbing, wiring, and accessories. There are two titanium alloy (Ti-8Al-1Mo 1V) duplex-annealed panels for each engine. Each panel is readily removable by unlatching and rotating the panel to the removal position. Ordinary hand tools suffice for panel removal. Either panel may be removed without removing or adjusting the other. Either one of the panels can be rotated to the open position and secured there by tubular braces. This gives easy access to the accessory area without removing the panel completely.

Quick access for oil servicing is provided by a separate small access door in the cowl panel. Access for ground fire extinguishing is provided by push-in panels. Two large, permanently installed, aluminum alloy subpanels are provided to allow fire burn-through relief should an engine compartment fire occur. The very light gage of the non-structural panel combined with the low melting point of aluminum results in very early failure of the panel when it is subjected to a fire environment. A small hinged access door is also provided for adjusting the fuel control unit during an engine ground run.

2.10 Engine Compartment Fire Protection (RFP 2.22.1, 3.2.15.4)

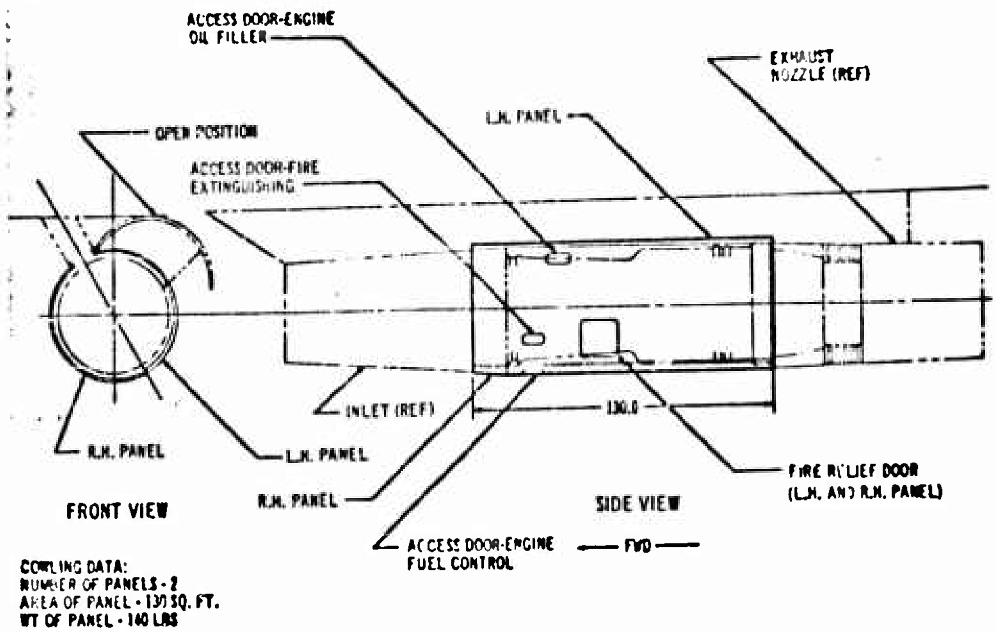
2.10.1 COMPARTMENT DESIGN

The engine compartment is designed to reduce to the minimum the probability of fire by: (1) separating of combustibles from ignition sources, (2) providing drains to prevent accumulation of combustible fluids and, (3) providing an appropriate pressure differential across zones to prevent the movement of combustible gases from one zone to another. Fire protection is provided by a continuous element fire detector system and a single nozzle, high-rate-



1 C.S.D. INPUT AND OUTPUT SHAFT SEAL
2 C.S.D. VENT
3 FUEL PUMP PAD
4 HYDRAULIC PUMP DRIVE PAD
5 FUEL CONTROL PAD
6 STRUT SUMP
7 FUEL COLLECTION CAN
8 ENGINE OIL TANK SCUPPER
9 AUGMENTOR COMBUSTION CHAMBER AND MANIFOLD
10 ENGINE COMBUSTION CHAMBER
11 MANIFOLD PRESSURIZING AND DUMP VALVE

2.12 *Engine Drain System*



2-13 *Engine Cowling*

discharge, fire extinguishing system. All components located in the fire zone are fire resistant in accordance with CAR 4b.

The engine is supported from the wing by a strut which provides a physical separation of an engine fire or explosion from the wing primary structure and fuel tanks. The forward vertical bulkhead and the lower spar of this strut are constructed of stainless steel to isolate the engine compartment from the strut. All compartments within the strut containing fluid-carrying lines are draft sealed, vented, and drained. Draft seals are provided between the strut and the wing structure.

Two large engine-fire burn-through panels provide pressure relief of the engine compartment and allow passage of fluid and flame out of this compartment should an engine fire occur.

All fluid carrying lines and electrical leads enter the engine compartment through steel fittings and are fireproof in accordance with CAR 4b. All components located in a designated fire zone are explosionproof. Compartment ventilation is essentially zero and limited to that necessary to provide adequate venting for altitude changes. A small, positive pressure differential is provided between the accessory section and the free air space to prevent flow of gas into the accessory section.

The components in the engine combustor section, such as afterburner fuel lines, fuel drain can, and thermocouple harness, are designed to withstand puddle fires.

2.10.2 FIRE DETECTION

A continuous element fire detector system is installed in each engine compartment. The element is engine mounted and attached with quick opening, hinged clips. The element run covers the bottom of the engine and other critical areas which require fire detection coverage. The detail routing of the element run provides maximum protection from damage by maintenance personnel or adjacent engine components. Chafing or breaking the element will not cause false fire warnings. Indication of either a fire within the engine compartment or an abnormal temperature condition is provided to the flight deck. An abnormal temperature condition such as the rupture or leakage of a high pressure, high temperature air duct, is indicated by a flashing red light. A fire condition is indicated by a continuous glow of the red light and an alarm bell.

2.10.3 FIRE EXTINGUISHING

The engine compartment fire extinguishing system is a single nozzle, high-rate-discharge system. Two supply bottles containing an extinguishing agent (triflorobromomethane) are installed on each side of the airplane with provisions for discharging the agent into the engine compartment. On each side of the airplane, either bottle or both bottles may be discharged into either accessory compartment. A fire extinguisher switch for each accessory compartment and two transfer switches, one for the two extinguisher bottles on the left hand side and one for the two extinguisher bottles on the right hand side, are provided on the flight deck.

2.10.4 FIRE SWITCH

A fire switch is provided for each engine. Actuation of the switch accomplishes the following:

- Closes fuel shutoff valve
- Closes hydraulic suction shutoff valves
- De-excites generator and disconnects generator from electrical system
- Cuts out hydraulic pressure warning lights
- Arms the engine fire extinguishing system
- Closes air bleed shutoff valves
- Closes engine anti-ice valves

All functions can be returned to normal in flight.

VOLUME A-VI

PROPULSION

3.0 ENGINE INLET SYSTEM (RFP 3.2.9.6; 2.25.3)

3.1 Inlet Considerations Summary

The supersonic inlet presents one of the major design problems confronting the supersonic transport, both from performance and safety standpoints. Boeing submits in this proposal an inlet concept that is new to supersonic aircraft. In its essentials, this design embodies the well known axisymmetric inlet design techniques. A variable diameter, non-translating centerbody is provided to control throat area independent of capture area. Safety of flight is guaranteed by the use of simple, pressure-actuated auxiliary doors. These doors prevent an unstable condition from enduring in the event of inlet unstart.

Test programs show that it is definitely possible to build an operating inlet and associated control system capable of adapting itself to its airplane environment. Such an inlet will provide the desired performance during takeoff, climb, acceleration, and subsonic cruise regimes. The inlet will also adapt itself to stall, engine-out interference, unstart, and gust effects. It is recognized that the inlet stability and shock system characteristics are such that airplane safety cannot be solely dependent on the automatic controls. Automatic controls are present to maintain the trim schedules essential to efficient and economic flight.

Should the automatic control system fail to maintain stable inlet operation, secondary doors will be opened by the air pressures acting directly upon them. These doors are redundant in quantity to ensure adequate air flow capacity and reliability. Although efficiency in this case is reduced, the inlet operates in a stable region. Airplane safety is in no way impaired.

3.1.1 INLET SELECTION

The supersonic inlet chosen for the proposal airplane is an axisymmetric external-internal compression inlet.

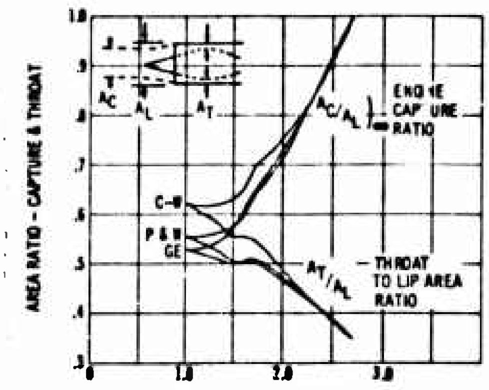
The proposed engines for the SST have high transonic airflow requirements relative to cruise requirements because of the need for high transonic thrust on the airplane. The capture area requirements at supersonic cruise size the cowl lip. The high ratio of transonic to cruise airflow requires a large variation in inlet throat area for efficient operation. Fig. 3-1 shows the airflow demand of three offered engines as a ratio of capture area required at each Mach number to the capture area at free stream M 2.7 cruise. The corresponding throat areas are also shown. The required area ratios are readily obtainable with the chosen inlet which has a variable diameter centerbody. Fig. 3-2 shows how the airflow demand of the three engines, shown in Fig. 3-1, compares with the capture area (air supply) of the inlet. A variable diameter centerbody design was chosen as the configuration best fitted to the airflow schedules of the proposed engines.

The choice of the axisymmetric inlet over the two dimensional type was based on the following inherent advantages:

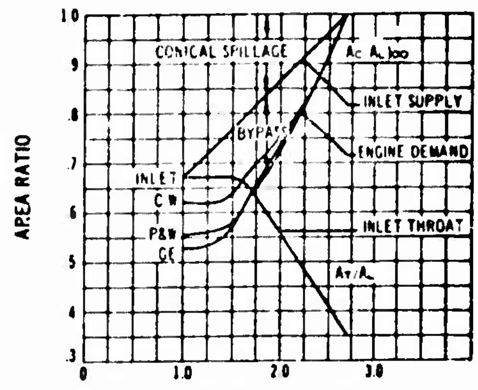
- Higher pressure recovery with lower boundary layer bleed
- Lighter weight
- Lower circumferential distortion at engine face
- Combatible with the podded engine concept
- Separable component, more easily maintained as a unit.

In addition to the ease of airflow matching to any engine airflow demand, other advantages of having the variable diameter centerbody in the inlet are:

- The physical location of the throat remains in a narrow region fore and aft. This greatly simplifies throat Mach number and normal shock position sensing.
- The fixed fore and aft position of the centerbody permits fixed lengths of inlet Mach sensing lines and centerbody bleed ducting.
- The sliding leaf feature of the centerbody design



3-1 *Engine Airflow Demand*



3-2 *Engine Match – Variable Diameter Center Body Inlet*

provide an automatic reduction in bleed at lower Mach numbers consistent with the demands of the inlet for optimum recovery.

3.2 System Description

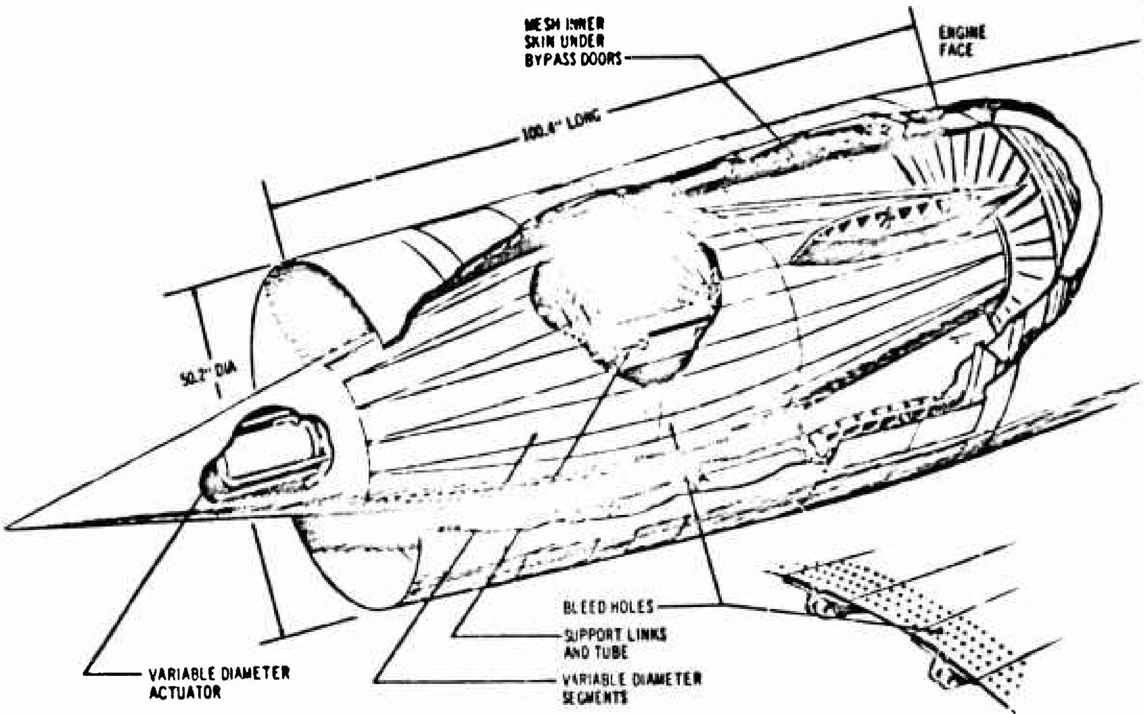
The inlet incorporates a variable diameter centerbody for controlling the throat area and variable bypass doors for matching inlet to engine airflow. The inlet is shown in Figs. 3-3 and 3-4.

3.2.1 INLET COWL ASSEMBLY

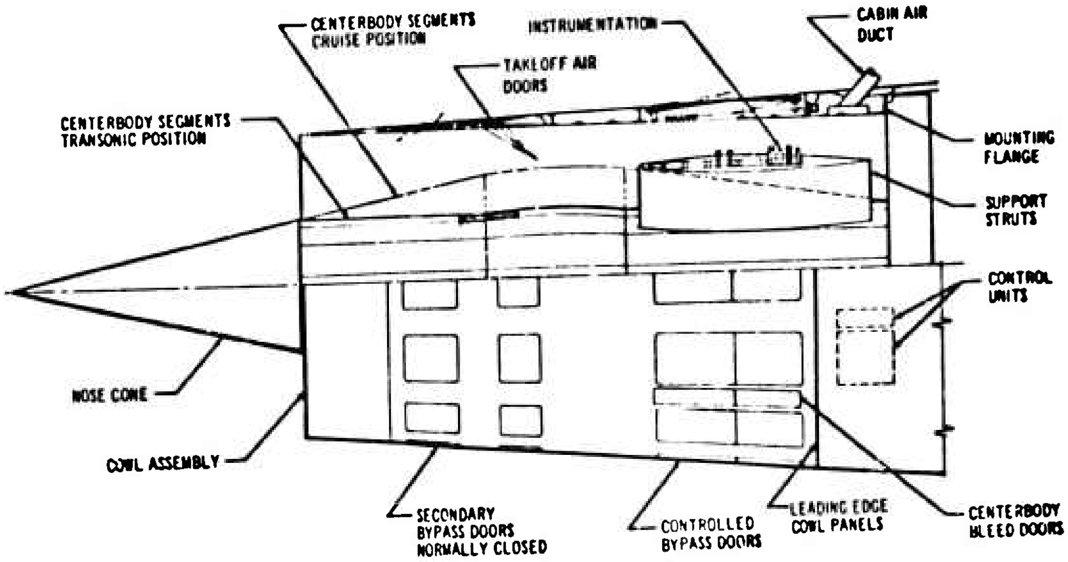
The inlet cowl assembly is supported by the engine forward flange. All loads pass from the inlet into the engine and through the engine mounting system to the strut and wing. The aft bay of the inlet assembly contains the actuators for the controlled bypass doors and other elements of the inlet control system. Loads on the actuators are reduced and equalized by mechanical interconnects between the doors. The controlled bypass doors are located immediately forward of the actuator bay. Also in this area are four doors located 90° apart which provide overboard passage for the centerbody boundary layer bleed. These doors are mechanically connected to the variable centerbody for closure in relation to airplane speed.

In addition to the controlled bypass doors discussed above, the inlet assembly contains a set of suck-in takeoff doors and secondary bypass doors (Fig. 3-4).

These doors are closed in normal cruise operation. The leading and trailing edges of the closed doors form tailored slots to pass the cowl inner surface boundary layer bleed efficiently overboard.



3-3 *Inlet Cutaway*



3-4 ***Inlet Assembly (Profile)***

The takeoff doors are similar to those on Boeing 707 fanjets. They are spring-loaded to the closed position and open inward in response to low pressure within the inlet during static and low speed operation.

The secondary bypass doors are spring-loaded to the closed position for all normal operation of the inlet. These doors open outward to relieve the pressure in the forward part of the inlet caused by any shock system expulsion at Mach numbers above 2.0.

The inlet cowl assembly is constructed of titanium alloy Ti-8Al-1Mo-1V duplex annealed material. The assembly can be removed as a unit by disconnecting the mounting bolts at the engine face flange, the cabin air supply line, the plumbing to the pump on the engine gear box, and the instrumentation lines.

Air for the cabin supply and the cabin heat exchanger is collected in a short manifold at the top of the inlet just forward of the mount flange.

3.2.2 CENTERBODY ASSEMBLY

The centerbody is a high strength steel and titanium assembly supported from four streamlined struts that connect to the inner surface of the inlet cowl assembly. These struts position the centerbody, pass all structural loads to the cowl, provide passage for instrumentation and control lines, and also passage for the centerbody boundary layer bleed air to the four exit doors referred to previously.

The centerbody assembly consists of:

- A nose cone, in which is housed the variable diameter control unit and the Mach sensor probes. (The cover for this area is easily removable for servicing.)
- Fourteen variable diameter segments with 14 radially spaced leaves.
- Two sets of links and collars which connect the actuator and the variable diameter segments.
- The main support tube and the necessary frames and stiffeners.

The actuator is powered by engine fuel at 1500 psi through a separate pump located on the engine. This pump also powers the controlled bypass doors. The variable area centerbody has the capability of increasing the inlet throat area 91 percent from the cruise position.

3.3 Operation

The operation of the supersonic inlet is independent of the engine controls. No controls are required on the flight deck to govern the inlet's variable area functions. The automatic power control for the variable diameter centerbody and the controlled bypass air doors is governed by self-contained pressure sensing units. A complete and detailed description of the inlet operation is given in Section 5.

3-5 *Design Inlet Recovery*

3.4 Performance (RFP 2.25.1f; 2.25.3)

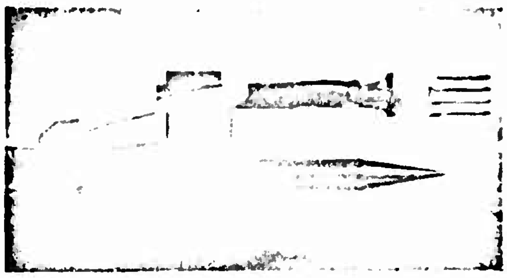
Fig. 3-5 shows the installed total pressure recovery of the full-scale inlet as a function of flight Mach number. This pressure recovery includes the effects of the local flow fields. The total pressure distortion at the engine face during all normal flight conditions will be within the engine manufacturer's requirements for continuous operation as defined in the engine specification. The change in flow incidence angles is small because of the sheltering effect of the wing at the underwing inlet locations. (See Figure 11, Par. 3.4.1.1, Flow Field Determination.) Over the Mach number range where the inlet is operating with partial internal compression ($M \approx$ 1.8 to 2.7) the change in the local flow angle of incidence at the inlet lip will be about one-half degree for the inboard inlet and one and one-half degrees for the outboard inlet. Even during engine-out conditions, for example, the airplane yaw angles will momentarily be less than 2 degrees which produces total pressure distortions of less than 14 percent. Because the distortion is predominantly radial, rather than circumferential, and because it is taken from small scale model data, it is expected that the actual full-scale distortion levels will be even lower. During a sudden 2.5 "g" pullup maneuver (which will occur very rarely), the change in the outboard inlet angle of incidence to the local flow will be about four degrees. The inboard inlet angle of incidence change will be less than two degrees. This is considered to be the most critical inlet distortion condition. At this condition the outboard inlet pressure recovery will drop off about ten percent, and the distortion level will be about 20 percent. This is tolerable for the short time such a maneuver will last.

The relative locations of the inlets are such that the flow field of one inlet does not affect the operation of the other. Even in the event of an inlet unstart the expelled shock will not disturb the adjacent inlet. This has been verified by wind tunnel tests with two operating inlets. (See Par. 3.4.1.2.)

In the case of an inlet unstart the pressure actuated secondary bypass doors will open and the automatic control will open the controlled bypass doors (see Section 5.0), to stabilize the expelled shock system and prevent inlet buzz. The automatic control system will immediately restart the inlet. The effect on the airplane of the pressure fields created by an expelled shock have been studied in wind tunnel tests. A discussion of these effects on airplane stability is contained in Volume A-V, Aerodynamics.

3.4.1 ENVIRONMENT

Supersonic inlets are sensitive to Mach number and flow direction (incidence) conditions at the inlet face and to transient changes in these conditions. Since the engine performance is dependent upon inlet total pressure recovery, it is necessary to evaluate the installed inlet recovery, in order to determine airplane performance. A major part of this evaluation is the determination of the flow field under the wing at the inlet lip and the inlet performance in this flow field. A further requirement is that the inlets be completely independent of one another so that flow conditions created by one inlet cannot disturb



Flow Field Rake

the neighboring inlets.

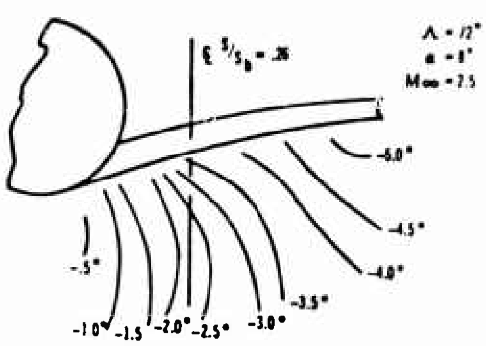
3.4.1.1 Flow Field Determination

In normal flight the outboard inlet will see no more than 1.5 degrees angle of incidence, based on experimental and analytical flow field data, while the inboard inlet will see almost no angle of incidence.

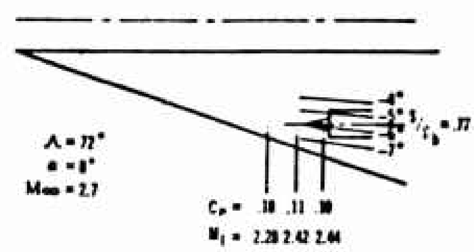
Flow field surveys in the vicinity of the inlets under the wing were made in the wind tunnels using a complete model of the airplane. A pressure rake was used to measure Mach number and flow direction beneath the wing. Figs. 3-6 and 3-7 show the rake and the rake mounted on the model. The flow direction relative to the airplane body centerline is measured in the form of downwash and sidewash components. Figs. 3-8 and 3-9 show lines of constant sidewash for the inboard and outboard inlet locations. Outward flow is indicated as negative sidewash on the curves. Fig. 3-9 also shows the pressure coefficient (C_p) measured on the wing and the local Mach number which corresponds to this C_p.

The flow field under the wing has been calculated using theory for a swept flat plate at the same sweep

3-7 Fig. 3-7 Airplane Flow Field Survey Model



3-8 Lines of Constant Sidewash – Inboard Inlet



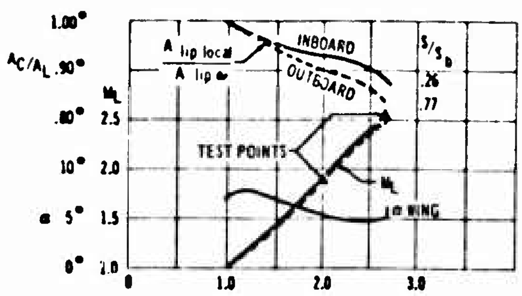
Lines of Constant Sidewash – Outboard Inlet

angle as the wing. Fig. 3-10 shows the theoretical Mach number and reduction in stream tube area for the two inlet locations (outboard and inboard), as a function of airplane Mach number for the airplane angle of attack curve shown. Data from the flow field test are also shown in Figs. 3-10 and 3-11.

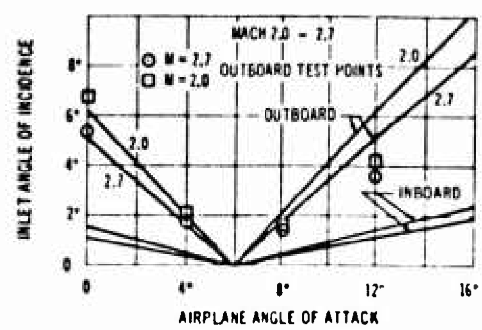
Fig. 3-11 shows the calculated and measured inlet angle of incidence versus airplane angle of attack for a Mach number of 2.0 to 2.7. From the airplane angle of attack curve in Fig. 3-10, together with Fig. 3-11, it is seen that for Mach numbers above 1.6 (inlet operating with partial internal compression) the inlets will see less than four degrees incidence for airplane angles of attack of less than eight degrees above the cruise angle. As explained in Par. 3.4, this corresponds to a 2.5 "g" pull up maneuver, which is the most critical condition imposed on the inlet. This maneuver should never occur in commercial operations. It is shown that the inlet can safely be operated under this condition.

3.4.1.2 Spacing of Adjacent Inlets

The relative spacing between adjacent inlets must be such that the inlets are completely independent of one another. In the case of an airplane with separate pods, the spacing is set by the requirement that an inlet unstart on one pod



3-10 *Flow Under Wing – Flat Plate Theory*



3-11 *Inlet Angle of Incidence*

not affect the inlet on the neighboring pod.

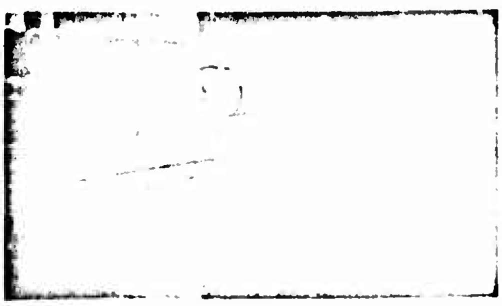
Boeing tests have been run with two operating, axisymmetric inlet models to establish the relative locations of adjacent inlets required to prevent the expelled shock of one inlet from affecting the other. The test model is shown in Fig. 3-12 as installed in the supersonic wind tunnel. Fig. 3-13 shows an experimentally derived zone representing a boundary of unsatisfactory locations for an adjacent inlet. An inlet will be undisturbed if located ahead of this zone. An inlet can be operated behind this zone and tolerate the unstarting of the adjacent front inlet only if it has a higher throat Mach number, corresponding to a recovery reduction of 8 to 10 percent. On the proposal airplane the inlets are positioned forward of the zone so that there will be no mutual interference or recovery reduction.

Fig. 3-14 shows a sequence from a high-speed movie of the shock systems of two inlets in a coplanar position. The expelled shock is clearly shown for one inlet, but the adjacent inlet is unaffected because it is located in the satisfactory zone. In this test the bypass door system of the unstarted top inlet was simulated in closed position. In Fig. 3-15 the top inlet was unstarted, but with the bypass doors simulated open. In both cases there was no interference between inlets, and in the second case the strength of the expelled shock was much reduced.

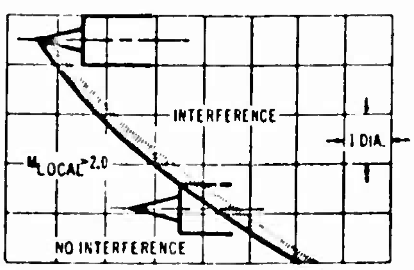
3.4.2 RECOVERY

For over five years The Boeing Company has carried out theoretical analyses and experimental testing of various inlet concepts for the SST.

Some of the models used in these test programs are shown in Fig. 3-16. Sizes range from 0.8 inch diameter cowl lip models for inlet aircraft stability, control, drag, and interference studies to 10 inch diameter cowl lip size models for boundary layer bleed, stability, and performance tests. Axisymmetric models, half-axisymmetric



3-12 *Inlet Interference Test Models*



3-13 *Region of Inlet Mutual Interference*

models, and two dimensional models of different sizes have been used.

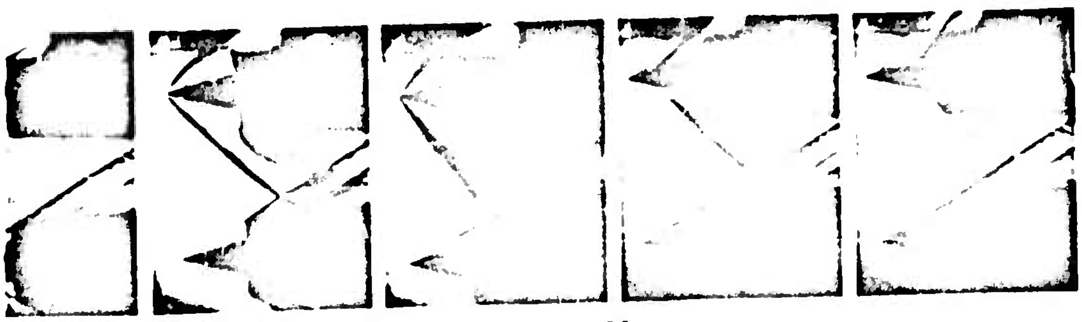
A summary of recovery levels attained with the axisymmetric, external-internal compression inlet models similar to Fig. 3-3 is shown in Fig. 3-17. Recent NASA data are shown for information. Also shown are initial test data for fixed versions of the proposed variable diameter centerbody inlet. The free stream (uniform flow without airplane effects) recovery curve used as the basis for installed inlet performance calculations for the proposal airplane is also presented. The airplane installed inlet performance effects are considered later, but Fig. 3-17 shows the basic performance level assumed for the inlet.

Fig. 3-18 shows the predicted full-scale installed design inlet recovery as a function of airplane flight Mach number. This recovery includes the effects of the airplane flow fields and the flight attitudes at various Mach numbers. This recovery was determined as follows: Initially the inlet was designed and developed by testing proper sized models in a uniform flow field. Measurements of the local flow conditions in the potential inlet locations on aerodynamic wind tunnel models provided angle of incidence and Mach number variations to be expected. These incidence and Mach number conditions were simulated in the wind tunnel by placing curved plates in front of the model. The resulting inlet performance effects were studied. For specific inlets these tests were followed by tests of inlet models in a uniform field and then in the pressure field of a model wing. Such tests have shown good correlation between predicted installed performance and performance in the wing flow field.

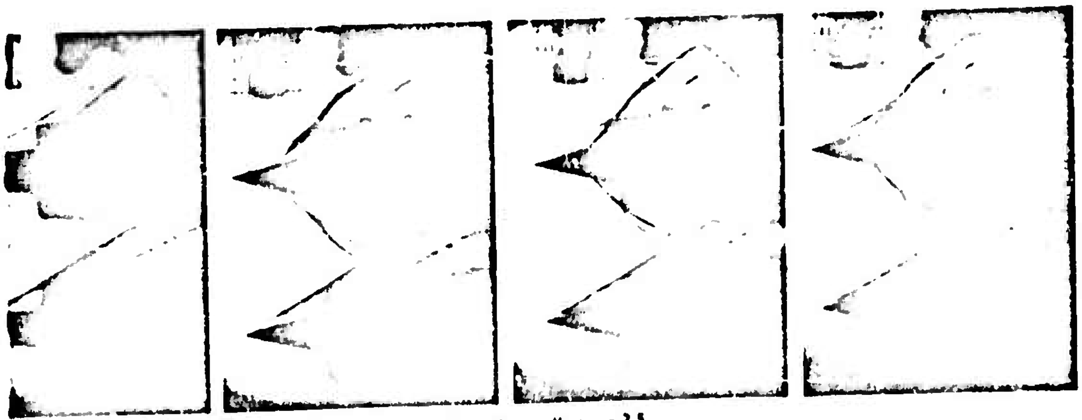
3.4.2.1 Inlet Design (RFP 2.25.16)

The specific inlet internal aerodynamic design for the GE4 J4C engine is described as follows. The scale model tests of this inlet are still in progress, but the axisymmetric inlet models which have been tested are very similar to the proposed inlet.

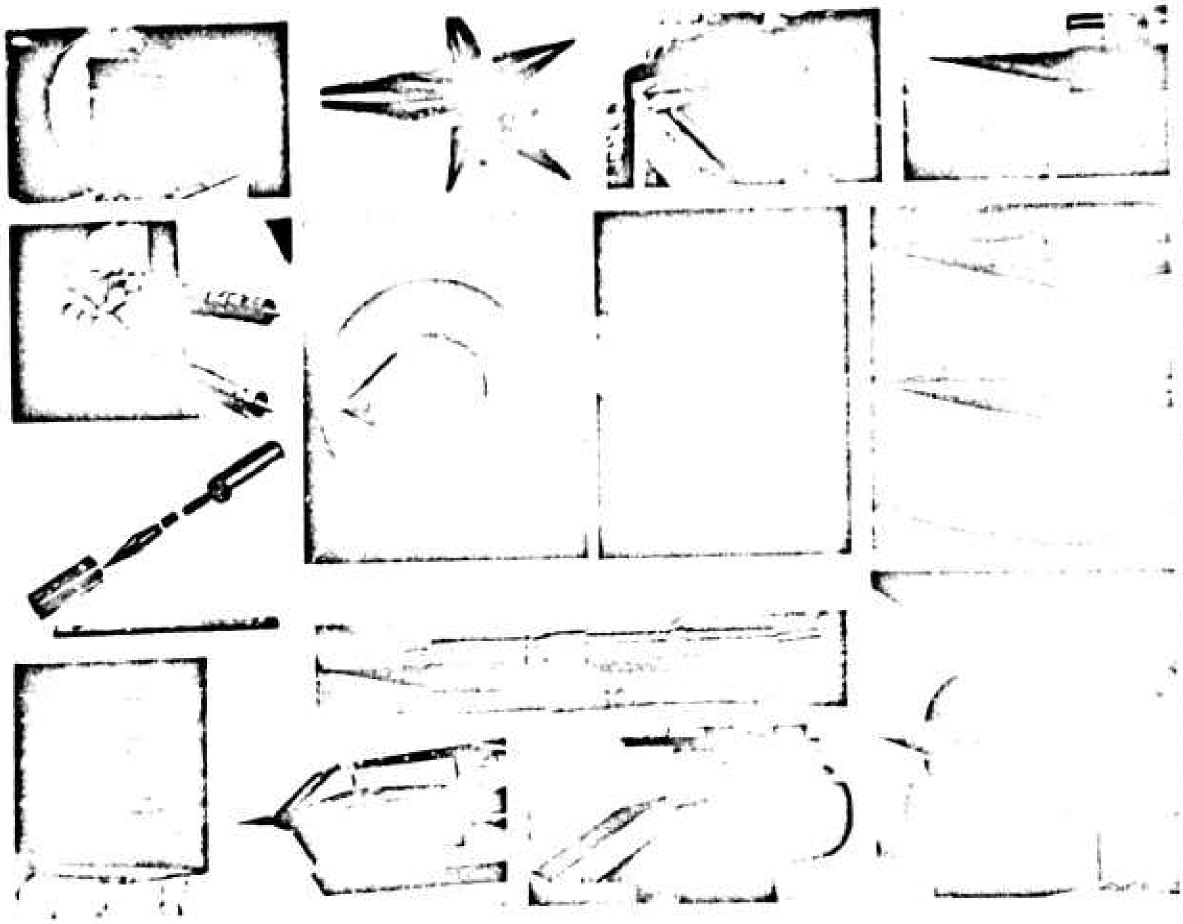
The inlet capture area ratio curve is shown in Fig. 3-19 together with the GE4 J4C engine capture ratio schedule for standard and non standard days at maximum dry power and above. The inlet internal contours are shown in Fig. 3-20 for Mach 2.7, 2.4, 2.2, and at sub-



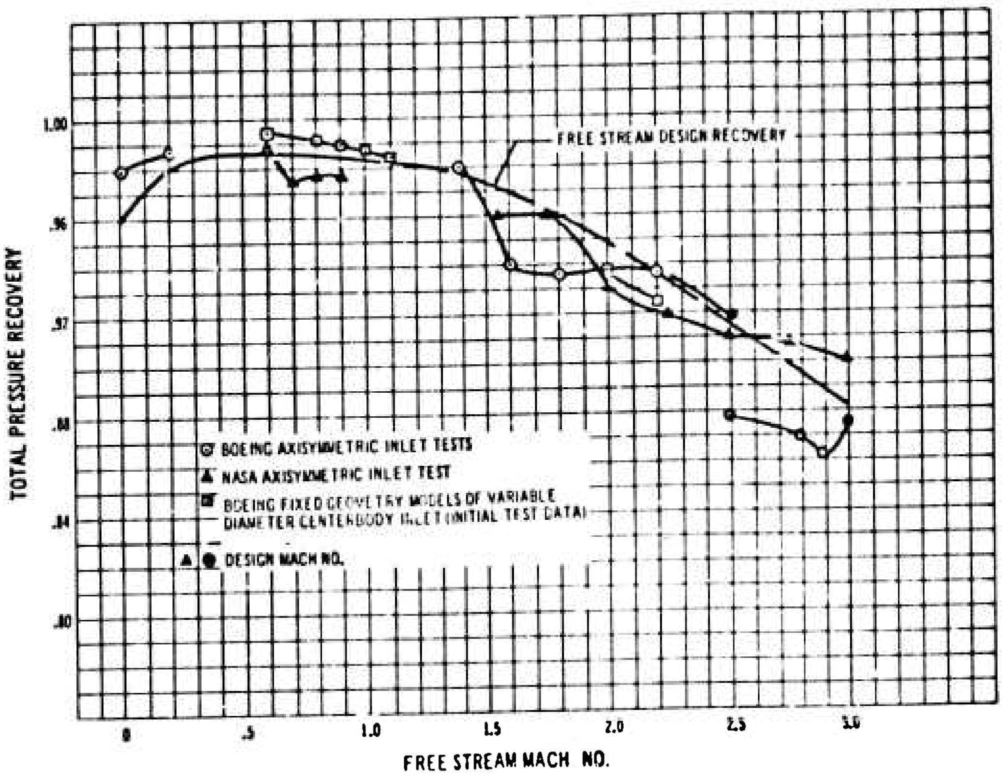
3-14 *Adjacent Inlet Tests – Unstart with Bypass Door Closed* $M_{inlet} = 2.5$



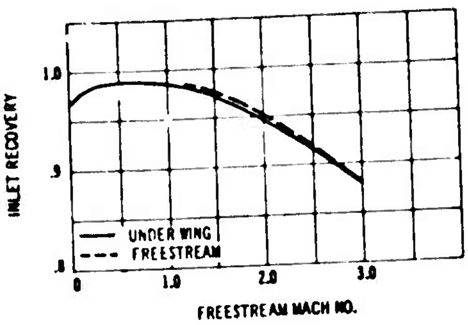
3-15 *Adjacent Inlet Tests – Unstart with Bypass Doors Open* $M_{inlet} = 2.5$



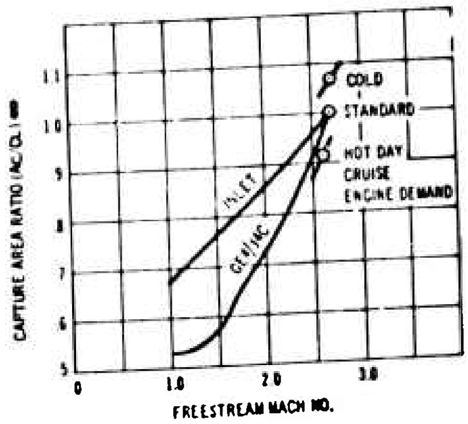
3-16 Various Inlet Test Models



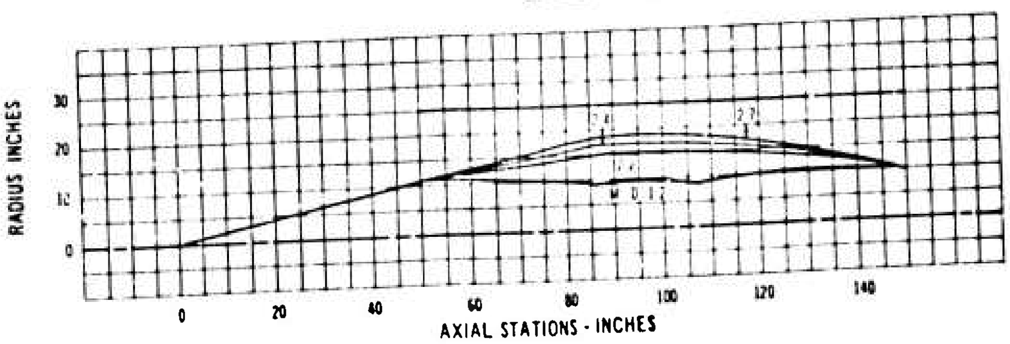
3-17 *Free Stream Inlet Recovery*



3-18 *Design Inlet Recovery*



3-19 *Engine – Inlet Match*

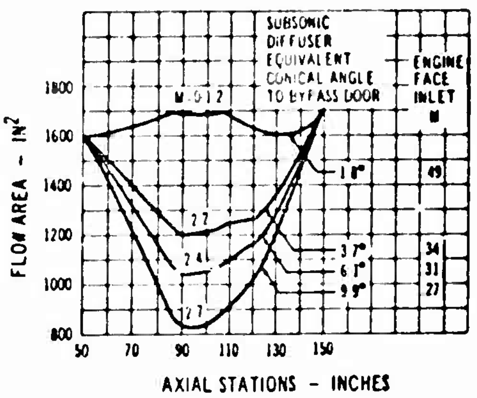


3-20 *Inlet Internal Contours*

ıic and transonic speeds. The corresponding internal w areas are shown in Fig. 3-21, with the equivalent nical included subsonic diffusion angle and Mach numrs at the engine face. The diffusion angles are within lues considered acceptable for inlet design. The internal ntours were designed by computing the flow characistics inside the inlet to obtain acceptable shock patns, wall static pressure gradients, and throat velocity ofiles throughout the Mach number range. Examples of is numerical flow field work are included in Fig. 3-22, owing the proposed inlet internal flow at the design airane Mach number of 2.7 and at one off-design condition. he average throat Mach number is 1.3 when the inlet is erated in the mixed compression mode (normal shock vallowed) above airplane Mach 1.8. This throat Mach umber provides high pressure recovery with adequate stality margin to handle upstream airflow disturbances such s gusts or airplane maneuvers which cause inlet Mach umber or flow direction changes. These numerical soluons of the flow equations, using the method of charactertics and shock wave theory, have been confirmed by wind unnel test.

At the proposal airplane design Mach number of 2.7, stalled inlets will operate on the airplane in an average cal Mach number field of 2.5. The Boeing Mach 2.5 ilet model has contours similar to the inlet chosen for he proposal airplane. This model has been tested extenively and will be referred to as the basic inlet model. The model has the same centerbody angle, same length two cowl lip diameters), generally the same internal conours, and the same design Mach number as the proposed nlet.

The variable diameter centerbody inlet has good contours for the Mach 2.5 design condition and for the off-design supersonic cruise and acceleration range. In the ransonic and subsonic range the variable diameter cenerbody inlet always has adequate area at the lip station o provide the proper lip velocity ratio. Excess air is taken on board at these speeds and bypassed overboard at low



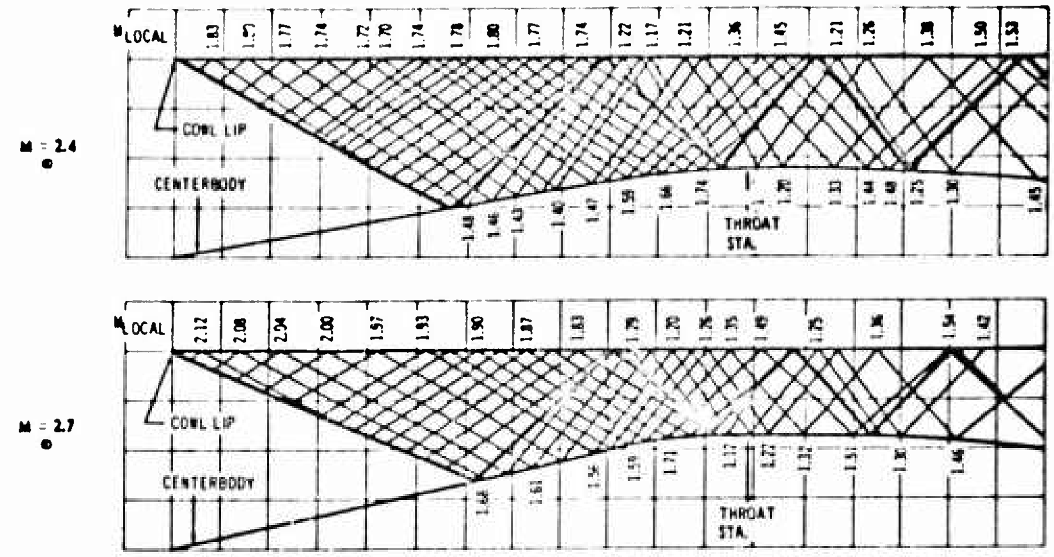
3-21 *Inlet Internal Area*

angles to the external stream. The variable diameter centerbody inlet is matched to give the optimum trade between pressure recovery and bypass drag.

The situation of engine shut down with windmilling brake applied is the design condition for sizing total bypass door area (secondary and controlled bypass). For this case essentially 90 percent of inlet air must pass through the bypass doors.

3.4.2.2 Mach 2.5 Free Stream Recovery Data

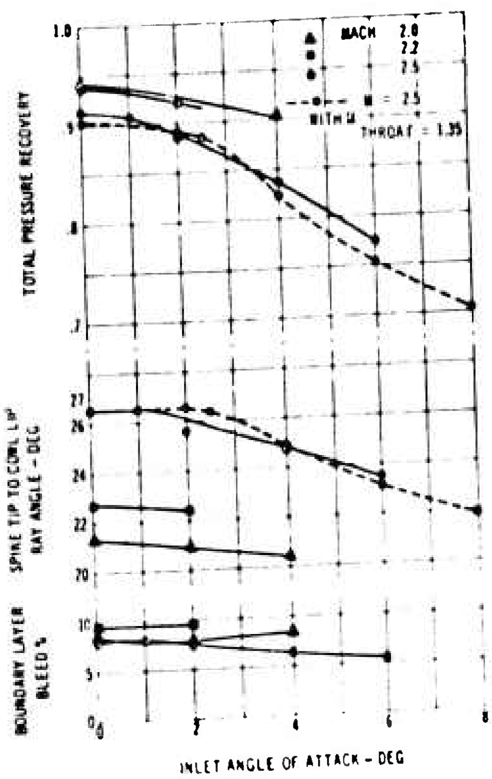
The free stream performance of the basic inlet test model is shown in Fig. 3-23 as a function of Mach number and angle of incidence. Fig. 3-24 shows pressure recovery versus Mach number. Recovery is 91.2 percent at Mach 2.5. To increase the stability margin against upstream flow dis-



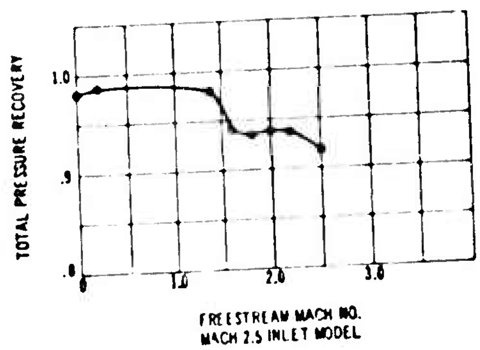
3-22 *Internal Flow Fields*

turbances the inlet operates with a throat Mach number of 1.25. In these tests the inlet operated with a swallowed shock up to an angle of incidence of 8 degrees. This corresponds to an airplane angle of attack of 20 degrees (Fig. 3-11), well beyond supersonic speed flight limits. These tests also show that the inlet capture mass flow ratio was constant up to 1.5 degree angle of incidence, as seen by the constant centerbody ray angle. This indicates that no throat area changes were required. As a result the inlet control need not react over this range.

A larger throat area model was also tested. The performance curve shown with broken lines in Fig. 3-23 is for this larger throat area, corresponding to a 1.35 design throat Mach number. With this contraction ratio the inlet is capable of accepting well over two degrees change in flow incidence before inlet control action is required,



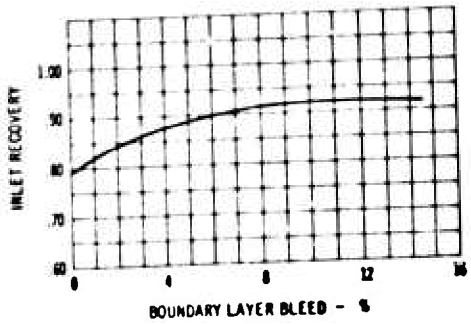
3-23 *Angle of Attack Performance*



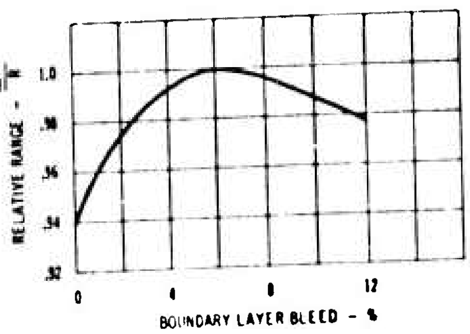
3-24 *Recovery vs. Mach No.*

but this is accompanied by a reduction in recovery.

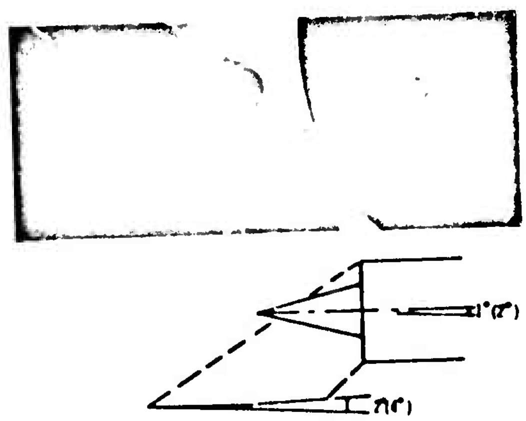
Boundary layer bleed requirements are shown at the bottom of Fig. 3-25. The amount of bleed is seven percent for this model size (3.4 inch diameter cowl lip). Model tests of a nearly identical Mach 3.0 inlet at 10 inch diameter cowl lip size, where representative design of the bleed removal systems could be accomplished, showed a decrease in bleed requirement to 5.75 percent. It is expected that the 5 percent bleed assumed for the airplane performance calculation will be attained with full scale inlets and tailored bleed configurations. Fig. 3-25 shows pressure recovery versus amount of bleed for the 3.4 inch diameter model. Fig. 3-26 shows the calculated airplane range versus percent bleed for the full scale inlet. These calculations assume that the proposal airplane inlet has the same critical inlet recovery as the model tests but with 2 percent less bleed. The trade between pressure recovery, engine performance, and bleed drag has been taken



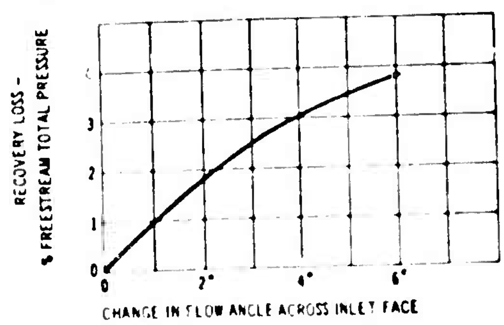
3-25 Bleed Effect on Recovery



3-26 Bleed Effect on Range



3-27 Simulated Inlet Distortion Tests



3-28 Recovery vs Inlet Face Flow Angle

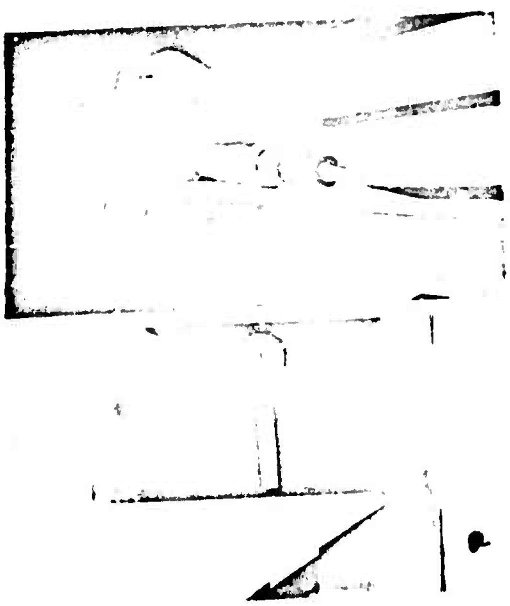
to account in Fig. 3-28.

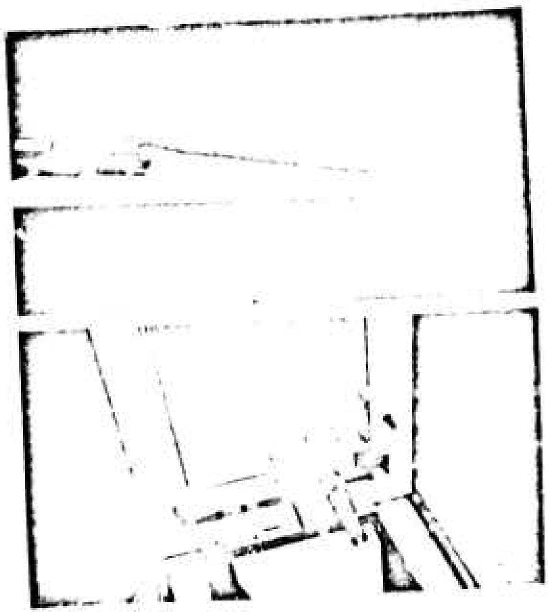
3.4.2.3 Flow Field Recovery

ents were conducted to determine the performance of an let in a non-uniform flow field. The field under the wing as simulated in front of the inlet by placing curved ates upstream of the inlet. Fig. 3-27 shows the inlet

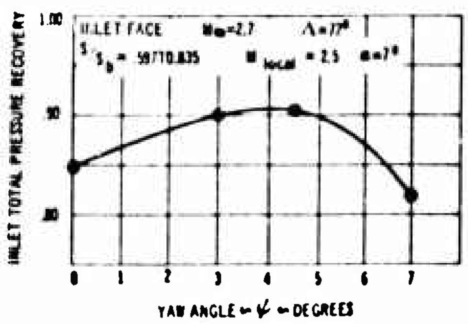
and the plate. The performance, in percent of free stream total pressure as a function of change in flow angle across the inlet face, is shown in Fig. 3-28. Using the data from the free stream inlet and flow field tests, the performance of the proposal inlet located under the wing was calculated (Fig. 3-18).

The design recovery curve for inlets located under





3-29 3-30 3-31 3-32 Inlet-Wing Effects Models



3-33 *Outboard Inlet Recovery vs Yaw*

3-34 *Vari...le Diameter Centerbody Inlet, Transonic Test Model*

the wing is compared with the free stream pressure recovery curve in Fig. 3-18. As shown by these curves the loss due to non-uniform flow is approximately equal to the gain due to the lower Mach number in the wing pressure field.

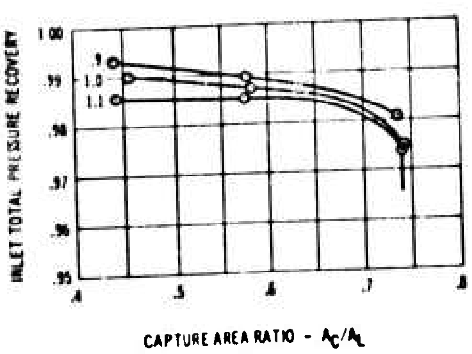
3.4.2.4 Inlet Under the Wing

As further support of the predicted performance of the proposal inlet, tests were conducted in the Boeing supersonic wind tunnels. The test configuration was the basic axisymmetric 12.5 degree centerbody inlet with a wing closely simulating the part of the 74 degree swept wing forward of the inlet. The inlet and wing are shown installed in the wind tunnels in Figs. 3-29 through 3-32.

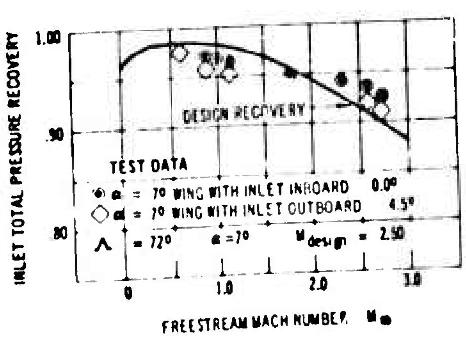
The test was conducted at 7 and 8 degree angles of attack at tunnel Mach numbers from 0.60 to 2... The inlet was first located on the wing centerline and then near the outboard edge of the wing, simulating the two inlet locations. The axis of the inlet was also turned inward relative to the wing axis of symmetry to establish the optimum inlet angle for the outwash of the local air at cruise. Fig. 3-33 shows inlet pressure recovery versus inlet angle. The optimum angle of the inlet is 4.5 degrees inward to the body centerline.

A transonic test using a [illegible] scale model [illegible] the variable diameter spike inlet [illegible] A photograph of the model [illegible] Fig. 3-34. Preliminary data from these tests are shown in Fig. 3-35.

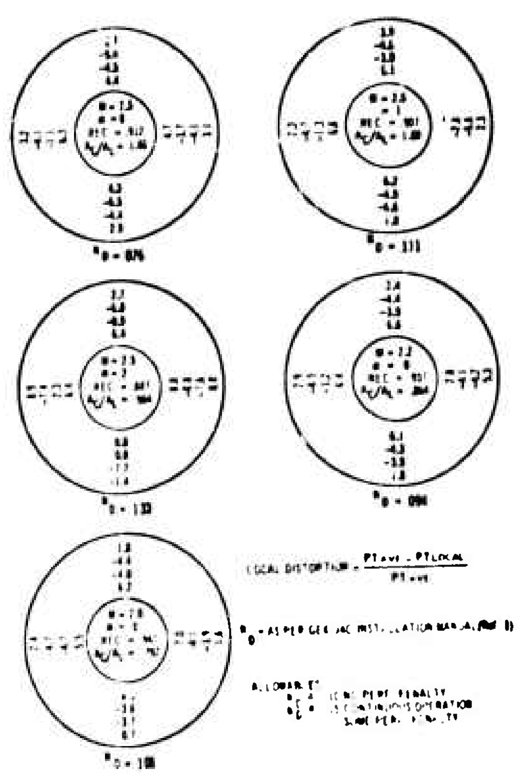
The results of both the transonic and supersonic tests of the inlet installed under the wing are plotted in Fig. 3-36 along with the design inlet pressure recovery. Past test data have shown that testing with larger scale inlets



3-35 *Subsonic Performance of Variable Diameter Centerbody Inlet*



3-36 *Inlet Under Wing Recovery*



3-37 *Local Distortion and Circumferential Distortion Indexes* will give inlet performance increases of 1 to 2 percent. With this increment applied to the test data shown, the design performance should be exceeded by a sufficient margin to permit operating with the 2 percent stability margin required for control dynamics.

3.4.3 DISTORTION

The total pressure distortion at the compressor face is a function of the angle of incidence of the inlet to the local flow, the amount of flow distortion in the air at the inlet lip, and the internal geometry of the inlet flow passage. The inlets are installed under the wing, reducing local angle of incidence changes with airplane angle of attack and are oriented with respect to the local flow to be essentially at zero angle of incidence. The flow direction change across the outboard inlet face is very small, about 1.5 degrees, and across the inboard inlet about 0.5 degree.

Fig. 3-37 shows typical compressor face total pressure distortion test results for the basic inlet model operating at various Mach numbers and angles of incidence (α). Recovery and capture area ratio are noted, together with Mach number and angle of incidence for each diagram shown. For each situation the General Electric distortion index (N_D) is shown as computed per directions in the GE4 J4C installation manual (Ref.3). At the supersonic Mach numbers listed for zero angle of incidence, the distortion index is below 0.10 as required for continuous engine operation at no performance penalty. The distortion is predominantly radial which is characteristic of a spike-type axisymmetric inlet. Axial flow compressors are usually less affected by radial than circumferential distortion.

Airplane maneuvers or attitude changes during supersonic cruise will be on the order of $\pm$ 2 degrees to cover all normal operations. This will result in inlet distortions well within the 0.15 value allowed for continuous cruise with small performance reductions. The engine-out yaw conditions will be less than [illegible] degrees. With full-scale inlets it is expected that distortion levels will be even lower, due to reduction in scale effects and because at the large scale more can be done to control the boundary layer. The sidewash angles (seen as inlet flow angularity) increase at transonic and subsonic Mach numbers but the distortion levels at transonic and subsonic speeds have been found to be relatively unaffected by inlet flow angles. The inlet distortion will be acceptable.

3.4.4 EXCESS AIR DRAG

The inlet drags associated with boundary layer bleed, excess air spillage and bypass air are discussed in Section 12.

VOLUME A-VI

PROPULSION

4.0 EXHAUST SYSTEM (RFP 3.2.9.5)

4.1 General Description

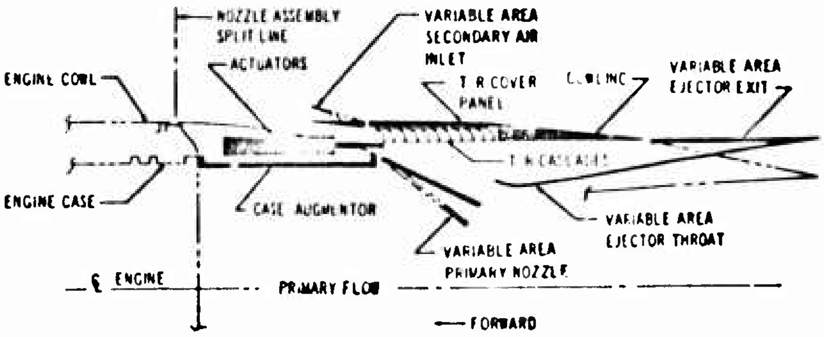
The exhaust system is a separate assembly that can be both installed or removed with the engine in place on the airplane.

The exhaust system assembly (Fig. 4-1) consists of the aft section of the engine augmenter case; the variable area convergent-divergent ejector nozzle; the integrated thrust reverser; the variable area secondary inlets for nozzle ventilation and cooling air; the actuators, controls, and associated plumbing; and the exterior cowling required to continue the aerodynamic contour of the pod from the aft end of the engine cowl panels to the nozzle exit.

The design of the exhaust system assembly requires a carefully coordinated program on the part of both The Boeing Company and the engine manufacturer. The Boeing Company will establish requirements to make the design compatible with the airframe configuration, such as external cowling profile for maximum aerodynamic performance, the operation and exhaust flow patterns of the thrust reverser, and the exhaust system controls.

To provide maximum propulsive performance through the broad speed range of the supersonic transport, the selected engine, the General Electric GE4 J4C, requires a variable area convergent-divergent ejector nozzle. Because the proper operation of the variable area feature of the nozzle is vital to engine performance guarantees, the complete exhaust system assembly will be designed and produced as an engine component by the engine manufacturer.

Some of the mechanisms that operate the variable area components of the GE4 J4C turbojet engine nozzle



4-1 *Schematic Diagram Exhaust System - Forward Thrust* are under patent disclosure restrictions. As a result, details of components are not included in this discussion. Ref. 2 may be consulted for detailed information.

4.2 Exhaust Nozzle

4.2.1 DESCRIPTION

The primary, convergent section of the nozzle consists of variable position flap and seal segments, a mounting ring, and actuating linkage. This section forms the jet nozzle throat. The secondary, divergent section consists of variable position, flap and seal segments supporting structure, and actuating linkage. This section forms the ejector walls and the external boattail surface. The ejector throat and exit areas are variable in order to guide and provide maximum control of the exhaust gas expansion. Aspiration and cooling air flows over the aft side of the primary nozzle segments and is taken into the ejector through an annular gap provided at the nozzle throat.

4.2.2 OPERATION

The exhaust nozzle and reverse control is shown in Fig. 4-2. A 3000 psi hydraulic system using the same type of fluid as the engine lubricating oil powers the control system. The system is self contained on each engine and has its own fluid reservoir, engine-driven pumps, and manifold system.

The primary nozzle area is governed by a closed loop positioning control. The control consists of a hydro-mechanical computer and servo valve, synchronized hydraulic actuators, and a mechanical position feedback system. The nozzle is positioned as a function of thrust lever setting and turbine temperature as shown in Fig. 4-3. At low thrust settings the area is established in accordance with the "floor" schedule. At high thrust settings the area varies between the mechanically established limits of the "floor" and "roof" schedules as a function of turbine temperature. In this region a turbine-temperature-signal amplifier introduces a bias to the control which varies the exit area to hold a constant turbine temperature. The "floor" and "roof" schedules maintain manual control of exit area in the event of a turbine-temperature-signal malfunction.

The ejector throat area, which is a function of the primary nozzle position, establishes the annular gap provided at the nozzle throat for efficient pumping of the aspiration and cooling air.

The ejector exit area and boattail angle are governed to provide the proper expansion ratio for nozzle efficiency. Studies are currently underway to position the segments by pressure balancing.

Nozzle position indication is provided on the flight engineer's panel.

4.2.3 PERFORMANCE

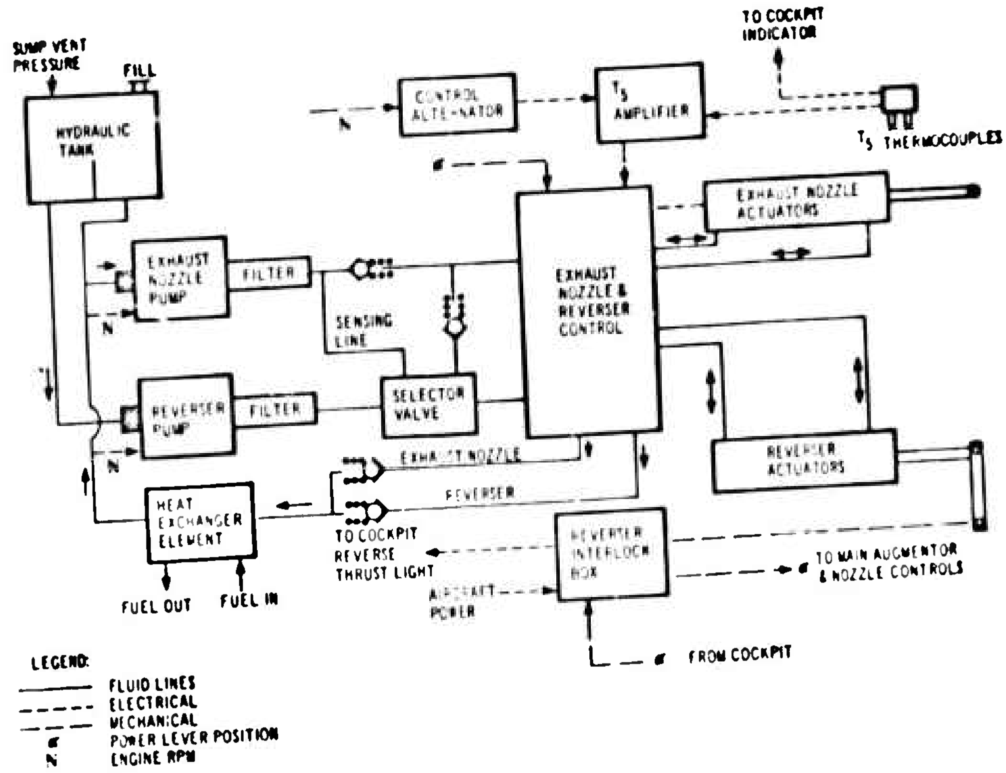
The function of the nozzle is to achieve maximum thrust minus nozzle drag from the engine exhaust gases. The performance is defined by a gross coefficient C_{T_g} where:

$$C_{T_g} = \frac{\text{Gross Thrust-Nozzle Drag (Including Ram Drag of Secondary Air)}}{\text{Ideal Gross Thrust of Primary Plus Secondary Air}}$$

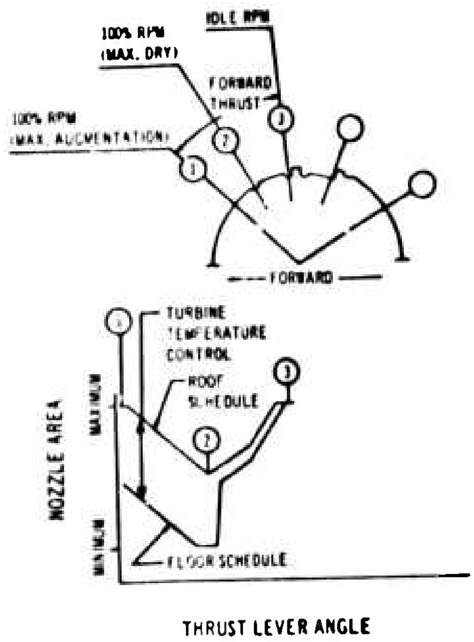
The nozzle geometry is scheduled to a position which gives the maximum gross thrust coefficient at any flight condition.

The nozzle on the GE4 J4C engine has divergent walls, which establish the nozzle expansion ratio, A_9/A_8, and the external boattail angle. These walls can be placed in one of four positions approaching the ideal expansion ratio for a given Mach number. Fig. 4-4 shows the expansion ratio-Mach number relationship provided by the nozzle. Shown in dotted form is the schedule indicated by theory.

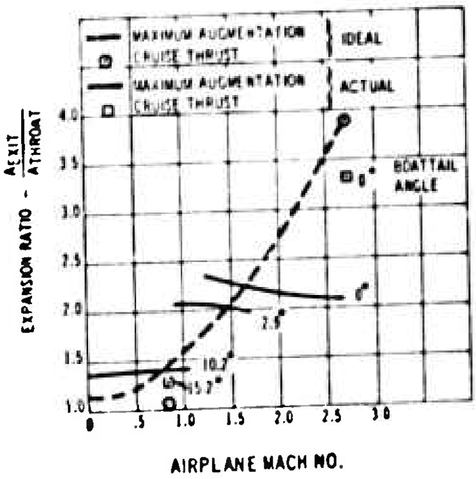
Experimental evaluation and analysis of this nozzle have been conducted by General Electric. The internal



4-2 *Nozzle Area and Reverser Control System*



4-3 *Nozzle Area Control Schedule*

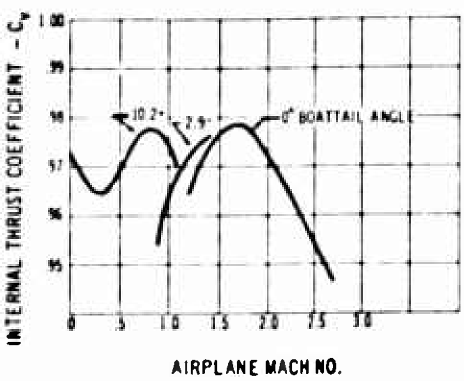


4-4 *Nozzle Expansion Ratio*

performance of the nozzle at each mechanical position is shown in Fig. 4-5. Fig. 4-6 shows the boattail drag effect expected at each nozzle position.

The composite performance estimate, C_{F_n} is shown in Fig. 4-7.

Analysis of the influence of the external flow field environment caused by the close proximity of the wing on nozzle gross thrust coefficient has been made. The slight decrease in nozzle pressure ratio caused by the increased pressure field under the wing aft section is approximately counter-balanced by the more favorable pressures acting



4-6 *Nozzle Internal Performance Max Augmentation*

on the nozzle boattail. Hence in all aircraft performance evaluations the noz- is assumed to be operating in free stream.

4.2.4 EXHAUST FLOW FIELD

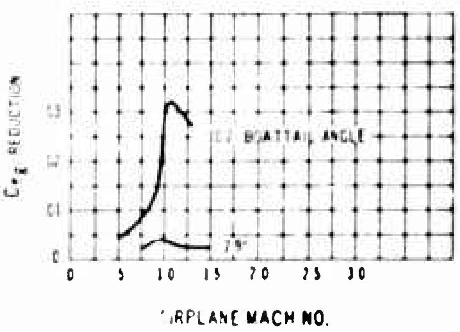
4.2.4.1 Surface Heating Influence of the Exhaust Stream (RFP 3.2.9.5)

The location of the propulsion pods under the wing and forward of the tail required an investigation of the heating effect of the jet stream on adjacent surfaces.

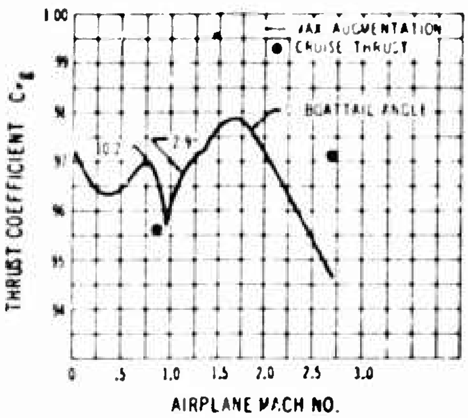
These conditions were analyzed:

- Ground checkout of the augmenter on a hot day at a total jet temperature of approximately 3000° F.
- Maximum dry operation during a hot day on the ground at a jet temperature of approximately 1600° F.
- Maximum augmentation at Mach 0.9, 25,000 feet, on a hot day at a jet temperature of 3000° F.

To investigate the ground conditions, a model test was set up with a jet exhausting alongside of a simulated body (Fig. 4-8). The jet was run at various temperatures up to the 2500° limit of the burner. Surface temperatures were measured at several distances along the simulated body. The body was placed at various distances from the jet centerline and at several angles referenced to the jet centerline. Test data for jet temperatures of 1000° F., 1500° F., and 2500° F. were taken as a function of jet diameters downstream and radial distances from the jet centerline. The data were then cross-plotted to arrive at the first two conditions above, as shown in Figs. 4-9 and 4-10.



4-7 *Boattail Drag Effects on Nozzle Thrust Coefficient*



4-7 *Nozzle Thrust Minus Drag Performance*

Jet wake temperatures, rather than surface temperatures, are presented for the third condition (Fig. 4-11). The above data were used to determine that surface temperatures of adjacent airplane structure do not exceed 200 F. (This determination is described in Section 6 of Volume A-IV, "Structures.")

4.3 Thrust Reverser (RFP 3.2.9)

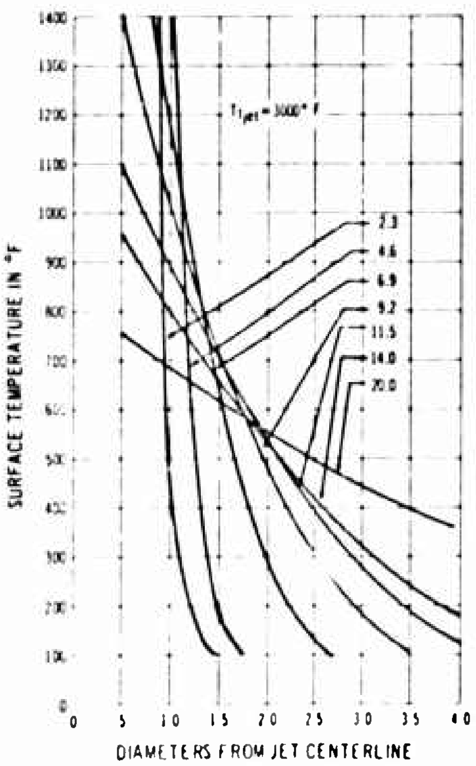
4.3.1 DESCRIPTION

The thrust reverser is an integral part of the nozzle structure, saving weight and reducing complexity. The reverser exit ports are located in the nozzle support structure. The variable position flap and seal segments forming the primary nozzle area control are also used for thrust reversal blockage. A schematic of the thrust reverser in the reverse thrust position is shown in Fig. 4-12.

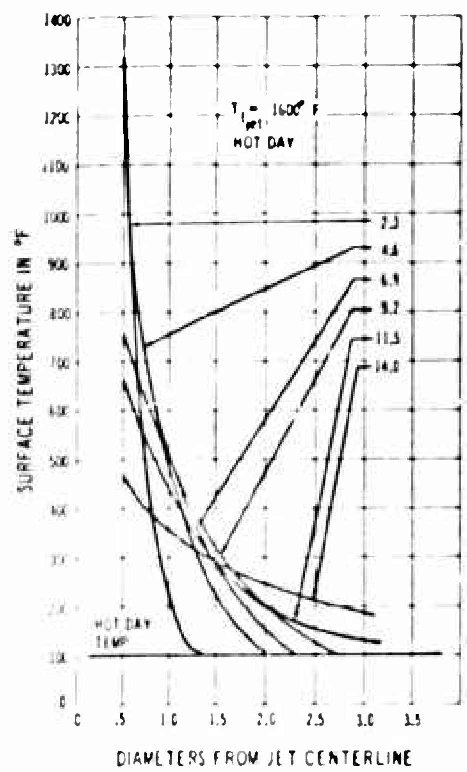
Thrust reversal is accomplished by moving a section of exterior cowling to uncover cascade assemblies located in the nozzle support structure. The primary nozzle variable position flap and seal segments are moved aft and deflected inward to block the normal exhaust flow path and to direct the exhaust outward through the cascades. The cascade assemblies turn and direct the gases into an efficient reverse thrust pattern.

A partial reverse position is incorporated into the thrust reverser design for additional flexibility in engine thrust control. A schematic of the thrust reverser in the partial reverse thrust position is shown in Fig. 4-13. Partial

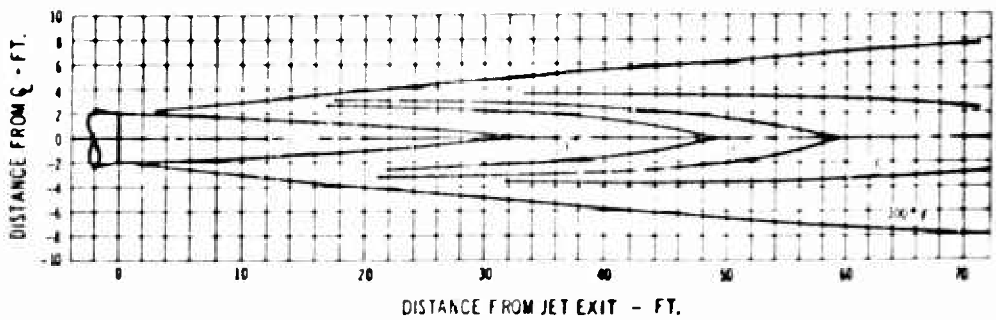
4-8 *Exhaust Effect on Adjacent Structure (test model)*



4-9 Exhaust Temperature, 3000°F Jet



4-10 Exhaust Temperature, 1600°F Jet



4-31 *Jet Temperature Profile Max Augmentation – M ∞ = 0.9 – 25,000 Feet*

thrust reversal is accomplished by opening the cowling to uncover the cascades and moving the variable position segments so that they partially block the normal exhaust flow path. The exhaust flow is divided so that a portion continues aft through the ejector nozzle and the remainder is directed outward through the reverser cascades.

4.3.2 OPERATION

The thrust reverser is governed by a closed loop, three position control portion of the exhaust nozzle and reverser control system. This consists of a signal mechanism and servo valve, synchronized hydraulic actuators, locking devices, and a mechanical position feedback and safety interlock system. The reverser is positioned as a function of the thrust lever setting. See Section 5 for a description of thrust reverser control and special features.

4.3.3 PERFORMANCE

The efficiency of a thrust reverser installation on an airplane at a given engine power setting is generally expressed as the ratio of the net thrust with the unit in the reverse position to the net thrust with the unit in the forward position.

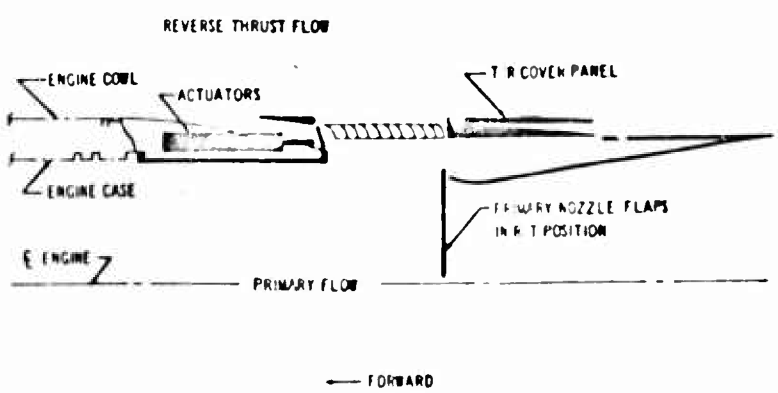
$$\lambda = \frac{F_n\ (\text{Reverse})}{F_n\ (\text{Forward})}$$

The net thrust with the unit in the reverse position experienced by the airplane differs from the gross thrust of the unit in the reverse position due to ram drag and interference effects of the reverse thrust flow field on the drag of the airplane. For example, the normal drag of extended flaps during the landing roll on the runway can be negated by the reverse thrust flow field under the wing.

The allowable gross thrust of the unit in reverse is influenced by three factors (1) the airspeed at which the reverse gases are ingested into the engine inlet, (2) the power setting of the engine with the unit in the reverse thrust position, and (3) the angle in relation to the engine centerline through which the reverse gases are turned.

- Ingestion of the reverse exhaust gases into the engine is caused mainly by the coanda attachment of the gases to the pod contour and by the restricted area between the wing and the ground plane in which the reverse gas must flow. At the higher airspeeds, momentum of the free stream forces the reverse gases to turn around and flow behind the airplane. As the airspeed is reduced, the turning point moves forward with respect to the pod until the engine inlet openings are reached and ingestion occurs.

  When ingestion occurs, the engine power must be reduced to decrease the mass flow of the reverse gases and allow the turning point to fall behind the inlet openings. Hence, as the airplane slows down, engine power is decreased proportionately.
- At the initiation of reverse thrust at touchdown airspeeds, the maximum possible reverse thrust force is desired and ingestion is not a problem. The highest practical engine power is employed. For augmented engines, this is maximum dry



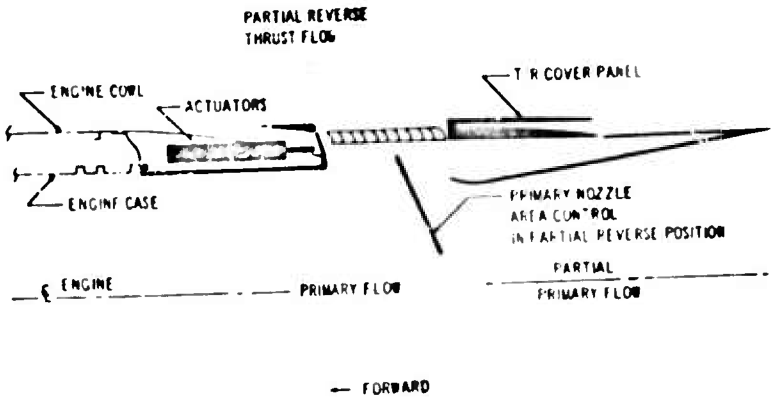
4-12 *Schematic Diagram Exhaust System – Reverse Thrust* power. The use of augmented power is not feasible since cooling the reverser components with secondary air is not possible.

- Boeing has established through experience that the maximum angle, in relation to the engine centerline through which the reverse gases are turned can be established by test only. For maximum reverse thrust, the gases should be turned forward to the extent that ingestion will occur at maximum dry power and low air speeds.

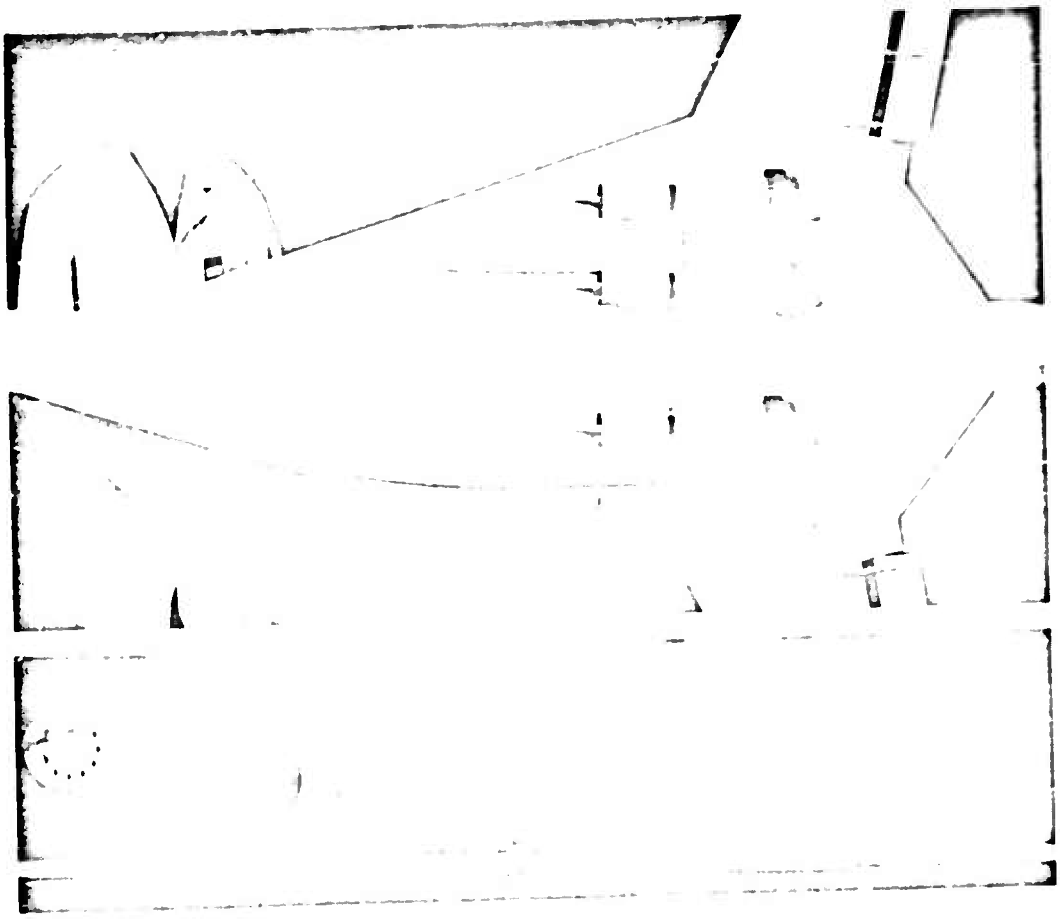
A model of the Boeing supersonic transport used for reverser development is shown in Figs. 4-14 and 4-15.

Three of the four pods have vacuum at the inlets and steam air at the reversers. Figs. 4-16 and 4-17 show the model installed and in operation.

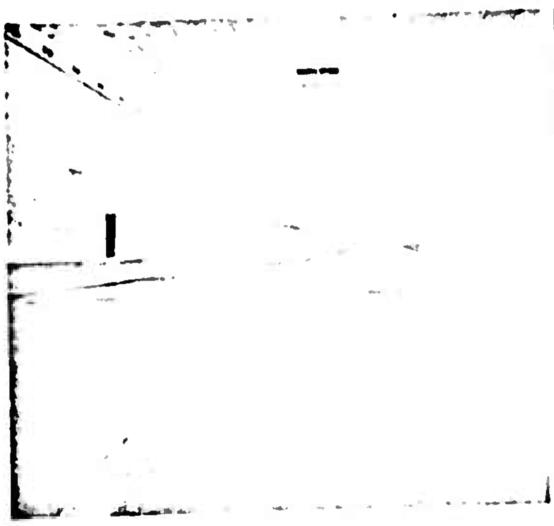
Experience gained from the Boeing Model 707 and 720 commercial transports and the supersonic transport reverser development programs establishes that an efficiency factor (λ) of 40 percent and a critical ingestion



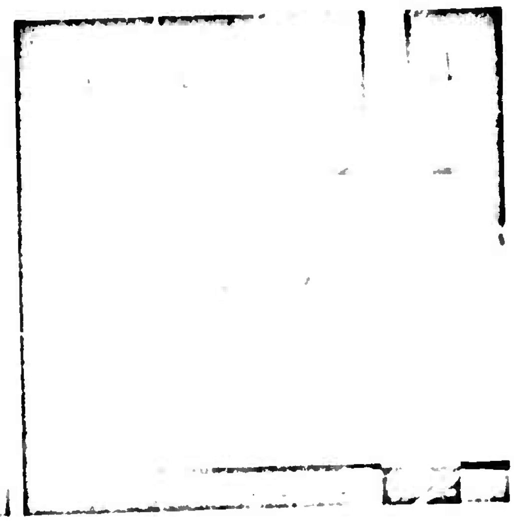
4-13 Schematic Diagram Exhaust System – Partial Reverse Position



SST Thrust Reverser Steam Model



4-16 *Thrust Reverser Model in Operation*



4-17 *Bottom View Through Transparent Ground Plane*

speed of 50 knots can be expected from the reverser configuration. Figs. 4-18 and 4-19 show the reverse thrust power schedule and the resulting reverse thrust available from touchdown speed to full stop. The curves are based on the following:

- Reverse thrust efficiency, λ = 0.40.
- Maximum dry power is used from touchdown speed to the critical ingestion speed of 50 knots.
- Continuous engine throttling from the maximum dry power setting at 50 knot ingestion speed down to 80 percent RPM (17 percent maximum dry power) at full stop.

The stopping distances noted in Volume A-V, Aerodynamics, are based on these data.

4.3.4 EXHAUST FLOW FIELD

The external zones into which the reverser exhaust gases can be discharged influenced the reverser configuration. The underwing podded installation results in some degree of surface impingement by the hot reverser gases on the wing and body and undercarriage. Careful control of the reverser exhaust flow fields is exercised to limit surface temperatures of adjacent structures to acceptable levels. Control of the placement of the exhaust gases is also required to avoid pressure buildups under the body and wing that would tend to pitch up the body. Temperature and pressure measurements are taken during the reverser development wind tunnel tests.

Fig. 4-20 shows the exhaust flow directions. The exhaust flow from the outboard engines is directed outboard and forward in an unrestricted flow path. The exhaust flow from the inboard engines is divided. The outboard portion is directed forward (behind the undercarriage) and under the outboard pod to mix with the outboard pod flow. The inboard portion is directed under the body to mix with the flow from the opposite inboard engine.

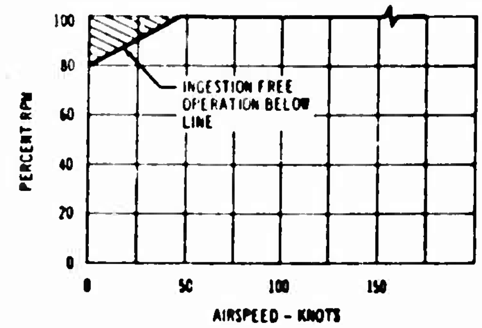
4.4 Secondary Air System

4.4.1 DESCRIPTION

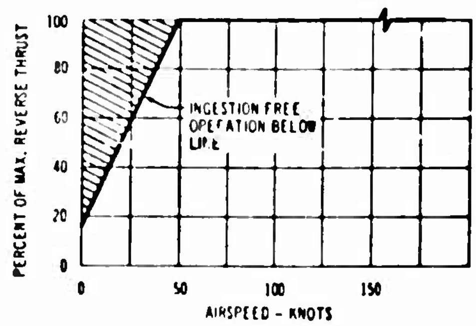
The aspiration and cooling requirements of the nozzle are supplied by variable area, normally flush scoops located in the exhaust system assembly exterior cowling. A schematic of the secondary air system is shown in Fig. 4-21.

The system consists of flush-type scoops with movable lips, ducting, actuators, plumbing, and controls. Two capture areas are established by the system. With the scoop lip in the flush position, the scoop capture area satisfies the dry (non-augmented) supersonic cruise secondary air requirements of the nozzle. With the scoop lip in the displaced outward position, the capture area is enlarged to satisfy the augmented, subsonic, and transonic secondary air requirements of the nozzle.

Air captured by the scoops is ducted inward to the forward end of the nozzle. From there it is directed over



4-18 *Reverse Thrust Power Schedule*



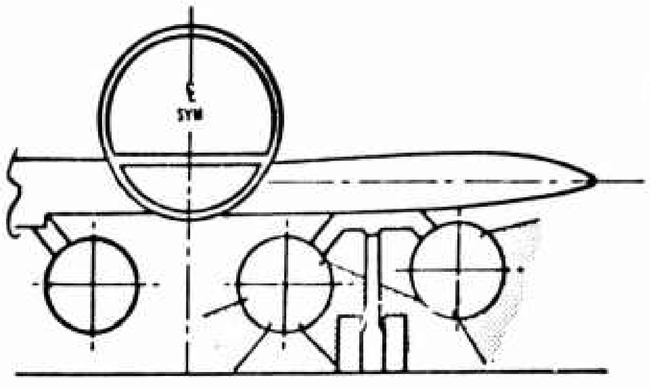
4-19 *Reverse Thrust Available*

the aft side of the primary nozzle segments and through the annular gap at the nozzle throat to fill and cool the ejector. A portion of the air flows between the exterior cowling and the ejector wall to enter the ejector nozzle area through slots provided downstream of the nozzle throat.

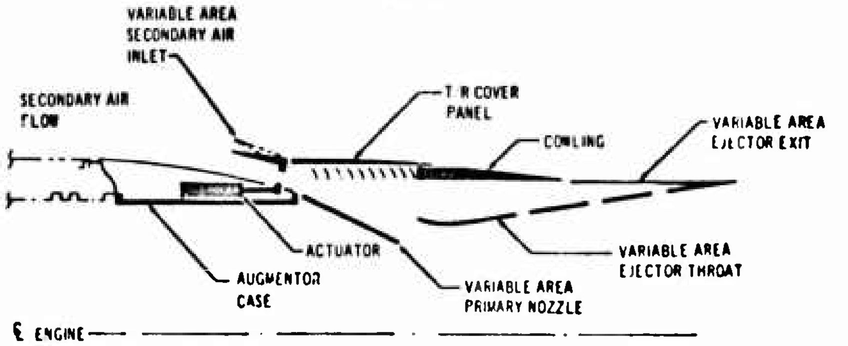
4.4.2 OPERATION

The secondary air system control is a portion of the exhaust nozzle and reverser control system. This consists of a signal mechanism and valve, hydraulic actuators, and the required plumbing.

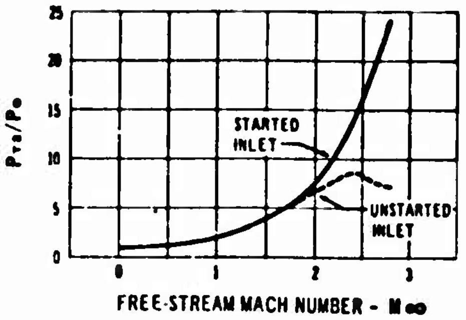
The signal mechanism senses the ratio of compressor inlet total pressure to the free stream static pressure. This provides, in essence, an airplane velocity indication as shown in Fig. 4-22. At pressure ratios indicative of airplane velocities of Mach 2.5 or higher, the signal mechanism will actuate the lip to the flush position, provided that the engine is in the dry power regime.



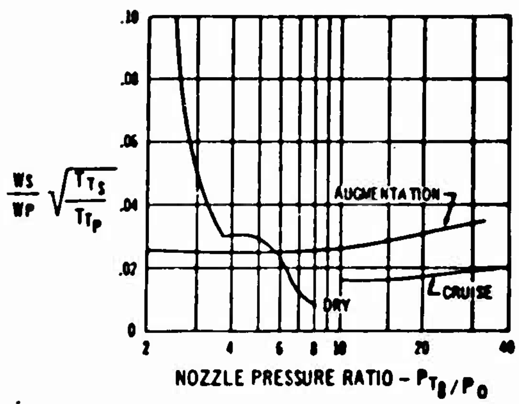
4-20 *Reverser Exhaust Pattern*



4-21 *Schematic Diagram Secondary Air System*



4-22 *Nozzle Secondary Air Inlet Control Pressure Ratio*



4-23 *Corrected Secondary Flow Used in Calculating Nozzle Performance*

4.4.3 PERFORMANCE

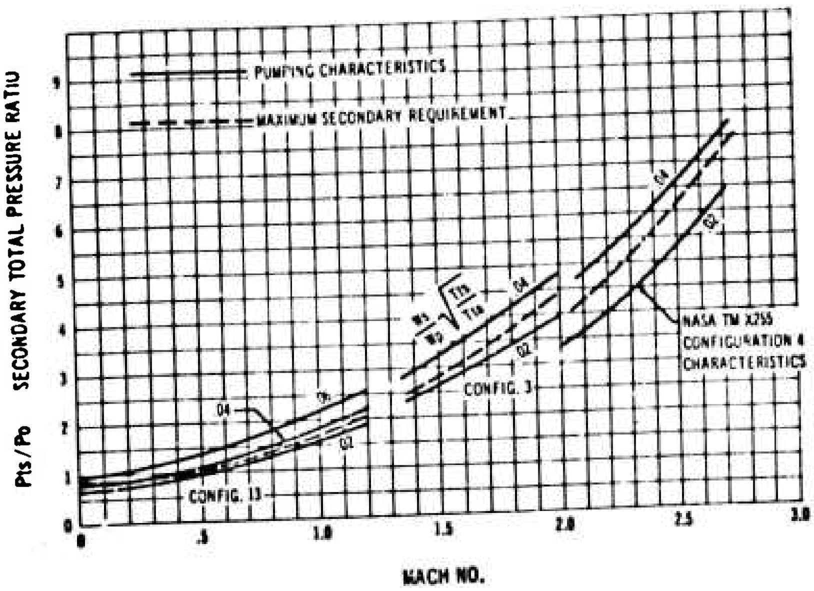
The secondary air system is designed to provide nozzle ventilation and cooling air flow as indicated in the curves in Fig. 4-23. In the dry power, low pressure ratio range, the rather steep rise in airflow requirements is established as a means of helping to prevent over-expansion of the primary airstream. At these low pressure ratio conditions, the optimum nozzle performance is obtained through the use of a greater amount of secondary air, thereby requiring less boattail angle.

The pumping characteristics of an ejector determine the required pressure level of the secondary air for a specific flow. These characteristics are mainly a function of primary pressure ratio and gap size between the primary nozzle and ejector throat.

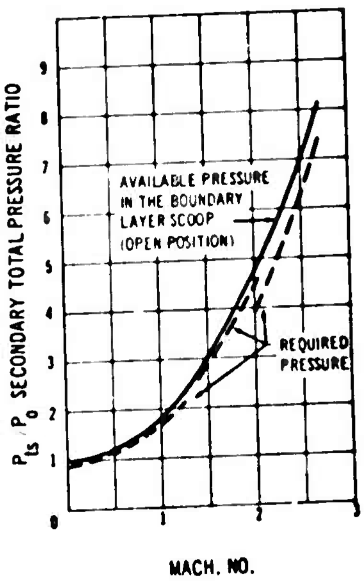
Fig. 4-24 shows an envelope of the pumping characteristics of a series of ejector models taken from NASA data (Ref. 3) matched to an engine over the Mach number range. The models selected in the development of this envelope agree closely in primary and exit geometry to the GE4 J4C nozzle.

Superimposed on the pumping characteristics curves is the curve of maximum secondary flow for the proposed nozzle, which indicates the required secondary pressure level.

Fig. 4-25 shows how the pressure recovered in the open position by the scoop meets the nozzle requirements. Calculations show the maximum scoop area requirement to be approximately one square foot per engine.



4-24 *Ejector Pumping Characteristics*



4-25 *Secondary Air Scoop Pressure*

VOLUME A-VI

PROPULSION

5.0 CONTROLS

5.0 CONTROLS (RFP 3.2.9.2 and 2.25.1c)

5.1 General Description

The propulsion system controls govern the operation of the engine inlet, starting system, exhaust system, and fuel system throughout all modes of flight.

The supersonic transport, operating over a broader speed range than today's subsonic aircraft, requires a more complex propulsion system; therefore special emphasis is being maintained during the development of the control system to ensure simplicity and reliability. One set of thrust levers, for example, controls all modes of thrust. "Piggy-back" reverse thrust levers are not employed. Maximum advantage has been taken of natural forces to avoid special and complex manual control requirements.

5.2 Inlet Controls (RFP 2.25.4)

Control of the supersonic inlet is accomplished by an automatic control system governing the position of the variable diameter centerbody and the controlled bypass doors. Natural forces act on secondary air inlet doors for takeoff and on secondary bypass doors to arrest shock expulsion. The system, except for the fuel supply pump, is self-contained within the inlet and requires no signal from the flight deck.

5.2.1 AUTOMATIC CONTROL SYSTEM

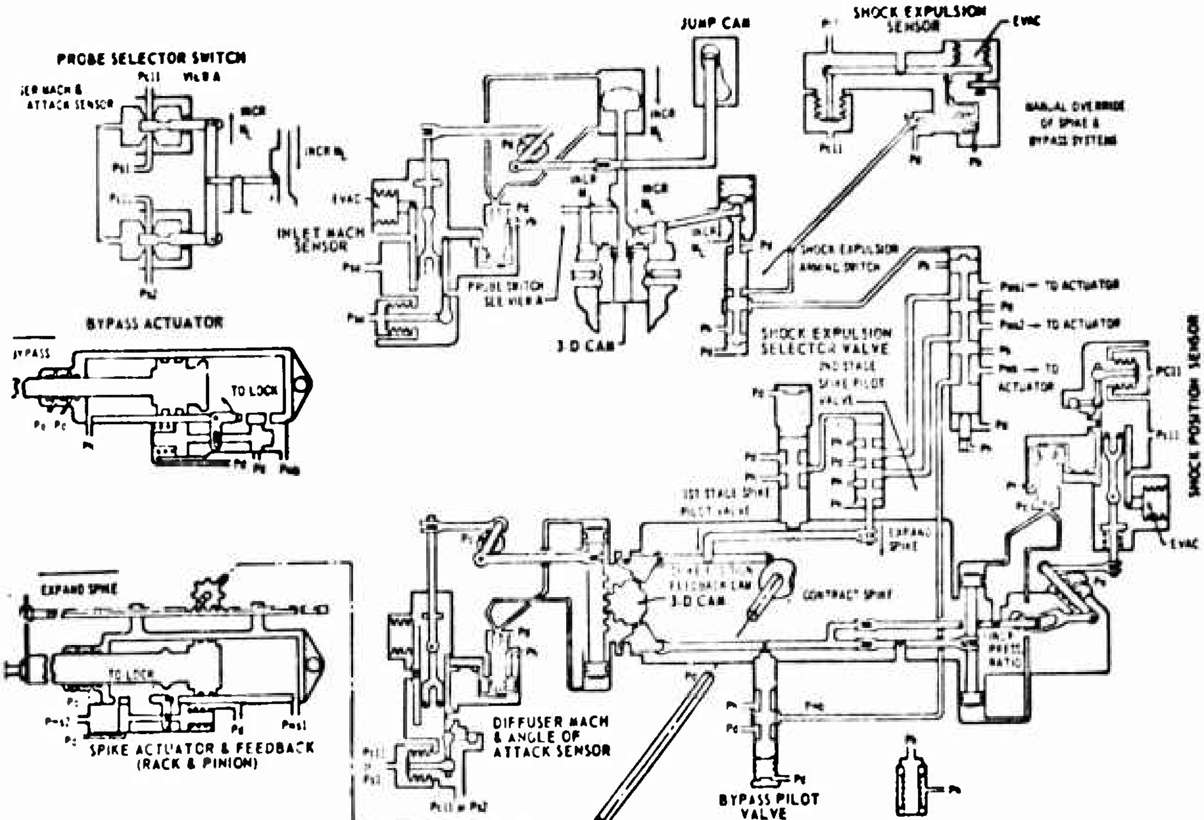
The automatic control is a hydromechanical unit which senses air pressures similarly to engine fuel control units and, in fact, operates with fuel as the fluid medium. An engine fuel control unit senses various pressure inputs, and schedules fluid flow as a function of the positioning of a three dimensional cam. Hydromechanical fuel control units have established an excellent record for reliable service. Airline maintenance of fuel control units is a routine operation. Technical coordination with various control unit vendors such as Hamilton Standard Division. United Aircraft Corporation; Marquardt, and the Garrett Corporation is in progress. The schematic of the automatic control system proposed for the Boeing supersonic inlet by Hamilton Standard is shown in Fig. 5-1.

Inlet Mach number, angle of incidence, and other pressure signals through the inlet are fed to the automatic controller to position the variable inlet geometry for optimum performance during all modes of flight. The automatic controller consists of a throat control loop to govern the position of the variable diameter centerbody and its airbleed exhaust doors, and a normal shock control loop to govern the controlled bypass doors. The overall loop gains and effective time constants of the throat and normal shock control loops are chosen to handle upstream and downstream disturbances.

The automatic control system is designed for safety and reliability. Redundancy of doors and actuators in the bypass system ensures that single failures will not cause loss of normal shock control. Design of the doors and pivots is such that, with loss of hydraulic power for the control and actuators, the controlled bypass doors will open to a fail-safe position.

5.2.1.1 Throat Control Loop

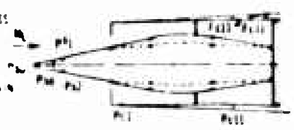
The throat control is an open loop comprised of a Mach sensor, centerbody position feed-back, and an attitude bias. Demand signals generated by the three dimensional cam (scheduler) as a function of Mach number and attitude inputs are mechanically fed into the servo system, initiating the necessary movement of the centerbody actuator. Completion of the actuator motion is effected by feed back linkage from the actuator to the servo, establishing the required throat area for inlet Mach number and attitude. Centerbody position as a function of inlet Mach number is shown on Fig. 5-2. The angle of incidence bias is shown on the same figure. Inlet Mach number and angle of incidence are measured by total and static pressures at the centerbody tip.



5-1 *Automatic Powered Control Schematic*

5-1 (Cont.) **LEGEND**

SPIKE TOTAL PRESSURE
SPIKE STATIC PRESSURE
SPIKE STATIC PRESS. FOR ANGLE OF INCIDENCE
FORWARD COWL STATIC PRESS
DIFFUSER TOTAL PRESS
DIFFUSER STATIC PRESS
HYDRAULIC SUPPLY PRESS
HYDRAULIC DRAIN PRESS
HYDRAULIC OVERBOARD DRAIN
METERED HYDRAULIC PRESS

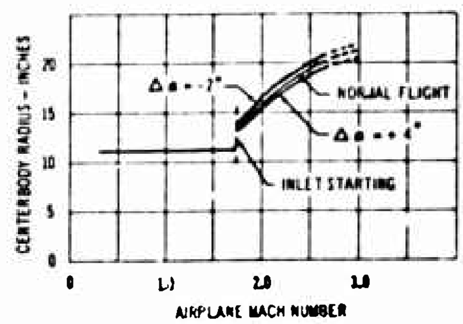


5.2.1.2 Normal Shock Control Loop

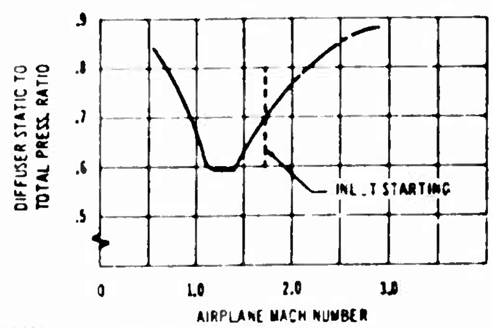
The normal shock control is a closed-loop system and consists of a reference signal scheduler, a normal shock position sensor, a servo, a start-unstart control, and a bypass door actuator. The closing loop is made by the aerodynamic feedback from the bypass door station to the diffuser Mach sensor. Operation of this system is divided into two distinct modes: one for the subsonic condition and external compression condition (up to Mach 1.8) and the other for the mixed compression condition at higher speeds (Mach 1.8 to 2.7). Switching of the operation is done by the inlet starting or unstarting initiated by the signal scheduler and the start-unstart control.

During the subsonic or external compression mode of the inlet, the Mach number sensed by the diffuser Mach sensor is compared with that generated by the reference signal scheduler. If an error exists, it will be fed into the servo, initiating the necessary movement of the bypass door actuators. Termination of bypass door actuation is accomplished by aerodynamic feedback from the bypass door to the diffuser Mach probes when the required inlet flow is established.

Operation of the loop during the mixed compression mode of the inlet is similar except that the reference signal now generated by the scheduler is for positioning the inlet normal shock. This provides the maximum pressure recovery with an adequate stability margin. The design stability margin is two percent of pressure recovery. Diffuser duct Mach number is the sensed signal



5-2 *Centerbody Radius Schedule*



5-3 *Diffuser Mach Schedule*

to control shock position. The same probes are used throughout the Mach range. The diffuser Mach number schedule for bypass door control is shown on Fig. 5-3.

5.2.1.3 Shock Expulsion Sensing System

In the event that the inlet should unstart, a shock expulsion sensor activates the servo to open the controlled bypass doors and throat area. This immediately restarts the inlet. After the shock has re-entered the inlet, the normal shock position sensor will resize the throat area and close the controlled bypass doors. Should some persistent fault exist so that the inlet successively unstarts, a counter on the shock expulsion arming system will energize to lock the controlled bypass doors open.

5.2.2 SECONDARY BYPASS SYSTEM

Secondary bypass doors located near the inlet lip remain closed at all speeds when the inlet is operating normally. At supersonic cruise speeds, if the inlet unstarts, the high pressure created at the front of the inlet immediately forces the secondary doors open. This stabilizes the normal shock at the inlet lip and prevents inlet buzz. The controller immediately restarts the inlet by opening the throat and the controlled bypass doors. This allows the spring loaded secondary doors to return to their normal closed position.

In the event that the controlled bypass system is inoperative and the inlet does not restart, the secondary bypass doors remain open and position the normal shock near the inlet lip to establish an inefficient but stable and safe flight condition.

These doors will open in the event that the engine is shut down and the windmill brake is applied (Par. 5.3.6).

5.2.3 TAKEOFF DOORS

During static, takeoff, and low speed, high engine power conditions, the engine will demand more air than can flow through the inlet lip. The negative pressure generated inside the inlet open the takeoff doors, providing more mass flow.

5.2.4 INLET CONTROL SYSTEM OPERATION

The percent cruise capture area as a function of the airplane Mach number is shown for the engine, the inlet, and the inlet throat in Fig. 5-4. The difference between engine requirements and the inlet capture area establishes the controlled bypass area requirement. The difference between the inlet capture area and 100 percent represents equivalent air to be externally spilled. The arrows with letters A through F correspond to letters in Fig. 5-5 which shows schematics of the inlet system and inlet flow patterns for several Mach numbers throughout the airplane speed range.

- SCHEMATIC A (TAKEOFF DOORS OPEN)

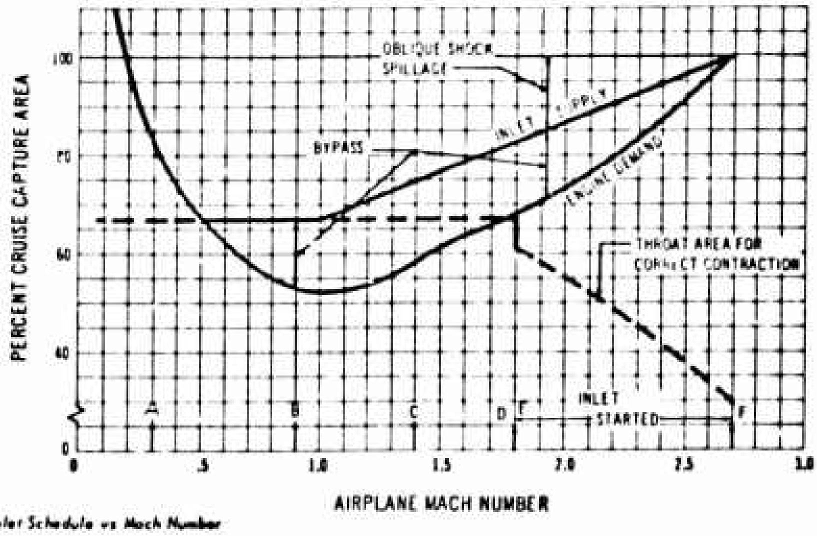
During takeoff and subsonic flight from Mach 0 to 0.3, the capture area demanded by the engine is larger than the supply of the inlet. The takeoff doors will suck in during this flight regime. The centerbody is fully contracted and the controlled and secondary doors remain closed. Above about Mach 0.3 the inlet supplies adequate air to the engine and the takeoff doors close.

- SCHEMATIC B
  (M 0.9, SUBSONIC BYPASSING)

At speeds above about Mach 0.5 the inlet supplies more air than the engine demands. Excess air is shown entering the inlet and is bypassed overboard through the controlled bypass doors. The centerbody is fully contracted

- SCHEMATIC C
  (M 1.4, TRANSONIC BYPASSING)

As the airplane speed increases, the controlled bypass doors progressively open. The maximum opening occurs at about Mach 1.1. The oblique and bow shock influences are gaining strength. Part of the air is deflected overboard in front of the inlet by the conical centerbody as external spillage. Excess air entering the inlet is bypassed over-



5-4 *Inlet Schedule vs Mach Number*

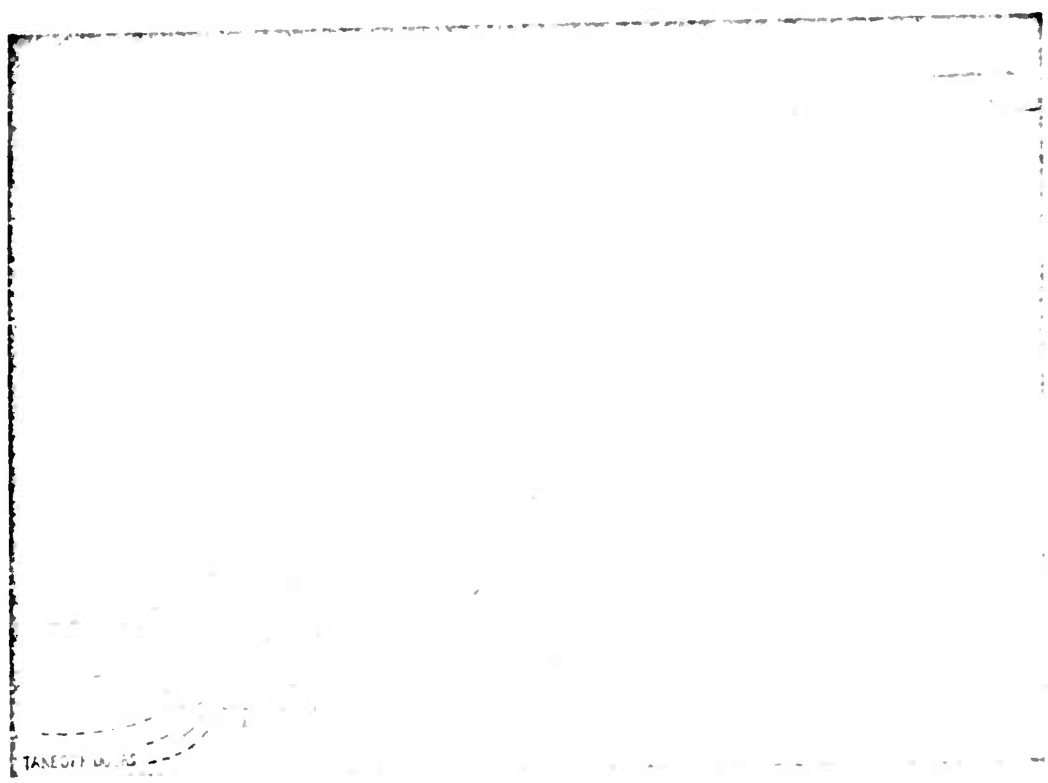
board through the controlled bypass doors.

- SCHEMATIC D
  (M 1.8, DURING INLET STARTING)

At a speed of about Mach 1.8, the inlet will switch from the external compression mode (normal shock in front of the inlet lip and subsonic flow inside the inlet), to the external-internal compression mode (normal shock swallowed and supersonic flow inside the inlet). The bypass doors open wide momentarily to allow the normal shock to enter the inlet. The centerbody diameter is increased slightly and the bypass door closed partially, locating the internal normal shock close to the throat of the inlet.

- SCHEMATIC E
  (M 1.8, AFTER STARTING)

After starting, the inlet continues to operate in the external-internal compression mode. Air is being spilled externally through the external conical shock system; excess air captured by the inlet is discharged through the



5-5 *Inlet Functional Schematic*

controlled bypass doors. The internal normal shock is maintained close to the inlet throat by the automatic control system.

- SCHEMATIC F

(M 2.7, DESIGN CRUISE MACH NUMBER)

As the airplane speed approaches the design Mach number, the automatic control system continues to expand the centerbody and close the controlled bypass doors. At the design point the oblique shock off the centerbody is approximately on the inlet lip with minimum external and bypass spillage.

5.3 Engine Controls (RFP 2.25.1e)

The engine control system provides the means of transmitting the pilots' desired thrust variations to the engine. The pilot positions the thrust levers on the aisle stand to the desired setting. This signal is mechanically carried to the power control system on the engine. In response to this signal, the main fuel and stator portion of the power control system controls fuel flow to the main combustors and positions the variable stators in the compressor; the augmentation control portion of the power control system controls fuel to the augmentor spraybars; the nozzle and thrust reverser control portion of the power control system positions the variable area components of the nozzle for proper area control and locates the thrust reverser components for the correct operating regime. The power control system also protects the engine from exceeding RPM and turbine temperature limits during stabilized operation as well as during acceleration and deceleration.

The thrust levers and the start levers are connected to the engine by means of cable control systems and lin[illegible] 5-6). The cable systems incorporate auto-tensio[illegible] to facilitate engine replacement. The cables, p[illegible]s, and linkages for the thrust and start controls are made of corrosion resistant steels.

The automatic thrust control portion of the automatic flight control system operates servomechanisms on the thrust control cables to vary thrust. Manual thrust controls by means of the thrust levers may override at any time.

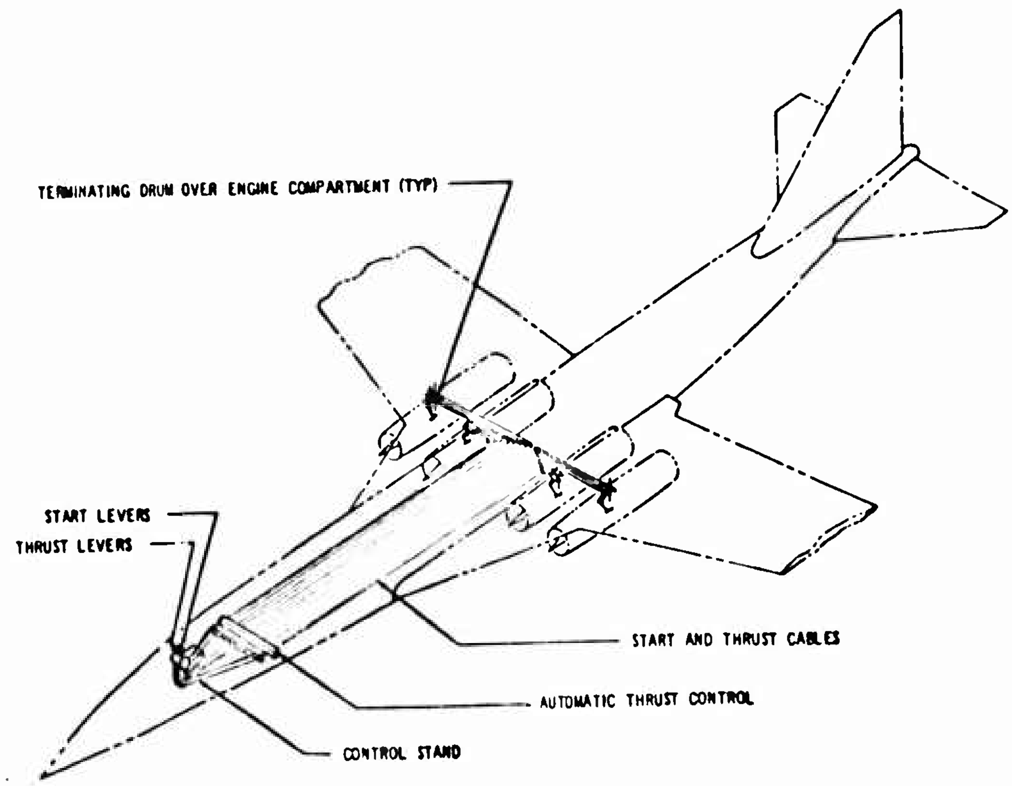
As a safety precaution, the thrust and start levers are physically separated, as shown in Fig. 5-7 Engine shutdown by using the start lever is accomplished by moving the lever to CUTOFF position, thereby closing the stopcock on the engine. Engine shutdown by using the thrust lever is accomplished by moving the lever to the IDLE position and activating the proper switches on the engineer's panel to shut off the fuel supply to the engine. Fig. 6-8 shows, schematically, the main features of the engine control system.

5.3.1 START CONTROLS (RFP 2.25.1d and 3.2.9.3)

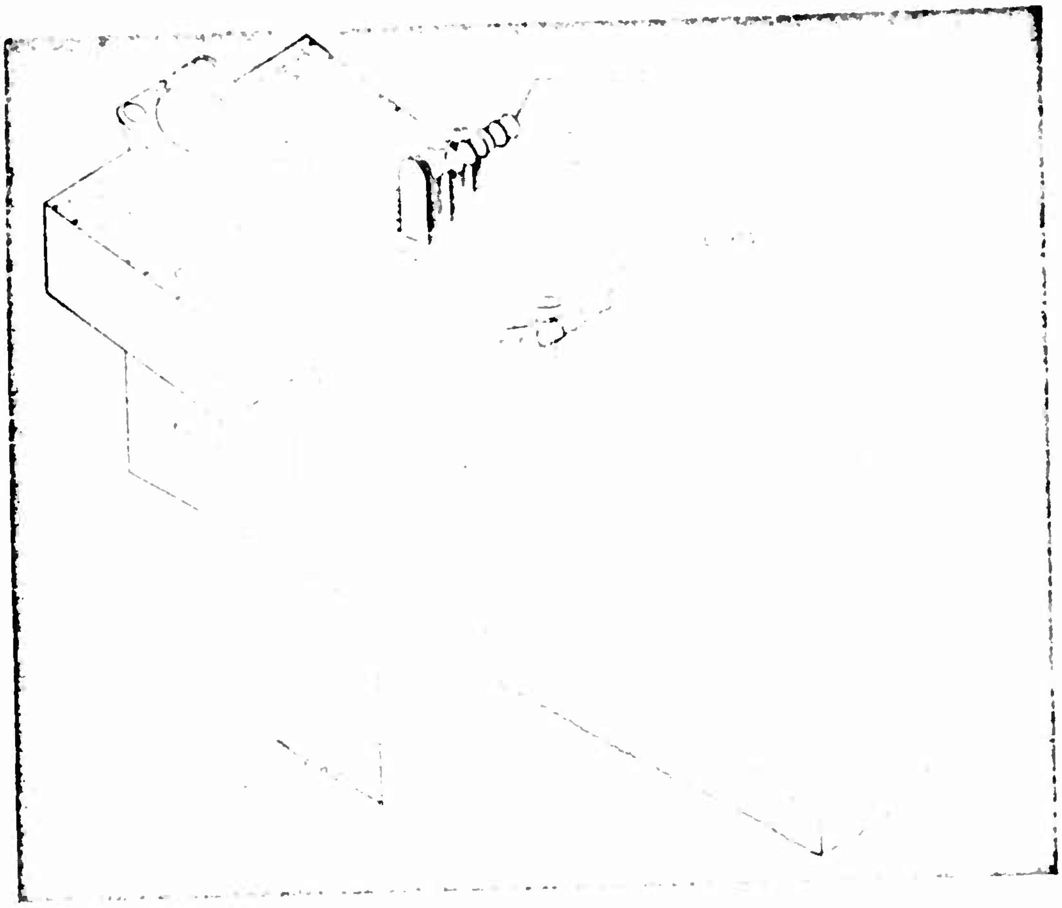
Starting and cutoff of each of the four engines is controlled by individual levers located on the control stand. Each start lever may be set at START, IDLE and CUT-OFF positions to control the start and cutoff functions. With the thrust levers in IDLE, moving the start lever from CUTOFF to START transmits a mechanical input into the primary fuel and stator control unit on the engine. As a result of this input, the control unit mechanically opens the engine fuel stopcock allowing fuel to flow through the main fuel pump to the control unit metering system and then to the fuel nozzles. Lightoff of the engine is accomplished by an electric signal from the flight deck which energizes the engine ignition system. After the start is accomplished, the start lever is moved into IDLE.

A pneumatic starter is used for starting each engine. A pressure regulating valve, installed in the pneumatic supply duct, controls the air supply to the starter.

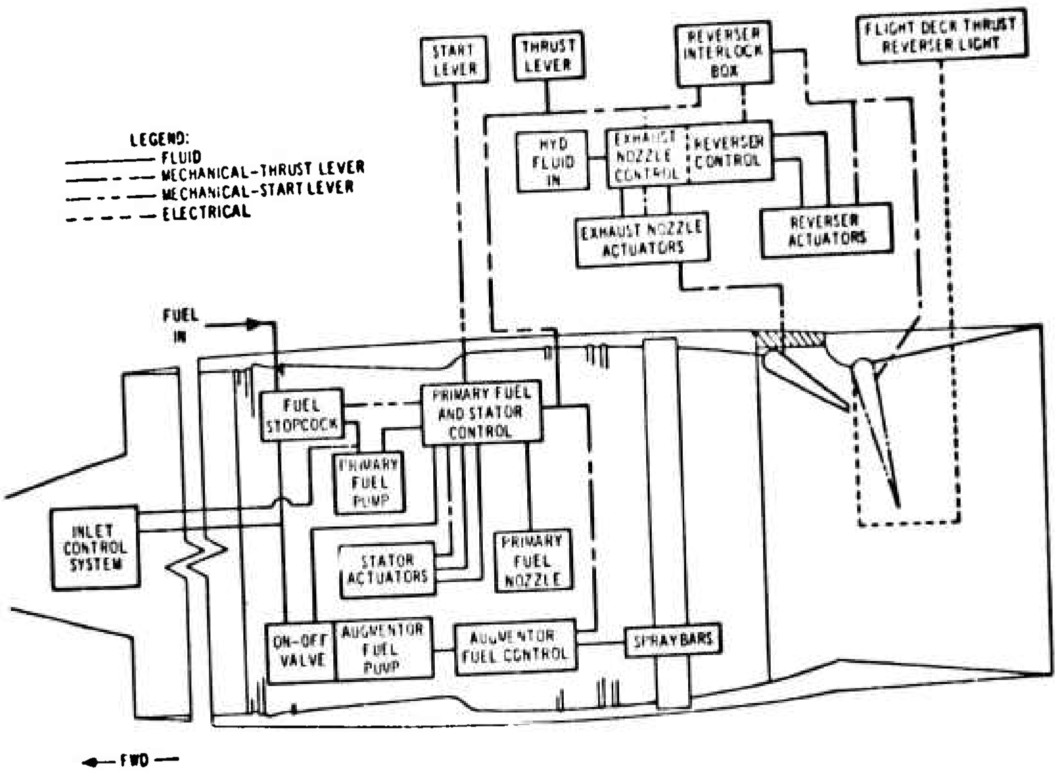
A schematic diagram of the electrical circuit of the system is shown in Fig. 5-9. A single, guarded toggle switch for each engine controls the solenoid of the starter



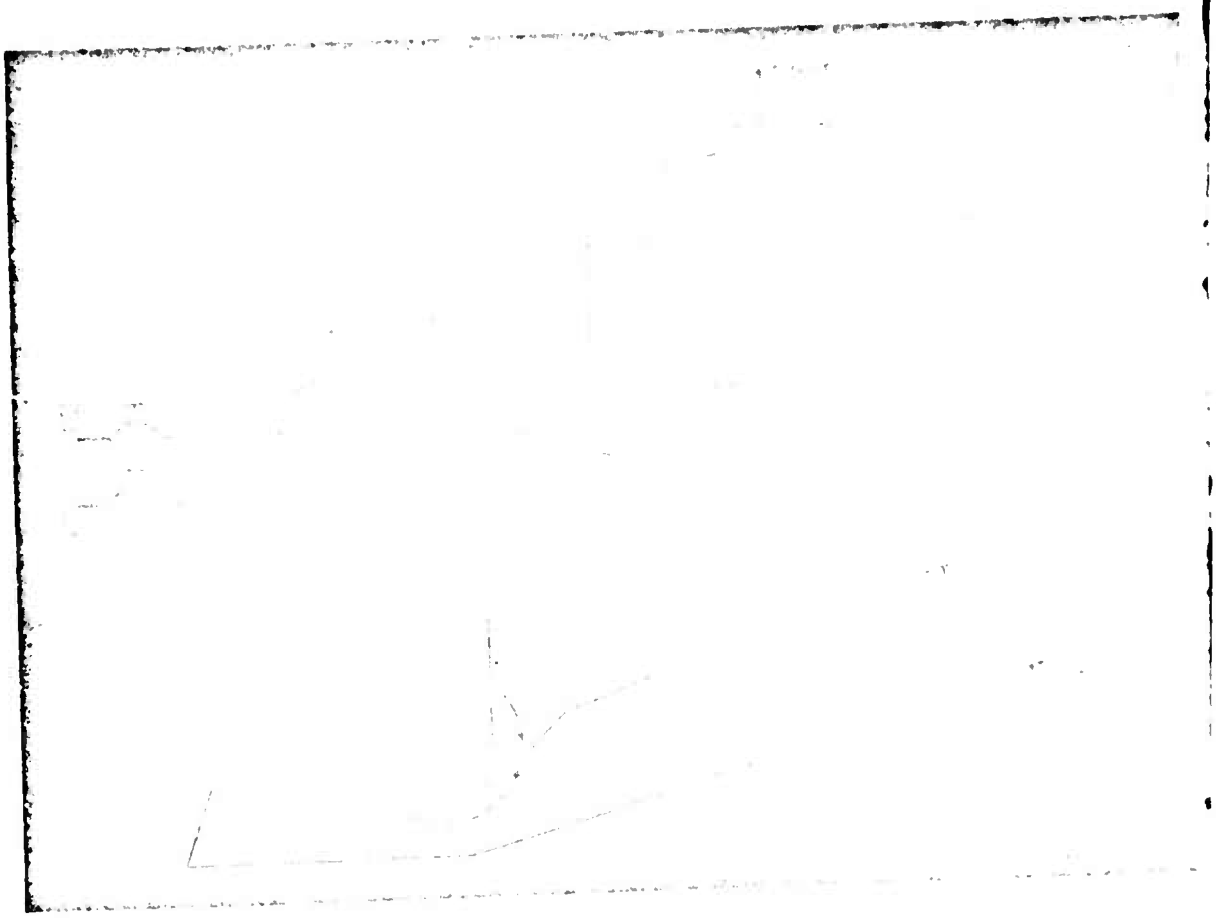
5-4 *Airplane Engine Control System*



5-7 *Pilots' Control Stand*



8-8 *Engine Control Schematic GE4 J4C Engine*



5-9 Start System Electrical Circuit

pneumatic valve. The switch, installed on the control stand, has three positions: GROUND-START, OFF, and FLIGHT-START. The switch is held in GROUND-START (momentary position) to arm the ignition system and to energize the starter valve. The open starter valve directs air to the pneumatic starter to begin rotation of the engine. Ignition is then obtained when the start lever is moved to START. Electrical energy to the starter valve is interrupted by a cutoff switch on the starter which automatically opens the start circuit when the cutoff speed is reached. The starting cycle can also be terminated at any time by releasing the start switch.

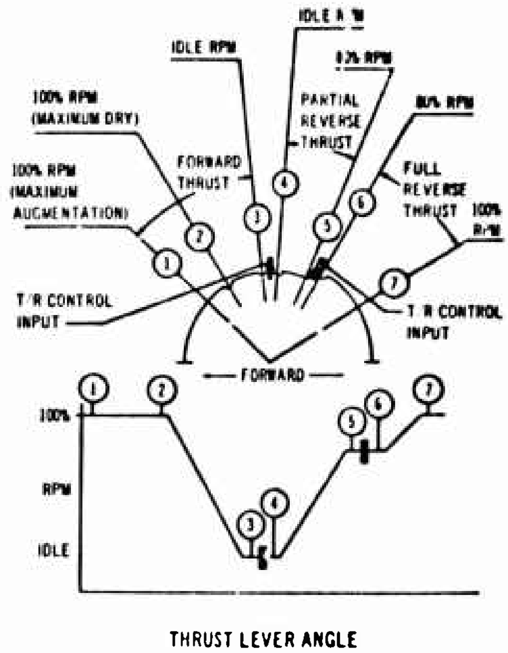
Moving the starting switch to FLIGHT-START (maintain-contact position) energizes the engine ignition system regardless of the position of the start lever but does not operate the starter. This position of the start switch is used for starting when the engine is windmilling, or when ignition is desired during takeoff, landing, or flying through se ere turbulence.

Engine motoring is obtained on the ground without ignition by moving the start switch to GROUND-START with the start lever in CUTOFF.

5.3.2 THRUST CONTROLS (RFP 2.25.1a)

The thrust of each of the four engines is controlled by individual thrust levers located on the control stand immediately forward of the start levers. The thrust levers control engine thrust from full reverse at the aft end of the lever travel through idle to full augmented forward thrust at the extreme forward lever position. Fig. 5-10 is a diagram of the thru 'ever positions.

For starting, the thrust r.mains at the idle stop position, shown as (3) i -10. After the start, moving the thrust lever forw. smits a mechanical signal to the engine primary fu l stator control unit and to the exhaust nozzle and thrust reverser control to increase engine power in forward thrust and to govern nozzle areas accordingly.



5-10 *Thrust Lever System*

Advancing the thrust lever to position (2) establishes maximum dry power at 100 percent rotor RPM. Advancing the thrust lever to position (1) establishes maximum augmented power at 100 percent rotor RPM. Retarding the thrust lever to position (3) reduces the engine to idle power. At position (3) the idle stop is encountered.

Lifting over the idle stop to position (4) actuates the thrust reverser to the partial reverse thrust position with the engine at idle RPM. Moving the thrust lever to position (5) accelerates the engine to approximately 80 percent RPM, with the thrust reverser remaining in the partial reverse position. At position (5) a high lift stop is encountered by the thrust lever.

Lifting the thrust lever over the high lift stop to position (6) actuates the thrust reverser to the full reverse thrust position, with the engine remaining at approximately 80 percent RPM. Moving the thrust lever to position (7) accelerates the engine to 100 percent RPM dry power, with the thrust reverser in the full reverse thrust position.

In the full reverse thrust position, the engine fuel control governs fuel flow so that engine overspeed will not be a problem. The thrust reverser control is interconnected with the engine fuel and stator control to close the stators slightly from their normal schedule. This increases the compressor stall margins, decreases the engine sensitivity to temperature distortion at the inlet, and allows some ingestion of the exhaust gas without causing surge.

Ingestion is a function of engine power and airplane velocity. At some airplane velocity during full power reverse thrust deceleration, a critical ingestion point is reached which will cause surge unless power is reduced. The pilot should move the thrust lever from position (7) toward position (6) to reduce power. It is anticipated that the 80 percent RPM, full reverse thrust position (6), can be maintained down to airplane turnoff velocities. From this point the idle RPM, partial reverse taxi procedure, position (4), may be used.

5.3.3 FLIGHT IDLE THROTTLING

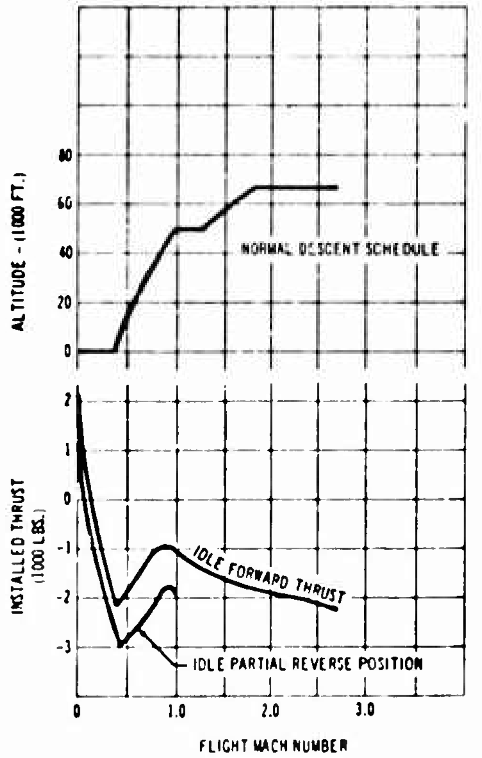
To aid in slowing the airplane during descent, an RPM unlocked rotor fuel schedule regime at flight idle power is incorporated in the engine control. The pilot may retard the throttle fully to the idle position (3). The engine rotor RPM decays, engine fuel flow is reduced, and the stators reposition to reduce airflow. The engine inlet automatically compensates for the reduced airflow condition. The reduction in fuel flow during the descent results in the saving of a substantial amount of fuel. The reduction in airflow establishes a moderate level of negative thrust per engine. In Fig. 5-11, the installed thrust of the engine as a function of Mach number is shown throughout the normal descent schedule for the RPM unlocked rotor idle forward thrust condition and the idle partial reverse thrust condition.

At airplane speeds above Mach number 1.5, the fuel flow rate at idle power does not supply the continuous cooling requirement of the airplane and engine systems combined. The fuel delivered to the engine during this condition will not exceed 125° F. When high airplane speed, idle power conditions are to be held for long periods, the air bled from the engine inlets for the cabin air system is switched from fuel-air heat exchanger cooling to ram air cooling.

5.3.4 PARTIAL REVERSE THRUST CONTROL

The partial reverse position provides the pilots with better control of airplane speed during normal descent, landing, and taxiing.

- During normal descent for landing, at airplane velocities below 300 knots indicated air speed, the pilot may place the thrust levers in the idle RPM, partial reverse thrust position (4). This produces a greater level of negative thrust than unlocked rotor at idle power, as shown in Fig. 5-11. During manual glide slope control the pilot may intermittently use the same position for speed control.



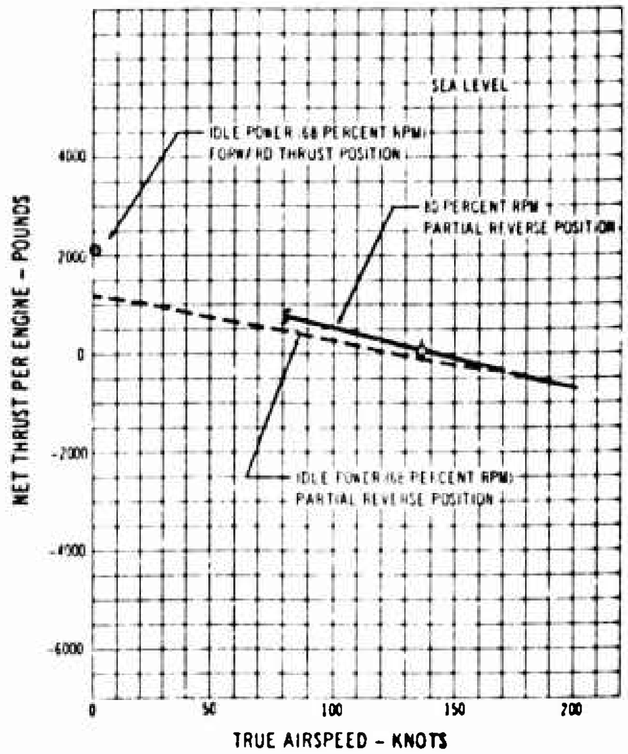
5-11 *Unlocked Rotor Idle Descent Performance*

- At the beginning of the flare on final landing approach, the pilot may pull the thrust levers through the idle detent and against the high lift stop at position (5). This produces zero to slightly negative thrust, as shown in Fig 5-12. Maintaining higher than idle RPM at touchdown enables the pilot to get faster and equalized accelerations to full power reverse thrust on command. Approximately 3 seconds of engine acceleration time are eliminated. The 80 percent RPM on the acceleration schedule at position (5) allows for thrust control rigging tolerances by bringing all engines to equalized RPM. This prevents asymmetric reverse thrust caused by these tolerances or by variations in the acceleration rate of the engines when accelerating from idle RPM.
- The pilot may use the partial reverse thrust position during taxi conditions. The idle thrust to weight ratio of today's commercial turbofan transports causes high taxi speeds unless brakes are used. On long taxi runs significant brake heat is generated. The idle thrust to weight ratio on the supersonic transport makes this more severe. Taxiing with one or more engines in idle reverse thrust blows foreign matter up off the runways to be ingested by the engines. The use of the partial reverse position for taxiing eliminates the ingestion hazard and reduces the forward thrust of the engines at idle by approximately 50 percent.

5.3.5 SAFETY INTERLOCK SYSTEM

The control system incorporates a mechanical safety interlock. This device was conceived and developed by The Boeing Company and installed on all Boeing jet transports. The significant features of the safety interlock are as follows:

- Power cannot be increased in the forward thrust



5-12 *Landing Flare and Taxi Thrust* lever regime unless the reverser is in the forward thrust position

- Power cannot be increased in the partial reverse thrust lever regime unless the reverser is in the partial reverse thrust position.
- Power cannot be increased in the full reverse thrust lever regime unless the reversers are in the full reverse thrust position.
- In the event that, at any power condition, the reversers should depart from the position dictated by the thrust lever position, the engine power will be reduced such that the net effect on the airplane will be equivalent to one-engine-out operating conditions.

The safety interlock between the thrust lever position and the thrust reverser position provides the pilot with an immediate signal in the event of malfunction. The movement or resistance of a thrust lever, coupled with position indicating warning lights on the flight deck, enables the pilot, in the event of a sudden change of thrust associated with the reverser, to determine which engine is affected and what modes of thrust are still available to him. The reverser position indicating light mounted on the pilots' center panel goes on when the reverser has left the forward thrust position.

5.3.6 WINDMILL BRAKE CONTROL (RFP 2.25.5)

In the event of inflight shutdown at high speed the engine may rotate at high RPM. Oil starvation, engine component deterioration, and seizure of the rotor would be potential hazards. To minimize the problem, a windmill brake is employed. The compressor outlet guide vanes are rotated in the engine to an overlapping position. The engine rotation is reduced to approximately 20 percent RPM. At this RPM engine windmilling is comparable to that on present day aircraft. Control of the windmill brake is accomplished by moving the engine start lever to cutoff.

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6.0 ENGINE STARTING (RFP 2.25.1g, 3.2.9.3)

High reliability was a primary objective throughout the selection of the engine starting system. Simplicity and the ability to use any one of several air sources are features of the chosen system.

The development and test plan to ensure a safe and reliable engine starting system is presented in Par. 8.5 of this volume.

6.1 System Description (RFP 2.10)

A pneumatic starter is used for starting each engine. A pressure regulating and shutoff valve installed in the pneumatic supply duct controls the air supply to the starter. The installation of the starter and valve is shown in Fig. 6-1. The manner in which the flight deck controls are operated to start the engines is explained in Section 5.

Accessory loads during starting, such as torques required to overcome the mechanical friction and the inertia of the hydraulic pumps, the constant speed drive, and the generator, have been taken into account in calculating starting power requirements and starting time. The engine driven hydraulic pumps incorporate a bypass system to unload the pumps during the start. The engine-driven generators are also unloaded during the start. To aid in preventing "hot" starts, the starters are sized to provide good acceleration through the engine lightoff range.

Starting can be accomplished with air from conventional airline ground carts having a pressure of approximately 50 psia, or from an operating engine.

The starter is installed on the engine gearbox. A quick-attach-detach coupling, supplied as a component of the starter, facilitates removal and installation. To accomplish this, a single clamping bolt, requiring a standard tool, attaches the starter to the engine. The adapter and clamping ring portion of the coupling remain with the engine when the starter is removed so that there are no loose parts.

The starter exhaust is discharged directly into the engine compartment and then overboard through a vent.

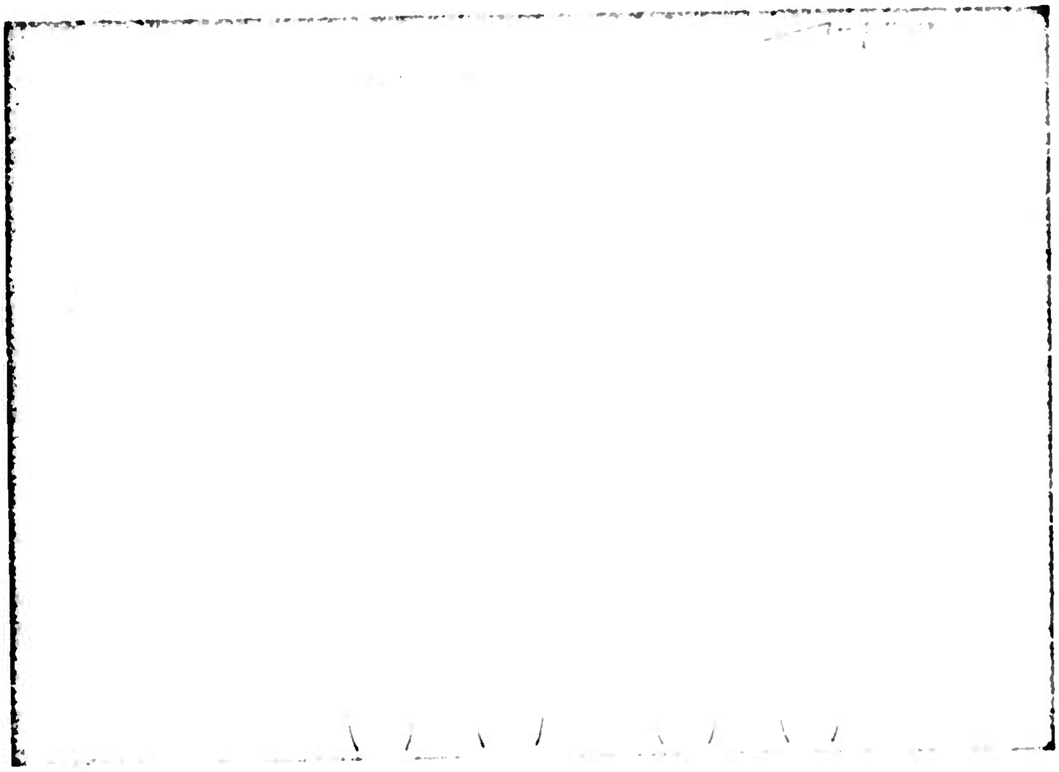
The starter and valve are installed in an area in which the temperature is 350° F. or less throughout the airplane operating range. They are designed with a 100° F. margin above this temperature to ensure long periods between servicing. The method used to control the accessory environmental temperature is explained in Section 2. No external cooling of the starter or starter oil is required. The starter is lubricated with oil of the same specification as that designated for the engines. When the lubricating oil in the engine area is heated to 40° F. or above, the engine may be started at an ambient temperature as low as —65° F.

6.2 SYSTEM CHOICE AND TRADES

During the selection of the starting system described in this section, several other starting systems were considered. The system chosen is believed to be the right one for the GE4/J4C engine. However, other systems would be reviewed in detail if a different engine, especially one requiring greater starting energy, were selected. In the paragraphs which follow, a few of the alternate systems are touched on briefly in order to justify the low-pressure pneumatic system which has been chosen.

Cartridge starters are not believed to be the proper choice for today's commercial aircraft. The present cartridge must be stored in a controlled environment such as in the pressurized cabin. The products of combustion, in addition to creating a smoke problem at a terminal, are toxic and corrosive. Cartridge starting is more costly than pneumatic starting. Cartridges safer and easier to handle are being developed. Progress in this field will be closely monitored.

Alternating current electric starting in conjunction with a hydraulic constant speed drive was ruled out because a 40 KVA starter generator cannot be constructed to efficiently develop the 200 horsepower needed for a



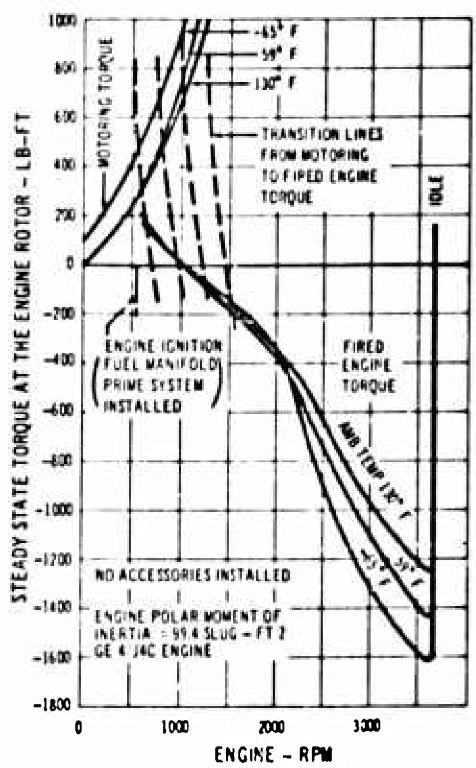
6-1 *Pneumatic Starter Installation* reasonable GE4 J4C starting time. A recent endeavor by Boeing and associated vendors to design an electric starting system for the Model 727 was abandoned after the program had progressed well into development and testing. Electro-mechanical integration problems, cost, and weight trends showed that a reasonable solution could not be achieved. Recognizing the advantages of a self-contained electrical starting system, Boeing will monitor development work in this field for new approaches that may show promise for the SST.

A review of energy requirements comparing an impingement starter with a gearbox-mounted pneumatic starter shows that impingement starting requires two to three times as much energy. General Electric indicated that check valves, weight, and blade problems were also associated with this type of starting.

Starters utilizing combustors to increase the available air energy were also considered. Experience with this type of starting demonstrated that its complexity and reliability left much to be desired. Since it was found that a satisfactory engine start could be obtained with a simple pneumatic starter and an existing ground cart, the use of a combuster was no longer considered. In any case, if an increase in energy is required, the choice would be to add the combuster to the existing ground equipment rather than to the airplane.

Starting systems using stored pneumatic energy were briefly considered. The storage cylinder would be roughly twice the size and weight of the present storage cylinders used on the 707's. The larger storage cylinder volume, combined with the shorter time allowed to recharge the bottle during flight, would greatly increase the inflight pumping requirements. The higher onboard air temperatures to supply the compressor inlet air pose a further design problem.

A pneumatic starter system using higher pressure was considered. The advantage of lighter aircraft components is offset by the requirement for new and more costly



6-2 *Engine Starting Torque Requirements* ground starting equipment. Cross-starting with the higher pressure system requires greater engine power, with the resulting objectionable increase in airport terminal noise. Should the starting requirements of the ultimate SST engine differ appreciably from those used in the proposal, consideration can be given to the higher pressure system.

Hydraulic, direct-drive mechanical, and pneumatic constant speed drive starters were also considered.

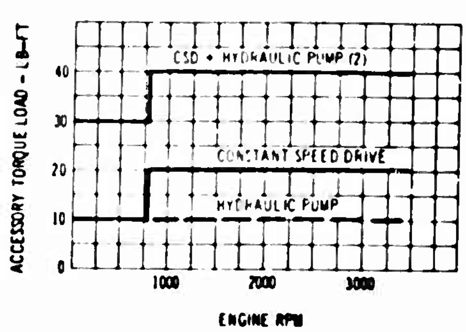
6.3 STARTER AND ASSOCIATED SYSTEM COMPATIBILITY

6.3.1 COMPATIBILITY WITH SELECTED ENGINE

The steady state starting torque characteristics of the GE4 J4C engine at three ambient conditions are shown in Fig. 6-2. The airframe accessory torque loads (Fig. 6-3) reflect the loads due to two unloaded hydraulic pumps and one constant speed drive with generator carrying no electrical load. The starter torque available is shown in Fig. 6-4 for four ambient conditions: −65° F., +59° F., +130° F. at sea level; and on a hot day (standard day temperature +61° F.) at 10,000 feet pressure altitude. Fig. 6-5 shows a typical performance plot for the engine-starter combination and the excess torque available for accelerating the engine rotor and airframe accessories, using either two ground carts or a single ground cart. Performance of typical ground carts supplying the air energy to the starter is shown in Fig. 6-6. The performance figures include temperature and pressure losses in the airplane ducting. Fig. 6-7 shows the starting time as a function of engine RPM for the four ambient conditions under consideration. Fig. 6-8 is a cross-plot of these data and shows directly the effect of ambient temperature on starting time. A point representing starting time at 10,000 feet altitude on a hot day is also shown on the plot.

6.3.2 COMPATIBILITY WITH GROUND EQUIPMENT

Starting may be accomplished by using the output from a single ground cart, from two ground carts, or by cross starting from an operating engine. Airline ground carts of the type now in existence can be used.

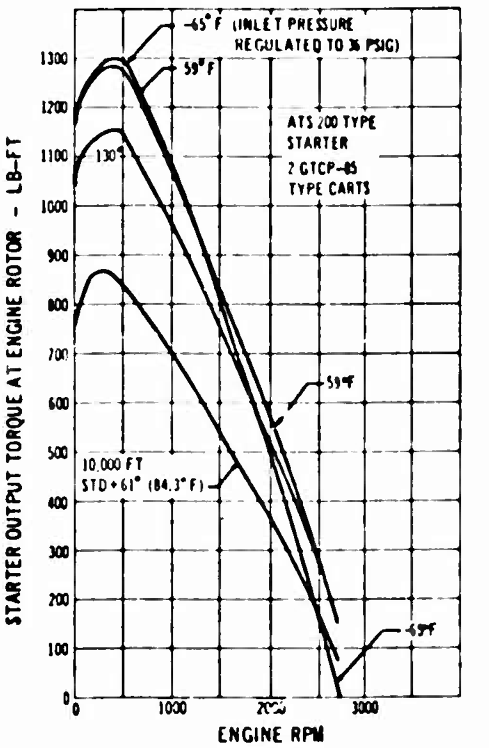


6-3 *Accessory Torque Load During Start*

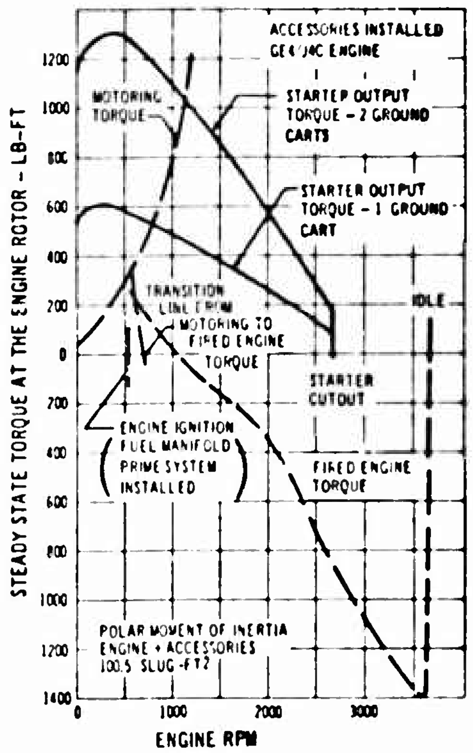
The engines may be started by using the output equivalent of two ground carts. This will result in an engine start in 38 seconds on a standard day, which is well within the RFP requirement and comparable to present day jet aircraft starting time.

Succeeding engines may each be started in the 38 second time period by continuing to use the ground equipment or by cross starting from an operating engine. Cross starting is accomplished by setting the engine RPM between 75 and 80 percent depending upon the starting time desired and permissible noise level.

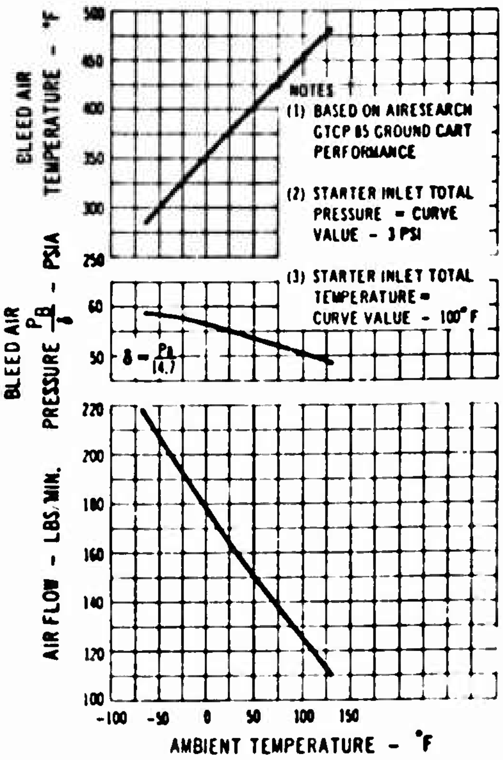
The engines may be started using a single ground cart. In this case starting time will be 70 seconds on a standard day. With one engine started, each succeeding engine can be cross started in 36 seconds. The resulting total starting time for all four engines will be approximately three minutes.



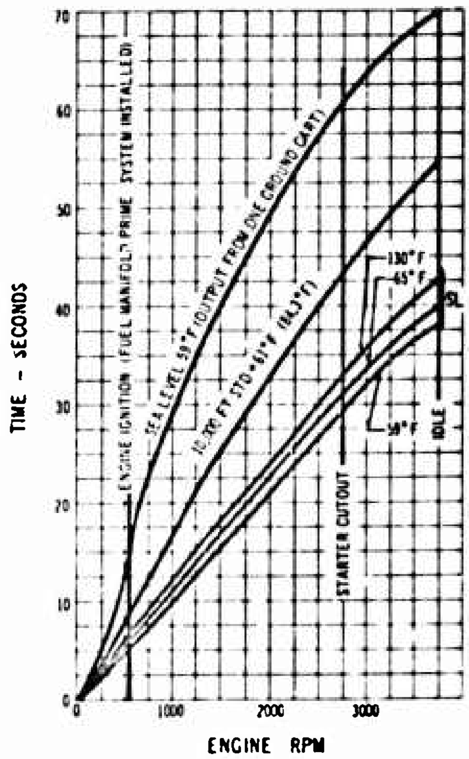
6-4 Pneumatic Starter Output Torque



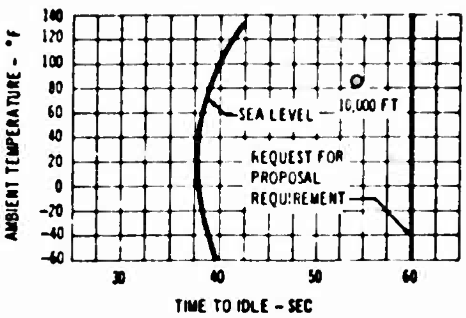
6-5 Pneumatic Starter Performance
Sea Level 59° F Ambient Temp



6-6 Ground Cart Airbleed Performance



6-7 Starting Time - Pneumatic Starter



Starting Time Comparison

Within a five year period, improvement of existing ground equipment will result in a 15 to 20 percent increase in power available from a single ground cart. This increase will further improve the engine starting time.

6.4 RELIABILITY AND SAFETY

The probability of successfully starting all four engines is 99.81 percent, taking into account all components of the system directly related to starting—starters, regulating valves, check valves, and electrical switches. The regulating valve is provided with means to operate the valve mechanically if it should fail to operate electrically; this feature is included in the reliability analysis. A detailed analysis of the starting system reliability is given in Section 9 of this volume.

Boeing has worked closely with manufacturers of starting equipment to improve the level of safety. The starter incorporates a cutout speed (overspeed) switch which will normally cause the starter valve to close in order to complete the starting cycle. However, in case of a malfunction which would allow the starter to overspeed, the RPM will be limited to a value less than that causing blade failure by the aerodynamic design of the starter impeller. If for any reason the blades should separate from the hub, they will be contained within the starter scroll. In addition, a failed hub is contained within the scroll up to the maximum cutout speed. These containment features provide a high level of safety.

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7.0 FUEL SYSTEM (RFP 3.2.9.4)

7.1 Description

Fig. 7-1 shows the system schematic of the principal components of the fuel system that perform engine feed, pressure fueling, dumping, and defueling.

Fuel is stored in four main tanks and two auxiliary tanks, one in each movable wing section. The fuel reserve is equally distributed between the four main tanks. Each main tank feeds directly to its engine; a cross-feed manifold permits fuel to be delivered from any tank to any engine or combination of engines. The auxiliary fuel is fed to the cross-feed manifold at pressure above the "no-flow" value of the main tank pumps. This arrangement provides automatic backup of auxiliary tanks with the main tanks and uninterrupted flow on auxiliary fuel run-out.

A combination of thermal insulation, scheduled fuel usage, and natural oxygen depletion through an open-vent system eliminates coking and the formation of tank deposits. The need for inerting and purging is avoided by locating tanks in cooler portions of the airplane and by placing vent exits to avoid full stagnation temperatures. Transient overshoots to Mach 2.9 will not be hazardous.

The fuel system and its components are designed to operate satisfactorily under all conditions within the operating envelope of the airplane, considering the effects of aerodynamic heating, insulation, and location in the airplane. The system is designed to operate with commercial kerosene at fuel temperatures from −65°F to 170°F in the tanks and up to 250°F at the engine inlet. However, the fuel temperature must not be less than 10°F above its freeze point.

All fuel system components are explosionproof and designed to limit maximum temperatures to a safe level during normal and failure conditions.

7.2 Fuel Management and Center-of-Gravity Control

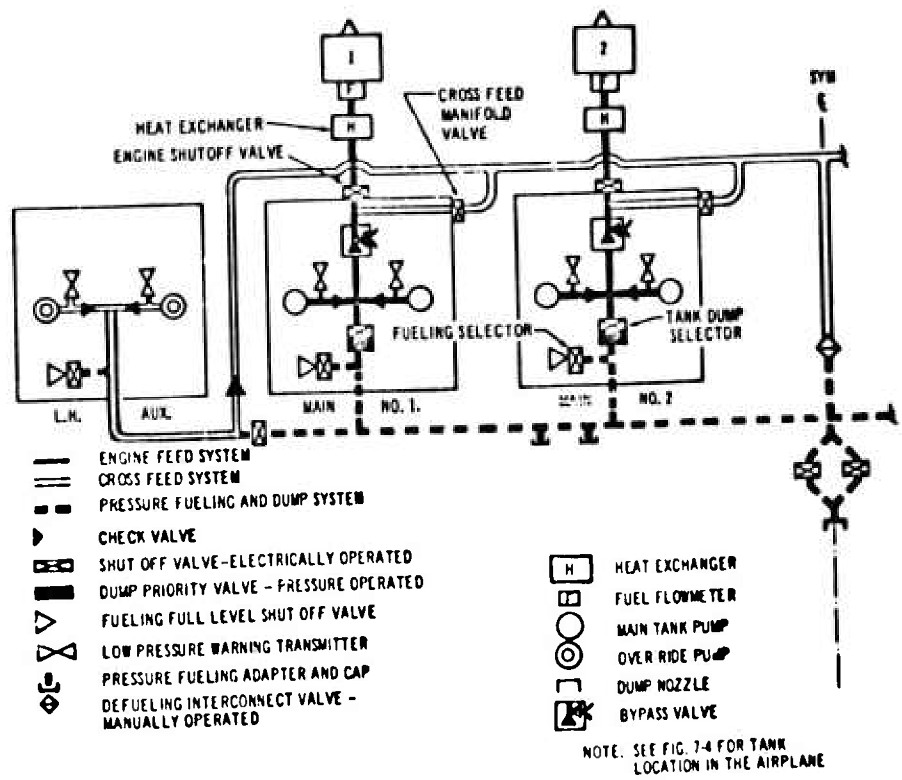
The movable wings in conjunction with the system of fuel management and center-of-gravity control give the Boeing SST configuration the ability to minimize trim drag by maintaining the center of gravity near its aft limit during supersonic flight. The center-of-gravity position is maintained without special fuel management or fuel transfer. Fuel tanks are balanced about the center of gravity and fuel is fed from them in such a way that a minimum amount of attention from the crew is required during normal operation.

7.2.1 FUEL MANAGEMENT (RFP 3.2.9.4a)

The simplicity of the system permits the flight engineer to manually control the system without the use of computers or excessive switching.

The sequence in which fuel is drawn from the tanks is as follows: (1) during takeoff and early climb each main tank feeds fuel directly to its engine, (2) during climb and early cruise the auxiliary tanks feed fuel directly to number 3 and 4 engines, and (3) when the auxiliary tanks are empty, the main tanks feed fuel to engines so that the later cruise, descent, and landing are performed using fuel directly from the main tank to engine.

Because of the higher heating rate in the auxiliary tanks, this fuel is used early in the flight to obtain maximum use of the main fuel supply as a heat sink for cooling airplane systems such as air conditioning, electrical power, and hydraulic systems.



7-1 *Fuel System Schematic*

7.2.2 CENTER OF GRAVITY CONTROL (RFP 3.2.9.4b)

Center of gravity travel of the aircraft caused by fuel usage during some typical missions is shown on Fig. 7-2 and Fig. 7-3.

In the event of a failure of an engine or an extended period of uneven fuel consumption, the center of gravity can be easily controlled by the flight engineer, using the cross-feed manifold and tank gages or fuel-consumed flowmeters to maintain specified ratios of fuel in the tanks, similar to current Model 707 operations.

During dumping operations the rates from each tank are proportioned to provide automatic control of center of gravity within the design limits.

7.3 Fuel Tanks (RFP 3.2.9.4.c)

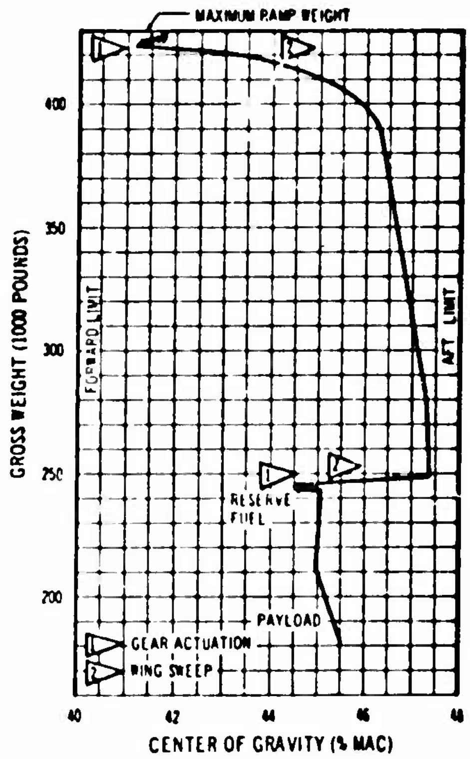
7.3.1 DESCRIPTION

Structural cavities within the airplane body, inner wing, movable wing and wing center section are used to hold fuel, as shown on Fig. 7-4. Fuel capacity will be 235,840 pounds (35,200 U.S. gallons) of commercial aviation kerosene. The tank fuel capacities in U.S. gallons are:

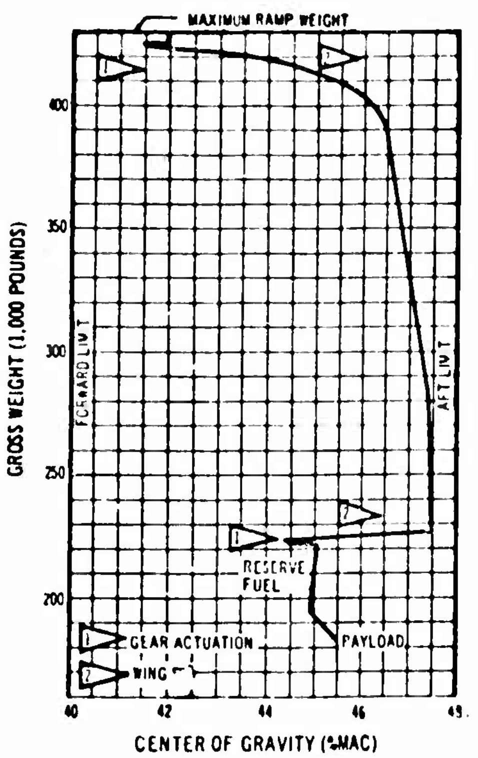
Main No. 1	8800
Main No. 2	8800
Main No. 3	4250
Main No. 4	4250
Left Hand Auxiliary	4550
Right Hand Auxiliary	4550

An expansion space of at least 3 percent of the fuel volume is provided in each tank.

Manual sump drain valves, installed at the low point of each tank, allow removal of water and sediment or complete drainage of the tank. The auxiliary tanks, which will be emptied in normal operations, drain directly to the pumps to minimize puddles which may boil off.



7-2 Center of Gravity Travel 30,000 Lb. Payload



7-3 *Center of Gravity Travel-Full Fuel*

Insulation in the form of air space of 1.25 to 3 inches between the outside skin and the tank surface assists in keeping fuel temperatures within acceptable limits. Estimates of tank temperature are given in Par. 7.10.

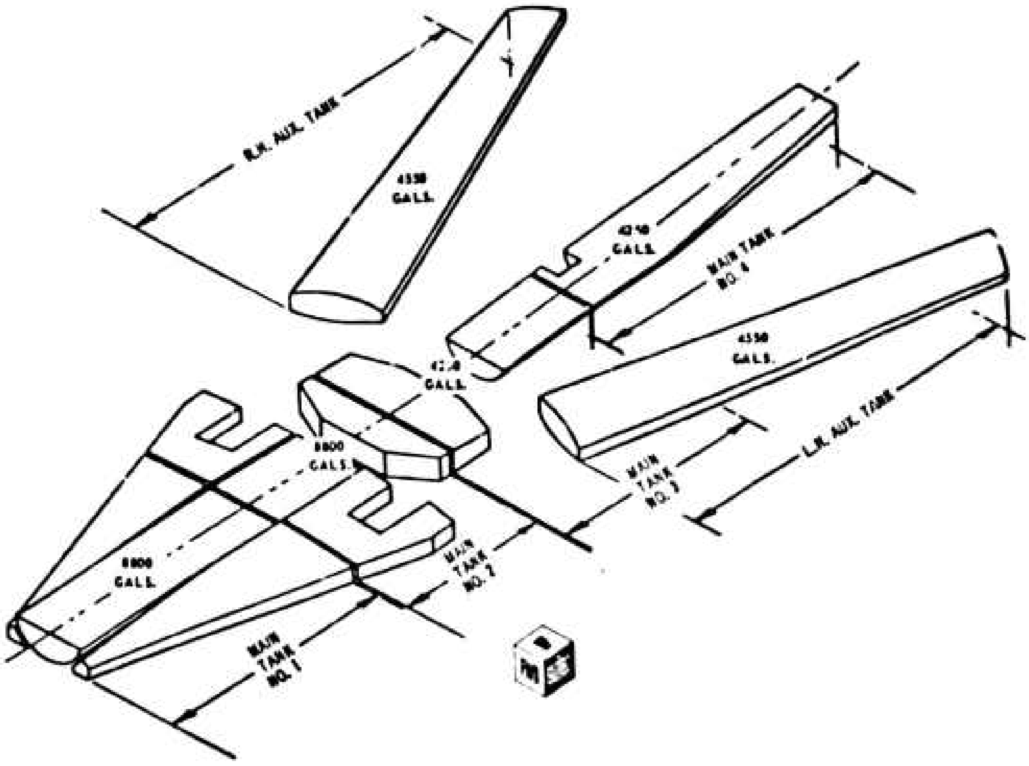
7.3.2 FUEL CELLS AND METHODS OF SEALING (RFP 2.22.2, and 3.2.15.9)

The auxiliary fuel tanks lie between the front and rear wing spars divided by wing ribs. Flow passages and limber holes through the structural ribs allow passage of fuel and air and minimize unusable fuel. A fuel vent surge tank compartment located outboard of the wing fuel compartment avoids external spillage during maneuvers.

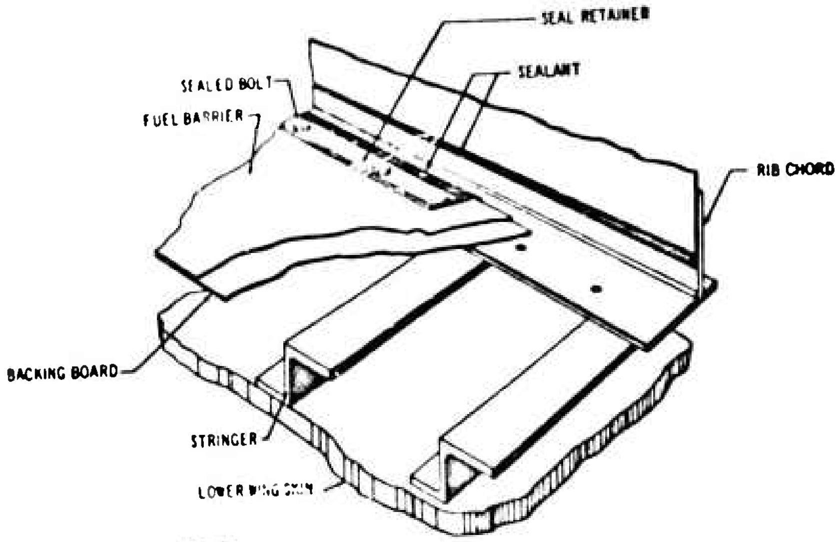
The fuel in the inner wing, center section, and body is divided into four main tanks. Each main tank supplies fuel to one engine under all operating temperatures, attitudes, and flow rates.

Primary structural bulkheads are used to compartment the main tanks into cells. This helps to avoid the undesirable effects of fuel sloshing (center of gravity travel and fuel heads) caused by longitudinal accelerations during flight maneuvers. The fuel tanks are designed to withstand survivable crash loads without rupture. The configuration of the airplane is such that wheels-up landings will not scrape the fuselage surrounding the fuel cells. The contact areas are propulsion pods, main gear pods, and the lower section of the ventral fin or nose. The lower surfaces of the body and inner wing are also designed to avoid rupturing the fuel cells in a water ditching.

The inner wing and center section fuel cells are of semi-bladder integral construction, using lower and upper surface liners sealed to the structure by mechanical seals and integral tank sealants (Fig. 7-5). The liners are teflon-coated. For auxiliary tanks, the same methods of sealing are used on the lower surface. Spars, ribs, and the upper wing panel form the remainder of the fuel barrier. The



7-4 *Fuel Tank Arrangement*

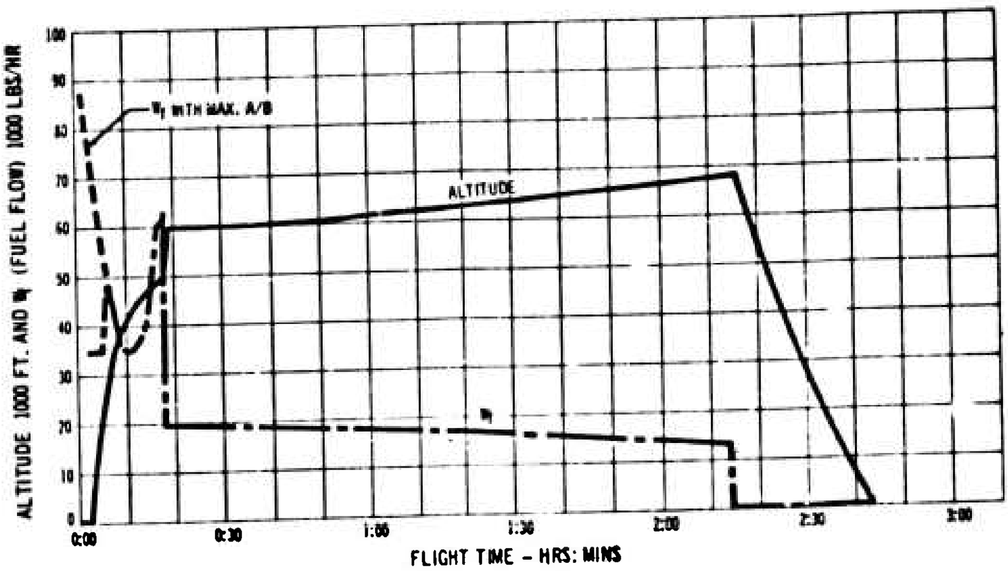


7-5 *Wing Fuel Sealing*

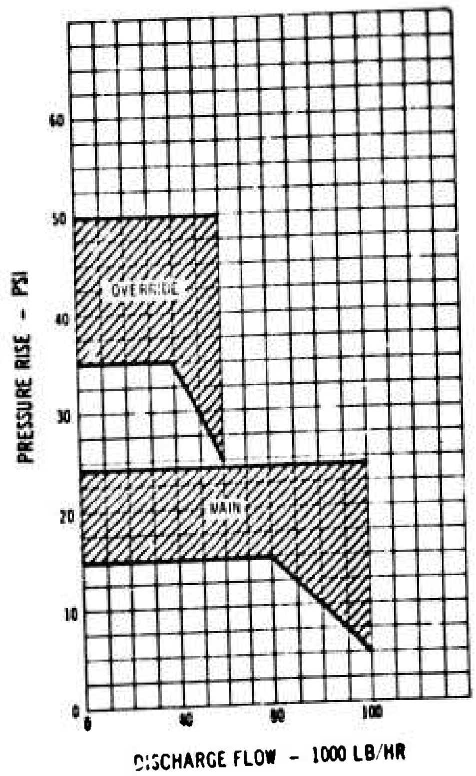
structure is sealed with high temperature sealants. Fittings in the body bladder cells are supported by the structure to mount pumps, valves, probes, interconnects, and access doors.

The bladder assemblies will meet the requirements of MIL-T-25783, "Military Specification, Tanks, Fuel, Aircraft and Missile, Non-Self-Sealing, High Temperature." Boeing's design objective, however, is to extend cell life to 30,000 hours minimum. The bladder assemblies are fully supported for positive pressures with backing board and laced in position to withstand negative pressures. The maximum dry cell wall temperature has been determined to be 250°F.

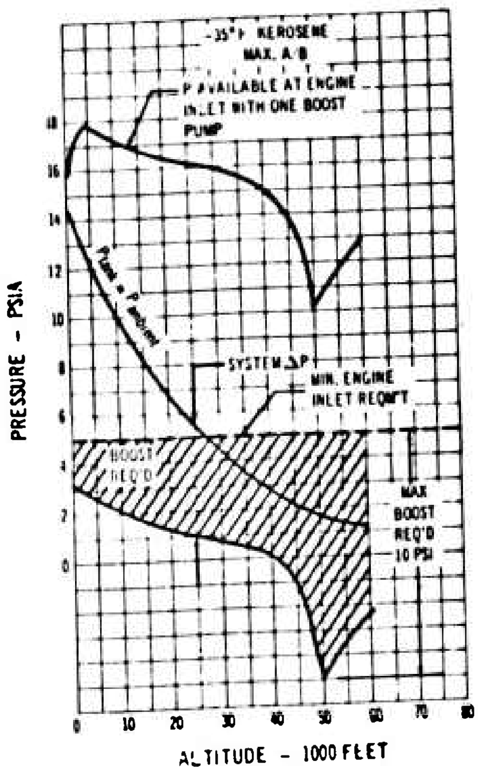
Cavities surrounding the fuel tanks provide compartmentation to enable service personnel to locate the source of leakage and to have maintenance access. Drainage of the cavities is designed to direct leakage to a safe overboard location. All cavities and integral tank structure are suitably protected to avoid corrosion.



7-6 *Single Engine Fuel Demand During Typical Mission*



7-7 *Boost Pump Performance*



7-8 *Main Tank Fuel Feed Performance*

7.3.3 COKE REMOVAL (RPF 3.2.9.4c and 3.2.9.4d)

A combination of thermal insulation, scheduled fuel usage, and natural oxygen depletion prevents the formation of coke in the fuel tanks, thereby eliminating the requirement for coke removal. A complete discussion of this subject is presented in Par. 7.10.

7.4 Engine Feed System

The engine feed system consists of one main tank per engine, connected directly to the engine it serves. A cross-feed manifold permits any tank to feed any engine or combination of engines.

The two wing auxiliary tanks connect to the cross-feed manifold to deliver fuel to the selected engines. Each tank contains two electrically driven centrifugal pumps, each capable of providing the fuel flow and pressure required by the engine. The auxiliary tank pump characteristics are such that they will override the main tank pumps and supply fuel to the selected engines. The main tank pumps serve as a backup during auxiliary tank usage to provide uninterrupted supply of fuel when the auxiliary tanks are depleted. Engine fuel requirements and airplane pump characteristics are shown in Fig. 7-6 and Fig. 7-7. Overall system characteristics using −35° F. kerosene are given in Fig. 7-8. This low temperature causes a maximum pressure drop because of high viscosity.

All boost pumps are readily removable through a boost pump dry bay without draining and entering the tanks (Fig. 7-9).

With all boost pumps inoperative and fuel temperature at 125° F, the pressure at the engine pump inlet will be at least 5 psi above the true vapor pressure at maximum augmented power up to 8000 feet altitude. At maximum dry power flow the inlet pressure will not be less than the true vapor pressure of the fuel throughout the entire operating envelope of the airplane.

Components are arranged with enough redundancy so that a single functional failure will not compromise the fuel system operation.

The system operates on commercial aviation kerosene but is compatible with all commercially available jet fuels.

The fuel feed line for each auxiliary tank also serves as the refuel and dump line. The auxiliary tank fuel line has a high temperature hose running through the center of the wing pivot. Little deflection is required to accommodate wing sweep (Fig. 7-10).

Except for coarse screens at the boost pump inlets there are no filters between the tank and engine fuel inlet in order to avoid blockage from ice or other contaminants. The primary fuel filtration process is accomplished by a filter in the engine fuel system.

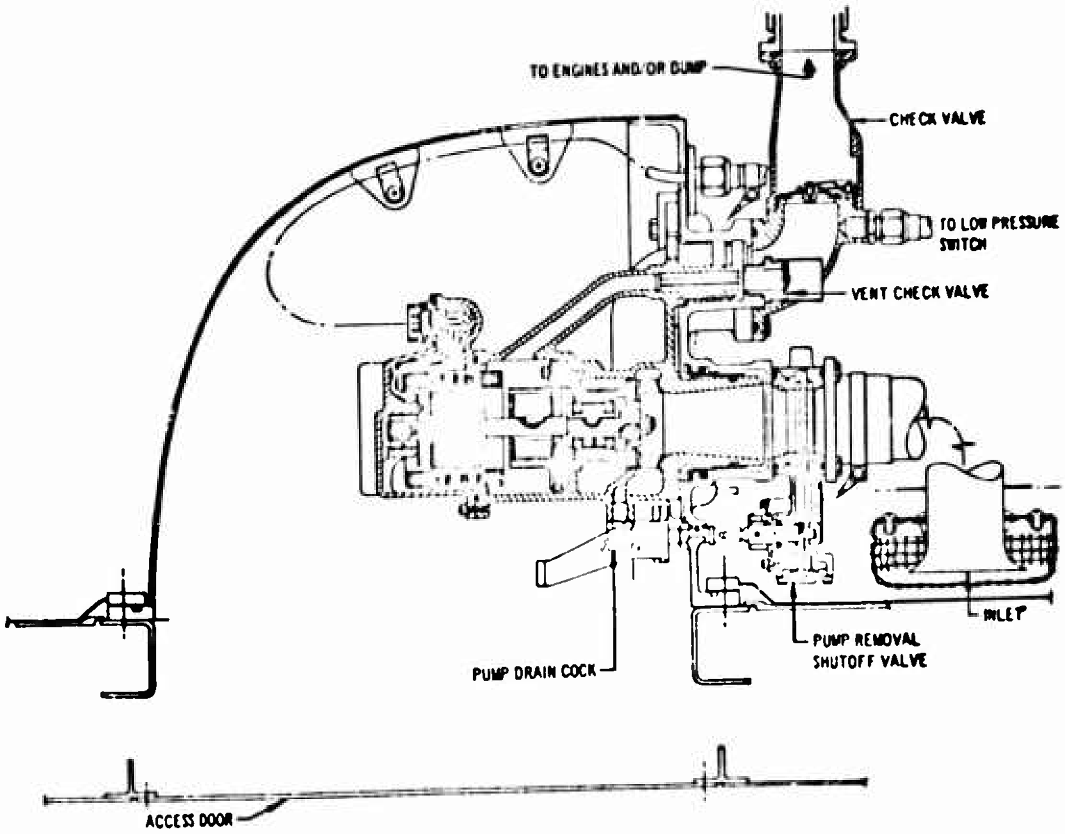
A fuel deicing system is not required because the heat load from air conditioning, hydraulic, and electrical system heat exchangers keeps the fuel at temperatures adequate to prevent icing for all operating conditions.

7.5 Refuel, Defuel, and Dump System

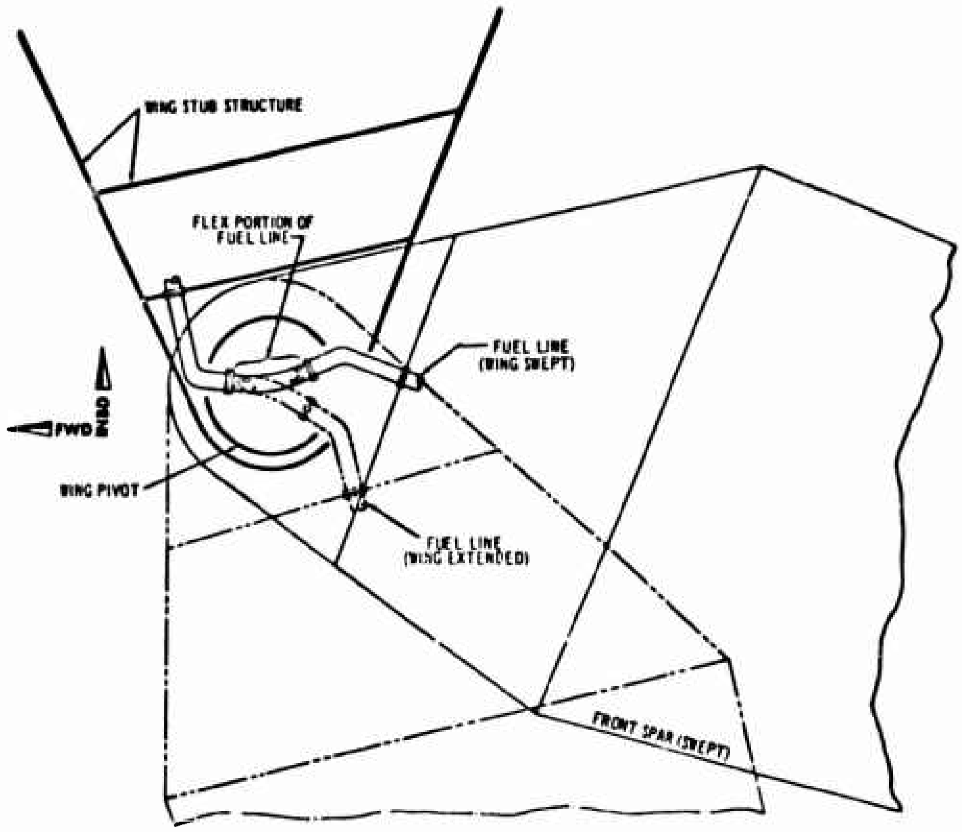
7.5.1 REFUELING

The system delivers a minimum of 1600 gallons per minute with 50 psi at the nozzles. A single manifold, common to the fuel dumping system, is used to service all tanks. A single level control valve, hydraulically operated by a float pilot valve, is located in each tank. Automatic shutoff will occur at the 100 percent fuel volume level. Electric fueling selector valves will control flow to each tank for partial or selective fueling. The refuel switch will also open and close the dump selector valve for the auxiliary tanks. Controlled valve closure rate will prevent any damaging surges. Since the fueling rates are within the limits used on present airplanes, hazardous fuel electrification will be avoided.

An illuminated refueling station is located on each side of the body near the inner wing. The control panel



7-9 Fuel Boost Pump Installation



7-10 *Wing Pivot Fuel Plumbing*

is located in the refueling station on the right hand side only.

Two refueling adapters with caps are installed at each servicing station. These adapters mate with MS29520 nozzles.

The right hand station includes the following equipment: (1) panel with fuel quantity gages and push-to-test system, (2) electric switch for start-and-stop refueling for each tank, and (3) fueling power switch. The refueling station door is designed so that it cannot be closed when any tank selector switch is in the open position.

Special check valves provide line drainage into a main tank to reduce the quantity of unusable fuel.

Protection of tank structure is accomplished by sizing the vents to receive the resultant flow of the refueling system in the event of a level control valve failure. Orifices are installed in the fuel plumbing for each tank to restrict the flow to the design value and to provide balanced rates to each tank for minimum overall fueling time.

7.5.2 DEFUELING

For defueling, the engine feed boost pumps discharge the fuel through the pressure-fueling adapters. Defueling rate is approximately 200 gallons per minute per tank with boost pumps operating. By opening the electric tank dump valves the tanks may be defueled to the dump reserve level.

A manual valve between the engine cross-feed system and the pressure fueling manifold permits complete defueling. This valve is accessible from outside the airplane and is designed so that the access door cannot be closed with the valve in the open position. Fuel may be pumped, or removed by ground equipment suction, down to the unusable volume by opening the manual valve and cross-feed valve for the tank or tanks to be serviced. Fuel may also be transferred between tanks on the ground by use of the defuel and the pressure fueling system.

7.5.3 FUEL DUMPING

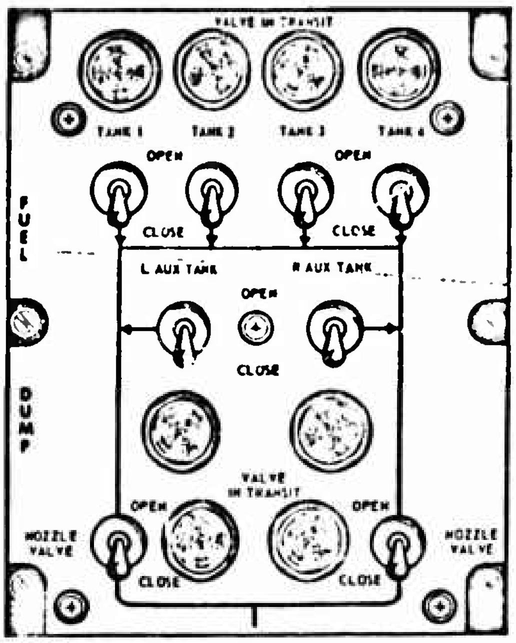
Engine feed boost pumps are used to dump fuel overboard through a fixed nozzle located in the body tail cone of the airplane. A priority valve in each main tank between the feed system and the dump system ensures the required flow of fuel and pressure to the engines under all operating conditions. A "jet-pilot" system for the priority valve automatically shuts off the fuel flow from each main tank before the CAR 4b reserve level is reached. The fuel in the auxiliary tanks may be completely dumped. Dumping from selected tanks is controlled by the flight engineer by use of individual control switches. Two parallel line valves electrically operated and located in the manifold near the aft end of the body ensure dumping capability (Fig. 7-1). The control panel, located at the flight engineer's station, has switches and in-transit lights for each tank and each nozzle valve. The panel has an access door which cannot be closed unless all switches are in the closed position (Fig. 7-11).

The complete system dump rate is 6200 pounds per minute, which is in excess of the 4300 pounds per minute required by CAR 4b.

7.6 Venting (RFP 3.2.9.4d)

The vent system is unpressurized and uses open tank ports and exits. The body tank system is manifolded and uses a single vent exit in the aft body. Each wing tank is vented separately and has its exit on the under surface of the outboard wing. A schematic of a typical body tank vent system is shown in Fig. 7-12.

Since the system requires no inerting, the vent outlets operate at ambient or slightly negative pressure. Evaporation or boil-off is not a problem with commercial kerosene because fuel heating is controlled by insulation and proper sequence of fuel usage. The outlets are also designed to be ice-free. Tank cavities are vented and drained overboard. The source pressures for the cavities are tailored to match, or be below, the tank pressure.



7-11 *Fuel Dump Panel*

A surge tank is located near each outlet to prevent spillage overboard during maneuvers. Fuel collected in the surge tank drains into an adjacent tank. A minimum of three percent air space is provided for each tank.

The vent system is large enough to prevent pressure in any tank from exceeding the structural design limits under the following conditions of operation: (1) failure of a pressure fueling level control valve at the maximum refueling rate, (2) maximum emergency descent with tanks empty, and (3) maximum rates of climb under all operating conditions. Failure of a level control valve is the condition which sizes the vent lines.

7.7 Plumbing and Fittings

All plumbing is located as close to the neutral axis as possible. Where this is not feasible, tubing is designed to accommodate length changes through bends and flexible couplings (axial and angular). Teflon-lined clamps are used because of their long life and to allow tube movement. Where no tube movement is present, rigid couplings, using multi-bolt, swaged tube flanges are used. All tube bracketry is adjustable to facilitate tube installations. All tubing installations are designed to minimize time for replacement. No tubing is welded in place or swaged in the airplane. The major portion of the tubing is routed inside of tanks, as is done on all Boeing jets, in order to minimize external leakage and reduce maintenance. Fig. 7-13 illustrates the typical fuel system fittings.

Tubing within the tanks will be fabricated from aluminum alloy. Fireproof tubing will be used in designated fire zones or where ambient heat would reduce the strength of aluminum. Fire-resistant hose assemblies and shrouds will be used where necessary.

7.8 Instrumentation (RFP 3.2.11.5)

Fuel system instrumentation is located in three places on the airplane: (1) flight engineer's station, (2) pilots' center panel, and (3) fueling station.

7.8.1 FLIGHT ENGINEER'S STATION

Four categories of fuel system instrumentation are used

7-12 *Vent Schematic*

at the flight engineer's station: (1) engine fuel feed, (2) fuel temperature, (3) fuel consumed, and (4) fuel dumping.

The arrangement of the fuel-feed panel simulates the functional arrangement of components being monitored, making it easy to observe and control the system and minimizing the probability of crew error. Similarly, switch action and layout correspond to fuel flow direction. The panel is a straightforward schematic of the operation of the system rather than its physical layout.

On the engine fuel feed panel (Fig. 7-14) six quantity indicators with a push to-test system indicate the fuel quantity in pounds remaining for each main and auxiliary fuel tank. A toggle switch for each boost pump turns the pump on and off, and low pressure lights allow each to be monitored for minimum pressures. Four lock-toggle switches operate the four engine shutoff valves and four rotary switches operate the four cross-feed valves. Each valve switch will have an in-transit light for monitoring valve actuation.

A fuel temperature system of four indicators shows temperature at each engine fuel inlet; a selector switch allows individual fuel tank temperature to be determined.

The panel contains a fuel consumed flowmeter for each engine and a total fuel tank quantity gage.

On the fuel dump panel (Fig. 7-11), fuel tank selector toggle switches for each main and auxiliary tank will open valves for dumping from selected tanks. Toggle switches also control the two dump nozzle valves. The panel has in-transit lights for each valve and markings to show line arrangements. The fuel quantity gage and tank selector valve may be used as a backup method of fuel cutoff.

7.8.2 PILOTS' CENTER PANEL

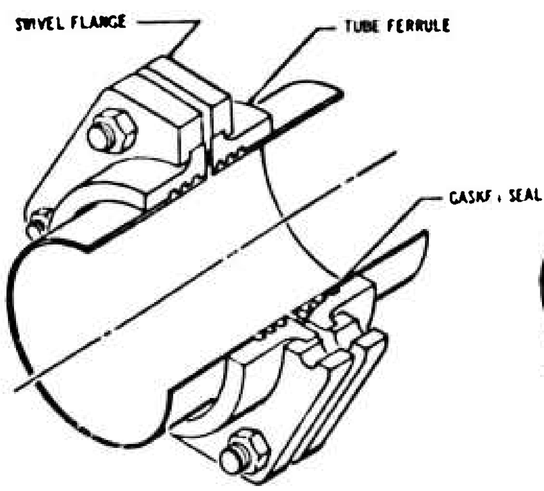
The center panel has four engine fuel flow rate indicators reading mass flow in pounds per hour.

7.8.3 FUELING STATION

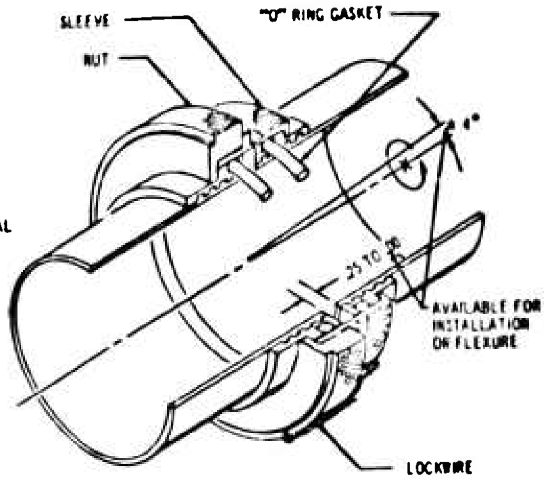
The left hand fueling station has two single point receptacles with caps, two ground jacks, and illumination. The right hand station contains two single point receptacles, control, illumination, and fuel quantity gages as shown in Fig. 7-15. There are six quantity gages, one for each tank, six shutoff valve switches, and six intransit lights. There is a fuel tank quantity gage test switch and a power switch for the gages and the station light. Drip sticks are also installed in each tank to provide a supplementary means of checking fuel quantity.

7.9 Inerting (RFP 3.2.9.4d)

Inerting of fuel tank and cavity areas is not required. Extensive test data have been accumulated which show that the selected system configuration, cruise speed, and altitude eliminate the need for inerting.

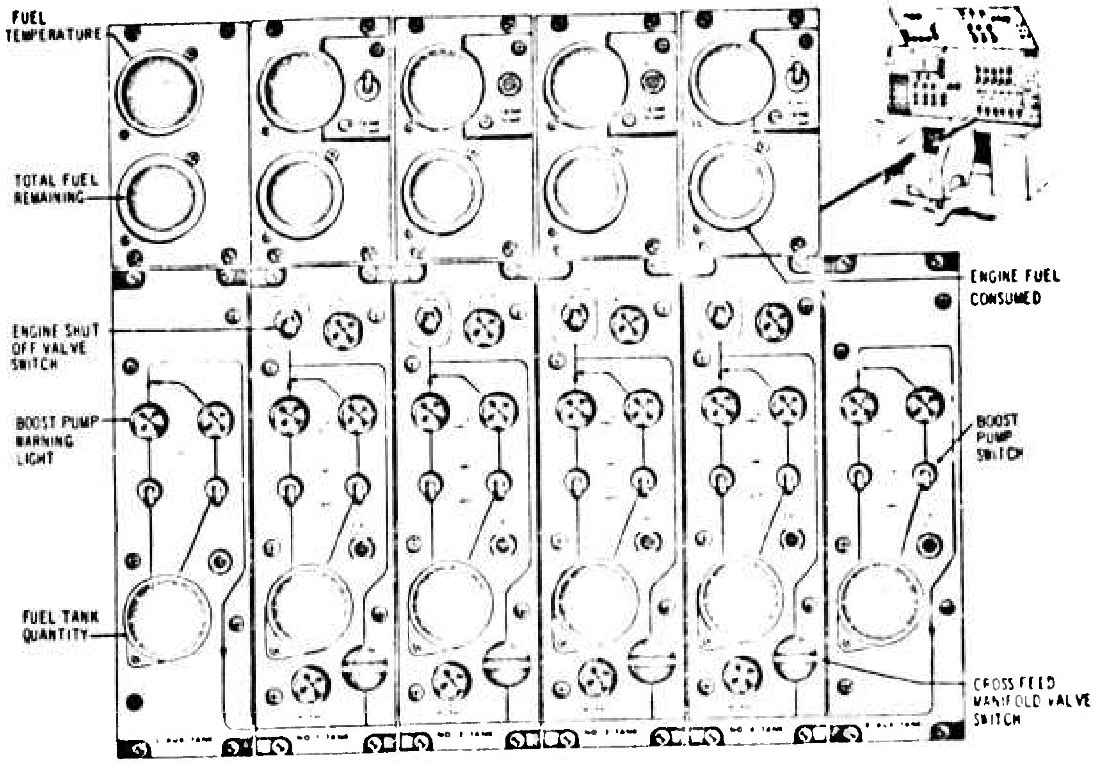


BOLTED FLANGE FITTING

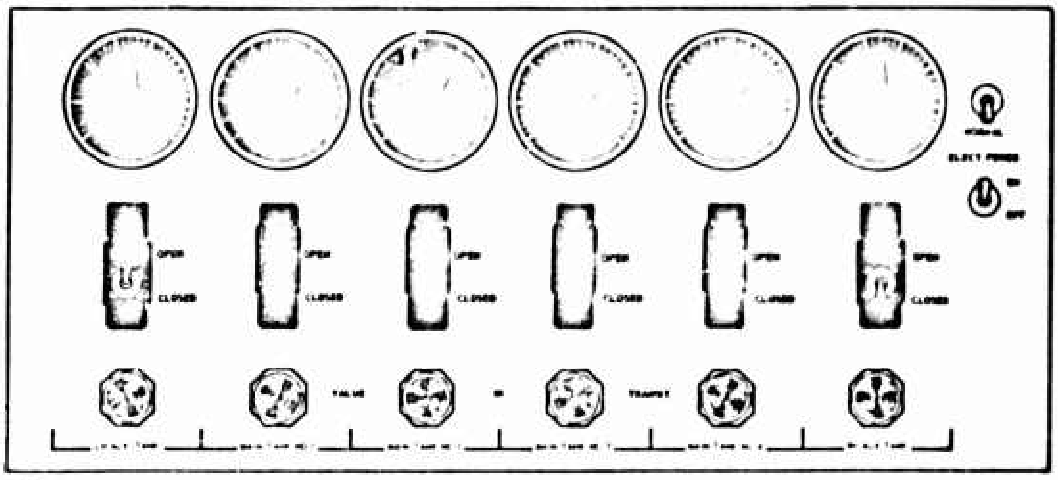


FLEXIBLE FITTING

7-13 Fuel Tube Fittings



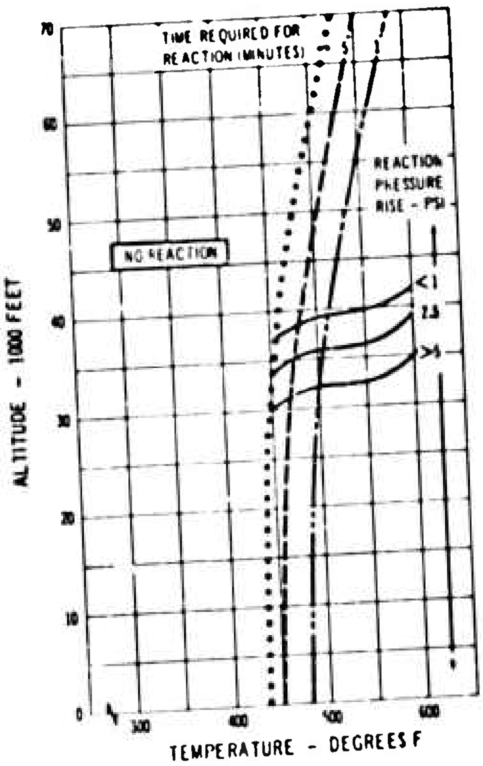
7-14 Engine Fuel Feed Panel



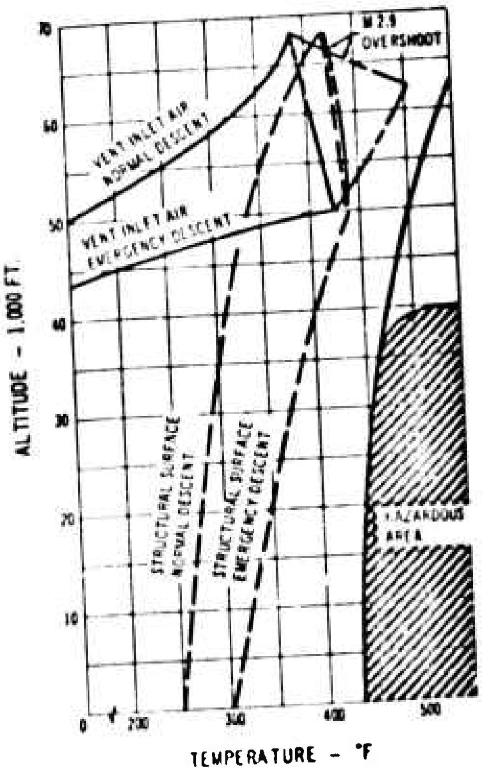
7-15 *Fueling Station Control Panel*

The relationship of autoignition temperatures of fuel-air mixtures to pressure altitude, reaction pressure, and residence time at these temperatures is shown in Fig. 7-16. This plot is based upon the most conservative data from more than 20,000 separate tests conducted by Boeing. With an ambient vent, increasing altitude decreases the pressure of the fuel-air mixture and hence increases the temperature at which the system will operate with no re tion.

Computations of surface and vapor temperatures of tanks and other cavities where combustible vapors may exist show that the temperatures will always be below hazardous levels for airplane operation within the proposed envelope, including transient overshoots to Mach 2.9. Fig. 7-17 shows the computed temperatures for Mach 2.7 cruise and for both normal and emergency descents. The vapor temperature shown is that of the air entering a wing tank vent. Vent air is obtained from the boundary layer adjacent to the skin and thus is close to skin temperatures. The structural temperature shown is of the hottest spot within the vapor regions. The hottest surface areas during descent will initially be those near the skin and then will transfer to the midpoint of structural members, such as the front spar, in the latter stages of descent.



7.16 Autoignition Reaction Zones



7.17 Tank Surface and Vent Air Temperatures During Descent

Testing of venting during descent is currently under way at Boeing. Hot air, programmed at 50°F higher than calculated temperature for conservatism and flow rates to simulate emergency descent from Mach 2.7, has been introduced into a heated fuel tank after a simulated mission. For further conservatism, the tank is maintained at cruise temperature during descent. Tank pressures are programmed to simulate an open vent system. No reaction has occurred in these runs.

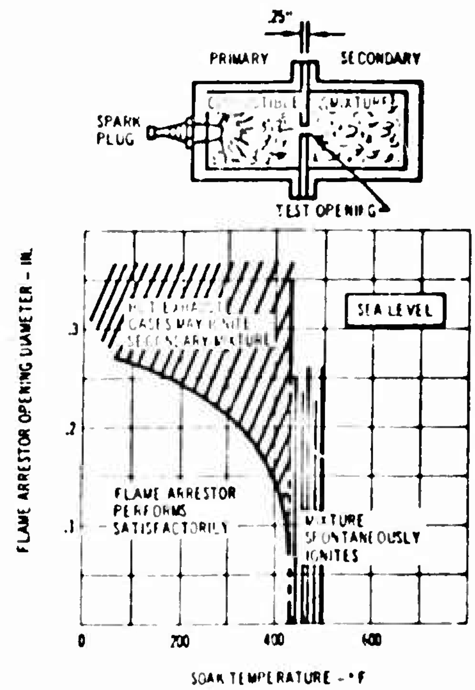
7.9.1 EXPLOSION PROOFING (RFP 2.22.1, 3.2.15.4)

Although the ambient temperatures are higher in supersonic flight, present day explosion proofing techniques are adequate and inerting will not be required for this purpose. The most critical condition occurs in the design of boost pumps where passages connect the motor compartment and tank for cooling and lubrication. Fig. 7-18 shows the results of a Boeing test series to determine the adequacy of flame arresters at elevated temperatures and sea level pressure. As shown, flame arrestor sizes may be selected which will prohibit transfer of an explosion, up to the spontaneous ignition temperature of the fuel vapors (approximately 430°F at sea level). As previously shown on Fig. 7-17, vapor and surface temperatures in the fuel equipment area always remain below the spontaneous ignition temperature for all conditions of operation. Thermal protective devices are incorporated on equipment where surface temperatures may exceed spontaneous ignition levels due to a malfunction.

7.10 Fuel Characteristics (RFP 2.17, 3.2.9.4e)

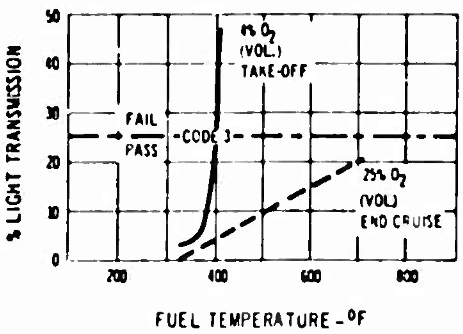
The fuel system is compatible with commercial aviation kerosene, jet fuels, and forecast improvements in kerosenes.

The engines proposed for the SST program require fuel of slightly greater thermal stability than that of some commercial aviation kerosenes. Advantage is taken

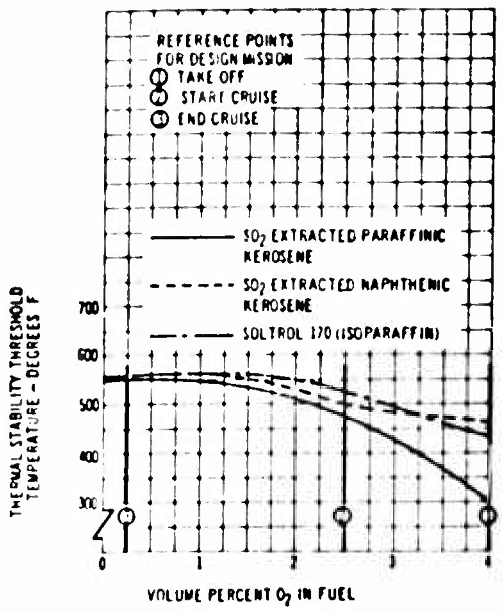


7-18 Effect of Temperature on Flame Arrestor Hole Size of this need for the better commercial kerosenes to make use of the fuel as a heat sink for the aircraft systems. Certain U.S. refineries are presently delivering the higher stability kerosene at no increase in cost. It is reasonable to expect that this fuel can be generally available before the SST operational period.

Test work has shown that a reduction in the oxygen content of the fuel effectively reduces the amount of deposits collected on screens or plated out on heat exchanger tubes. Information from The Phillips Petroleum Company and The Texaco Company on the correlation of oxygen content with thermal stability is plotted in Fig. 7-19 and Fig. 7-20. With an open vent system which maintains the vapor space pressure at ambient, the oxygen in the fuel will be removed by the decrease in vent pressure during climb as shown in Fig. 7-21. A comparison of these figures shows that the fuel oxygen

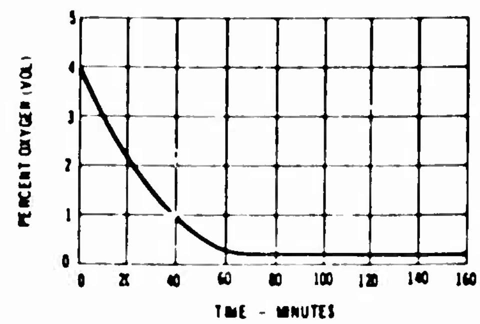


7-19 *Effect of Dissolved Oxygen on Fuel Thermal Stability*

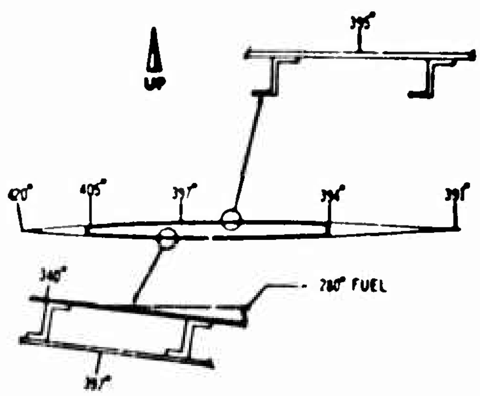


7-20 *Threshold Temperature at Various Oxygen Concentrations*

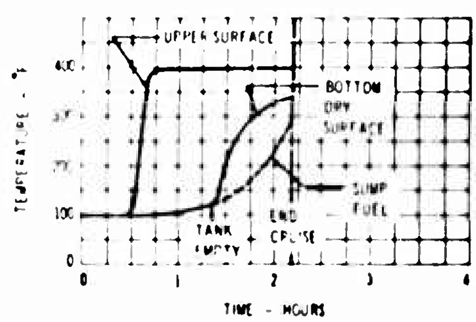
content has been sufficiently reduced early in cruise to increase the thermal stability level by approximately 250 F. This factor, in conjunction with improved fuel thermal stability, ensures minimum engine maintenance caused by thermal degradation of fuel.



7-21 *Fuel Oxygen Content During Flight*



WING TANK CROSS-SECTION AT END CRUISE



7-22 *Wing Tank Temperatures in Flight*

7.10.1 COKE PREVENTION

Boeing tests (Ref. 4) have shown that the formation of coke in the fuel tanks can be prevented by a combination of thermal protection, tank coating material, and fuel management. In accordance with these results, double-walled tanks with teflon internal bottom coatings are used where required. The fuel management procedure uses the total movable wing tank fuel approximately one hour before the end of cruise. The main tanks in the body retain the reserve and fuel for descent and end of cruise. With this protection and managment, the resulting fuel temperatures at the end of a maximum range cruise at Mach 2.7 are as shown in Fig. 7-22 for the wing tanks and in Fig. 7-23 for the main tanks.

7.11 System Thermal Characteristics (RFP 3.2.9.4e)

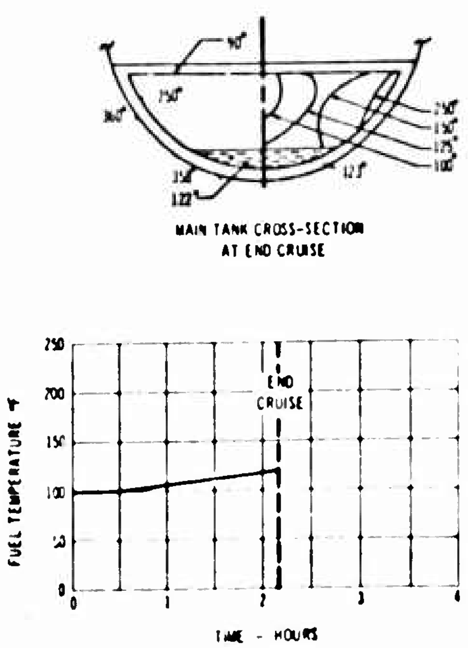
Insulation and fuel management procedures allow the fuel to provide a major portion of the heat sink capability required aboard the airplane within the 250 F cruise limit for fuel delivery to the engine.

Typical underground fuel storage temperature was determined to be approximatelly 50 F. Fig. 7-24 shows a history of fuel temperature if 50 F fuel were loaded. A small amount of auxiliary cooling is required during this mission.

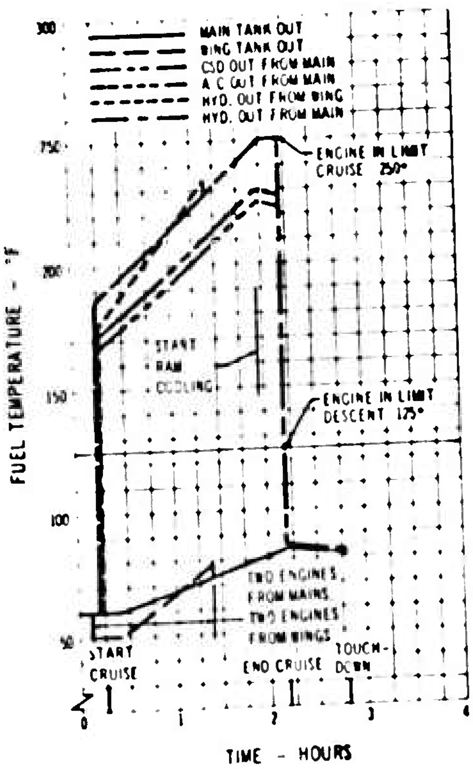
Fig. 7-25, a history of the fuel temperature during a maximum range supersonic mission with 95 F fuel loaded, shows the temperatures obtained through the various heat exchangers and the requirement for auxiliary cooling. The 95 fuel loading temperature is considered the maximum to be expected in commercial operations.

If 110 F fuel is loaded and a maximum mission is flown, the resulting fuel temperatures will be as shown on Fig. 7-26.

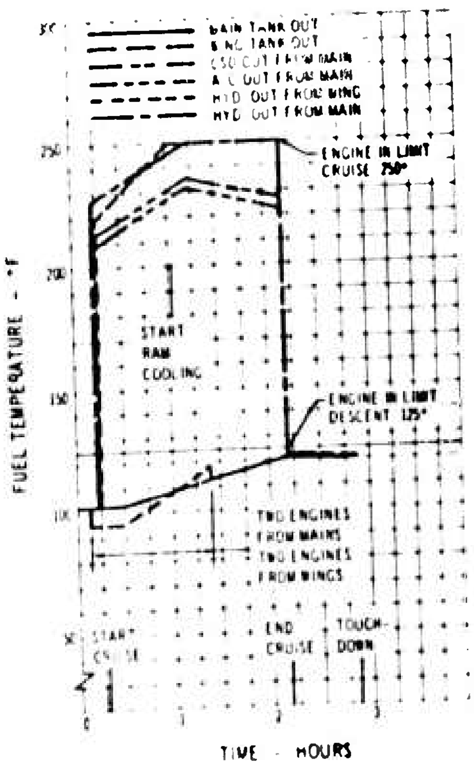
During descent the fuel temperature into the engine is limited to 125 degrees F. to provide adequate cooling for the engine. This is accomplished by removing all cabin conditioning and hydraulic heat loads from the fuel. This arrangement has the effect of allowing the fuel to be delivered to the engine at tank temperature. (See Volume A-VII for a description of systems cooling).



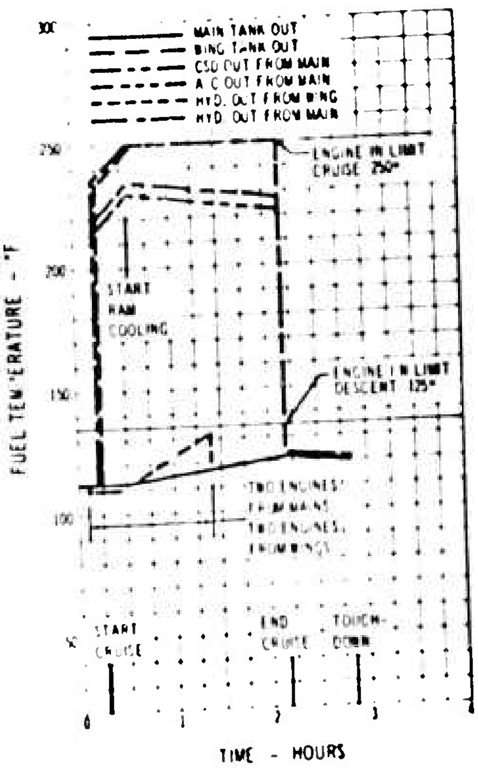
7-23 Main Tank Fuel Temp. in Flight



7-24 Temperature Profile for Typical Flight 50 °F Fuel Loaded



7-25 Temperature Profile for Typical Flight 95 °F Fuel Loaded



7-28 *Temperature Profile for Typical Flight, 110° F Fuel Loaded*

VOLUME A-VI

PROPULSION

8.0 TESTING AND DEVELOPMENT PROGRAM (RFP 3.2.18, 3.2.19)

8.1 General

This section summarizes the testing and development of the major propulsion system components through each of the following stages:

- Model tests
- Component tests
- Development tests
- Component qualification
- System development and qualification
- Preflight
- Certification
- Acceptance

Information on test facilities and scheduling is given in Volume A-IX, Test and Certification Plan.

8.2 Engine Installation

Development testing will be done to substantiate the design details of the overall engine installation. This test program includes simple laboratory tests of various components, static ground tests of a complete engine installation, tests on the completed airplane, and flight tests which conclude with the airplane's certification and customer acceptance tests.

8.2.1 COMPONENT DEVELOPMENT TESTS

The design concepts of a large number of both functional and structural components used in the installation will be tested to develop and prove that the design details are sound and practical. These tests will be made at the earliest possible time at Boeing facilities and also at facilities of the vendors and subcontractors participating in the development of the propulsion section of the airplane.

An objective of these tests is to ensure complete compatibility between items tested and the very high altitudes and extremes in temperatures associated with supersonic operation.

8.2.2 ENGINE GROUND RIG TESTS

Initial ground tests of engines will be performed on two Boeing ground rigs in the Seattle area approximately 12 months before first flight. The engines will be of the same type and model as the flight engines. One engine rig is located at the mechanical engineering laboratory in the Seattle area and the other rig is at the Tulalip remote test site.

The complete propulsion pod for the airplane, including inlet, engine with augmentor and nozzle, engine mounts, accessories, engine fuel system, cowling, reverser, and strut will be tested in these facilities to confirm the propulsion system design prior to first flight and to support the flight test program.

The test rigs will be used to perform developmental testing of components and accessories and to evaluate engine performance and operation. Items to be tested include the inlet, starter, oil system, fuel system components, engine instrumentation, reverser, nozzle, and controls. Measurements will be made of inlet pressure recovery and distortion, accessory cooling, engine environment, fuel and oil flow rates, thrust, internal temperatures and pressures, starting and acceleration times, etc. Engine operation in both the augmented and dry modes will be fully tested. A portion of the airplane fuel feed system will be used to supply fuel to the test engine.

The following engine installation tests are planned:

- ENGINE OIL COOLING

Tests to evaluate engine oil cooling system performance (temperatures, pressures, and flows), both steady state and transient, with hot fuel and normal fuel temperatures.

- CONSTANT SPEED DRIVE AND GENERATOR OIL COOLING

Tests to evaluate system oil cooling performance (temperatures, pressures, and flows), both steady state and ransient, with hot fuel and normal temperature fuel.

Tests to evaluate drive pressurization system performance (engine bleed air).

Tests to obtain generalized heat exchanger performance.

• ENGINE AND COMPONENT COOLING

Tests to determine component temperatures in the engine environment (ignition system, fuel control, fuel pump, dump valve, nozzle control, bleed valves, etc.), and to evaluate pod temperature environment.

Tests to determine accessory gearbox and drive system environmental temperature as well as the ambient temperatures for constant speed drive, generator, starter, hydraulic pump, transducers, transmitters, etc.

• FIRE DETECTION

Tests to establish fire detector locations and temperatures.

• ENGINE INSTRUMENTS

Tests to evaluate thrust measuring system accuracy and response; to evaluate other engine instrument systems such as RPM, fuel flow, exhaust gas temperature, oil temperature, oil pressure and oil quantity, for response and accuracy, and to establish limits as required.

Tests to evaluate instrument systems for constant speed drive and generator oil cooling systems.

• ENGINE CONTROL SYSTEM (RFP 2.25.1a)

Tests to evaluate engine control system response to determine engine performance as a function of thrust lever position, to determine engine response to thrust lever movement, during both acceleration and deceleration.

• ENGINE FUEL SYSTEM

Tests to evaluate engine performance, both with and without assistance from boost pumps; to determine system pressures, temperatures, and flow during steady state and transient operations, to evaluate surge pressures, and to develop generalized performance for fuel oil heat exchanger.

• POD DRAINAGE

Test to evaluate propulsion pod drainage provisions.

• MISCELLANEOUS (RFP 3.2.16, 3.2.4.7d)

Near field and far field acoustic measurements will be made during the test period. Exhaust gas velocity and temperature profiles in the region of the airplane fuselage, wing, and tail section will be obtained.

The test rig will be in operation until propulsion installation problems are solved, a period of approximately 30 months.

8.2.3 AIRPLANE GROUND TESTS

Ground tests will be run on the airplane prior to first flight to confirm that the various systems are working properly and are satisfactory for flight.

All engines will be operated during the preflight tests to evaluate the following functions:

- Engine starting (see Par. 8.5)
- Engine acceleration
- Augmentor and nozzle operation
- Engine instrumentation including the thrust measuring system
- Flight deck controls
- Engine case temperatures
- Pod cooling
- Engine oil system
- Reverser operation (see Par. 8.4)

8.2.4 AIRPLANE FLIGHT TESTS (RFP 2.25.1)

The tests outlined in this section are those required to evaluate the efficiency and confirm the design of the engine installation during the flight tests.

In flight tests will measure engine performance and operational characteristics, fuel flow, airflow, exhaust gas temperature, pressures, nozzle area, air bleed temperatures, and pressures.

The air start envelope for the engine will be established for both the main gas generator and the augmentor. The following operational items will be evaluated during the flight test program:

- Engine performance characteristics
- Engine accelerations and decelerations
- Engine thrust lever arrangement
- Augmentor operation
- Propulsion pod and engine cooling
- Engine oil system breathing, scavenging, oil consumption, oil cooling, heat rejection, etc.
- Compressor surge characteristics
- Vibration surveys
- Accessory operation
- Engine anti-icing
- Engine instruments
- Pod drainage
- Engine fuel system

8.2.5 CERTIFICATION TESTS

The propulsion pod will be tested to obtain data for certification to the maximum capabilities of the engine installation within the flight envelope of the aircraft.

Tests listed below are typical of those to be performed to ensure compliance with the indicated sections of CAR 4b.

- INFLIGHT RESTARTS (CAR 4b.742)

The envelope of airspeeds and altitudes within which satisfactory engine relights can be obtained will be demonstrated.

- ENGINE ACCESSORY COOLING (CAR 4b.450 THROUGH 4b.452, 4b.606, AND 4b.625)

It will be demonstrated that the engine accessory cooling system provides the cooling necessary to maintain the components within established limits and that component environment is acceptable throughout the flight envelope. Probable failures will be simulated.

- ENGINE INSTRUMENTS (CAR 4b.613, 4b.730, 4b.731, 4b.734 THROUGH 4b.736)

The engine instruments will be demonstrated to serve their intended functions throughout the flight envelope and under conditions of daylight and night operation. Instrument locations and markings will be in accordance with Part 4b.

- ENGINE OIL SYSTEM (CAR 4b.450 THROUGH 4b.452)

The primary responsibility for the engine oil system lies with the engine manufacturer.

It will be demonstrated that Boeing's installation is compatible with the engine oil system, and that secondary systems, such as oil pressure, temperature, and quantity indicating systems, serve their intended function.

- ENGINE ANTI-ICING (CAR 4b.461)

Ice protection system design for the basic engine is an engine manufacturer responsibility.

Flight tests will be conducted to demonstrate that adequate icing protection is available for both the engine and inlet during subsonic operation. These tests will consist primarily of dry air tests where surface temperatures are measured during flight conditions appropriate to the stage of flight.

- ENGINE OPERATING CHARACTERISTICS (CAR 4b.400 and 4T.139)

The response of all engines to thrust lever movement will be demonstrated throughout the flight envelope. These tests will be conducted with no airbleed or power extraction and with maximum airbleed and power extraction.

Satisfactory engine operation will be demonstrated with the airplane in sideslips, in stalls, during maximum-rate turns at high altitude, during go around (refused landing) conditions, during airplane acceleration from minimum speed, at maximum operating speeds, and at maximum dive speeds.

Engine operation on the ground will be demonstrated to be satisfactory during crosswind and tailwind conditions (winds not to exceed maximum velocities to be certified for takeoff and landing. 90° cross wind, 30 knots minimum).

• FIRE PROTECTION (CAR 4b.484, 4b.485, and 4b.489)

Fire isolation will be demonstrated on a model in the wind tunnel.

Proper settings for fire warning system will be verified during the engine and accessory cooling tests.

• FIRE TEST PLAN (RFP 3.2.9.1e)

Proof of the propulsion system fire integrity is accomplished by separate test of the applicable major components. Fire testing of a complete propulsion pod is not proposed since the pod employs the design knowledge gained from subsonic pod testing and service experience. It would also be impractical to ground test a pod at supersonic speeds. The primary fire integrity principles used throughout the design are:

- Cowling and firewalls prevent flame impingement on critical portions of the airframe. Non hazardous flame paths are proven by wind tunnel model tests.
- Limiting oxygen availability by controlled ventilation such that fires will be self-extinguishing or of low intensity.
- Burn out panels are installed in strategic locations to allow "burn out" of high intensity fires if they occur.

8.2.6 ACCEPTANCE TESTS

Performance tests will be accomplished to demonstrate that the engine installation meets all required performance characteristics as set forth in the airplane detail specification (Volume A-II).

8.3 Engine Inlet

Inlet development testing, starting with models and proceeding through full-scale wind tunnel tests, (preferably with operating engines), and actual airplane tests, is planned to define the design details of the inlet and its control system.

8.3.1 MODEL TESTS

Model tests of the inlet and the inlet propulsion pod-wing combination will be conducted as follows.

8.3.1.1 Small-Scale Model

Small-scale inlet tests will be conducted in Boeing wind tunnels to obtain design data to define the inlet internal and external geometry, to define the inlet control parameters and sensor locations, to determine optimum movable geometry schedules, and to establish the suitability of the inlet location relative to the airplane wing and fuselage. The inlet models will be tested over a Mach number range of 0.2 to 3.0 at various angles of attack and yaw. The inlet internal geometry, bleed requirements, and control requirements will be established, primarily through development tests of the inlet alone. Periodic tests of the wing inlet combination to verify satisfactory inlet operation over the entire flight envelope will also be conducted. Inlet drag tests (cowl, spillage, and bypass) will be conducted in the large wind tunnels to establish external lines and bypass door control schedules. Several small scale models will be built for the development work.

This effort will extend from program go-ahead through the flight test phase until full development has been accomplished.

8.3.1.2 One-Fifth Scale Model

A one fifth-scale model of the inlet will be built and tested over the entire flight envelope to provide additional development data, with particular emphasis on scale effects, boundary layer bleed details, and inlet control development. The Boeing transonic and supersonic wind tunnels and one of the NASA tunnels will be used for this work. The purpose of this testing will be to provide inlet performance and stability and inlet control design data to be used for engineering drawing release. The model will be fully controllable, with movable inlet geometry, variable takeoff bypass stability system, and an automatic inlet control. These tests will be run in conjunction with the inlet control subcontractor to evaluate control concepts and sensor requirements.

8.3.1.3 Small-Scale Static and Low-Speed Testing

Small-scale (approximately one-eighth) static inlet models will be used to develop the takeoff door configuration. The takeoff doors provide auxiliary air to the engine during takeoff and during low-speed high-power operation. These models will be tested in the Boeing mechanical engineering laboratory low speed (up to Mach .2) wind tunnel and will provide the data necessary for detail design of that portion of the inlet system. Measurements will be made of airflow, pressure recovery, and distortion at the engine face over a Mach range of 0 to 0.2. Auxiliary door area shape and location will be some of the variables tested. An engineering laboratory water table will also be used for evaluation of the auxiliary air system.

8.3.2 Full-Scale Tests

8.3.2.1 NASA Lewis Laboratory and Arnold Engineering Development Center

A full-scale inlet will be tested at the NASA Lewis Laboratory or equivalent facility to determine the effect of model scale on the inlet performance and stability. The inlet boundary layer bleed configuration will be tailored during this test phase. The inlet control sensor type and location will be established. A prototype inlet control will be used to demonstrate the operation of the inlet control system. Pressure recovery, stability margin, compressor face distortion, bleed flow rates and pressures, time constants, and response rates of the control system will be some of the characteristics measured. Mach range will be from 1.6 to 3.0 or as limited by the test facility. The full-scale inlet fabricated for the Lewis Laboratory tests will be first bench-tested at the Boeing mechanical engineering laboratory to confirm the variable geometry actuation and control operation.

If incompatibility between the inlet and the engine becomes apparent, further full-scale tests are planned at the Arnold Engineering Development Center.

8.3.2.2 Qualification Testing

Qualification tests of the engine inlet will prove structural and mechanical integrity of the inlet design. The inlet structure will be fatigue tested in the structural dynamics laboratory. In-flight temperatures, pressure loads, and vibrations will be simulated.

Mechanical tests of the takeoff doors, bypass doors, balance panels, internal variable geometry elements and actuators will be performed in the mechanical engineering laboratory. For these tests the components will be subjected to simulated in-flight temperatures and loads.

8.3.3 GROUND TEST RIG AND AIRPLANE GROUND TESTS

Further inlet ground tests will continue as part of the engine rig and airplane preflight test program.

8.3.4 AIRPLANE DEVELOPMENTAL FLIGHT TESTS

Engine inlet performance will be evaluated throughout the airplane operating envelope. Total pressure distribution across the engine inlet plane will be determined, and the inlet control system will be evaluated for proper scheduling, adequate response, and adequate flow stability. Operation during an adjacent engine shutdown and during reverse thrust conditions will be demonstrated. The effects of critical, single failures in the inlet control system will be demonstrated.

8.3.5 CERTIFICATION TESTS (CAR 4b.450)

The engine inlets will be demonstrated to supply the required quantities of air, within the limits of pressure, temperature, and velocity distribution specified by the engine manufacturer for proper engine operation during all normal flight conditions.

The effects of failures of the inlet actuating and control systems will be demonstrated.

For supersonic flight, it will be demonstrated that inlet airflow is maintained to a degree satisfactory for safe flight. It will be further demonstrated that for abnormal conditions where flow disruption occurs, normal inlet airflow can be re-established.

8.3.6 ACCEPTANCE TESTS

Performance and operational tests will demonstrate that the inlet installation meets all requirements as set forth in the airplane detail specification (Volume A-II).

8.4 Exhaust System

Development testing will be performed to establish and substantiate the design details of the exhaust system. Because the nozzle with its integrated thrust reverser must satisfy the requirements of the airframe manufacturer as well as the engine manufacturer, the design and development program must be closely coordinated. This test program includes the listing of the responsibilities of the airframe manufacturer and the engine manufacturer, the laboratory and component development tests, the static ground tests of complete assemblies, the acoustic tests to evaluate engine noise, the exhaust flow field measurements, the tests on the airplane, and the flight tests, including the airplane certification and customer acceptance tests.

8.4.1 AIRFRAME MANUFACTURER'S RESPONSIBILITIES

The airframe manufacturer will establish the requirements for, and evaluate the effects of, reverser operation in all operating regimes of the airplane. The major items for consideration are: (1) effect on airplane performance characteristics; (2) exhaust ingestion; and (3) exhaust gas structural impingement flow patterns as to temperature, pressure, and induced vibration frequencies.

The Boeing Company will be responsible for:

- Defining the exhaust system and reverser requirements – this includes control of external contour and all performance and operational requirements.
- Monitoring the exhaust system development program to ensure that the system is compatible with the airplane.
- Integrating the exhaust system into the airplane.
- Approving the engine manufacturer's exhaust and reverser control system design to ensure its compatibility with the airplane control system.
- Demonstrating the performance of the exhaust system by flight test.
- Coordinating with applicable governmental agencies.

8.4.2 ENGINE MANUFACTURER'S RESPONSIBILITIES

The engine manufacturer will design the exhaust system and determine and evaluate the effects of exhaust nozzle and reverser operation on the engine performance, in accordance with the flight profile performance requirements established by the airframe manufacturer. Additional exhaust system considerations include structural integrity, reliability, maintainability, and economy.

The engine manufacturer will be responsible for:

- Delivering an exhaust system of maximum propulsive efficiency consistent with reliability, maintainability, and engine performance guarantees.
- Establishing the exhaust system performance in all modes. Capability will be demonstrated in small-scale tests in the engine manufacturer's nozzle test facilities. Full-scale static ng tests will be conducted to demonstrate reverse thrust capability, exhaust gas flow characteristics, and acceptable noise level.

- Developing full-scale hardware. This will consist of full-scale tests to evaluate the structural and mechanical integrity of the exhaust system.
- Designing and fabricating the flight hardware. This includes preparation of detail designs, drawings, etc., and manufacture of the flight units.
- Conducting type certification tests of the production exhaust system.
- Obtaining Boeing approval of the nozzle and reverser control system.
- Coordinating with the airplane manufacturer.
- Coordinating with applicable governmental agencies.

8.4.3 LABORATORY TESTS

Models of the engine exhaust nozzle will be tested at Boeing to obtain nozzle thrust coefficient data with and without external flow. These data will be used to verify the performance of the engine manufacturer's proposed nozzle.

The test models will duplicate the airplane pod configuration, and measurements will be made of the thrust minus drag of the nozzles including the effects of boattail and base drag, internal thrust and secondary air momentum, and reverser hardware.

Models of the engine exhaust nozzle will be tested in the Boeing acoustic engineering laboratory to obtain jet noise data. These data will be used to monitor the structural and community noise characteristics of the engine.

8.4.4 FULL SCALE QUALIFICATION TESTS (RFP 2.25.8, 2.25.9, 2.25.10, 3.2.16, 3.3.9)

Full-scale nozzle and reverser development tests will be run by the engine contractor using suitable ground test engines for developing the structural hardware, actuators, mechanism, and control system.

A 75 hour, flight test status, prototype qualification test of the nozzle and reverser will be run using a ground test engine. The nozzle and reverser will be subjected to repeated simulated flight cycles to demonstrate structural and mechanical integrity. The specific test cycle to be used will be established at a later date. The test will be conducted by the engine manufacturer at his facility.

A type certification test of the reverser and nozzle will be conducted in conjunction with the 150 hour endurance type certification tests of the engine which will be conducted by the engine contractor in accordance with CAR 13 or applicable revisions for the SST engine. An altitude test (or simulation) to substantiate the structural capability and performance characteristics of the exhaust system will be a portion of the certification.

Full-scale tests using a ground test engine will be conducted to determine the airplane noise environment and airport noise environment for ground operations of the SST. Simulation of noise-critical airplane structural areas will be required to obtain some of these noise data. The tests will be run at engine operating conditions from 50 percent of maximum dry power to maximum augmented power.

Full-scale tests in addition to those listed in Par. 8.2.2, using the ground test engine, will be conducted by Boeing to measure the exhaust environment on the airplane fuselage, wing, and tail sections.

8.4.5 AIRPLANE TESTS

Tests specified in Par. 8.2.3, 8.2.4, and 8.2.5 include the exhaust system.

8.5 Starting System (RFP 3.2.9.3)

Starter system testing will be performed to establish and substantiate the design details. This test program includes: (1) laboratory testing to develop components, (2) full-scale static rig testing to qualify components and system, (3) service testing to evaluate components, and (4) airplane flight testing to verify system design.

The starter must satisfy the requirements of the airplane manufacturer as well as the engine manufacturer. Testing and development will therefore be closely coordinated between Boeing, the engine manufacturer, and the starter supplier.

The supplier will conduct the starting system component development tests and the qualification tests.

A starting system will be installed and tested by Boeing on two engine ground test rigs. Airline in-service testing of the starter valve will be conducted. Testing of the complete airplane system will be conducted in conjunction with the airplane development, certification, and acceptance tests.

8.5.1 STARTER COMPONENT DEVELOPMENT TESTS

Component tests will include: (1) verification of starter impeller containment, (2) starter and control valve performance, and (3) cutout switch performance. Reliability development testing will be with engine qualified oil. The temperature extremes will be −65°F to +450°F.

Starter valve development tests will be primarily those to develop its reliability under the extremes of vibration and temperature environment. Emphasis will be on cycle endurance testing.

8.5.2 QUALIFICATION TESTS

Starter and valves will be subjected to qualification tests. The qualification test requirements will be per Boeing specification and will include: (1) cycle testing with the starter and valve cold-soaked to −65°F and hot soaked to +450°F, (2) cycle endurance testing for a minimum of 2000 cycles, (3) environmental tests at the in-flight temperature conditions, (4) starter containment tests, and (5) vibration. The starter will be coupled to a flywheel representing the inertia of the engine ro[illegible]d accessories. Applicable sections of MIL S-5[illegible]e included in the Boeing procurement specification.

The starter valve will be subjected to a total of 10,000 cycles of operation throughout a range of ambient temperatures from −65°F to +450°F as a part of the qualification testing.

8.5.3 STARTER GROUND RIG TESTS

The starting system including starter, check valves, and regulating and shutoff valve, will be used for starting throughout the Boeing ground rig test program to further aid in developing a highly reliable starting system.

8.5.4 VALVE IN-SERVICE TESTS

The pneumatic regulating and shutoff valve will be airline service tested in order to improve its reliability. Four valves will be tested on operational jets for at least one year prior to first flight of the SST. Although not completely representative of the installation in the supersonic transport, the rigors of daily in-service use will uncover any areas of weakness in the valve for which corrective action may be required.

8.5.5 AIRPLANE START SYSTEM TESTS

Engine starting tests will be conducted on the prototype and on the certification airplane. Instrumentation will be installed to obtain data on the actual starter air temperatures and pressures and the in-flight environmental temperature of the starter to verify the qualification test parameters selected.

8.6 Fuel System (RFP 3.2.9.4 and 3.2.18)

Fuel system developmental tests will be run to provide criteria for design of the fuel system, to establish the potential in-service problem areas and provide solutions to these problems, and to confirm the suitability of the system prior to in-flight operation. Flight tests will be conducted to verify and certify the fuel system design.

8.6.1 MODEL TESTS

Vent exit testing will be conducted in the Boeing supersonic and transonic wind tunnels to determine the location and configuration of the tank and cavity vent openings. The major portion of this testing will be conducted with an aerodynamic pressure model in conjunction with other aerodynamic testing. However, additional testing of larger scale vent openings will also be conducted, including icing characteristics.

Thermal environment vent testing will be conducted with small-scale fuel tanks with structure and vent air heated to anticipated descent conditions.

8.6.2 COMPONENT TESTS

Evaluation testing of various fuel system components will be conducted on vendor designed equipment not previously evaluated by Boeing. This testing will include the wing pivot fuel line, boost pumps, couplings, and other components at the higher temperature environment required for the SST. This will include determination and evaluation of any explosion-proofing techniques that may be required.

8.6.3 DEVELOPMENTAL TESTS

Thermal testing will be conducted with a full-scale test rig of representative sections of body and wing tanks. They will have provisions for simulation of the structural deflection of the wing c[illegible] and the pressure and temperature environment expected in flight and will include all fuel system components within the tank. The tests will simultaneously determine deposition characteristics, sealing, maintenance techniques, fuel management, and the characteristics of fuel flow out of the tank. Rapid descent tests will be conducted to confirm the results of the small-scale vent tests. The structural deflection of wing cells and pressures and temperature environment will be varied through representative cycles for long periods of testing.

Initial thermal environmental testing will be conducted in a small tank to establish test coating and material characteristics.

8.6.4 COMPONENT QUALIFICATION TESTS

Vendor-designed components will be qualified by the vendor to Boeing specifications.

Slosh, vibration, and structural deflection testing will be conducted on fuel tanks.

8.6.5 SYSTEM DEVELOPMENT TESTS

Testing of the fuel feed system will be conducted with components representative of main tanks and fuel lines in conjunction with the engine ground r[illegible] test. Additional fuel feed system test work at elevated temperatures and altitudes will be conducted in conjunction with the full-scale tank used for the thermal environment testing.

A test setup of the pressure fueling system will be used to establish orifice sizes and to investigate surge pressures due to start and stop of fuel flow.

8.6.6 PREFLIGHT TESTS

Fuel system tests conducted on the airplane before first flight area are:

- Fuel tank quantity gage calibration, sump volume, and trapped fuel.
- Engine feed system performance and operation.
- Refuel, defuel, and dump system operation.
- Vent system overflow and tank pressure tests.

8.6.7 CERTIFICATION TESTS

8.6.7.1 Fuel Feed System (CAR 4b.410 and 4b.413)

Fuel feed system tests will primarily consist of a series of ground tests to demonstrate that this system, when operated in conjunction with the complete fuel system, will supply the required quantities of fuel at the desired pressure for all combinations of airplane operating conditions.

The effects of malfunctions, fuel types, and contaminants will also be determined where possible by analysis or ground test.

Flight tests will be conducted as necessary to verify the results of ground tests and analysis.

8.6.7.2 Fuel Tank and Cavity Venting Systems (CAR 4b.426)

Ground testing of the venting systems will be accomplished in conjunction with the ground tests of the fuel feed system outlined above.

Flight testing of the venting systems will be conducted to supplement the ground test results.

8.6.7.3 Fuel Dumping (CAR 4b.437)

It will be demonstrated that the minimum flow requirements are met, that minimum reserve fuel cannot be dumped, and that fuel does not impinge upon or enter the aircraft during operation in the design operating envelope.

8.6.7.4 Fuel Gaging (CAR 4b.613 and 4b.736)

The fuel gaging (quantity) system will be calibrated during ground tests in the level attitude. Trapped fuel quantities and fuel tank expansion space will be determined.

Flight checks will be made during the flight test program on an instrumented aircraft to determine the effects of flight attitudes and accelerations on the quantity indicating system.

8.6.7.5 Fuel Management (CAR 4b.740 and 4b.741)

Aircraft balance will be maintained in flight by the scheduled use of fuel from each tank.

The adequacy of the procedures to be proposed in the Airplane Flight Manual for fuel management will be evaluated during the flight test program.

8.6.8. ACCEPTANCE TESTS

Performance tests will be accomplished to demonstrate that the fuel system meets all required operational characteristics as set forth in the airplane detail specification.

VOLUME A-VI

PROPULSION

9.0 RELIABILITY, MAINTAINABILITY, SERVICEABILITY

In order to enhance the earning power of the SST, Boeing is following its standard procedure of considering propulsion system reliability, maintainability, and serviceability during the design phase.

9.1 Reliability (RFP 2.12, 4.10)

Details of programmed reliability activities can be found in Section 5.0 of Volume M-II.

9.1.1 ENGINE (RFP 2.25.7, 3.2.9.1)

The substantiation (by extended endurance testing) of an initial in-service engine time between overhauls of 600 hours, as specified in the RFP, is considered a realistically attainable objective. The General Electric Company has, however, indicated in its preliminary data that it plans to achieve a time between overhaul of 600 to 1000 hours at the start of airline service, with an eventual 4000 hours between overhauls with no mid-point inspection.

Further discussion of engine reliability and maintainability objectives is contained in Para. 11.6.4.2 of this volume.

Reliability, safety, and maintainability will be carefully considered during Boeing's evaluation of the engine manufacturers' proposals. After FAA selection of the engine associate, Boeing will take the initiative, in cooperation with the FAA, to ensure development of mutually satisfactory reliability, safety, and maintainability objectives and requirements, with particular emphasis on the ultimate customers' requirements.

9.1.2 ENGINE STARTING SYSTEM (RFP 3.2.9.3)

The probability of successfully starting all four engines is estimated to be 99.81 percent, taking into account all components in the system directly related to starting – starters, regulating valves, check valves, and electrical switches. Based on 2.5 hours per flight, the corresponding malfunction rate is predicted to be one per 1400 flight hours. An analysis of the starting system reliability is shown on Fig. 9-1. This high degree of reliability derives from extensive Boeing experience in the development of

FAILURE PROBABILITY

	QUAN AIRPLANE	ENGINE START	NORMAL START	ASSISTED START
CHECK VALVE	5	0.00001	[illegible]	[illegible]
STARTER	4	0.00044	0.00176	[illegible]
REGULATOR VALVE	4	0.00044	0.00176	[illegible]
SWITCH	8	0.00002	0.00016	0.00008
		TOTAL	0.00373	0.00189

NORMAL START

SYSTEM FAILURE PER FLIGHT	0.00373
PROBABILITY OF SUCCESSFUL START OF 4 ENGINES	0.99627

MAINTENANCE ASSISTED STARTS
(MANUAL OPERATION OF REGULATOR VALVE)

SYSTEM FAILURE RATE PER FLIGHT	0.00189
PROBABILITY OF SUCCESSFUL START OF 4 ENGINES	0.99811
PERCENT OF SUCCESSFUL STARTS	99.81%

9-1 *Starting System Reliability (Includes all components of the start system)* pneumatic starting systems. The regulator valve has two manual override features. The ball switcher (pilot valve), which is normally operated by an electrical solenoid, can be operated directly by pushing a button. The other override feature is the main butterfly valve, which is normally operated pneumatically but can be operated manually by applying a standard wrench at the end of the butterfly valve shaft.

9.1.3 FUEL SYSTEM (RFP 3.2.9.4f)

The following features of the fuel system contribute to its inherent reliability:

- There are two pumps in each tank, either of which will furnish the engine's fuel requirements.
- Electrical power is supplied to the individual pumps of each tank from separate A.C. power buses.
- The engine pump will suck the fuel from the main tanks in case of total electrical failure.
- The system is designed to prevent inadvertent reversal, mismating, or improper connections of lines and electrical connections.
- Thermal pressure relief is provided where trapped volumes may exist.
- Surge pressures resulting from valve closures are held below proof pressure of the system by control of valve operating rates.
- Components requiring orientation to provide correct flow direction are designed so that they cannot be installed improperly.
- Wherever possible, fuel lines are routed through fuel tanks to minimize external fuel leakage.
- Ground defueling is controlled by a shutoff valve, manually actuated on the ground only. The ground refueling access door cannot be closed unless the valve is in the closed position.
- Boost pumps, valves, and gages have check-out capabilities.
- Engine shutoff and cross-feed manifold valves are powered by the essential electrical system and a special battery bus.
- The fuel manifold allows fuel from any tank to be used by any engine.

A representative analysis of the effect of the failure of specific components on mission capability is shown in Fig. 9-2.

9.1.4 INLET CONTROL (RFP 3.2.9.6)

A numerical reliability analysis has been made by Hamilton Standard Division of United Aircraft Corporation to estimate the reliability of its proposed inlet automatic control system. Data were taken from 11,000,000 hours of Hamilton Standard experience with jet engine fuel controls which contain similar types of components. This experience has shown the following: average premature removal period, 8200 hours; average inflight shutdown period, 250,000 hours (est.); mean time between partial failures (performance degradation), 1250 hours. Utilizing the fuel control experience as a guide, the mean time between failures (MTBF) for total failures may possibly be as much as 30 times as high as for partial failures, or about 38,000 hours MTBF.

Due to difficulties experienced in separating actual failure occurrences from reported occurrences, the estimates are believed to be quite conservative, so that the true MTBF may be much higher than stated. In addition, the reliability program proposed by Hamilton Standard is expected to result in significant reductions in failure rates from those experienced on jet engine fuel controls. Boeing will establish a design goal for the inlet control of 3,000 hours MTBF (performance degradation). Test requirements will be established in the procurement specification to provide a reasonable assurance of achieving this objective.

EQUIPMENT FAILURE AND HUMAN ERROR MODE AND EFFECTS ANALYSIS

...ONENT	FAILURE OR ERROR MODE	EFFECT ON: SYSTEM OPERATION	EFFECT ON: AIRPLANE OPERATION	HAZARD A	HAZARD B	HAZARD C	HAZARD D	FAILURE RATE	DETECTION METHOD	COMPENSATING PROVISIONS
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

EXAMPLE

9-2 Sample Failure Analysis

9.2 Maintainability (RFP 2.11)

9.2.1 ENGINE (RFP 3.2.9.1)

The propulsion pod is separated into three major components: the inlet, the engine, and the exhaust sections. The inlet and exhaust sections may be removed from the aircraft individually for shop overhaul or other heavy maintenance and replaced with serviceable units. Fig. 9-3 depicts typical ground handling techniques for propulsion pod components.

Structural provisions for attaching the pod handling fittings are provided in the wing lower surface. The entire propulsion pod may be removed or installed, or the exhaust and inlet sections may be removed and installed individually, by use of the proper fittings.

Depending on customer need, spare engine buildup may consist of either or both inlet and exhaust sections in addition to the basic engine. Differences in buildup installation are held to a minimum to enable neutral engine buildup conversion to any position with a minimum of maintenance effort.

9.2.1.1 Engine Replacement

The power plant assembly is attached to the engine strut with three cone bolts which provide self alignment during installation. All engine plumbing to the aircraft, except fuel, runs through a disconnect panel. Hydraulic lines use self-sealing, quick-disconnect couplings. Electrical disconnect is accomplished at a common bulkhead by self-locking plugs and receptacles which require no safety wire. Engine controls are connected with the aircraft by a quick-disconnect, self-tensioning coupler eliminating the need for cable rigging during engine change.

Power plant assembly replacement is accomplished in the following sequence:

- Remove engine side cowling.
- Attach hoist lugs to lower surface of wing.
- Disconnect plumbing.
- Disconnect engine mechanical controls.
- Disconnect electrical connectors at firewall.
- Attach hoist and lift engine using lower wing surface fittings to unload cone fittings.
- Position transportation trailer, remove cone bolt nuts, and lower engine.

Installation, in addition to reversing the above procedure, requires torquing of cone bolt nuts.

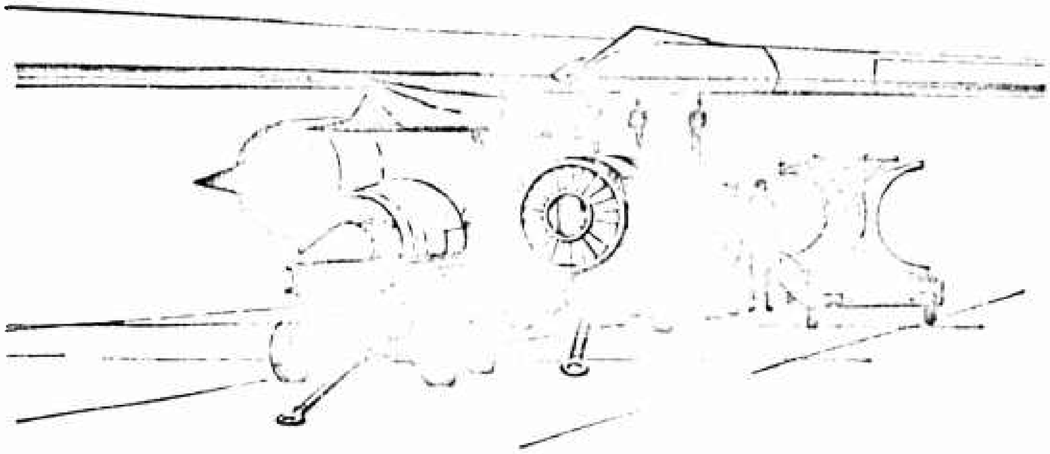
9.2.2 INLET SECTION

Opening of engine side cowl panels provides access to the inlet assembly attach bolts, hydraulic disconnect points, door position switch wiring connector, and various pneumatic sensing lines requiring disconnect prior to inlet removal. The position of all inlet doors can be visually verified. The actuators for the powered bypass doors are conveniently exposed by opening the engine section cowling. The actuator for expanding the centerbody is in the nose outside the cowl front lip and accessible through the removable nose cone. Elements of the automatic powered subsystem control are also easy to reach by opening the pod cowling. If preflight operational checks of the inlet control system are desirable, a simple on-board test unit will be developed. Ground equipment must be used when complete maintenance checkout and calibration of the inlet system is required.

9.2.3 EXHAUST SECTION

Boeing will monitor maintainability during design and development. The following objectives have been established:

- Time between overhaul of all components on the exhaust section, including controls, will be the same as the engine. Target is 4000 hours.
- All components (e.g. bushings, bearings, seals) will have replaceable wear surfaces where this provides a significant advantage over complete component replacement.



9-3 *Propulsion Pod Ground Handling*

- Means will be provided for powered ground operation of the actuators without running an engine.
- Control valves and devices will be replaceable without disrupting cable systems. Reference points or rig pins will be provided for rigging. All rigging measurements will be linear and be taken between flats or index marks on the appropriate components.

Maintenance procedures of the exhaust section will be similar to those for present commercial jet transports. The entire exhaust section is removable from the engine case aft face for overhaul or heavy maintenance. Access to the attach bolts is through the engine side cowl and augmentor case cowling. All hydraulic and electrical disconnects are also accessible through the engine side and augmentor case cowling. Pre-rigging of the exhaust assembly components prior to engine installation provides minimum system adjustment at installation. The secondary air inlet door actuators and nozzle area control and reverser actuators are accessible by removal of the augmentor case cowling.

Ground operation of the exhaust system secondary air inlets, nozzle area control components, and reverser cover panel is accomplished by a low capacity, external hydraulic supply cart attached to suitable ground service connections at the engine. Such ground operation permits routine line maintenance, security, and rigging checks.

9.2.4 FUEL SYSTEM

SST fuel system maintenance requirements are similar to those of current commercial jet transports. The equipment now in use for fuel tank purging and inspection is directly applicable.

Body fuel tank maintenance techniques are similar to those used on the Model 707 center wing tank. Fuel cells are replaceable individually through access plates in each tank bay, and the interconnect concept is the same as the Model 707 installation.

Wing tank maintenance and inspection is accomplished through conventional access plates in the wing lower surface. Ample fuel dams are installed in wing tank insulating space to permit isolation of any fuel leak to an area between ribs.

Fuel tank boost pumps, drip sticks, and fuel temperature probes can be replaced without entering or defueling any tank. The wiring from the fuel quantity tank sensing unit requires only one connector and cable to carry the total signal from each tank. This permits use of a splice-free wiring system, eliminating a source of system malfunction.

Fuel system plumbing is replaced in a conventional manner. The use of mechanical connections eliminates the need for welding, swaging, or cutting on the aircraft. All motor-operated valves in the fuel system are either totally exposed or semi-submerged, requiring no tank penetration or plumbing disconnect to replace motor and gate valve assembly. Manual override handles are provided on all valves for ground operation without electrical power.

Routine line maintenance requirements for fuel pressure checks may be accomplished with the fuel boost pumps. All plumbing not inside fuel tanks is accessible through access plates to accomplish required leak checks.

Ground fuel transfer for maintenance purposes can be accomplished by use of the fuel system electric boost pumps, by proper selection of dump and refuel valves, or by use of a manual valve in the cross-feed system. Defueling is accomplished by connecting to the refuel manifold; no special equipment is required.

9.2.5 ENGINE ANALYZER—MAINTENANCE ANALYSIS AND RECORDING

As a means of reducing maintenance costs and improving schedule adherence, the use of a flight maintenance analysis and recording system is under consideration for the engines and engine accessories used on the SST. By monitoring and analyzing engine performance, this system assists in pinpointing probable failures for preventive action. The potential benefits from the system may be considerable. However, present installations in military and commercial aircraft are experimental, and further evaluation of effectiveness and reliability is required.

The system provides information in two ways. First, the on-board display affords a quick look at the data being accumulated and indicates any significant out-of-tolerance conditions to the flight crew. Second, the data are recorded on magnetic tape for later detailed analyses at ground facilities by general-purpose computers, such as those generally available at airline installations.

9.3 Line Maintenance and Inspection

The power plant installation is designed for ease of line maintenance, inspection, and servicing. The side cowl panels are equipped with quick-opening latches and tubular supports. The panels can be removed or secured in the open position without special tools, thus exposing the complete engine, including all accessories, for inspection and maintenance.

With the airplane in the normal parked position, all engine components are less than 10 feet above the ground, and all engine driven accessories are less than seven feet above the ground. Any accessory can be replaced without removal or loosening of another accessory. Filters and sump plugs requiring periodic servicing are readily accessible. Particular attention was given to eliminating cowl wear points. Where this is not possible, easily replaceable rub strips are provided on cowl wear surfaces, and all cowling hinge points are bushed for repair ease.

Positive means of indicating positions of the cowl latches, of the inlet and exhaust system doors, and of the removable portion of the inlet centerbody will be provided for ease of conducting walk-around inspection.

9.4 Serviceability

Rapid, easy access is provided all along the engine, from inlet actuators to exhaust flange, by opening the engine cowl panels. Opening the panels exposes all engine components and accessories which require servicing. Access to engine mounts, wire bundles, plumbing, ducting and engine instrumentation is also provided. All items within the engine compartment, such as the oil filter, hydraulic fluid filter, and the fuel control filter, which require periodic removal and inspection, are removable without disturbing any other system or engine accessories.

The fire extinguisher pressure gages and discharge indicators can be inspected from the ground through a sight glass without opening a panel.

Access for oil filling through a separate door eliminates the necessity for opening the main cowl panels.

The design objectives for engine components is a minimum of 3000 hours between overhaul. The detail specifications for purchased equipment on the engine require qualification testing which will ensure satisfactory operation to meet this design objective.

VOLUME A-VI

PROPULSION

10.0 NOISE SUPPRESSION (RFP 2.6.1, 2.6.2, 2.6.3, 3.2.9, and 3.2.16)

10.1 General

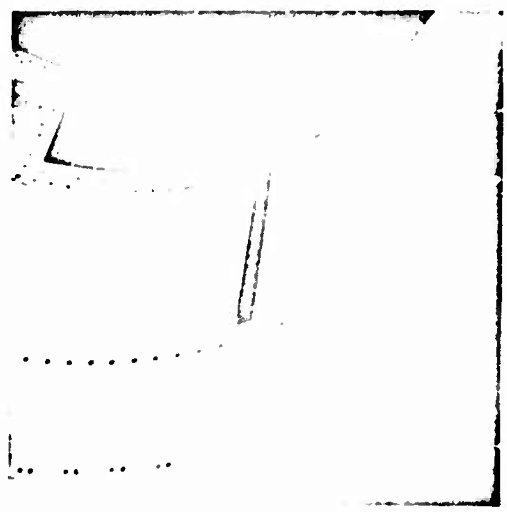
Special noise suppressors will not be required to hold the community noise generated by the engine to levels comparable to jet engines in international operation today. As noted in Volume A-V, Aerodynamics, and Volume A-VII, Systems, the takeoff, landing, and ground noise requirements of the RFP are satisfied. Noise characteristics will be closely scrutinized by the engine and airframe manufacturers throughout the design of the engine, the inlet, and the exhaust system.

To determine whether noise may be further reduced, the capability of Boeing's acoustic research group and the acoustics model jet facility are employed in a continuing effort in support of the SST program. Potential approaches for noise suppression are presented in this section.

10.2 Potential Approaches for Noise Suppression

10.2.1 INLET NOISE

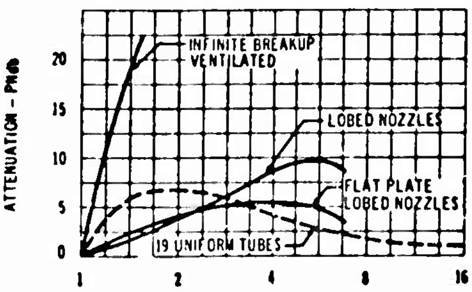
Inlet noise consists primarily of discrete frequencies generated in the compressor blading. There are three main approaches to reducing this noise: (1) source reduction by tailoring compressor design; (2) transmission blockage by choking inlet flow, and (3) transmission attenuation by wall absorption along the inlet duct. The first approach is the subject of intensive investigation at Boeing, at the engine manufacturers, and at research laboratories. It has been demonstrated that varying the axial spacing between the compressor blade rows and canting the stators can markedly improve perceived noise in the speech interference range, although overall sound levels may remain unchanged. Varying the number of blades in one compressor stage in relation to the number in adjacent stages has produced favorable results. The effect of other compressor aerodynamic design variables with respect to noise generation is being studied. Although under continuing study the second approach, transmission blockage, has not been accepted as practical because of distortions induced at the compressor inlet face. The third approach, transmission attenuation, has been successfully applied in a relatively short inlet duct, as shown in Fig. 10-1, and should produce even better results in a duct of longer length. The absorptive lining may be of two types: broadband absorptive material, such as fiber glass; or tuned-resonant lining, which is effective over a relatively narrow frequency range but is immune to damage from watersoaking and similar operating conditions.



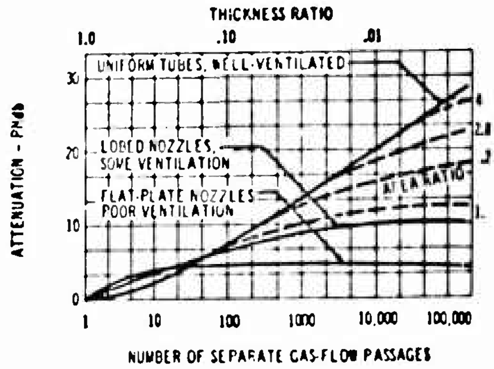
10-1 *Sound Absorptive Inlet*

10.2.2 JET NOISE

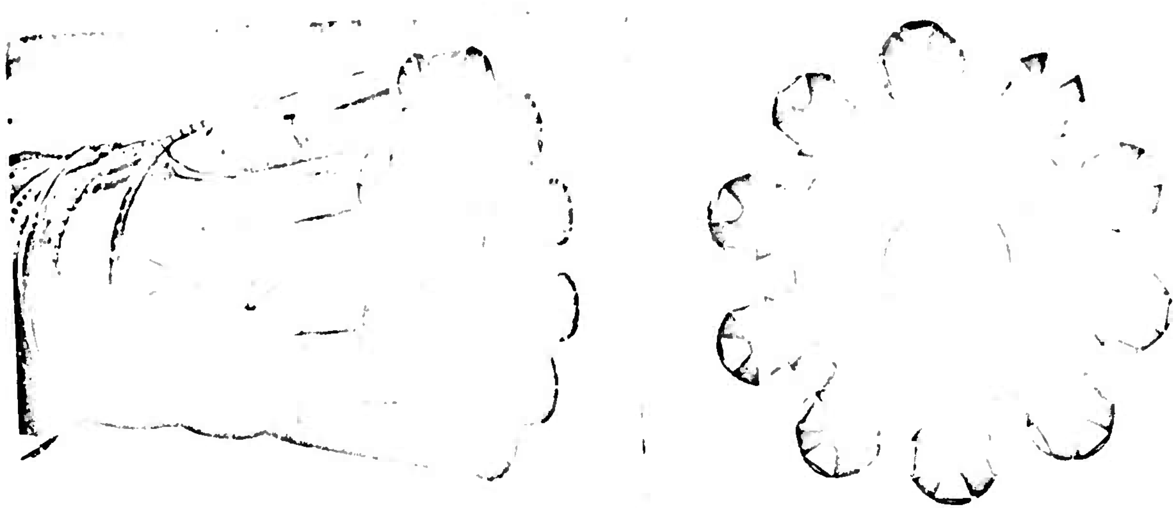
There are few practical suppression techniques which can be applied to a given high exhaust velocity engine. They are variations of the concept of accelerating induced secondary air and mixing with the primary stream, with the consequent reduction of the relative velocities between the jet and ambient air. Figs. 10-2 and 10-3 illustrate some of these findings. The two suppressors most successful to date are: (1) the divided-flow nozzle, which contains what may be thought of as internal ejectors, illustrated by Fig. 10-4; and (2) the divided-flow (or corrugated-boundary) nozzle plus external ejector shell. Variable area, convergent-divergent ejector nozzles such as used by two of the engine manufacturers proposing for the supersonic transport can be adapted to the second approach. Large volumes of secondary air can be pumped into the nozzle by the proper tailoring of the ejector. This air is accelerated and mixed with the primary flow at the nozzle throat by special flow-dividing air passages moved into position when suppression is desired (Fig. 10-5). Another approach is to displace alternate segments comprising the ejector exit variable area control radially in order to provide a corrugated exit shape for mixing the exhaust gasses with the free-stream flow (Fig. 10-6). These approaches are under study to determine the potential performance in noise reduction and nozzle efficiency. No credit has been taken for any potential noise suppression in the calculations included in this proposal. If other techniques are found to show promise, they will be thoroughly investigated. Any application must be consistent with airplane requirements, such as small thrust loss, small aerodynamic drag, light weight, and compatibility with thrust reverser and augmentor operation.



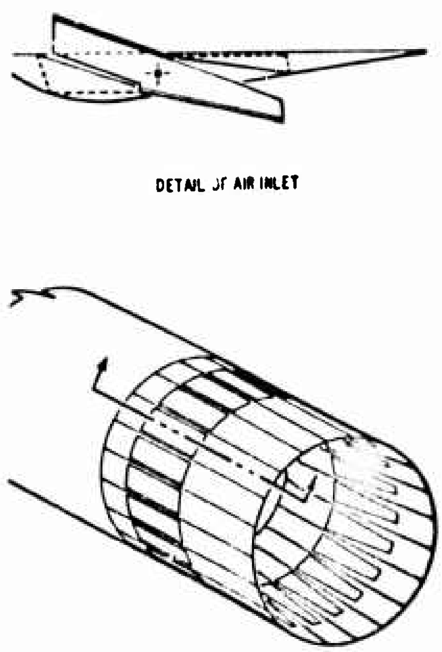
10-2 *Effect of Area Ratio on Noise Attenuation*



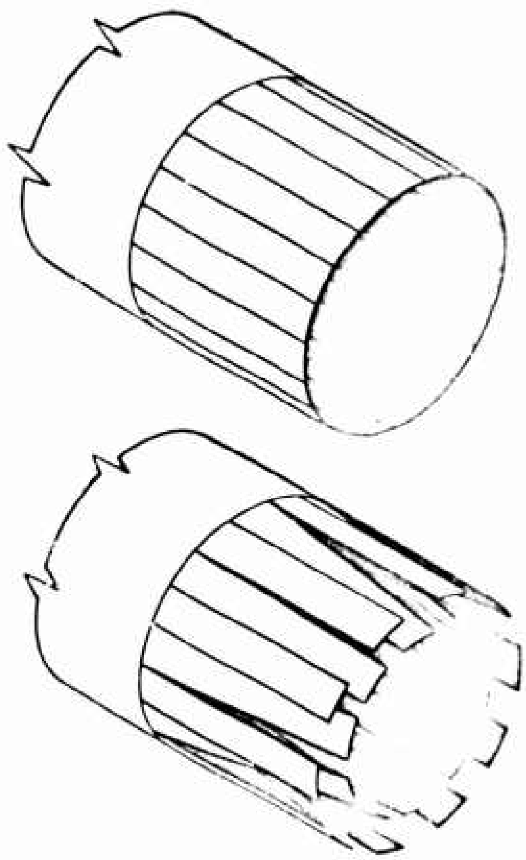
10-3 *Effect of "Break-Up" and Ventilation on Noise Attenuation*



10M Divided Flow Noise Suppressor



10-5 Injector Type Noise Suppresser



10-6 Petal Noise Suppresser

D6-2400-12

VOLUME A-VI

PROPULSION

11.0 ENGINE SELECTION AND DEVELOPMENT

11.0 ENGINE SELECTION AND DEVELOPMENT (RFP 2.25, 3.2.9.1)

11.1 Introduction (RFP 1.2)

For several years, The Boeing Company has studied the engine selection for the SST, investigating wide variations in engine cycles. The engine companies and NASA have also contributed significantly to the cycle selection effort. In the early Boeing studies, technology similar to that offered by the J58 and J93 engines was used. When designing either fixed wing or variable sweep airplanes for cruise speeds between Mach 2.5 and 3.0, and ignoring the transonic boom considerations, the engine choice consistently was a low bypass turbofan. Changes in turbine technology and a re-evaluation of sonic boom problems alter this result.

The subsequent NASA SCAT program (Contract NAS 1-2580), in which The Boeing Company participated, resulted in two broad conclusions: (1) high turbine in-temperatures and low engine specific weight are required, and (2) the turbofan was the optimum cycle for the SST. However, the engines used for the SCAT program were NASA study engines in which the fans had somewhat better cruise performance characteristics and lighter weight than the presently offered engines. Also, the SCAT mission requirements were different from the present Request for Proposal (RFP) requirements.

In determining the engine which The Boeing Company feels best meets the RFP mission, consideration was given to the engine-airplane technology and its technical substantiation, the advanced design features offered in the engine, and an evaluation of each engine manufacturer's capability to carry through a successful commercial engine program.

Based on the preliminary engine performance data supplied by the engine manufacturers on November 15, 1963, and subsequent modifications, the General Electric GE4 J4C turbojet designed for 2200°F. cruise turbine temperature and the Curtiss Wright TJ70 resulted in the lightest gross weight airplanes by about a 10 to 15 percent margin. However, the Curtiss-Wright nozzle performance, engine weight technology, and turbine cooling techniques have not been substantiated to the same degree as those offered in either the General Electric or Pratt & Whitney engines. Hence the TJ70 appears to be a greater risk than the General Electric engine. The GE4 J4C engine (Ref. 19) was selected as the basic engine for the Boeing proposal.

Although the GE4 J4C engine appears to be the correct choice for the proposal airplane based on the current RFP mission and engine data, the Boeing configuration lends itself to use of any of the offered engines in the event that a different engine is desired by the FAA or the airlines in the final evaluation. The performance of the Boeing SST airplane with the other proposed engines is covered in Volume A-V, Aerodynamics.

The following sections will discuss the specific characteristics of the engines offered, the general matching characteristics of the engine cycles, and the design airplane gross weight which results with each of the engines. A technical review and discussion of component technology is presented. An initial appraisal is made of the relative development status of each of the engines and of the demonstrated capabilities of the engine manufacturers. Comparative installation features are discussed. A more detailed review will be submitted in March, 1964, as requested in the RFP.

11.2 Discussion of Offered Engines

11.2.1 BASIC FEATURES

The basic characteristics of the engines proposed for the SST are summarized in Fig. 11-1. Each engine manufacturer's specification basic thrust and airflow size are shown. All the engines are scalable except the JT11F-4. The distinctive features and important design param-

ENGINE	TJ70	GE4/J4C	GE4/F6A	STF-188B (JTF15A-1)	JT11F-12	JT11F-4
Airflow – Lbs/Sec (Sea Level Static)	600	475	550	630	640	640
Maximum Dry Takeoff Thrust – Lbs	53,200	38,900	27,200	32,600	33,800	33,100
Maximum Takeoff Thrust – Lbs	53,200	51,800	45,000	51,200	51,100	50,400
Bypass Ratio	0	0	1.1	1.3	1.08	1.08
Fan Pressure Ratio	–	–	2.2	2.5	2.5	2.5
Primary Pressure Ratio	9.0	9.5	11.0	11.0	9.5	8.2
Maximum Cruise Turbine Inlet Temperature	2040° F	2200° F	2200° F	1900° F	1900° F	1900° F
Maximum Turbine Inlet Temperature	2200° F	2200° F	2200° F	2100° F	2100° F	2100° F
Engine Weight (Including Nozzle and Thrust Reverser) – lbs	6340	8351	7670	8485	9135	9555

NOTE: These Data are Based on Engine Company Inputs Through 12/24/63.

11-1 *Basic Data Offered Engines*

eters are discussed below:

TJ70 Curtiss-Wright

- Single Spool, two bearing rotor.
- Single stage, highly loaded, transpiration cooled turbine operating at 2100 F maximum continuous.
- Variable geometry turbine diaphragm and exit nozzle.
- Cusp plug nozzle. (Subsequent to November 15, 1963, this nozzle was changed to a convergent-divergent plug nozzle with variable flaps.)
- Reverser included with engine.

This engine has extremely high aspect ratio compressor blades which are a high risk item because of structural dynamic problems which may arise. In fact, the whole compressor design may have to be changed because the short chords may lead to poor low speed matching characteristics.

The engine has a high thrust weight ratio, but it incorporates a variable turbine nozzle, a variable primary nozzle and a variable divergent nozzle. These are features which usually cost extra weight in engine design. The engine nozzle originally proposed was a simple cusp plug which had poor measured performance in the Boeing test facilities. The nozzle was changed to the present design with no weight increase.

Variable geometry turbine nozzles have not achieved a high degree of development and are not used on any present engines. This feature is mandatory for a dry turbojet on an SST to obtain good low-speed performance.

GE4 J4C General Electric Augmented Turbojet

- Single spool, three bearing rotor.
- Variable stator compressor (9.5:1 pressure ratio in 7 stages).
- Two stage turbine, convection plus film-cooled, operating at 2200 F maximum continuous.
- Full augmentation afterburner.
- Convergent-divergent ejector nozzle with variable throat and variable exit.
- Reverser included with engine.

The engine is a conventional turbojet patterned after the J79 and J93. The main advancement is the high turbine in-temperature of 2200 F. This temperature is achieved through combined convection and film cooling techniques. The exit nozzle design is similar to that of the J93.

Turbine cooling is a continuous flow process during all engine operation.

GE4 F6A General Electric Augmented Turbofan

- Single spool, three bearing rotor.
- Single stage 2.2:1 pressure ratio front fan.
- Seven stage primary compressor giving overall primary pressure ratio of 11:1.
- 1.1:1 bypass ratio.
- Variable stators and inlet guide vanes.
- Two stage turbine with convection plus film cooling operating at 2200 F maximum continuous.
- Mixed flow augmentor. (Fuel injection and flame stabilization occur in the primary stream. Mixing is accomplished with the aid of a fixed geometry daisy chute mixer.)
- Nozzle is similar to that employed in the J4C as described above.
- Reverser included with engine.

A salient feature of this engine is the mixed flow augmentor which provides high augmentation during transonic acceleration and takeoff. The flame holders and fuel injection nozzles are located in the hot stream so that combustion can be initiated easily and efficient burning will result.

There are problems associated with a fully mixed after-burning fan engine designed to operate over a wide Mach number range. The fan pressure ratio and efficiency tend to be reduced at low corrected engine speeds due to low fan discharge pressures. Mixing losses can also be high.

Another distinguishing feature is the first compression-fan stage which produces low pressure ratio near the hub and very high pressure ratio near the tip, with a shroud in the middle. Whether this combination on a single rotor fan will have development problems remains to be determined.

STF 188B (JTF15A-1) Pratt & Whitney Duct-Burning Turbofan

- Two spool, four bearing rotor (two bearing supports).
- Two stage fan (front spool) with 2.5:1 pressure ratio.
- Five stage compressor (rear spool) with overall primary pressure ratio of 11:1.
- 1.3 bypass ratio.
- Two stage turbine operating at 1900°F maximum continuous. (First stage turbine is convection cooled. Second stage drives fan rotor.)
- Fan burning augmentor employing aerodynamic flameholder.
- Convergent-divergent blow-in door ejector nozzle with variable throat area control in the fan stream.
- Reverser included with engine.

This is an advanced duct-burning turbofan engine which employs the latest Pratt & Whitney engine technology. The duct burner is external to the primary engine case, which could present some engine case and turbine cooling problems. The burner employs an aerodynamic flameholder with low pressure drop, which contributes to high performance. The fan exit nozzle is variable and provides a high level of fan efficiency at all flight speeds. The ejector nozzle is the same type that Pratt & Whitney has under development for the TF30 engine (TFX) and has been evaluated extensively through wind tunnel model tests during the past few years.

A 1900°F continuous turbine flame temperature detracts from the performance of this engine. In part, it is compensated for by the extensive use of lightweight technology which is consistent with the 1970 time period.

JT11F-4 Pratt & Whitney Duct-Burning Turbofan

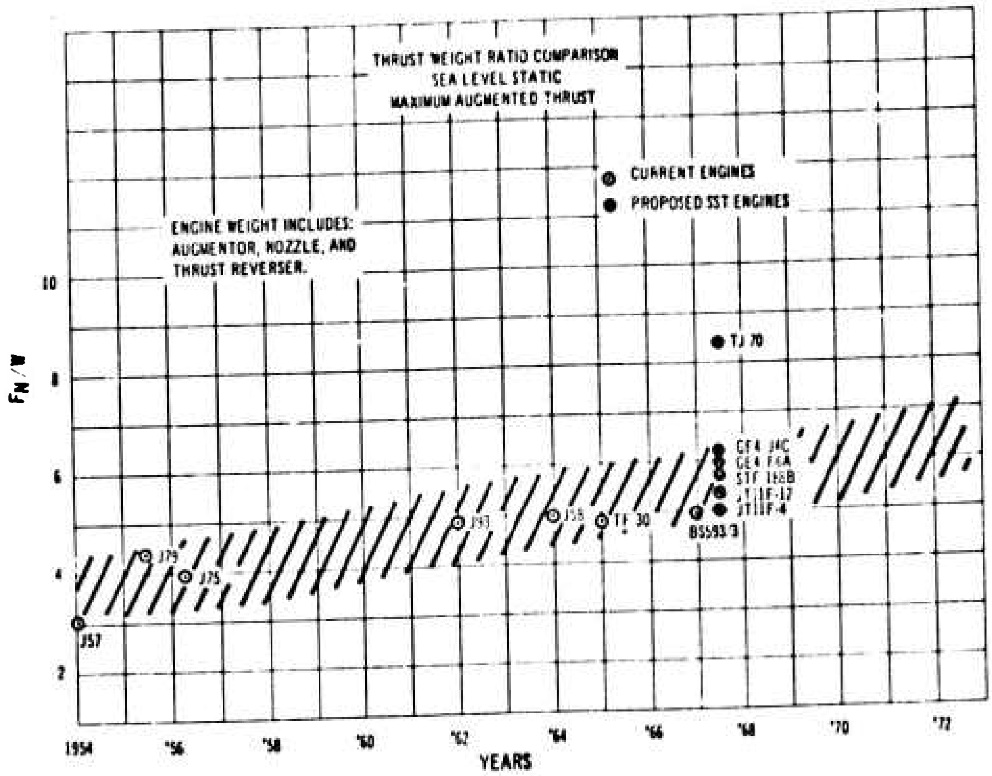
- Single spool, fixed airflow fan version of the JT11 (J58) engine.
- Two stage fan, 2.5 pressure ratio.
- Five stage compressor behind fan with overall primary pressure ratio of 8.2.
- 1.08 bypass ratio.
- Three stage turbine, first stage convection cooled, operating at 1900°F.
- Duct heater and nozzle same as STF 188B.
- Reverser included with engine.

This engine is a modification of the J58 Mach 3.0 turbojet engine which is currently under development by Pratt & Whitney. It is designed to use the existing compressor, burner, and turbine stages of the J58 with an additional turbine stage added to drive the compressor-fan rotor. The duct burner and nozzle arrangement is the same as that used on the STF 188B. This engine has the advantage of being available for early delivery (two and one-half years after go-ahead) for a prototype airplane. The engine weight will be high since it will not incorporate the latest state-of-the-art development and weight technology. This engine is not offered as a scalable engine.

JT11F-12 Pratt & Whitney Duct-Burning Turbofan

- Advanced lightweight scalable version of the JT11F-4.
- Primary pressure ratio increased to 9.5.

This engine could exist as a follow-on to the JT11F-4 or could be developed as a new engine designed initially for the SST mission. It is slightly heavier than the STF 188B but has comparable performance. The engine has



11-2 *Thrust, Weight Ratio Advancement* the same disadvantage in that the lower turbine flame temperatures offered by P&W detract from the performance.

11.2.2 TECHNOLOGY OF OFFERED ENGINES (RFP 3.2.9)

Important advances in technology are being offered in each of these engines. The probability of achieving these technology levels has been considered in The Boeing Company evaluation of the engines.

The advancements having the greatest significance are specific weight, turbine temperature, and nozzle performance.

11.2.2.1 Weight Technology

Fig. 11-2 shows the level of thrust weight ratio of the proposed engines and compares them with past and current supersonic engines in development or operation. The thrust weight levels of the proposed SST engines (except for the C-W TJ70) appear to be a logical progression in weight technology. The weight technology, represented by the cross-hatched area, considering the higher turbine inlet temperatures proposed, appears to be a reasonable goal for a 1970 operational date. The weight technology indicated for the TJ70 engine appears to be optimistic for a commercial engine.

This general improvement in weight technology is being achieved through higher compressor stage loading (which results in fewer stages of compression for a given pressure ratio), higher heat release burners (which shorten the combustion section), and improved turbine cooling.

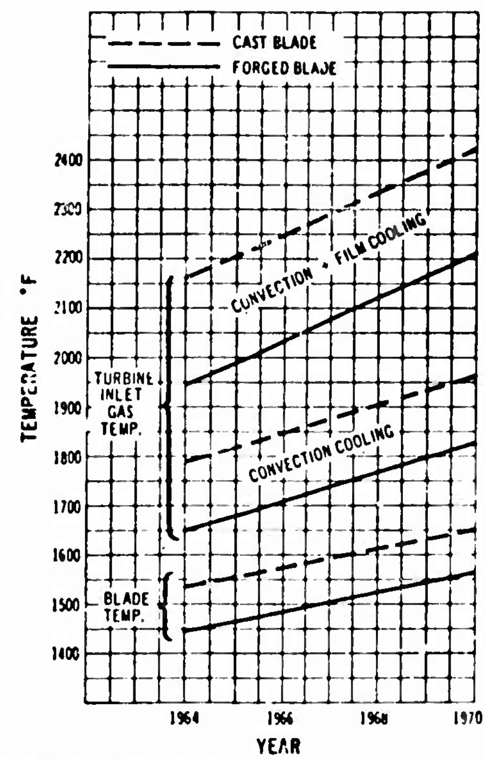
11.2.2.2 Turbine Temperature Technology

The results of Boeing and government-funded SST studies have shown that turbine flame temperatures well in excess of existing commercial practice will have to be used in order to make the program a success.

One of the fundamental differences between the offered engines is that General Electric and Curtiss-Wright are quoting 2200 F and 2100 F cruise turbine in-temperature (TIT) respectively, while the Pratt & Whitney quoted TIT level is 1900 F. The higher TIT provides greater transonic thrust and lower cruise specific fuel consumptions, and has a significant effect on airplane gross weight to perform the mission. It is therefore of prime importance to evaluate the level of TIT which is reasonable for the 1968-1970 time period.

The Boeing Company discussed this problem not only with the three engine companies involved in this proposal but also with specialists at Allison, Rolls-Royce, and Bristol to gain information on available turbine temperatures and turbine cooling techniques. The general conclusions of this survey are listed below:

- Commercial parts life of up to 10,000 hours is required in order to ensure that random failures will permit time between overhauls (TBO) values in excess of 3000 hours.
- When cast blades are employed, together with cast cooling passages, a maximum cruise flame temperature of 1800° to 1850 F can be tolerated today on the SST mission. These convective cooled blades would withstand 3000 hours TBO based on creep life expectancy. Higher temperature would require either new materials or other cooling methods than pure convection. Metallurgical improvements by 1968-1970 should raise this limit to the 1950 F to 2000 F range.
- Using forged materials with cooling passages, the allowable blade temperature for the same creep life will be lower by approximately 75° to 90 F. Hence, at a given gas temperature when using forged materials, the manufacturer must improve his cooling effectiveness to allow for the method of fabrication.
- Employment of film cooling or transpiration cool-



11-3 *Predicted Turbine Temperature for 5,000 - 10,000 Hour Creep Life*

ing could raise the creep life temperature limit of forged blades to above 2200 F, through the improved cooling effectiveness. This conclusion was supported by Rolls-Royce and Allison.

Fig. 11-3 shows the trend of turbine in-temperature and blade temperature with years, and also the effect of various types of cooling on the allowable temperature based on creep life.

- In most current commercial engines turbine blade replacements are caused by surface cracks due to thermal shock or by high temperature bending fatigue and not by creep life limitations. There is no analytical way to predict shock life. Only by running 5000 to 10,000 hours of cyclic testing can the thermal shock and bending fatigue characteristics of a particular turbine and cooling configuration be determined.
- When forged materials are used, the resistance to thermal shock is improved by about 75° to 90°F at the same cooling effectiveness.

Based on the findings of this survey, it appears that a 600 hour TBO can be achieved in the 1968-1970 time period using advanced blade materials, film cooling, and forged blading, when operating with cruise turbine flame temperatures of up to 2200 F on the SST mission. A TBO of 3000 hours is a reasonable target after some service experience. Provisions for visual inspection of the engine turbine between overhauls will probably be necessary.

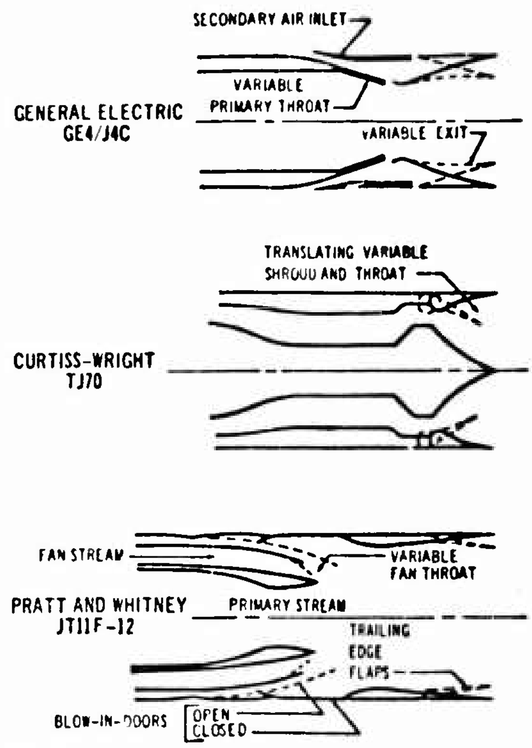
General Electric turbine blade cooling tests have been run on a J93 up to 2400 F TIT using the film cooling technique. At 2200 F the turbine blade metal temperatures are at or below the metal temperatures in the current commercial subsonic jet engines. General Electric has developed a turbine blade stem drilling process and quality control technique which is unique and has been successfully demonstrated on the J93. Pratt & Whitney has also conducted turbine cooling tests using convective cooling techniques on a modified J78 test engine. Curtiss-Wright has run turbine cooling tests using a transpiration cooling technique at 2370°F. Although transpiration cooling appears to offer the greatest potential, certain fundamental structural problems appear to make it a riskier approach than either convective or film cooling techniques.

11.2.2.3 Nozzle Technology

Each engine manufacturer has proposed a different nozzle design for his engines. General Electric has proposed a fully variable convergent-divergent (C-D) ejector nozzle; Pratt & Whitney, a fixed shroud blow-in door ejector; and Curtiss-Wright, an annular C-D nozzle. Sketches of these nozzle types are shown in Fig. 11-4.

The estimated nozzle gross thrust minus drag coefficients for the three types of nozzles are shown in Fig. 11-5. Values are shown for conditions of maximum thrust at all Mach numbers and cruise thrust for supersonic cruise and subsonic cruise. Selected test data have been plotted on these curves to indicate the level of development that has already been achieved for the different nozzle types. Gross thrust minus drag (C_F) is defined as nozzle thrust minus nozzle drag (including ram drag of secondary air and nozzle boattail drag) divided by the ideal thrust of the nozzle primary and secondary airflows. The Curtiss-Wright annular C-D nozzle performance is lower at subsonic cruise than the other nozzles, due primarily to higher boattail drags.

The test points shown (Fig. 11-5) were derived from three sources: NASA, Boeing, and the engine manufacturers. In all cases the models tested were not exact duplicates of the proposed nozzles. However, the throat to exit area ratios were closely approximated. It should be noted that the measured performance levels of these models do not necessarily indicate the full potential of the various nozzle concepts. Very little test data are available for the Curtiss-Wright nozzle because of limited



11-4 Proposed Exhaust Nozzles

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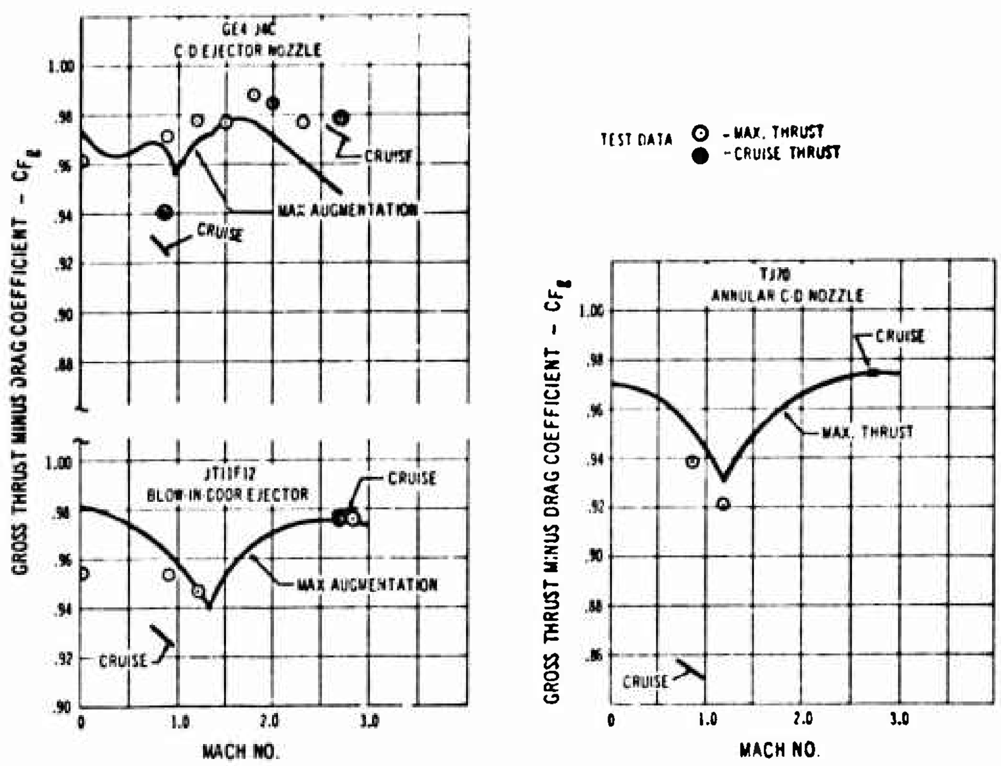

11-5 *Nozzle Performance Comparison*

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levelopment work.

11.3 Airplane Performance Comparisons

11.3.1 ENGINE-AIRPLANE MATCHING

The relative performance merits of the proposed engines can best be judged in terms of the resulting airplane capabilities. Before comparing the specific engines, the manner in which a propulsion system is matched to the airframe will be discussed.

In the matching process, the installed propulsion system is scaled to represent different sizes of the engine under study. The airplane takeoff gross weight and operating weight empty are also scaled to represent different sizes of the airplane under study. The wing area is varied to include the effect of wing loading on the aerodynamic performance. The body size and payload are held constant.

The performances of the engine-airplane combinations are then computed for the design mission. The flight profiles are determined by the sonic boom overpressure limits. The matched engine-airplane is that combination which achieves the design range at the minimum gross weight.

In addition to the sonic boom overpressure limit, other restrictions are applied which sometimes make the airplane heavier and the engine larger than would otherwise be the case. Among these are:

- Wing area has a lower limit, dictated by takeoff speed limitations or by other practical considerations.
- The engine must be large enough to provide adequate airplane acceleration under all flight conditions. For example, a minimum thrust margin, $\frac{F_n - D}{D}$ 0.3 on a standard day is required during climb and acceleration at the altitude determined by sonic boom limitations. This margin ensures that adequate thrust is available to accelerate to cruise speed on a hot day, and that the time required to accelerate will not be excessively long.

The engine-airplane matching results for the non-augmented turbojet, augmented turbojet, and augmented turbofan engines are generally as follows for a variable sweep airplane:

- **The Non-Augmented Turbojet**

The engine size is established as that necessary to provide the thrust margin to accelerate the airplane at the altitude dictated by the sonic boom overpressure limit. Because of its low thrust per pound of airflow, the size is large compared to augmented engines. At supersonic cruise conditions, the engine is operated near the maximum power available. This setting provides sufficient thrust to fly the airplane at maximum lift over drag (L/D) altitude and at minimum specific fuel consumption (SFC). Some excess thrust is available at this condition for maneuver or control.

Because of the size required for transonic thrust, the engine is considerably oversized for subsonic cruise and holding operations. The power required is a very small percent of that available, and the resulting SFC is considerably higher than the minimum value. If variable area turbine and exit nozzle geometry are provided, the penalty for this oversizing can be reduced. At takeoff, the maximum available thrust far exceeds the minimum needed to meet the field length and second segment climb requirements. If takeoff is made at part power, the takeoff noise can be lower than either the augmented turbojet or turbofan, and still meet the field length and climb requirements.

- **The Augmented Turbojet**

This engine airflow is usually sized at the supersonic cruise condition to achieve maximum range by attain-ing the best compromise between engine weight, SFC, and airplane L/D. The engine may cruise with or without augmentation, depending on the engine weight technology and the level of SFC.

If the engine is sized to provide sufficient non-afterburning thrust to fly at maximum L/D altitude, the SFC will be near minimum, but the installed engine weight will be high. If a smaller engine is used, the same thrust can be achieved with some minimum afterburning; however, the SFC is high. If this smaller engine is operated with the afterburner not lit, the SFC is near minimum, but the thrust is too low to fly at maximum L/D altitude. The selected engine size is the best compromise of these considerations.

With the engine sized for cruise, the thrust margin for acceleration and climb within prescribed boom requirements can be adequately provided by additional augmentation from the afterburner. Because this engine has more thrust per pound of airflow at cruise and at transonic acceleration than the non-augmented turbojet, it is not as greatly oversized for subsonic flight as the non-afterburning version. Consequently, the subsonic cruise and holding SFC's are nearer the minimum values. If some variable geometry is provided the subsonic SFC versus thrust relationship can be adjusted somewhat to reduce the SFC at the required thrust as in the case of the non-augmented turbojet.

At takeoff, the non-afterburning thrust is usually more than adequate to meet the field length and climb-out requirements. There is not as much excess thrust for reducing noise as there is with the non-augmented turbojet and therefore the takeoff noise levels tend to be somewhat higher. Community noise levels may not necessarily be higher. This is discussed in more detail in Par. 11.3.3.2.

- **The Augmented Turbofan**

The mixed flow augmented turbofan is usually sized at the supersonic cruise condition to achieve maximum range. Partial augmentation is used during supersonic cruise. The thrust setting is selected to provide the best compromise between installed weight, L/D, and SFC. This condition occurs at a thrust level which permits cruising at very near maximum L/D altitude. At lower thrust levels, the reduction in SFC is not sufficient to compensate for the L/D reduction at the reduced altitude. At Mach 2.7 the turbofan cruise SFC is higher than either of the turbojets.

The engine is not as greatly airflow oversized at subsonic conditions as either of the two versions of the turbojet. Consequently, the subsonic operation occurs closer to the minimum SFC. In addition the turbofan has a fundamental propulsive efficiency advantage over the turbojets at subsonic speeds, which results in a lower SFC. At takeoff, a low augmentation power setting is required to meet the engine-out, second segment climb gradient. Nevertheless, the basically lower nozzle pressure ratio makes the takeoff noise less than with the augmented turbojet, using dry takeoff thrust.

The duct-burning (or unmixed) turbofan may be sized by transonic thrust requirements. The airport noise tends to be higher than the mixed fan because of the high velocity of the primary jet.

11.3.2 PERFORMANCE OF PROPOSED ENGINES

In order to compare the in-flight performance of the engines the airflow sizes have been adjusted to that required to match the Boeing SST configuration.

The matched sea level static airflow sizes of the several proposed engines are shown in Fig. 11-6. The airplane gross weight required for the RFP mission is also shown.

Performance comparisons of the matched engines at cruise, transonic, subsonic, and takeoff conditions follow. The performance of the JT11F-4 is not shown be-

	AIRPLANE G.W. LBS.	BASIC AIRFLOW LBS/SEC.	MATCHED AIRFLOW LBS/SEC.	MATCHED ENGINE WT. - LBS
TJ70	412,000	600	535	5520
GE4/J4C	430,000	475	415	7077
GE4/F6A	459,000	550	454	6060
JT11F-12	473,000	640	555	7750
JTF15A-1	470,000	630	551	7360

11-6 *Engine Sizes*

cause it is a heavy engine of a fixed size which does not match the airplane requirements.

11.3.2.1 Cruise Performance Results

The supersonic cruise installed SFC and thrust of the various engines are shown in Fig. 11-7. The matched thrust required on the Boeing configuration during cruise is marked on the curve for each engine.

The lowest SFC at Mach 2.7 is achieved with the TJ70 non-augmented turbojet which operates at slightly less than maximum cruise thrust. This engine has the lowest SFC because it does not have the augmentor pressure losses. The GE4/J4C augmented turbojet operates at about seven percent higher SFC at the cruise power setting. This condition occurs somewhere between maximum dry thrust and minimum augmented thrust. In practice this will require a mixture of augmented and dry power settings on the four engines or a change in cruise altitude with some slight range penalty.

The offered engines with the next higher SFC's are the P&W JT11F-12 and STF 188B turbofans, which operate at well above minimum augmentation and have SFC's about five percent higher than the J4C. The GE4/F6A operates at a slightly higher SFC. The SFC change with thrust is somewhat flatter with the fans than with the turbojet. The general level of SFC is higher because of the lower thermal efficiency of the fan cycle at Mach 2.7.

The dashed curve shows the performance improvement of the JT11F-12 with 2200 F TIT. However, since Pratt & Whitney has not offered this level of TIT for the SST, this performance has not been used for airplane evaluation.

11.3.2.2 Transonic Performance

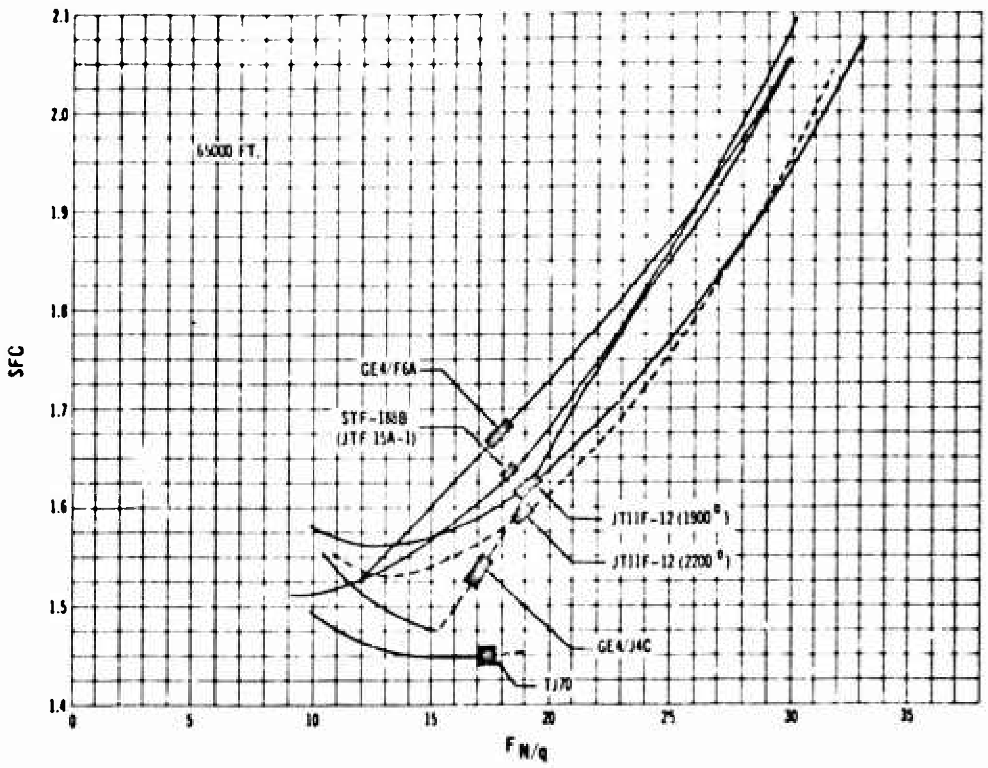
Figs. 11-8 and 11-9 show the transonic thrust and SFC for the proposed engines. All the engines offered have adequate thrust to meet the sonic boom limitations with a minimum of 0.3 thrust margin on a standard day. The TJ70 has the lowest SFC because it is non-augmented, while the GE4/F6A has the highest SFC because it is a fully augmented turbofan. The engine SFC during acceleration is a significant factor in the overall fuel consumed during the mission.

11.3.2.3 Subsonic Performance

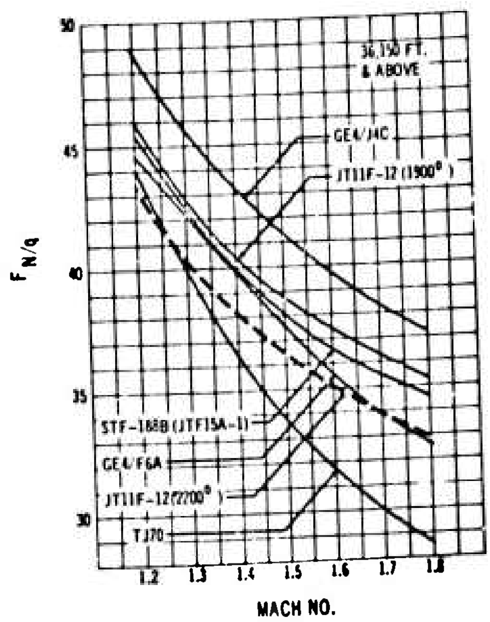
Fig. 11-10 indicates performance of the various engines for the cruise to alternate and holding conditions. In all cases the thrust required is much less than that available at minimum SFC. The turbofans provide the lowest SFC's for two reasons: (1) they tend to match nearer the minimum SFC, and (2) they have a basically lower SFC because of their better propulsion efficiency. The turbojets, both afterburning and non-afterburning, have about the same matched SFC's. It should be noted that even with the variable turbine nozzle feature, the TJ70 has the highest SFC at these conditions.

11.3.2.4 Takeoff Performance

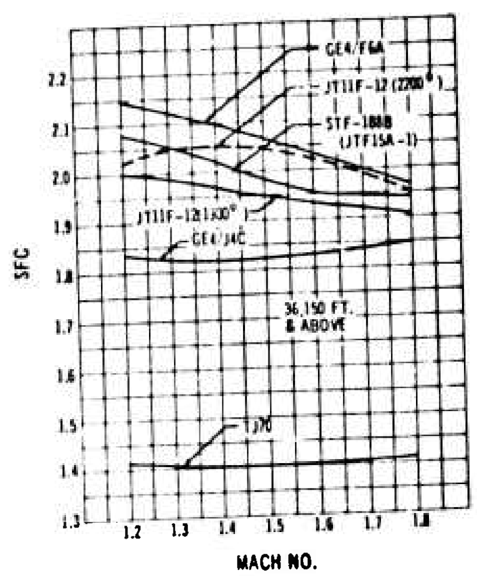
Fig. 11-11 shows the takeoff thrust for the engines, both augmented and dry, on a standard day. The minimum thrust required to meet the takeoff field length and second



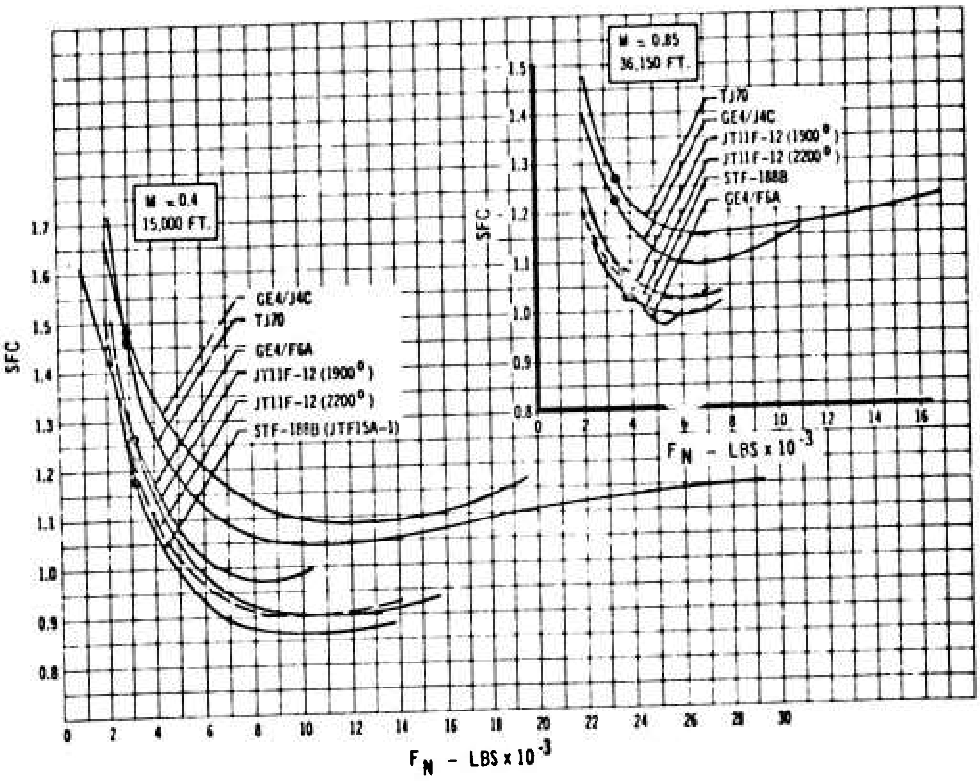
11-7 *Mach 2.7 Thrust vs Fuel Consumption*



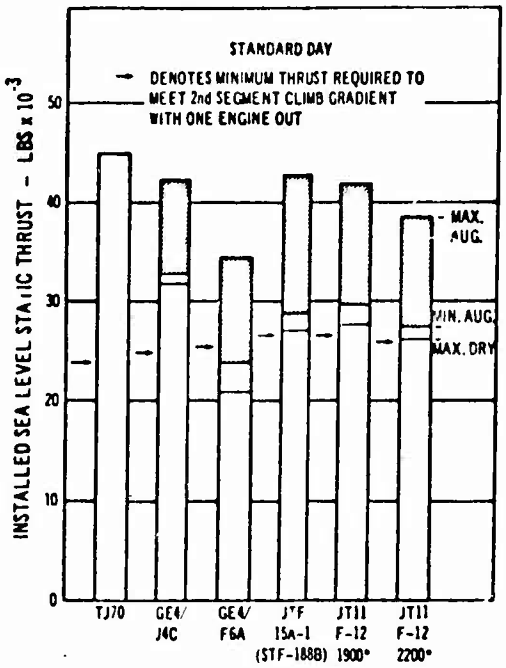
11-8 *Climb & Acceleration Thrust*



11-9 *Climb & Acceleration Fuel Consumption*



11-10 *Subsonic Thrust vs Fuel Consumption*



11-11 *Takeoff Thrust*

segment climb gradient, for the airplanes designed to meet the RFP mission, is shown by the arrow.

All engines except the GE4 F6A turbofan meet the takeoff thrust requirements at maximum dry power or below. The GE4 F6A turbofan will require partial augmentation. On a hot day, all the turbofans will require partial augmentation.

The airport noise at 1500 feet from the airplane, parallel to the runway, as a function of thrust of the engines is shown in Fig. 11-12. The comparison of community noise of the offered engines is discussed in Par. 11.3.3.2.

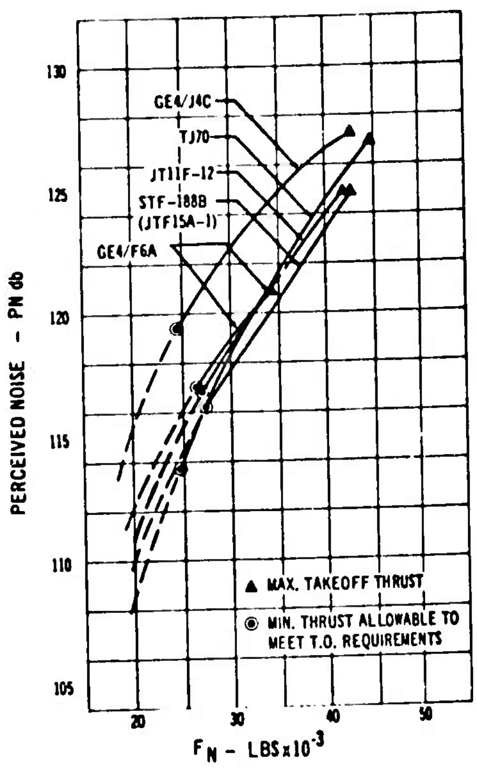
11.3.2.5 Installed Pod Drag

The installed pod drag of the proposed engines, sized to meet the RFP mission, is shown in Fig. 11-13 for the complete range of Mach numbers. At supersonic cruise the GE4 F6A engine has the lowest drag. The C-W TJ70 has the highest cruise drag.

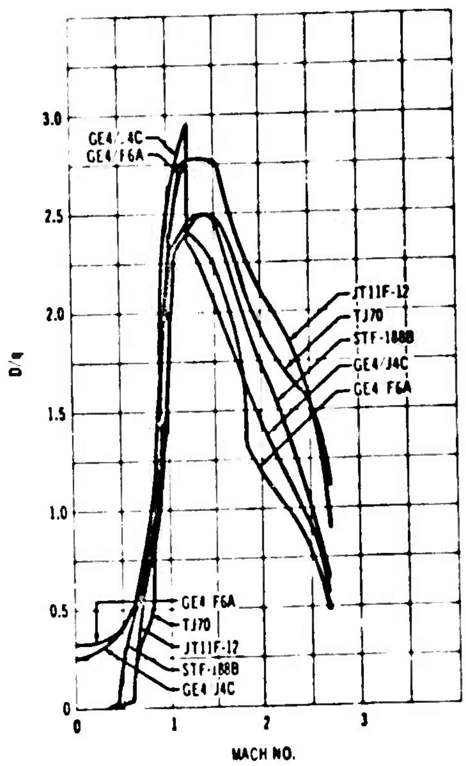
The transonic drags of the various pods are also shown in the same figure.

11.3.3 COMPARATIVE AIRPLANE PERFORMANCE

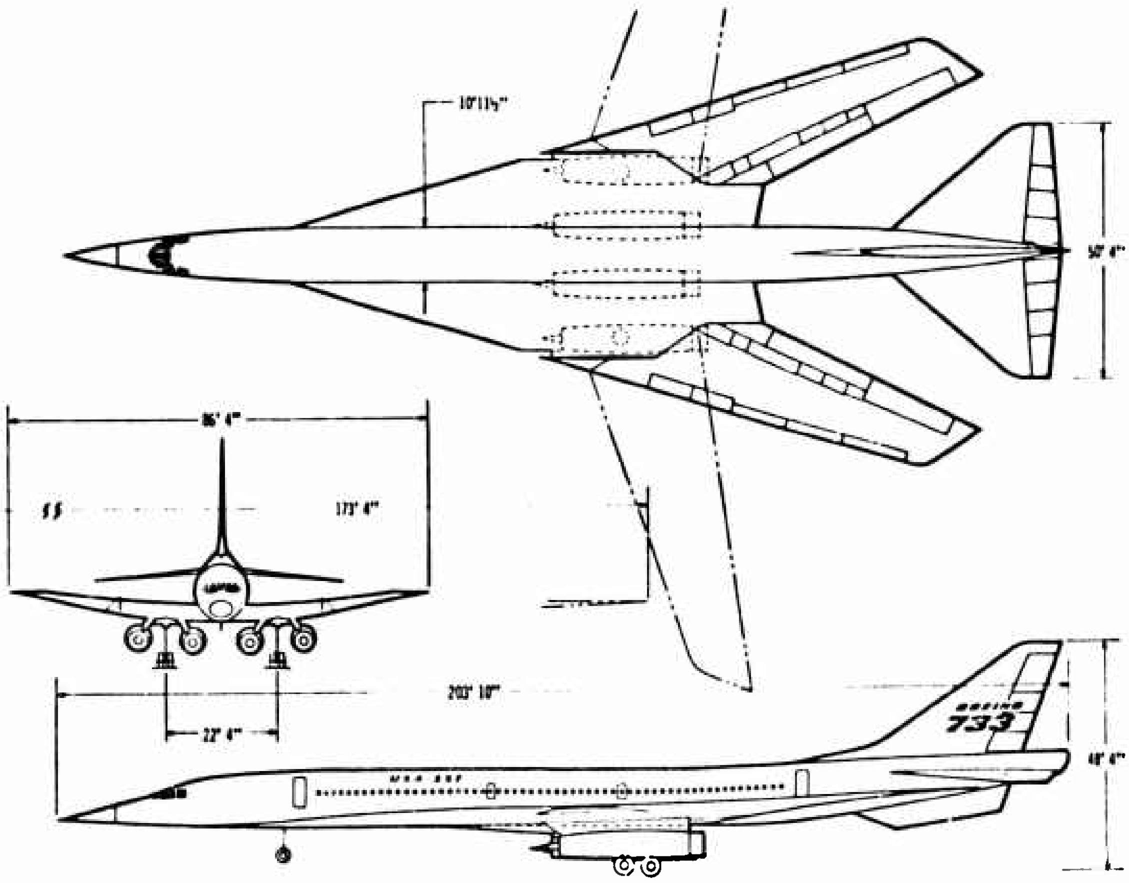
The optimum airplane performance which results from the offered engines matched to the Boeing SST configuration is discussed below. The airplane configuration which was used in these performance studies is shown in Fig. 11-14. This configuration was used to obtain relative performance comparisons with all of the offered engines. The changes in pod weight, drag, and installed performance with the various engines were accounted for. For this study the wing area was limited to a minimum of 4684 square feet by configuration considerations. The maximum wing loading was limited to 100 pounds per square foot (psf) to meet the 165-knot takeoff requirement in the RFP.



11-12 *Airport Noise Levels 1500 Ft. From Centerline of Runway*



11-13 *Installed Pod Drag*



11.14 Airplane Configuration Boeing Model 733

11.3.3.1 Gross Weight Comparisons

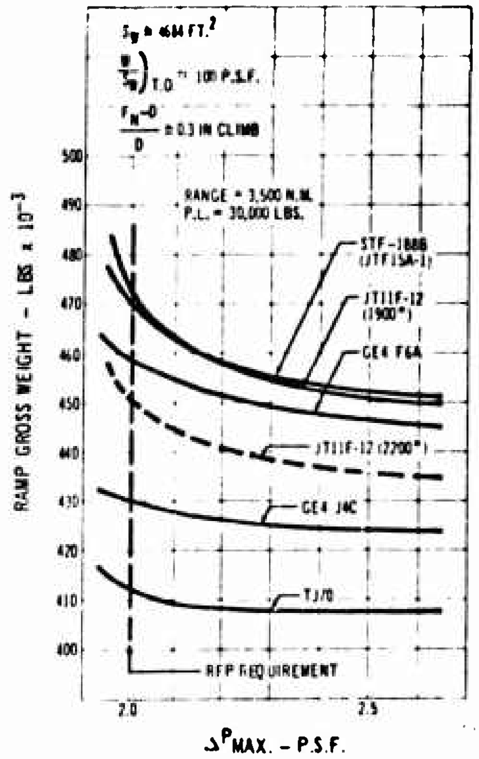
The most significant comparison is the airplane gross weight required to perform the 3500 nautical mile range, 30,000 pound payload mission at the selected cruise Mach number of 2.7, with the sonic boom overpressure limit of 2.0 psf. This comparison is shown in Fig. 11-15. At the overpressure limit the lowest gross weight is provided by the TJ70 non-augmented engine at a gross weight of 412,000 pounds. The GE4 J4C engine results in an airplane gross weight of 430,000 pounds. The JT11F-4 match is not shown, but the gross weight is well over 500,000 pounds.

Fig. 11-16 summarizes some of the pertinent data from each engine-airplane match for the RFP mission. All matched engine sizes are within the scaling range of the offered engines.

11.3.3.2 Noise Considerations

- **Takeoff Noise**

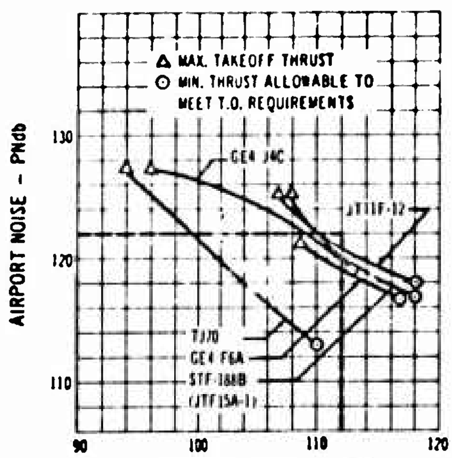
The extra thrust available at takeoff with these engines allows a trade between the takeoff ground roll noise and the noise over the community. Fig. 11-17 shows this trade based on Boeing-calculated noise characteristics for the offered engines. Higher takeoff thrusts result in higher airport noise but lower community noise because the airplane arrives over the community at a higher altitude. In all cases the community noise is shown for a position three miles from the brake release point with thrust reduced to that required for 500 feet per minute rate of climb. The airport noise is shown for a distance of 1500 feet parallel to the runway. If the allowable airport noise is set at 120-122 PNdb, all the engines will yield community noise levels less than 112 PNdb, the limit set in the RFP.



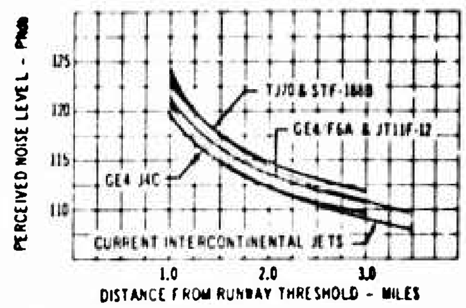
11-15 *Airplane Design Gross Weight vs. Maximum Sonic Boom Overpressure*

ENGINE	TJ70A-4	GE4/J4C	GE4/F6A	JT11F-12 (1900°)	JT11F-12 (2200°)	STF-188B (JTF15A-1)
SLS Airflow	535	415	454	555	491	551
Max. T.O. Thrust (SLS Std. Day)	45,200	42,800	34,500	42,300	38,700	43,100
Engine Weight (Incl. Nozzle & Thrust Reverser)	5,520	7,077	6060	7,750	6,500	7,360
Pod Weight	8,100	8,872	8480	10,249	9,020	9,891
Transonic Thrust Margin $\frac{(F_N - D)}{D}$	0.3	.54	0.349	0.3	0.3	0.3
T. O. Field Length at Max. T. O. Thrust	3827	4250	6,515	5,300	5,335	5,250
Total Fuel						
Take-Off & Climb	54,100	51,000	70,684	68,011	67,463	70,110
Cruise	116,700	131,155	140,837	144,735	133,800	144,484
Descent & Reserves	34,850	35,345	31,638	34,415	32,183	33,636
Airplane Ramp Gross Weight	412,000	430,000	459,000	472,000	450,000	469,000

11-16 *Matched Engine Data*



11-17 *Airport vs. Community Noise*



11-18 *Landing Approach Noise*

- **Landing Noise**

Landing noise is a function of the power required to maintain the landing approach glide slope. Fig. 11-18 shows the noise levels for the offered engines, assuming a three degree glide slope with wings fully extended, at various distances from the runway threshold. As shown, the landing noise level for the SST will not be higher than for existing intercontinental jet aircraft.

11.4 Other Considerations

11.4.1 COMPARATIVE INSTALLATION FEATURES

11.4.1.1 Engine Starting

Starting requirements vary considerably among the engines considered for the SST (Fig. 11-19).

The STF 188B and the TJ70 may be started with the type of starters and carts that are currently in commercial transport usage. Although the TJ70 engine is a single rotor, high inertia engine, the exceptionally high fired torque characteristics quoted by Curtiss-Wright (Fig. 11-20) allow starting with a relatively small starter and cart.

The General Electric J4C and F6A engines require a larger starter than the above engines. The larger starter requires two of the presently used GTCP-85 carts or one GTCP-100 series cart. The Pratt & Whitney JT11F-12 engine requires a large starter and two of the presently used GTCP-100 series ground carts.

	GE4/J4C	GE4 F6A	JTF15A-1 (STF188B)	JT11F-12	TJ70
NUMBER OF SPOOLS	1	1	2	1	1
POLAR MOMENT OF INERTIA OF THE ROTOR (SLUG-SQ. FT.)	100	88.8	15	60.2	145
IDLE RPM	3450	4000	4850	3400	3140
STARTER TO ROTOR GEAR RATIO	1.78:1	2:1	.884:1	1.155:1	2:1
STARTER TYPE*	ATS 200	ATS 200	ATS 100	ATS 400	ATS 100
STARTER PLUS VALVE WEIGHT (POUNDS)	75	75	38	75	38
CART TYPE*	GTCP-85	GTCP-85	GTCP-85	GTCP-100	GTCP-85
NO. CARTS REQUIRED	2	2	1	2	1
START TIME, S.L. STANDARD DAY (SECONDS)	37	35	21	31	33
*AIRESEARCH MFG. CO., STARTER AND CART MODEL TYPE DESIGNATIONS					

11-19 *Starter Requirements*

11.4.1.2 Nacelle Cooling

Generally the fan engines have a lower engine case temperature and thus present a less severe nacelle cooling problem than do the turbojet engines. This results from the excellent insulation provided by the relatively cool fan air which shrouds the primary engine. Fan engine case temperature in the diffuser case area will be approximately 400°F cooler than the equivalent area in the jet engine.

The accessory area will be compartmented for all of the engines to reduce the soaking temperature of the accessories. The compartment will be shielded from the engine case. The jet engines will require more compartment insulation than the fans.

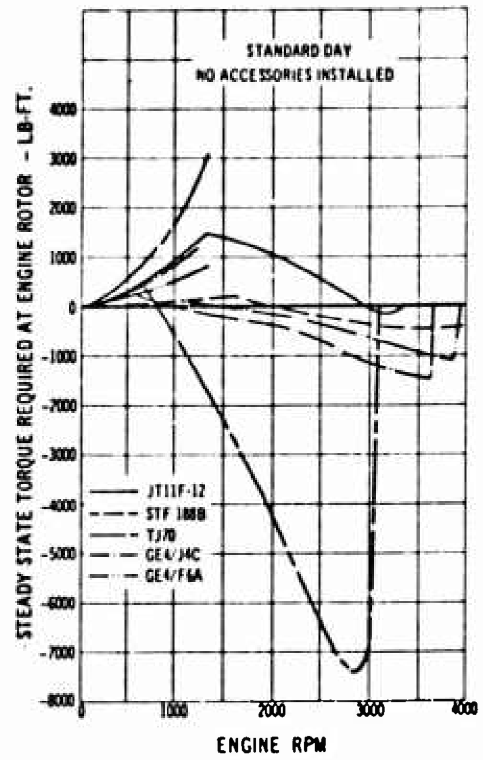
All engines require cooling air for the nozzle and reverser actuators.

11.4.2 ENGINE AVAILABILITY

11.4.2.1 Development Status

None of the engines being offered by the various engine manufacturers for the SST is currently under development.

The Pratt & Whitney JT11F-4 engine is closest to



11-20 *Engine Starting Torque Comparison*

an actual engine under development since it is basically a fan version of the JT11 engine, which is currently in the advanced development stage. The P&W STF 188B, however, and the JT11F-12 are new engine developments. Component research for these engines is progressing in burner development, ejector nozzles, and high temperature turbine technology.

The General Electric turbojet (GE4 J4C) and turbofan engines (GE4 F6A) are new engines designed specifically for the SST, using technology gained from the J93 program. The compressor section is a scaled version of an existing high stage loading unit which has been successfully run. General Electric has also been actively engaged in high temperature turbine work, nozzle and reverser design, and in afterburner technology.

The C-W TJ70 is a new engine designed specifically for the SST. Curtiss-Wright has been engaged in considerable component development work in transpiration cooling of turbine blades and high stage loading compressors to provide support to its concept.

Fig. 11-21 shows the months from go-ahead to preflight rating test (PFRT) and certification for the offered engines. The GE4 J4C and P&W JT11F-4 engines meet the airplane development schedule.

11.4.2.2 Production Schedules

With respect to engine certification and delivery of engines for the first production airplanes, the GE4 J4C and P&W JT11F-4 come closest to meeting the airplane requirements.

11.4.2.3 Engine Manufacturers' Capabilities

Pratt & Whitney

- **Previous Record**

Pratt & Whitney has an excellent record of producing high quality engines on schedule. Boeing's experience on the B-52, KC-135, and 707 commercial programs with

	MONTHS FROM GO-AHEAD (MAY 1 1964)	
	75 HR PFRT	CERTIFICATION
GENERAL ELECTRIC		
GE4/J4C	36	60
GE4 F6A	42	66
PRATT AND WHITNEY		
JT11F-4	30	60
JT11F-12	42	72
STF-188B	42	72
CURTISS-WRIGHT		
TJ-70	44	65

11-21 *Engine Development Schedules*

Pratt & Whitney's J57, TF33, JT3D, and J75 engines has generated a high level of confidence in the capability of Pratt & Whitney to produce engines that are efficient and reliable and to meet their commitments with respect to schedules, performance, and weights.

- **Engineering and Management**

The engineering and management personnel at Pratt & Whitney responsible for the development and production of the SST engine are the same personnel that were responsible for Pratt & Whitney's other successful programs. This continuity of experienced personnel which exists at Pratt & Whitney produces a depth of technical talent for which there is no substitute. The Pratt & Whitney engineering department also has the capability of solving problems quickly and efficiently that may arise in the field.

- **Test Facilities**

Pratt & Whitney has the world's largest privately owned installation for the development testing of air-breathing power plants. At these facilities compressors, burners, turbines, and full scale engines are run at speeds up to Mach 3.2 and altitudes up to 100,000 feet. Thirteen test cells at the facility are provided with air at the required pressures and temperatures for simulating ram air inlet conditions. Evacuated exhaust conditions are also provided. Three of the test cells are altitude chambers capable of testing full-scale engines at high altitudes. Total air-flow capacity is over 700 pounds per second. Exhauster capacity varies from 50 pounds per second at two psia to 550 pounds per second at 15 psia. Supplementing the main laboratory in the same area are a compressor laboratory, a fuel system laboratory, and two sea level test cells. The East Hartford plant test complex contains 28 full-scale engine sea level test stands for engine development and qualification testing. The Florida facility also has altitude and Mach number simulation capability for evaluating components of large size.

General Electric

- **Previous Record**

Boeing has had extensive experience with General Electric engines on the B-47 (J47) engine. This was one of the first jet engines developed in this country and resulted in the development of a bomber-type aircraft with speed capability in excess of the fighter aircraft of that period, a bomber which is still in first-line service.

Boeing has had no experience with more recent General Electric engines. The General Electric record on the J79 engine and CJ805 from all reports is very good and probably is what can be expected for the GE4 J4C. The J79 is installed in the aircraft which holds most of the world's altitude and Mach number records and was the first Mach 2.0 engine developed in this country. General Electric has been developing the Mach 3.0, J93 for the B-70 program. General Electric has considerably more supersonic operational engine experience than any other company. Reports on the field service record of General Electric with the CJ805-3 engine have been favorable both with respect to the engine and the service personnel.

- **Engineering and Management**

General Electric engineering and management are capable of doing an excellent job of developing the engine for the supersonic transport. It is felt that the overall experience and technical capability of the General Electric engineering staff is very high and more than adequate to perform the development job required for the GE4/J4C. The technology involved is an extension of the J79 and J93 experience, which will be applied directly to the development of the engine. Adequate technical personnel are available to concentrate on this program.

- **Test Facilities**

General Electric has several large test cells used for development and qualification of the J79 and J93 engines. This company has a large air supply and exhauster capability and numerous component development rigs. The General Electric ram test facility, currently being used to test the J93, can test engines at conditions from sea level static to Mach 3.0 at 70,000 feet. The test facility drive unit consists of a 250,000 cubic feet per minute compressor, a 32,000 hp synchronous motor, and a 3200 hp steam turbine. This unit is combined with a 100 million Btu per hour heater to create airflows of the required temperature.

Curtiss-Wright

- **Previous Record**

Boeing has no actual experience with Curtiss-Wright jet engines. Very limited experience was gained with the Curtiss-Wright turboprop engine on the XB-47D airplane. This airplane was built as a flying test bed and was not flown extensively. The only jet engine produced in quantity by Curtiss-Wright was the J65 which was a development from the British Sapphire engine.

- **Engineering and Management**

The Curtiss-Wright engineering and management staff has not been involved in a jet engine development program in the past five years. However, Curtiss-Wright has an outstanding but limited number of design personnel who do understand the technical problems of the SST.

- **Test Facilities**

Curtiss-Wright has several sea level test cells capable of testing engines up to 50,000 pounds of thrust. Component test rigs include five airblowing test stands for combustion chamber and related component testing. Curtiss-Wright has proposed that a large share of the full-scale component and engine testing be conducted at outside private or government-owned facilities.

11.4.3 RELIABILITY AND MAINTAINABILITY

The requirement for advanced technology to make the supersonic transport a success is well established. The need for high reliability and maintainability is also unquestioned.

It is very difficult, this early in the design stages of a new engine program, to rate the various engine design approaches, but in the area of reliability and maintainability, simplicity is certainly of major importance. Since high turbine temperatures are required, advanced cooling techniques must be employed, and the hot parts must be readily accessible. These two requirements, simplicity and accessibility, point in the direction of the turbojet engine. In attempting to evaluate the potential reliability and maintainability of the offered engines, certain fundamental design features in each engine stand out.

The C-W TJ70 engine is a simple, single spool, ry turbojet which incorporates the variable turbine ozzle in order to obtain competitive subsonic cruise iFC's. The transpiration cooling technique involves a abricated turbine bucket construction which has very imited test time. However, the simple turbojet lends tself to easy access for maintenance and inspection of he hot section. The high aspect ratio compressor blades ire probably susceptible to foreign object damage.

The General Electric turbojet is a simple, single spool engine with a conventional afterburner. The turbine film cooling technique is new, but has been under development for some time, and promises to reduce the metal temperatures to below present commercial jet levels. The experience gained on the J79 and J93 programs will be applicable. This turbojet lends itself to easy access for inspection and maintenance of critical parts, particularly in the combustion and turbine areas.

The turbofan engines, because of their annular fan ducts, tend to decrease the accessibility of the turbine area for inspection and maintenance.

The GE4 F6A fan has a cool duct over the turbine area and the burning is done in the mixed stream downstream of the turbine. The turbine cooling technique is similar to that in the GE4/J4C.

The Pratt & Whitney turbofan engines all involve annular fan combustion chambers which create a hot duct over the primary combustion and turbine sections. Otherwise, the engines are based on J58 and JT3D design experience. The quoted cruise turbine temperatures are 300 F lower than the General Electric temperatures, but the cooling technique is not as advanced (convective rather than film cooling). The ultimate turbine and hot parts life is a function of cooling design (metal temperatures) as well as flame temperatures.

The past record of the engine manufacturer certainly should be considered in evaluating the probable maintainability and reliability of an offered engine in production. In this respect, it is felt that Pratt & Whitney and General Electric have demonstrated their ability to attain reliability in their current commercial engine programs, while Curtiss-Wright has had no experience in the commercial turbine engine field.

It is expected that more detailed information regarding the reliability and maintainability of the offered engines will be contained in the engine proposals to be submitted on January 15, 1964.

11.4.4 ENGINE COSTS

The following estimated production prices and development costs have been received from the engine manufacturers on December 23, 1963, for production quantities of 1200 engines (200 se. plus spares). The unit price does not include any amortization of development costs.

Engine	*Manufacturer*	*Unit Price*	*Est. Dev. Costs*
GE4 J4C	General Electric	$ 950,000	$325,000,000
GE4 F6A	General Electric	1,025,000	375,000,000
TJ70	Curtiss-Wright	820,000	333,600,000
JT11F-4	Pratt & Whitney	1,664,000	350,000,000
JT11F-12*	Pratt & Whitney	1,824,000	500,000,000
JTF15A-1 (STF 188B)	Pratt & Whitney	1,824,000	500,000,000

*The price shown is for a version of the JT11F-12 engine limited to a continuous cruise Mach number of 2.7. P&W refers to this engine as the Boeing version of the JT11F-11 engine.

11.5 Overall Evaluation

A simplified scoring system was used to evaluate the offered engines. The factors considered in selecting the optimum engine and the weighted scoring system are shown below:

Factor	Scoring Value A	B	C	D	E
Airplane Performance	25	20	15	10	5
Engine Credibility	20	15	10	5	0
Engine Contractor Capability	20	15	10	5	0
Reliability & Maintainability	15	10	5	0	0
Engine Production Costs	10	5	0	0	0
Engine Availability & Schedule	10	5	0	0	0

The engine evaluation is shown below. The total score shows the GE4 J4C engine to be the primary choice with a score of 90 out of a possible 100. The remaining engines have the same total score, indicating that their suitability as alternate engines is about equal.

Factor	TJ70	GE4/J4C	GE4/F6A	JT11F-12	JTF15A-1
Airplane Performance	25	20	15	15	15
Engine Credibility	10	20	20	20	20
Engine Contractor Capability	10	15	15	20	20
Reliability and Maintainability	15	15	10	10	10
Engine Production Costs	10	10	10	5	5
Engine Availability	5	10	5	5	5
Total Score	75	90	75	75	75

11.6 Development of Selected Engine (RFP 3.2.9.1)

The General Electric GE4 J4C engine will be developed by the engine manufacturer to meet the guaranteed performance under all flight conditions and to meet the specified reliability and maintainability goals established for the SST. The test plan leading to engine certification, the engine production schedule, the growth potential of the engine, and the reliability and maintainability aspects of the engine are discussed in this section.

11.6.1 ENGINE DEVELOPMENT PLAN

Significant milestones of the development plan as proposed by the engine manufacturer are:

	Months After Go Ahead
• First dry engine run	17
• First complete engine run	23
• Flight test status qualification complete	36
• Type certification test complete	60

11.6.1.1 Ground Test

Sufficient ground testing is required to obtain engine performance equal to or exceeding guaranteed performance. The engine mechanical design and structural integrity will be proven. Endurance as well as cyclic testing will be performed under controlled inlet pressure and temperature conditions (altitude chamber and heated air tests) to simulate as much of the flight envelope as possible. Testing with various inlet distortion patterns will be performed to satisfy performance guarantees with respect to allowable inlet distortion. The detailed information on the number of test engines, manpower and facility requirements, and the test schedule is not available prior to the submission of General Electric's firm proposal.

It is planned that the engine contractor and the airframe contractor will conduct integrated propulsion pod tests at the Arnold Engineering Development Center to confirm compatibility of the exhaust nozzle-engine inlet combination.

11.6.1.2 Flight Test

The General Electric Company does not plan to flight test the GE4 J4C engine prior to its installation on the prototype SST. Boeing concurs in this, because no suitable aircraft is available on which the engine could be tested through the full flight spectrum in a manner which would be compatible with the SST. Subsonic flight testing does not appear to warrant the expense involved. Supersonic flight testing of the engine on an airplane other than the SST is of questionable value. Flight acceleration and cruise can be simulated on test stands under conditions which may be more realistic than on an airplane where the propulsion pod does not have exactly the same relationship to the airframe as it will on the SST.

The engine used for prototype airplane flight testing is planned as a pre-flight rating tested (PFRT) engine. Flight testing of the prototype airplane and the engine will occur simultaneously. The plan for this testing is covered in Section 8.

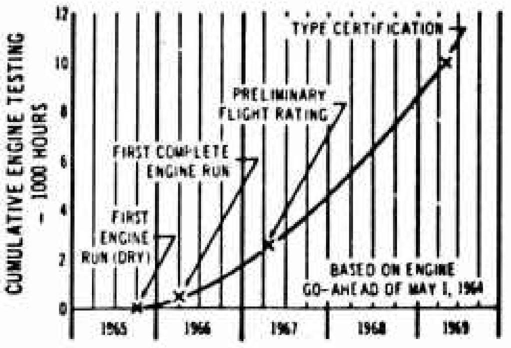
11.6.1.3 Certification Program (RFP 3.2.9.1c)

The cumulative engine development test hours leading to type certification are shown in Fig. 11-22. Scheduled dates for preliminary flight rating and type certification are noted. A total of approximately 10,000 test hours will be run to obtain type certification of the engine, including 4250 hours of heated inlet testing and 250 hours of altitude performance testing.

Further details of the General Electric certification test program are not available prior to the submission of General Electric's firm proposal.

11.6.2 ENGINE PRODUCTION SCHEDULE (RFP 3.2.9.1d)

Engine delivery schedules during the development and early production period are shown in Fig. 11-23. General Electric has given firm dates for PFRT, engine certification, and delivery of the first four prototype engines and the first four production engines. The remainder of the engine delivery schedule is as required by Boeing to match the airplane production schedule. This schedule is based on four engines per airframe plus 50 percent spares during the prototype flight test phase and 25 percent spares



11-22 *Engine Certification Test Program—GE4/J4C*

during the certification program. Also included are three engines required to support the Boeing propulsion system ground test program as defined in Section 8.

11.6.3 GROWTH POTENTIAL

Engine growth is required to allow the payload-range capability of this aircraft to be extended following the initial airframe-engine development program. General Electric has identified a two-phase growth plan for the GE4 J4C engine.

Phase I yields a four and one-half percent decrease in SFC at minimum augmented power at cruise by means of a 10 percent larger augmentor and nozzle diameter and a 50° F. increase in turbine inlet temperature. The augmentor efficiency is increased about four percent, due to a lower inlet velocity obtainable with the larger augmentor. Takeoff and transonic performance are essentially unchanged.

Phase II gives a six percent increase in takeoff net thrust, a 10 percent increase in transonic net thrust and a 12 percent increase in cruise net thrust, all at approximately the same SFC, by means of component performance improvements. The changes required for Phase II are in addition to the Phase I changes described above. The engine airflow will be increased approximately five to seven percent by readjusting the compressor blade angles and turbine nozzle flow area. Also, another 50° F. increase in turbine inlet temperature will be used.

If a different growth sequence becomes desirable because of test experience, the engine design can be modified to achieve other performance characteristics.

11.6.4 ENGINE RELIABILITY AND MAINTAINABILITY (RFP 2.25.6; 2.11)

The improved technology on reliability and the records of engine experience will significantly reduce reliability problems on the SST engine program.

Reliability is a product attribute that can be quantitatively specified, analyzed, predicted, and measured. For the SST engine, a high level of reliability, achieved early in the development phase, is a major objective. This recognizes economic and safety consequences, effects of operating environments, required engine time between overhauls, and reduced development time, due to the lack of military experience on a comparable engine operating in the same flight regime.

Maintainability is closely related to reliability since both influence important cost indices like maintenance hours per airplane flight hour as well as inspection and overhaul. The significance of reliability and maintainability requires that they both be emphasized.

11.6.4.1 Basic Approach

Reliability and maintainability programs at General Electric involve the establishment of goals, the predicting of engine and component capabilities, design of tests, measurement of test and operational results, and introduction of improvements. Reliability design goals for each of the subsystems and components are established by the use of a reliability apportionment system. Maintainability goals are similarly established for the design and development work.

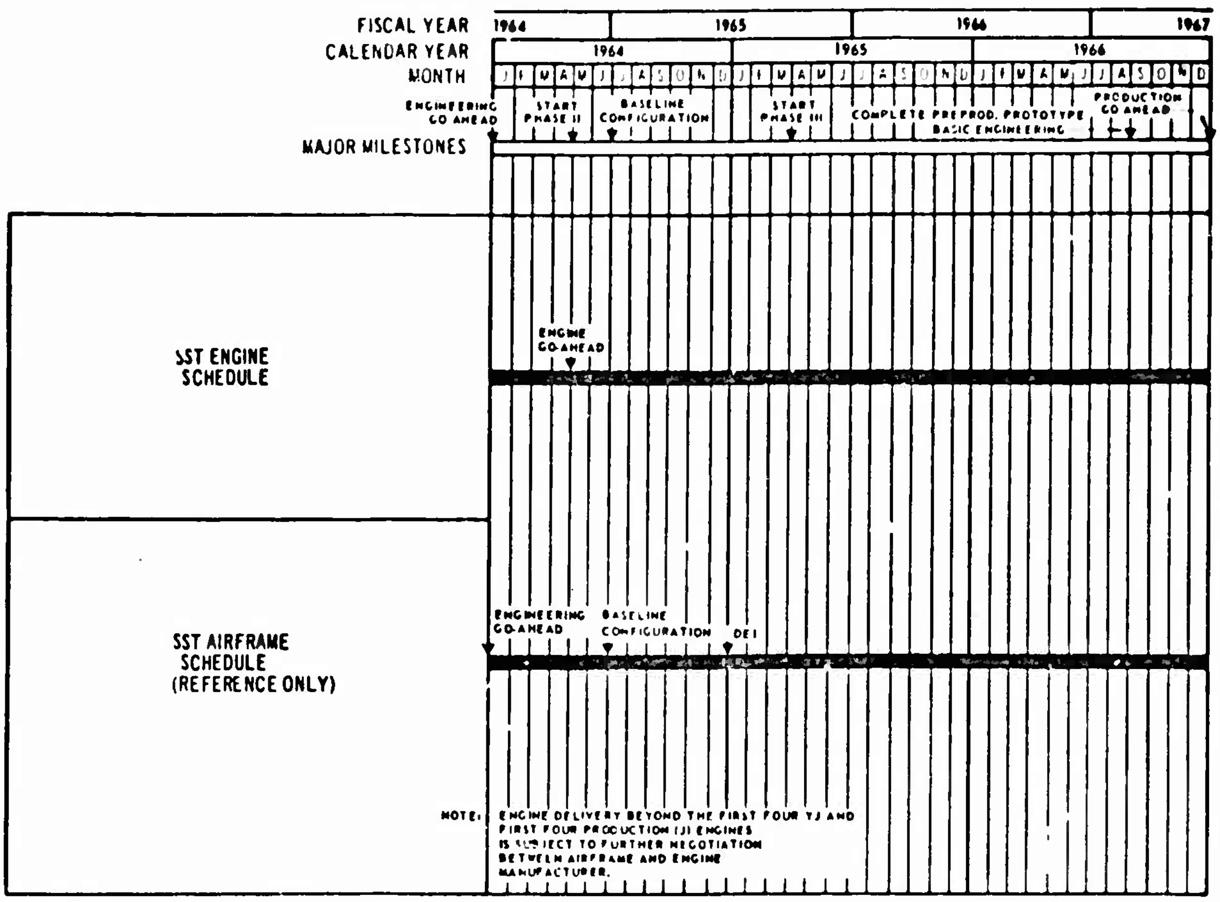
Detailed design reviews on reliability and maintainability will continue for all components and systems of the engine during various phases of the program. The key objective is to uncover and eliminate potential problem areas.

11.6.4.2 Proposed Objectives (RFP 2.25.7)

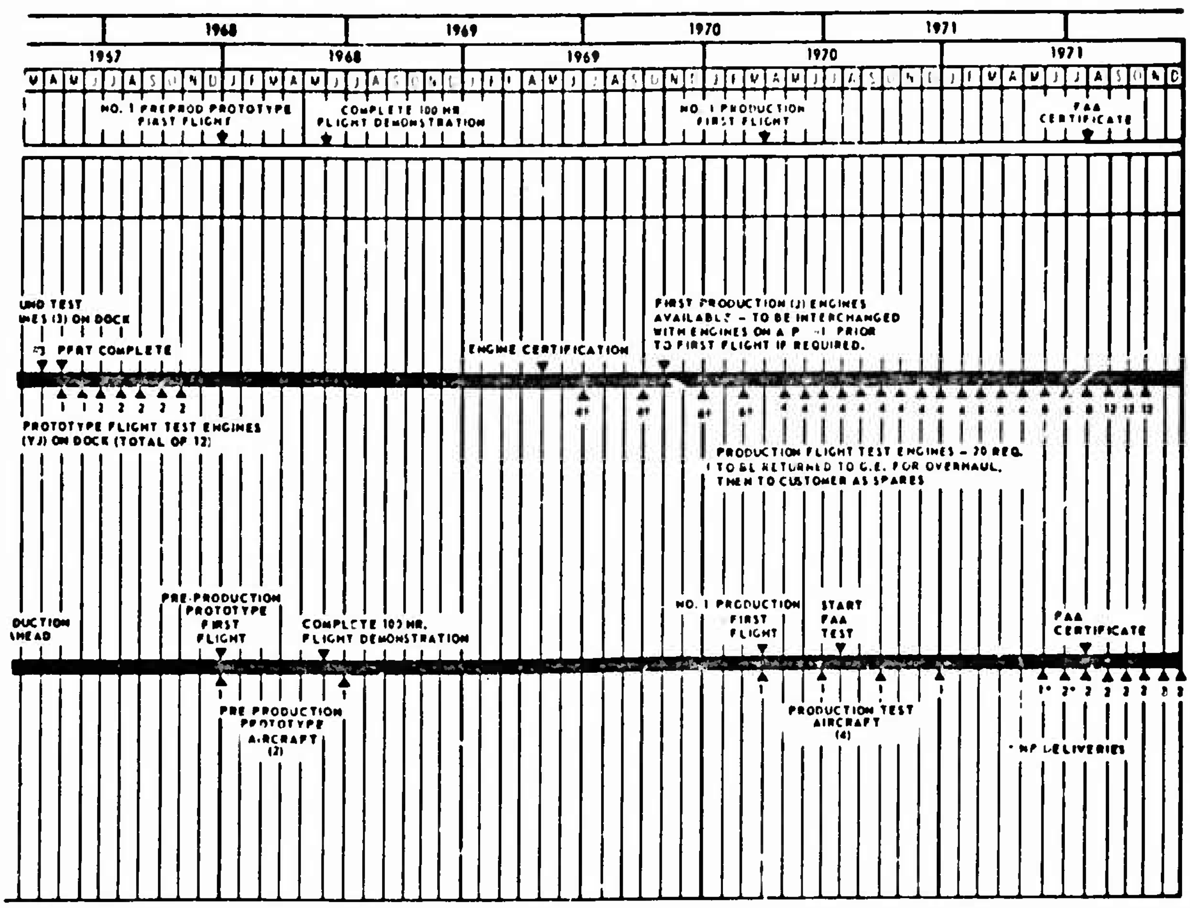
A reliability and maintainability program requires meaningful goals. A study has been performed by General Electric to obtain clear and concise product requirements with respect to reliability and maintainability. The critical factors chosen are believed to be optimum for an augmented engine which must operate in the flight environment of a supersonic transport. The analysis revealed that no single assessment factor would provide a true evaluation, so two reliability and three maintainability factors were selected, as shown below.

	OBJECTIVES	
	Start of Airline Service	*Growth During Airline Service*
• Mean Time Between In-Flight Shutdowns	3500 hours	10,000 hours
• Mean Time Between Premature Engine Removals	750 hours	5,000 hours
• Overhaul Manhours	......	2,500 hours
• Mean Time Between Inability to Obtain or Sustain Augmenting Power	700 hours	3,000 hours
• Maintainability Index (Applied Manhours per Flight Hours)	1.1	0.50

In addition to the above, a time between overhaul of 600 to 1000 hours is planned at the start of airline service. The eventual goal is 4000 hours, with no midpoint inspection required.



11-21 *Engine Delivery Schedule*


1968
1969
1970
1971
1967
1968
1969
1970
1971
NO. 1 PREPROD PROTOTYPE FIRST FLIGHT
COMPLETE 100 HR FLIGHT DEMONSTRATION
NO. 1 PRODUCTION FIRST FLIGHT
FAA CERTIFICATE
PPRT COMPLETE
ENGINE CERTIFICATION
FIRST PRODUCTION (2) ENGINES AVAILABLE – TO BE INTERCHANGED WITH ENGINES ON A P -1 PRIOR TO FIRST FLIGHT IF REQUIRED.
PROTOTYPE FLIGHT TEST ENGINES (YJ) ON DOCK (TOTAL OF 12)
PRODUCTION FLIGHT TEST ENGINES – 20 REQ. (TO BE RETURNED TO G.E. FOR OVERHAUL, THEN TO CUSTOMER AS SPARES
PRE-PRODUCTION PROTOTYPE FIRST FLIGHT
COMPLETE 100 HR. FLIGHT DEMONSTRATION
NO. 1 PRODUCTION FIRST FLIGHT
START FAA TEST
FAA CERTIFICATE
PRE PRODUCTION PROTOTYPE AIRCRAFT (2)
PRODUCTION TEST AIRCRAFT (4)
DELIVERIES

VOLUME A-VI

PROPULSION

12.0 PROPULSION SYSTEM PERFORMANCE (RFP 2.25; 3.2.9)

The installed performance of the GE4 J4C engine is presented in this section. The engine data include the effects of inlet pressure recovery, horsepower extraction, and air bleed. The total pod drag, except for skin friction, corrected to free stream conditions is also included in this section.

12.1 Inlet Total Pressure Recovery

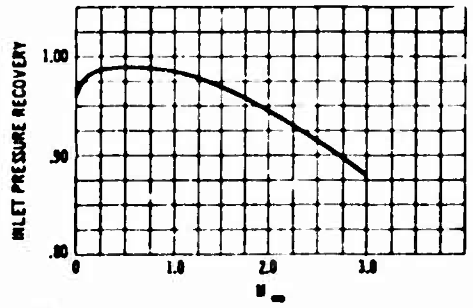
The inlet matched with the GE4 J4C engine is an axisymmetric inlet. The inlet total pressure recovery versus free stream Mach number used in computing engine performance is shown in Fig. 12-1. Inlet total pressure recovery is an average of the inboard and outboard engine locations. Five percent of the inlet airflow is bled from the centerbody and inner cowl surfaces for boundary layer control to achieve the level of inlet total pressure recovery shown. Substantiation and description of these performance figures is covered in Section 3.

12.2 Exhaust Nozzle Performance

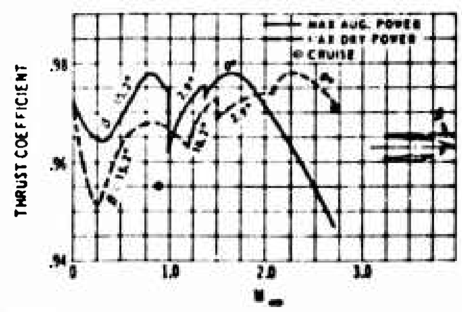
The estimated nozzle internal thrust coefficient supplied by the engine manufacturer is shown in Fig. 12-2 for various flight conditions. The nozzle coefficient shown includes the ram drag of the secondary air but does not include the external boattail drag. The nozzle boattail drag is discussed in Par. 12.4.

12.3 Power Extraction and Air Bleed

Power extraction is required for aircraft hydraulic and constant speed drive systems. A constant 100 horsepower has been extracted per engine to account for these requirements. It is recognized that during cruise conditions this is high; during other phases of flight this figure is, in general, adequate. There are short, high power extraction periods during which the figure is low. In terms of an overall flight, the 100 horsepower is a conservative value.



12-1 *Inlet Performance*



12-2 *Nozzle Internal Thrust Coefficient*

High pressure compressor bleed air is required for cabin air conditioning. Fig. 12-3 lists the engine compressor bleed extraction per engine for various flight conditions.

AIRPLANE OPERATING CONDITION	AIR BLEED EXTRACTION LB/SEC/ENGINE
TAKEOFF	1.7
CRUISE	0.9
HOLDING M_∞ = 0.4 15,000 FEET	1.5
CRUISE TO ALTERNATE M_∞ = 0.8, 16,089 FEET	1.2
CLIMB AND ACCELERATION	1.3

12-3 *Airbleed Requirements*

12.4 Installed Propulsion Pod Drag

Cowl wave drag, inlet spillage drag, cowl lip suction force due to spillage, inlet bypass drag, inlet boundary layer bleed drag, nozzle boattail drag, and strut drag were computed. Pod and strut friction drag are included in airplane friction drag and thus are not included in propulsion pod drag. The total installed pod drag coefficient is shown in Fig. 12-4 for the GE4 J4C engine as a function of free stream Mach number. All drag coefficients are based on the inlet lip frontal area. The various contributing drags are also shown.

12.4.1 COWL WAVE DRAG

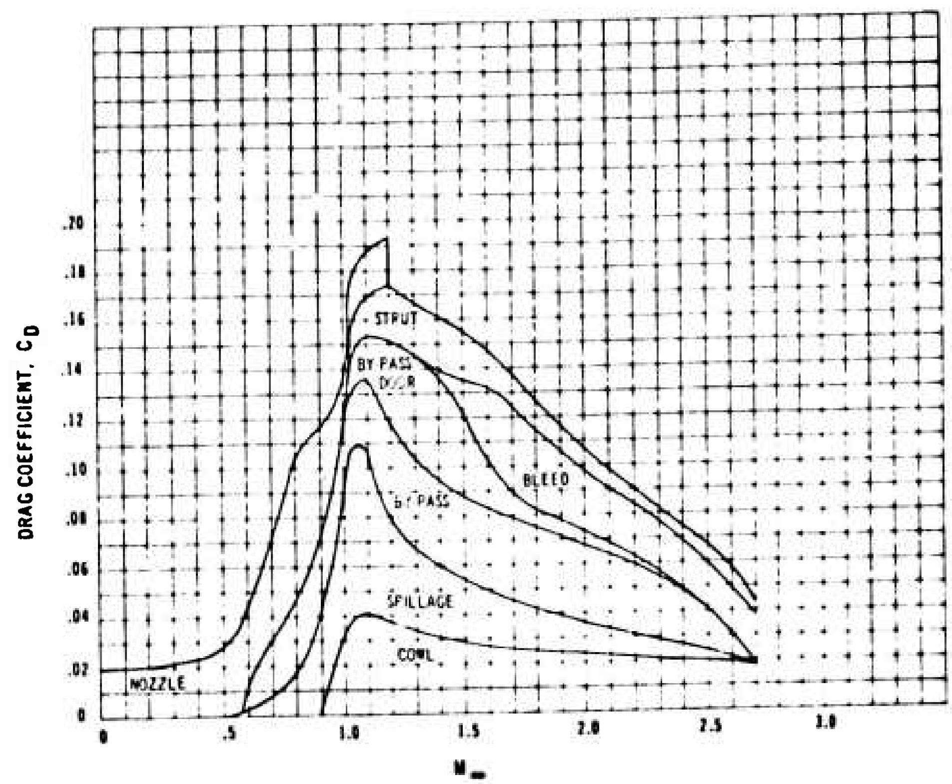
Cowl wave drag was computed by using a Boeing program which is an improvement of a Lighthill method for predicting surface pressures of axially symmetric bodies (Refs. 5-9). Inlet size was fixed for each engine, allowing for inlet boundary layer bleed and for local density in the wing pressure field. The cowl drag was computed for the under-the-wing cowl in a free stream ambient pressure field. This drag was then corrected for under-the wing pressure field as part of the airplane drag.

12.4.2 INLET SPILLAGE DRAG AND COWL LIP SUCTION FORCE

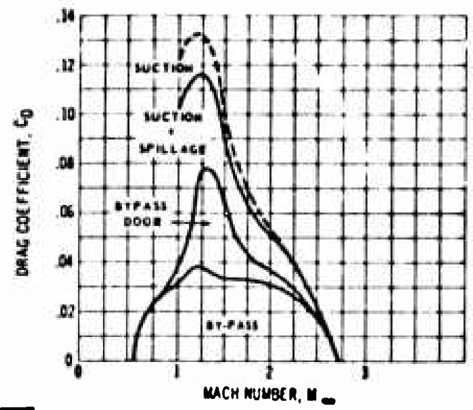
Inlet spillage drag was computed for all engines for a 12.5 degree non-translating centerbody with an independently variable throat, based on free stream spillage areas. Conical flow theory was used to compute spillage drag (Refs. 10 and 11). The cowl lip suction force associated with inlet spillage is included in the spillage drag. A breakdown of the spillage drag and the suction force is shown in Fig. 12-5.

12.4.3 INLET BYPASS SYSTEM MOMENTUM AND EXTERNAL BYPASS DOOR WAVE DRAG

The inlet supply air often exceeds the engine demand air. The excess air is expelled through low-angle bypass doors ahead of the compressor face. During transonic speeds, when the bypass doors are open, the discharge angle is 7 degrees relative to the cowl external surface. Wave drag for the external bypass doors was included in the bypass drag for an aspect ratio of one (Ref. 14). The drags are consistent with external bypass door drags in Refs. 15 and 16. The air momentum drag was computed for a convergent nozzle at 10 degrees (7 degrees plus 3 degrees cowl angle) relative to the axial direction. The maximum nozzle thrust coefficient is 0.965, which occurs at transonic speeds. The total pressure of the bypass air is 96 percent of inlet recovery total pressure (Fig. 12-1) at all Mach numbers.



12-4 *Installation Propulsion Pod Drag Coefficient*



12-5 *Inlet Excess Air Drag Coefficient*

12.4.4 INLET BOUNDARY LAYER BLEED DRAG

To provide good inlet performance after the inlet is started, boundary layer air is bled from both the centerbody and the cowl. At Mach 2.7 the amount bled and discharged overboard is five percent of the total inlet supply. The centerbody bleed is closed off below Mach 1.5. It is assumed that the cowl bleed is aerodynamically shut off below Mach 1.3 because of the low bleed pressure recovery. All bleed air is discharged overboard through convergent-divergent nozzles at 7 degrees from the axial direction. The nozzle exit-to-throat expansion ratio is 1.25. The bleed total pressure recovery is 0.3. The nozzle thrust coefficient at Mach 2.7 is 0.965.

12.4.5 EXHAUST NOZZLE BOATTAIL DRAG

Supersonic nozzle boattail drags were computed using the method of Ref. 6. For the GE4 J4C engine, the nozzle boattail angle schedule was the optimum which yielded the maximum installed climb thrust and minimum installed subsonic specific fuel consumption (SFC). Subsonic nozzle boattail drags, at maximum dry power setting, were based on test data presented in Ref. 12. Subsonic and transonic nozzle boattail drags for other power settings were based on data in Ref. 13. The nozzle boattail angle schedule used in computing installed performance is shown in Fig. 12-6.

POWER SETTING	MACH NO.	BOATTAIL ANGLE
MAXIMUM AUGMENTED	0 TO 1.0	10.2
	1.0 TO 1.2	2.9
	1.2 TO 2.7	0
DRY POWER	0 TO 0.9	15.2
	2.7	0

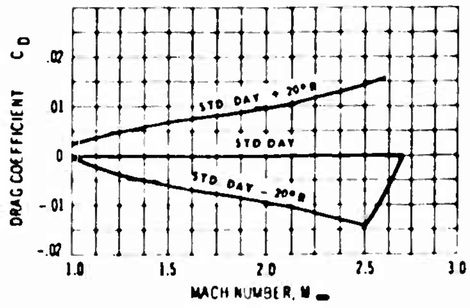
12-6 *Exit Nozzle Settings*

12.4.6 WAVE INTERFERENCE DRAG

The actual cowl wave drag is higher under the wing because of the local wing pressure field on the cowl and because of wave reflections between the cowl and the under surface of the wing. The pressure field under the wing increases the inlet spillage drag and decreases the inlet ram drag. This interference drag between the pod and the wing, and the associated wing lift, are included in the airplane performance and are more fully covered in Volume A-V, Aerodynamics.

12.4.7 NON-STANDARD DAY DRAG

For flight during hot or cold days the bypass system handles the difference in engine mass flow requirement. The resulting increase or decrease in bypass drag was computed using the method described in Par. 12.4.3. Fig. 12-7 shows the bypass drag coefficient increment for a plus and minus 20 degrees Rankine (R) day.



12-7 *Non Standard Day Bypass Drag Increments*

12.4.8 ENGINE SHUTDOWN DRAG

During supersonic cruise, if the engine is shut down, the windmilling brake will be applied, reducing the engine mass flow to 10 percent of normal. The controlled and secondary bypass doors will discharge the excess air from the inlet for stable operation. At Mach 2.7, the increase in pod drag coefficient for the bypass system and the nozzle boattail is 0.3884. The estimated internal drag coefficient increase is 0.0318 (based on braked J93 data).

During subsonic operations also, if the engine is shut down, the windmilling brake will be applied. The increase in pod drag coefficient for the bypass, external spillage, and boattail is 0.6782. The internal drag coefficient is 0.0080 (based on braked J93 data).

12.5 Engine Performance Data

The engine performance data were derived from Refs. 17 and 18. Addenda to these references have resulted from coordination between General Electric and Boeing.

The performance data are based on the 1962, U.S. Standard Atmosphere. Fig. 12-8 shows the design characteristics of the engine.

12.5.1 ENGINE OPERATION

The GE4 J4C engine is capable of continuous cruise operation at Mach 2.7 at maximum dry power. It is also capable of continuous cruise with augmentation.

12.5.2 STANDARD DAY INSTALLED ENGINE PERFORMANCE

• Takeoff: Maximum augmented thrust and maximum dry thrust are shown in Fig. 12-9 at sea level for true airspeeds up to 400 knots. The SFC for the above conditions are shown in Fig. 12-10.

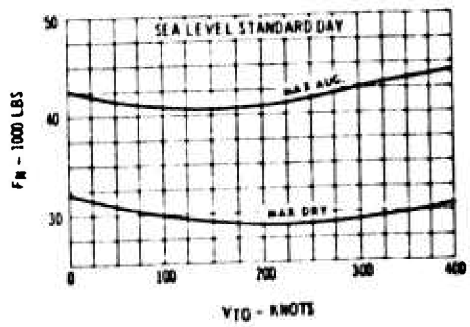
• Climb and Acceleration: Maximum augmented net thrust divided by incompressible dynamic pressure, (F_n/q) and SFC, versus Mach number for a range of flight speeds up to Mach 2.7 and climb altitudes above 15,000 feet, are presented in Figs. 12-11 and 12-12. Maximum dry F_n/q and SFC versus Mach number up to M = 0.9 are also included for altitudes from sea level up to 36,089 feet.

Partial augmentation F_n/q versus SFC for a range of Mach numbers at altitudes of 15,000, 25,000, 36,089 and 45,000 feet are shown in Figs. 12-13 through 12-16.

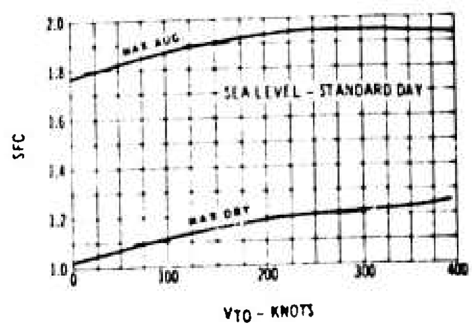
• Supersonic Cruise: The afterburning F_n/q versus SFC for a range of dry and augmented power settings are shown in Fig. 12-17 for M = 2.5, 2.7, and 2.9 at 65,000 feet. The altitude effect on F_n/q and SFC, from 55,000 to 75,000 feet, at M = 2.7, is shown in Fig. 12-18.

Sea level static standard day thrust (No Losses)	
Maximum Augmented	45,200 lbs.
Maximum Dry	34,000 lbs.
Engine dry weight, including exhaust nozzle and thrust reverser (airflow = 415 lbs./sec.)	7,077 lbs
Thrust/weight (sea level static)	
Maximum Augmented	6.4
Maximum Dry	4.9
Net thrust/Weight (transonic M_∞ = 1.5 45,000 FT)	
Maximum Augmented	2.9
Design Mach Number	2.7
Supersonic Cruise SFC, M_∞ = 2.7, 65,000 ft., Ram Recovery = .90	1.54
Subsonic Cruise SFC, M_∞ = .85, 36,150 ft., Ram Recovery = .986	1.23
Loiter SFC, M_∞ = .4, 15,000 ft., Ram Recovery = .986	1.47
Acceleration Net Thrust, M_∞ = 1.2, 36,089 ft.	23,3[illegible] lbs.
45,000 ft.	15,300 lbs.
55,000 ft.	9,500 lbs.
Reverse Thrust (% Maximum Dry Power)	40
Turbine Inlet Temperature (Nominal)	
Take-off	2200° F
Supersonic Cruise	2200° F
Transonic Acceleration	2200° F
Augmentation Temperature (Nominal)	
Take-off	3500° R Max.
Supersonic Cruise	3500° R Max.
Transonic Acceleration	3500° R Max
Compressor Pressure Ratio	9.5:1
Initial Tice Between Overhaul	600 TO 1000 HRS
Inlet Diameter	50.2 in

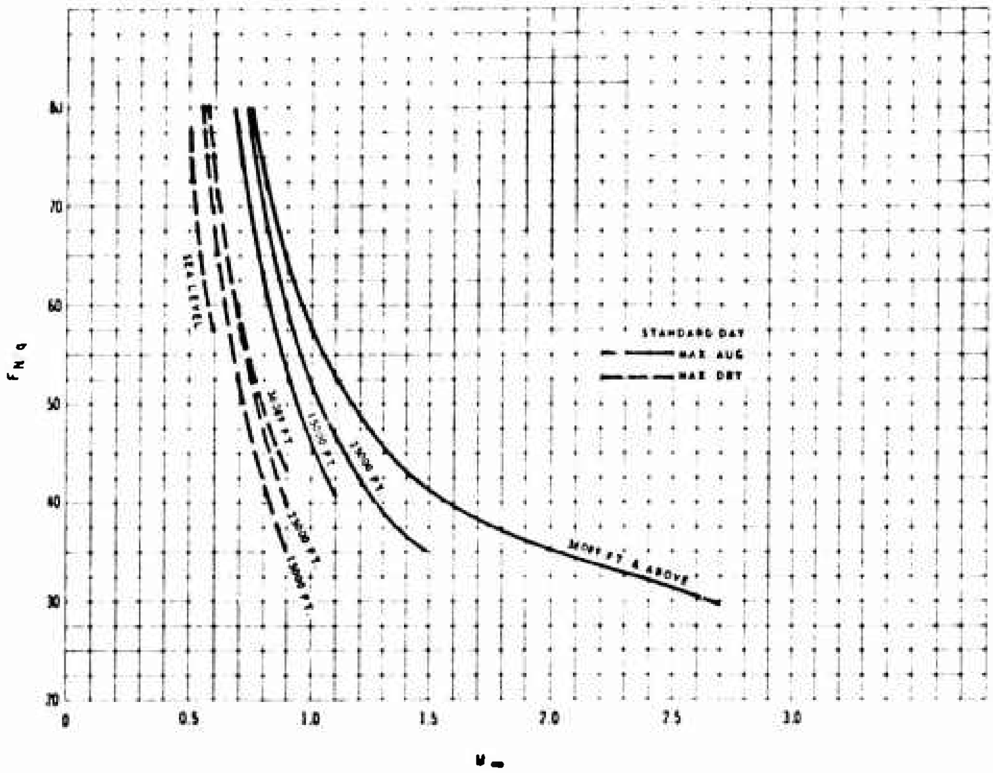
12-8 *Engine Characteristics*



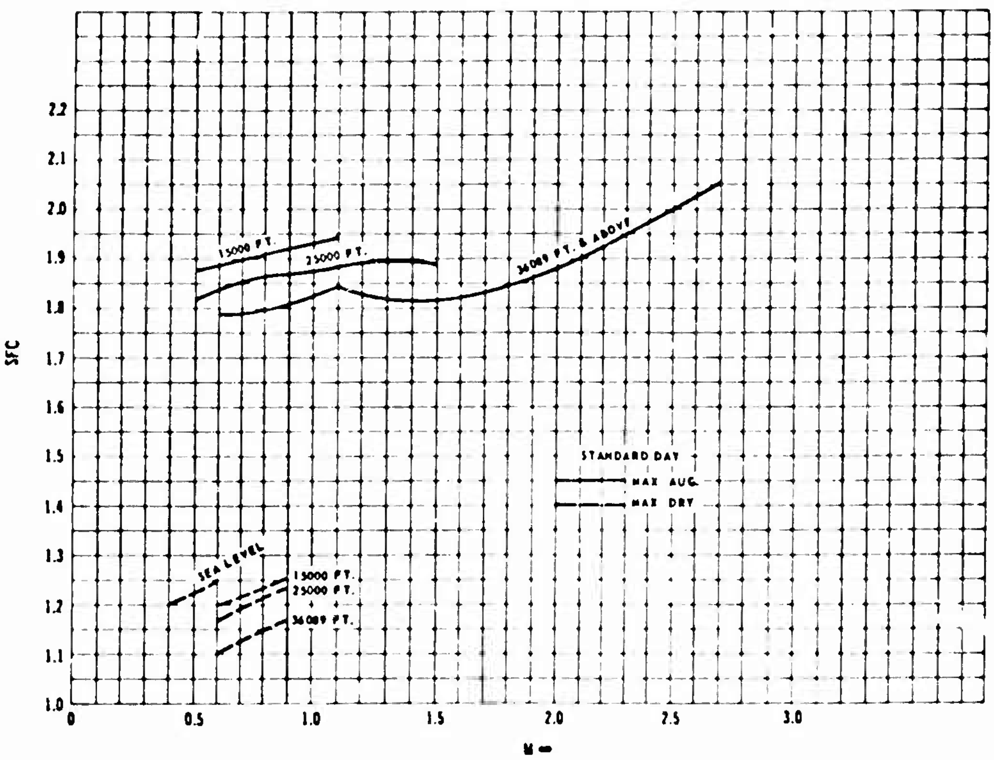
12-9 *Takeoff Thrust*



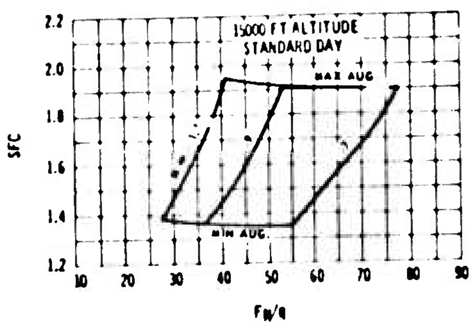
12-10 *Take-Off SFC*



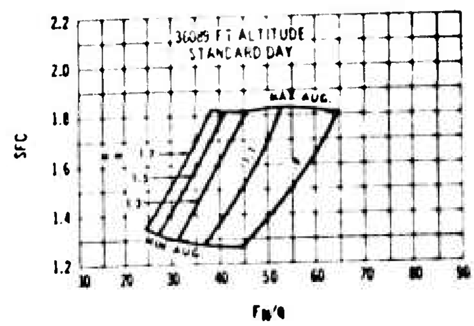
12-11 *Climb and Acceleration Net Thrust*



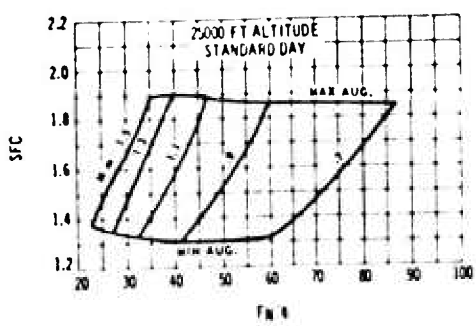
12-12 *Climb & Acceleration SFC*



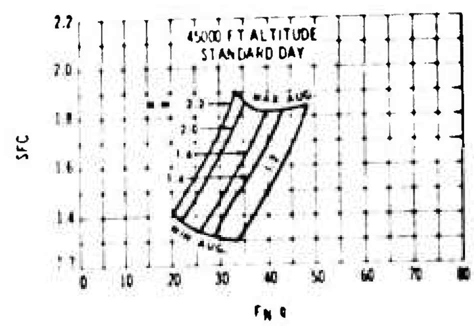
12-13 *Augmented Performance – 15000 FT*



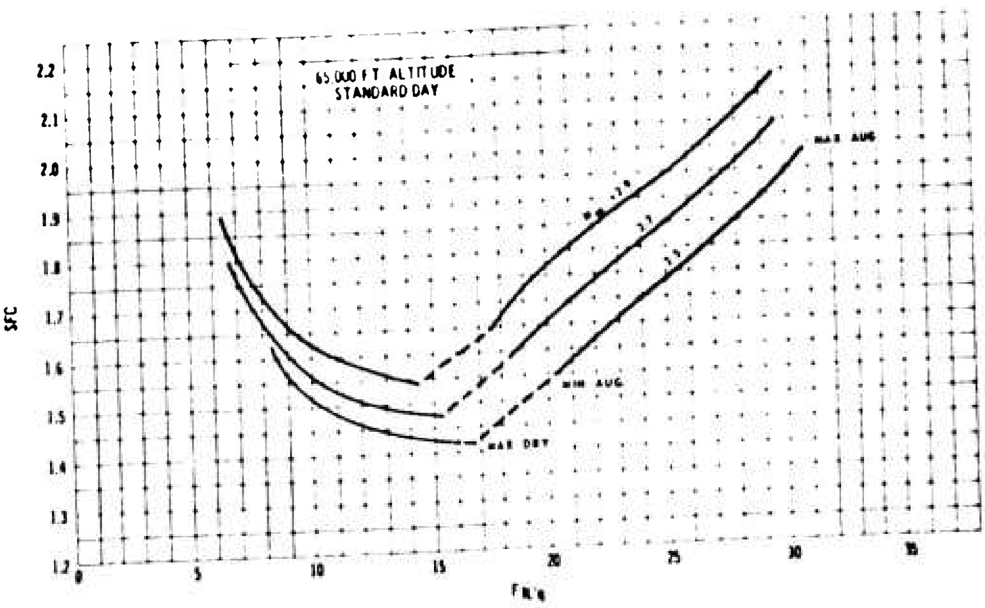
12-15 *Augmented Performance – 36089 FT*



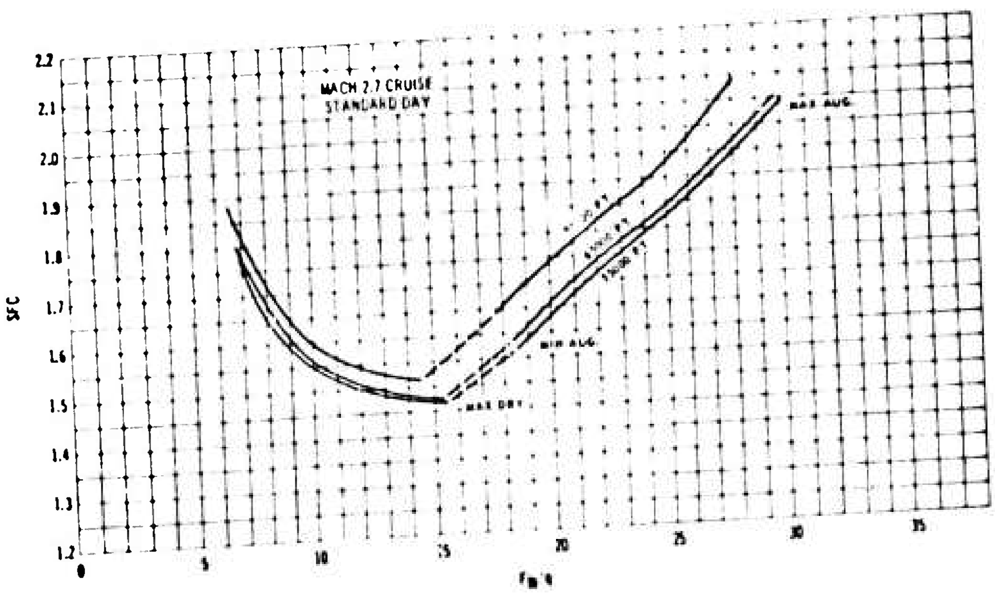
12-14 *Augmented Performance – 25000 FT*



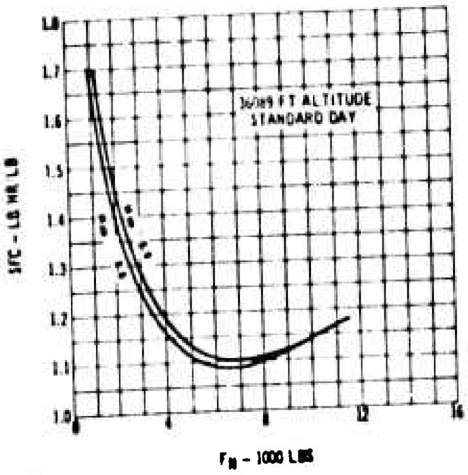
12-16 *Augmented Performance – 45000 FT*



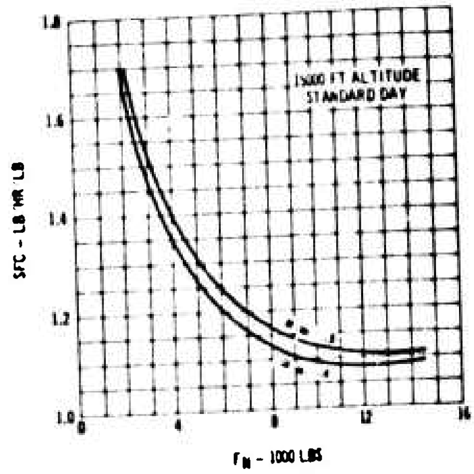
12-17 Supersonic Cruise Performance



12-18 Supersonic Cruise Performance



12-19 Subsonic Cruise



12-20 Subsonic Holding

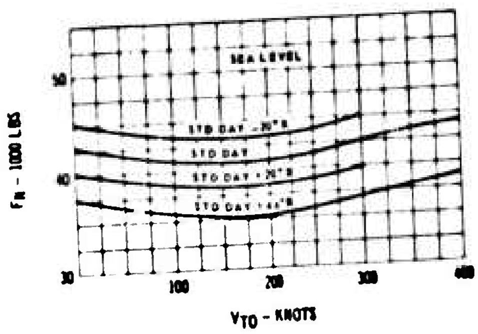
• Subsonic Cruise: The dry net thrust (F_n) versus SFC are shown in Figs. 12-19 and 12-20 for M ≈ 0.8 and 0.9 at 36,089 feet and for M ≈ 0.4 and 0.5 at 15,000 feet, which are typical cruise to alternate and holding operating conditions, respectively.

12.5.3 NON-STANDARD DAY INSTALLED ENGINE PERFORMANCE

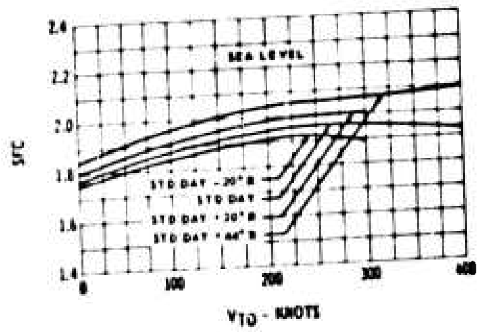
• Takeoff. Maximum augmented and maximum dry thrust for 39°, 59°, 79°, and 103°F at sea level up to 400 knots are shown in Figs. 12-21 and 12-22. SFC's for the above conditions are shown in Figs. 12-23 and 12-24.

• Climb and Acceleration. Maximum augmented and maximum dry F_n q and SFC for 15,000, 25,000, and altitudes over 36,089 feet for standard day plus 20°R and minus 20°R up to M ≈ 2.7 are shown in Figs. 12-25 and 12-26.

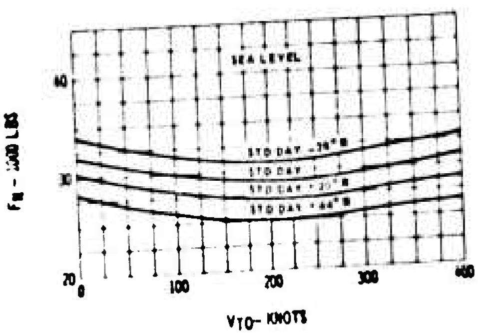
• Supersonic Cruise. The augmented and dry F_n q versus SFC for standard day plus 20°R and standard day minus 20°R at M ≈ 2.6 and 2.7, respectively, are shown in Fig. 12-27.



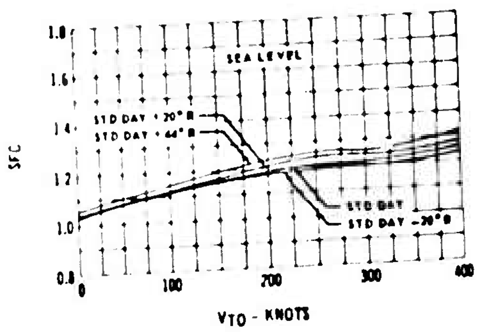
12-21 Max Aug. T.O. Thrust



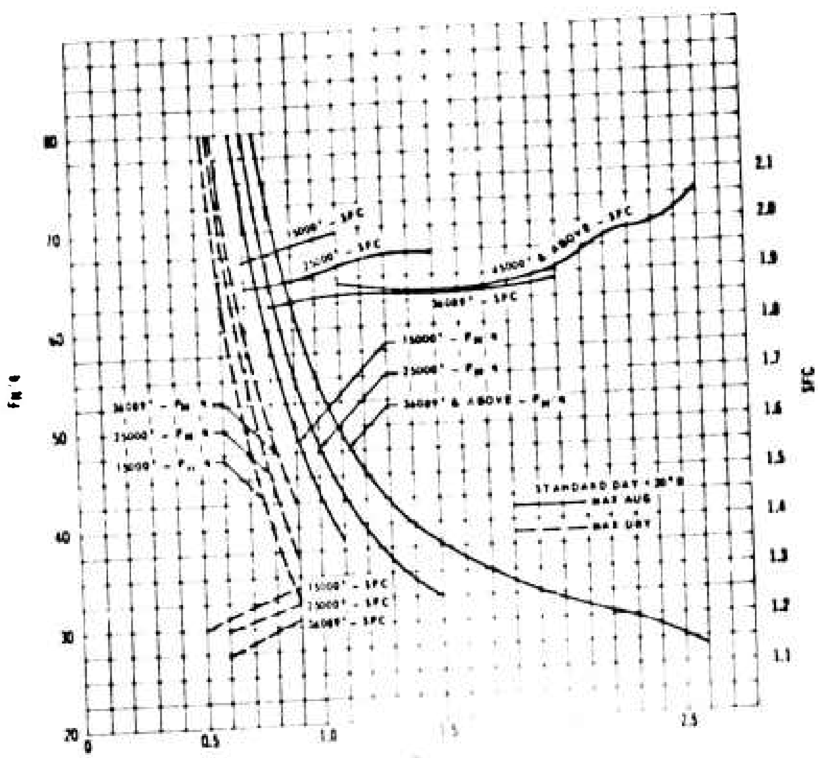
12-23 Max Aug. T.O. SFC



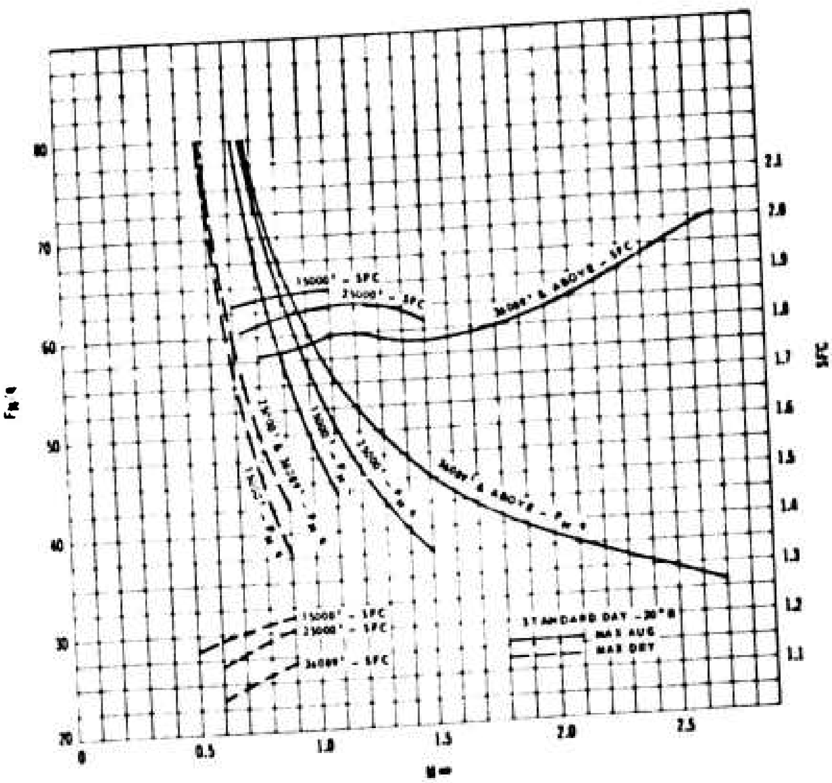
12-22 Max Dry T.O. Thrust



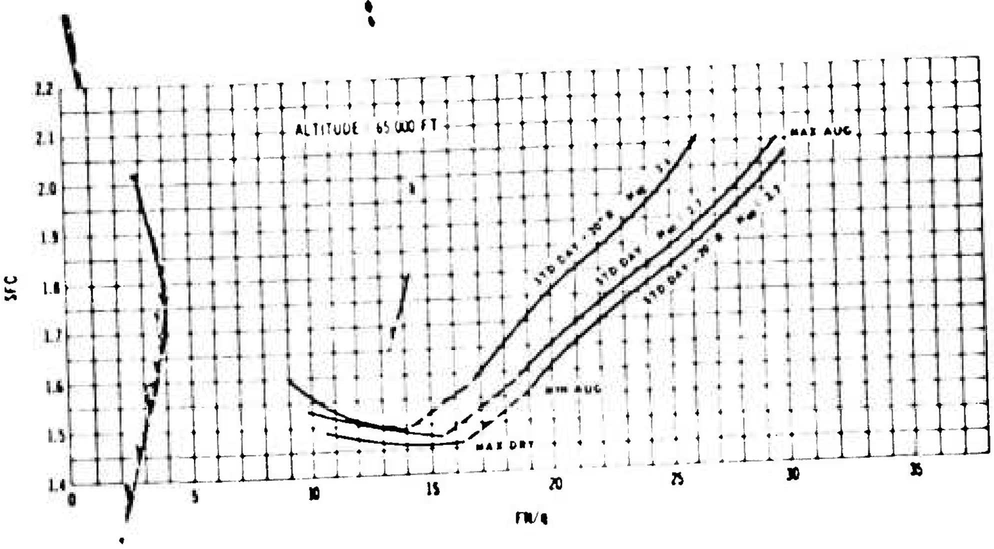
12-24 Max Dry T.O. SFC



12-25 Standard Day + 20 R Climb and Acceleration



12-26 *Standard Day -20°R Climb and Acceleration*



12-27 Non-Standa : Day Cruise Performance

13.0 REFERENCES

Copies of the following referenced data may be obtained by making a request to either:

The Boeing Company
Suite 120 Commonwealth Building
1625 K Street
Washington 6, D.C.
or
The Boeing Company, Airplane Division
P.O. Box 707
Renton, Washington
Attn: M. L. Pennell
Organization 6-2000
Mail Stop 73-60

The numbered references below are specifically referred to at appropriate places throughout the text of Volume A-VI.

The additional unnumbered references have provided data and concepts which have been evaluated and applied to the development of the propulsion system proposed in Volume A-VI.

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16. A. R. Vick: An Investigation to Determine the Discharge and Thrust Characteristics of Auxiliary-Air Outlets for a Stream Mach Number of 3.25; NASA TN D-1478, 1962

*17. General Electric Preliminary Performance Bulletin for the GE4 J4C Augmented Turbojet Engine, Specification, Number GEI 67870, January 15, 1964

*18. General Electric Performance Card Deck R63FPD377

*19. General Electric Engine Model Specification E2031, November 15, 1963

*Obtain from General Electric.

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  Report Nos. LR 16261-Series
  Contract No. AF 33(657)-9433
  Lockheed Aircraft Corporation
- Commercial Supersonic Transport Program Engine Cycle Study Program—Final Report
  (GE4 Engine Study)
  Report No. R63FPD222
  Contract No. AF33(6570-9232) Project No. 9056
  Flight Propulsion Division
  General Electric Company
  July 9, 1963
- Supersonic Transport Engine Data GE4 J4 Study A - Advanced Technology Turbojet Supersonic Transport Propulsion Operation, Advanced Engine and Technology Department
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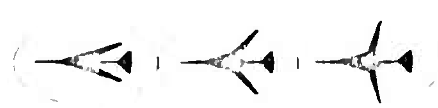

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- A-IV 10.2.5
- A-VI 8.0
- A-VII 4.4, 10.0
- A-IX 3.0

INDEX OF PROPOSAL DOCUMENTS TO REQUEST FOR PROPOSAL PARAGRAPHS (Continued)

INDEX OF PROPOSAL DOCUMENTS TO REQUEST FOR PROPOSAL PARAGRAPHS (Continued)

UNCLASSIFIED

CROSS REFERENCE INDEX

PROPOSAL DOCUMENTS - REQUEST FOR PROPOSAL (RFP)

This cross-reference index relates the paragraphs of the proposal documents to the Evaluation Factors and to the numbered paragraphs of the Request for Proposal. This index permits a reader to locate in the proposal documents each of the paragraphs which relate to any of the Evaluation Factors or any one of the paragraphs in the RFP.

The code uses the proposal document numbers as listed below:

Vol. No.	Subject
V-I	Summary
A-I	Airframe Work Statement
A-II	Model Specification
A-III	Aircraft Description
A-IV	Structural Report
A-V	Aerodynamic Report
A-VI	Propulsion Report
A-VII	Systems Report
A-VIII	Ground Support Equipment Report
A-IX	Test & Certification Plan
A-X	Aircraft Mockup and DEI Plan
A-XI	Flight Operation and Safety
M-I	Management
M-II	Management Controls
M-III	Product Support Plan
M-IV	Preliminary Production Plan
M-V	Development & Production Cost
M-VI	Direct Operating Costs

For example, if a reader is interested in Boeing's Development Plan as a part of the Master Plan as one of the Evaluation factors, he will find this subject discussed in:

Development Plan

V-I - paragraph 4.1
A-I - paragraph 2.1.6, 3.1.6
M-IV - paragraph 1.0, 1.3, 2.0, 6.0 and 7.0

If the reader wishes to know where paragraph 3.2.11.3 of the RFP, AUTOMATIC FLIGHT CONTROL SYSTEM, is described he will find the subject discussed in:

3.2.11.3 AUTOMATIC FLIGHT CONTROL SYSTEM

A-II - paragraph 8.4, 9.6
A-III - paragraph 9.7
A-VII - paragraph 3.3.3
A-IX - paragraph 4.2.6.3, 5.1.6.3
A-XI - paragraph 3.10

INDEX OF PROPOSAL DOCUMENTS TO EVALUATION FACTORS